A new model-free method performing closed-loop fault diagnosis for an aeronautical system

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Abstract: Fault diagnosis for a closed-loop controlled aeronautical system is considered. The geometry of the problem, the nonlinear model of the aircraft and two guidance laws are described. A new approach to perform fault detection, based on the study of the control signals, is then introduced. The method proposed allows the detection and isolation of actuator faults. Simulation results with measurement uncertainty illustrate the relevance of the approach. The influence of the guidance law and perspectives are discussed.

Keywords: aircraft, closed-loop system, fault detection and isolation, fault diagnosis, guidance and control.

1. INTRODUCTION

In-flight securement of aircrafts can be performed either by completing on-board equipment with additional measurement devices or by means of algorithmic procedures exploiting the available measurements. In this paper, we aim at identifying early unexpected changes (faults) in an aerial system before they lead to a complete breakdown (failure), without adding sensors. This objective should be fulfilled by applying Fault Detection and Isolation (FDI) methods to a class of aerial systems with typical sensors and actuators. The set of devices is fixed and no hardware redundancy is allowed. A nonlinear knowledge-based dynamical model of the vehicle in state-space form is also available, but will only be used to design the control law.

FDI for aerial and space vehicles have already been addressed in many papers (see, e.g., Patton (1991)). Most of the work of the diagnosis community has been focused on linear systems but recently there has been some trend toward fault detection methods for nonlinear systems (see, e.g., Witczak (2007)). In Marzat et al. (2009), a survey of the main FDI methods is presented, where both model-free and model-based methods are described and the most promising approaches for the aircraft securement problem in question are shortlisted.

However, all these approaches are passive as they solely check the consistency between the inputs and outputs of the monitored system. The control is ignored in this framework although it may lessen the impact of the failing components, by modifying the fault dynamics. Control information provides additional insight on the system behavior and should be exploited. This is mainly performed either by applying active methods or by studying the closed-loop behavior.

Active fault diagnosis injects an auxiliary input into the system. The goal of this additional test signal is to distinguish faulty behavior from the normal one (Campbell and Nikoukhah (2004)). Designing this input can be seen as a multi-objective optimization problem. The signal should be large enough to permit detection, but also small enough to avoid performance degradation. This study has been carried out for linear models only (Niemann (2006)). The use of feedback information to design the auxiliary input has been investigated recently (Ashari et al. (2009)). The major drawback of this approach is that the parameters of the two models (nominal and faulty) are supposed to be known, which is scarcely the case in real-world applications. Concerning aerospace systems, an interesting attempt is reported in Bateman et al. (2007), where an active diagnosis procedure for an UAV (unmanned aerial vehicle) with redundant flight control surfaces is proposed. Each actuator is excited with a specific signal such that the global resultant forces and moments are negligible if there is no fault.

To avoid the complex design of an additional input, an interesting alternative is to study the behavior of the system and its faults within the closed loop. The effect of feedback on robustness of fault diagnosis methods has been analyzed by Niemann and Stoustrup (1997) and more recently by Baïkeche et al. (2006). For linear transfer function models, it has been shown that model uncertainty or multiplicative faults make the generated residuals depend on the control signal. A trade-off must be achieved between fault detection and performance of the closed-loop system. This is closely related to dual control, where a control law is designed in order to enhance a particular property of the system such as observability or diagnosability while preserving performance (see, e.g., Wittenmark (1995)).

In this paper an alternative approach is described following the remark in Baïkeche et al. (2006): in a feedback-controlled system, the control input is the signal containing the most information concerning faults. The originality of the present work is to study the fault detection problem from the point of view of a successful mission. If control fails, then faults may affect the regulated system. This paper shows how it is possible to detect and isolate them by analyzing the adequacy of the system response. Contrary to the active or control-loop based methods, we do not use any auxiliary input or trade-off controller that can affect performance. The control law is designed to fulfill the mission requirements and the monitoring is based on residuals showing adequacy to control objectives. The approach can be said to be model-free, as these signals are only functions of the measurements and do not rely on the explicit model of the system.

Section 2 describes the aeronautical case study. The model of the system, including its sensors and actuators is presented. Guidance and control laws are described along with the geometry of the problem. The set of possible faults affecting the system is also specified. Section 3 explains the principles of the new approach and presents the expression of residuals for guidance and control laws previously given. Section 4 describes the flight scenario, and the results of the approach for the considered aircraft. Finally, conclusions and perspectives are detailed.

2. MODEL

The case study considered here is based on the aeronautical benchmark described in Marzat et al. (2009). It involves a six-degree-of-freedom surface-to-air missile but only longitudinal movement is considered here. The autonomous vehicle aims at intercepting a non-maneuvering target whose position and speed are measured. It could also be seen as the following of waypoints by an UAV.

In this planar case, two means of control are used. One is a rudder whose axis is perpendicular to the plane where the vehicle flies and the other is propulsion rate. The main sensor is an Inertial Measurement Unit (IMU) comprising gyrometers and accelerometers, which is part of an Inertial Navigation System (INS). These standard components are preset and there is no hardware redundancy.

2.1 State-space model

- b is the inertia term,
- $[x, z]$ is the position in the inertial frame,
- $[v_{bx}, v_{bz}]$ is the speed in body coordinates,
- θ is the orientation in the (x, z) plane,
- q is the angular velocity,
- δ_m is the rudder deflection angle,
- η is the propulsion rate,
- $Q = \frac{1}{2}\rho(v_{bx}^2 + v_{bz}^2)$ is the dynamic pressure,
- $\alpha = \arctan(\frac{v_{bz}}{v_{bx}})$ is the angle of attack,
- m is the aircraft mass,
- $f_{\min}$ and $f_{\max}$ are constants of the propulsion model,
- s_{ref} and l_{ref} are characteristic dimensions,
- $c_{(.)}$ are the aerodynamic coefficients.

With this notation, the dynamics is described by the following state equations.

$$\begin{cases} \dot{x} = \cos(\theta)v_{bx} + \sin(\theta)v_{bz} \\ \dot{z} = \cos(\theta)v_{bz} - \sin(\theta)v_{bx} \\ \dot{v_{bx}} = -qv_{bz} - \sin(\theta)g - \dfrac{Qs_{\text{ref}}}{m}\left[c_{x0} + c_{xA}\alpha + c_{x\delta_m}\delta_m\right] \\ \qquad + \dfrac{1}{m}\left[f_{\min} + (f_{\max} - f_{\min})\eta\right] \\ \dot{v_{bz}} = qv_{bx} + \cos(\theta)g - \dfrac{Qs_{\text{ref}}}{m}\left[c_{z0} + c_{zA}\alpha + c_{z\delta_m}\delta_m\right] \\ \dot{q} = \dfrac{Qs_{\text{ref}}l_{\text{ref}}}{b}\left[c_{m0} + c_{m\alpha}\alpha + c_{m\delta_m}\delta_m + \dfrac{l_{ref}}{\sqrt{v_{bx}^2 + v_{bz}^2}}c_{mq}q\right] \\ \dot{\theta} = q \end{cases}$$

where the aerodynamic coefficients $c_{(.)}$ are piecewise continuous nonlinear functions of the Mach value and angle of attack. The state vector is $\mathbf{x} = [x, z, v_{bx}, v_{bz}, q, \theta]^{\text{T}}$, the input vector is $\mathbf{u} = [\delta_m, \eta]^{\text{T}}$. This model belongs to the general class of nonlinear control-affine systems. Note that the design of the FDI methodology will not involve these equations, which are only used to tune the control law and simulate the process.

2.2 Measurements

The IMU provides measurements of non-gravitational acceleration and angular rate. The INS integrates these signals to estimate position, speed and orientation. The output vector is thus $\mathbf{y} = [x, z, \dot{x}, \dot{z}, v_{bx}, v_{bz}, \theta, q, a_{bx}, a_{bz}]^{\text{T}}$. Speed and position of the target are measured either with a ground radar or with an embedded tracking sensor. The target information vector is $\mathbf{c} = [x_c, z_c, \dot{x_c}, \dot{z_c}]^{\text{T}}$.

2.3 Guidance and control

Interception involves finding a control law such that the distance between the missile and target goes to zero. The classical approach defines, in a decoupled way, a guidance law and then an autopilot that implements the acceleration orders. The interception criterion can be expressed in various geometrical ways, which leads to several guidance laws. In order to describe them, we need the following definitions:

- The vector between the missile and the target is the *line of sight* (LOS): $\mathbf{r} = [x_c - x, z_c - z]^{\text{T}}$.
- The opposite of the LOS derivative is the *closing velocity*: $-\dot{\mathbf{r}} = [\dot{x} - \dot{x_c}, \dot{z} - \dot{z_c}]^{\text{T}}$.
- λ is the *LOS angle*, and the *LOS rate* is $\dot{\lambda} = \frac{\mathbf{r} \times \dot{\mathbf{r}}}{\mathbf{r}^2}$.
- The missile velocity in the inertial frame is denoted as $\mathbf{v}_{\text{M}} = [\dot{x}, \dot{z}]^T$.

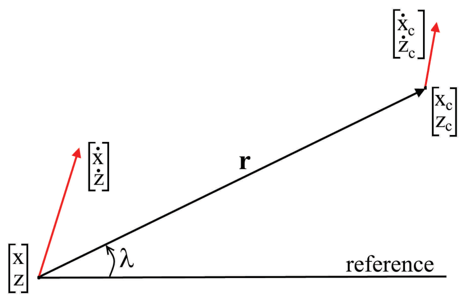

Fig. 1. Geometry of the problem

Guidance laws We consider two classical guidance laws, namely *pure pursuit* (PP) and *proportional navigation guidance* (PNG). The geometrical rules of both laws are given next (see Shneydor (1998) for more details).

- *Pure pursuit:* PP makes the velocity of the pursuer $\mathbf{v}_\mathrm{M}$ coincide with the LOS $\mathbf{r}$. This is the first guidance law that has been developed, inspired by how predators catch their prey. The simplest guidance consign is then to have the acceleration input proportional to the angle between the aircraft velocity and the LOS. This is known as *velocity pursuit*. Another version aims at aligning the axis of the vehicle on the LOS: this is *attitude pursuit*. In this study, velocity pursuit will be used.
- *Proportional navigation guidance:* PNG achieves parallel navigation. The geometrical rule is to keep the direction of the LOS constant relative to inertial space, i.e., the LOS is kept parallel to the initial LOS. This is also called *constant bearing* and the rule could be very concisely stated as the LOS rate being equal to zero. The guidance consign is taken proportional to the LOS rate.

There exist many variations of these two guidance laws (Lin (1991)). PP is less sensitive to noise than PNG, but PNG is more effective than PP because of its geometrical rule (PNG is almost following the optimal straight path). PNG is the most used law in the majority of guidance problems.

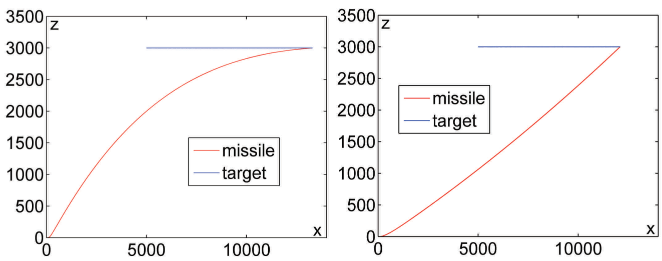

Fig. 2. Interception with PP (left) and PNG (right)

Control An autopilot has been designed in order to achieve the guidance acceleration. Propulsion rate is obtained by a proportional regulation of the velocity to a reference. A linear controller computes the appropriate rudder angle.

2.4 Failures

We consider two types of actuator faults: the loss of half of the thrust due to an engine problem, and the locking of the rudder. Sensor faults are not considered here and will be addressed elsewhere.

3. APPROACH

3.1 Principles

The starting point is to acknowledge that control signals hold relevant information concerning faults in closed-loop systems. The controller is effectively trying to compensate for faults so as to reach its initial goal. For our interception problem, the success of the mission is a safety-critical operation. The fault diagnosis module should give enough information to decide whether to abort the task. Therefore, the diagnosis problem can be seen as that of assessing the adequacy of the system to fulfill the control requirements. It is equivalent to assuming that there is no fault if the controlled system is successful. How to detect faults in this context is now explained.

The fault detection method checks whether the specific geometrical rules of the guidance law are satisfied. Guidance residuals, for the two laws described in Section 2.3, are given in Section 3.2. Another signal that shows the behavior of the propulsion rate and will be used to build a residual is proposed in Section 3.3. These signals have predictable behavior when the guidance objective is fulfilled (equivalent to the absence of faults with our point of view). Mathematical expressions for all the residuals are in Table 1.

3.2 Guidance residuals

Pure pursuit Two residuals can be computed for this guidance law.

- The vehicle velocity should be aligned with the LOS. The first residual r_pp1 is the angle between the speed of the aircraft and the LOS.
- In both attitude and velocity pursuit, the missile is required to turn at a rate equal to the LOS rate. Therefore, the angular velocity should be equal to the LOS rate for low maneuvering targets. This gives our second residual, r_pp2.

Proportional navigation The only computable residual r_png for this law is equal to the LOS rate, which should be equal to zero.

3.3 Propulsion signal analysis

The residuals described in the previous paragraph, monitor the behavior of the rudder. We also need checking the effectiveness of propulsion. A fault on the propulsion system is usually sudden. The relevant signal to analyze this process, r_prop, is the quotient of the remaining distance (the LOS module) to the current speed of the vehicle. When such a fault occurs, there will be an abrupt change in the convergence rate of this signal.

Table 1. Expression of the residuals

$$r_\mathrm{pp1} = \frac{180}{\pi} \arcsin\left(\frac{\mathbf{v}_\mathrm{M}}{\|\mathbf{v}_\mathrm{M}\|} \times \frac{\mathbf{r}}{\|\mathbf{r}\|}\right)$$

$$r_\mathrm{pp2} = \left\|q - \dot{\lambda}\right\|$$

$$r_\mathrm{png} = \left\|\frac{\mathbf{r} \times \dot{\mathbf{r}}}{\mathbf{r}^2}\right\|$$

$$r_\mathrm{prop} = \frac{\|\mathbf{r}\|}{\|\mathbf{v}_\mathrm{M}\|}$$

3.4 Residual analysis

This section presents the residual analysis procedure used to decide if faults have occurred.

Guidance residual If the control objective is achieved, we expect the guidance residuals to be small. They cannot be exactly equal to zero because of the configuration of the problem: the target is moving and the actuator response is not instantaneous. Nevertheless, we have a good idea of the expected response of the system when the guidance law is designed. Based on that prior knowledge and the uncertainty level of the sensors (given by the manufacturer), a simple threshold decision logic can be chosen. Note that, once the convergence of the residuals is achieved, the signals do not vary much. It seems then interesting to monitor the derivatives of the residuals (if the noise level allows it) or to use a statistical test to detect a drift in the mean. This is not the principal scope of this paper but the decision issue should be further addressed to enhance robustness.

Propulsion residual r_{prop} is another kind of signal. In normal conditions, this "time-to-go" is regularly decreasing following a globally linear slope. When a propulsion loss appears, the slope changes. Basically, the aircraft is slowed down by this fault and thus the interception may still be possible but later than expected. To identify this slope change, we propose the following adaptive slope tracker algorithm which estimates (with a least-square method) the parameters of the slope on an interval, and uses them to predict the expected values of r_{prop} on the next interval. The mean squared error between the prediction and the real value is computed on this next interval. We thus obtain one error value for each interval, except the first one. This error signal is expected to be small in normal operating conditions and large for the interval where the slope change occurs.

Algorithm 1. Online slope change detection

$n \leftarrow$ number of points in each interval
$k \leftarrow 1$
$e_{\text{prop}} \leftarrow 0$
$[a_1, a_2] \leftarrow [0, 0]$
loop

- On interval $[k, k+n]$,
  estimate $[a_1, a_2]$ such as $r_{\text{prop}} = a_1 t + a_2$
- On interval $[k+n, k+2n]$,
  $e_{\text{prop}}(k) = \frac{1}{n}\sum_{j=k+n}^{k+2n}[r_{\text{prop}}(j) - (a_1 j + a_2)]^2$
- $k \leftarrow k+n$

end loop

4. SIMULATION RESULTS

This section shows the results of the method presented in Section 3 on the model detailed in Section 2. The simulation time step is $0.01s$.

4.1 Flight scenario and fault occurrences

Flight conditions are given in Table 2 and faults considered in Table 3. Successful interception trajectories have been shown in Figure 2. Missed interception for both laws due to rudder locking is displayed in Figure 3. Note that the propulsion fault may just delay the interception, but could also make it fail if the missile becomes slower than its target. The rudder mainly controls the vehicle orientation, while the thrust influences the norm of the velocity. The guidance residuals contain only angle or angular velocity information and thus will strongly react to rudder locking. The propulsion residual is function of the velocity norm and will be affected by propulsion loss only.

Table 2. Flight conditions

	Nominal speed	Initial position
Target	150 m/s	$x = 5000$ m $z = 3000$ m
Missile	270 m/s	$x = 0$ m $z = 0$ m

Table 3. Faults considered

Component affected	Propulsion	Rudder
Fault	50% loss	lock-in-place
Interpretation	engine failure	power failure, freezing
Instant	18 s	20 s

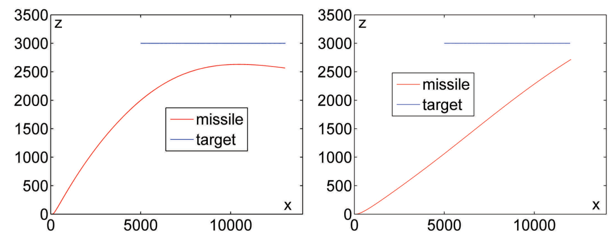

Fig. 3. Missed interception with PP (left) and PNG (right)

We now describe the uncertainty affecting the IMU and the target tracking device. IMU measurements are subject to various errors such as biases, scale factors and noise. Considering, for example, a one-axis sensor $\tilde{q}$ measuring the roll rate q, the measure is modeled as: $\tilde{q} = k_q q + b_q + N(0, \sigma_q^2)$ where k_q is the scale factor, b_q the bias and σ_q the standard deviation of a zero-mean Gaussian white noise. These three parameters (for each sensor) are characteristic of the IMU and should fall within a set of values provided by the manufacturer. The target tracking device is supposed to be only affected by zero-mean Gaussian white noise.

4.2 Results

The first case we consider is the locking of the rudder alone.

Pure pursuit guidance Remember that we have two residuals for the PP guidance law. The first one, r_{pp1}, is not much affected by measurement uncertainty because it only uses the measured position of the target and the velocity measurements provided by the aircraft IMU/INS. The second residual, r_{pp2}, is much noisier because its computation requires both position and velocity of the target. The two signals, filtered with a fourth-order Butterworth low-pass filter to reduce the noise effect, are shown in Figure 4. Both residuals are reacting to the fault in the same time scale.

Proportional navigation guidance The r_{png} signal is very noisy, due to the use of the measurements of target position and speed, along with IMU outputs. Therefore, if we use a simple threshold logic we have to choose a relatively high value to avoid a high false-alarm rate. Figure 5 displays the r_{png} signal in normal and faulty circumstances. It has been filtered the same way as the other residuals. It would probably be preferable to use a statistical test to detect an abrupt change in the mean of this residual. This remark also stands for the processing of r_{pp2}.

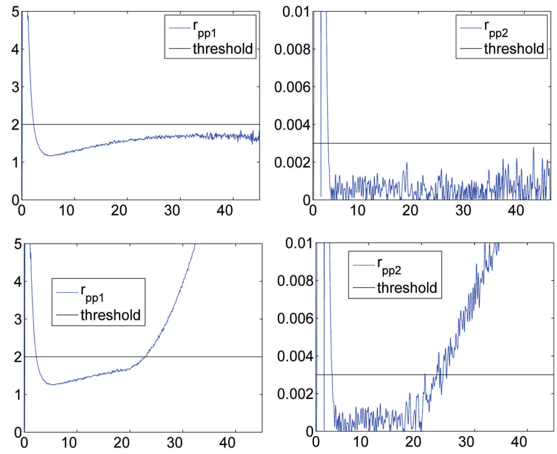

Fig. 4. Pursuit residuals: no fault (up) and rudder locked (down)

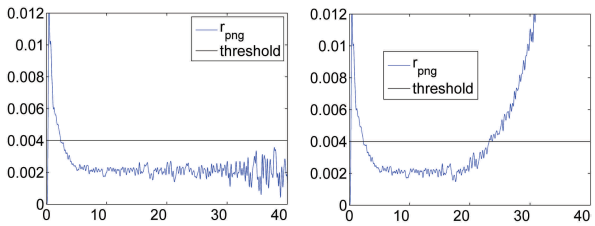

Fig. 5. PNG residual: no fault (left) and rudder locked (right)

Thrust residual Consider now the 50% loss of propulsion. The residual r_{prop} that should monitor the propulsion behavior depends little on the guidance law. This is why only the results with PNG are shown on Figure 6, as the results with PP should be similar. On this figure is also displayed the error signal e_{prop} obtained with Algorithm 1, which is actually our residual. The detection is achieved with a delay equal to the width of the interval window. Here a window of 100 points has been chosen, leading to a one-second delay.

Chosen thresholds and corresponding detection delays for all the residuals are shown in Table 4. Two choices are given for each threshold: an optimistic one and a conservative one. Note that the detection delay for the rudder fault may appear high, but it is justified by the incipient behavior of the fault.

Finally, we consider the two faults successively: first the rudder locking at 18s then the thrust loss at 20s. Figure 7 shows that each residual (e_{prop} to the propulsion and

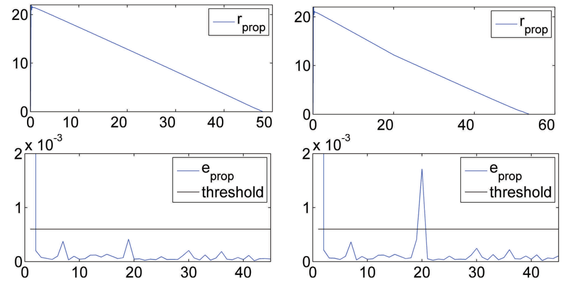

Fig. 6. Propulsion residual: no fault (left) and half propulsion (right)

Table 4. Residual analysis and detection delays

Residual	Threshold values	Detection delay
r_{pp1}	1.75°	3.6 s
	2°	5.3 s
r_{pp2}	0.002 s^{-1}	2.3 s
	0.003 s^{-1}	4.86 s
r_{png}	0.003 s^{-1}	2.5 s
	0.004 s^{-1}	5.19 s
e_{prop}	> 0.0005	1 s

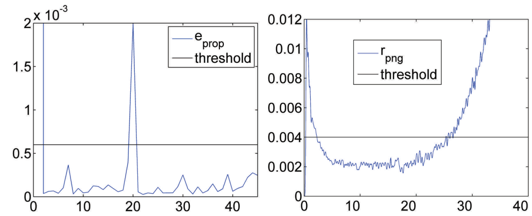

Fig. 7. Two successive faults: propulsion (left) and rudder (right) residuals

r_{png} to the rudder) is sensitive to only one fault. This demonstrates the possibility of fault isolation.

The PP law allows a more reliable decision than with only the PNG residual, because its two available residuals contain little noise. However, proportional navigation guidance has been found to be more efficient in most applications: miss distance and interception time are lower than those obtained with PP. Selection of either law becomes a trade-off between efficiency and safety.

4.3 Elements of comparison

The newly proposed method has to be compared with other approaches to evaluate its potential as objectively as possible. A first comparison strategy is to extract results from papers on closely related topics. However, models and simulation conditions differ so the exercise is limited in scope to some rather general statement. For example, the active method described in Bateman et al. (2007) is applied to an UAV in order to detect rudder locking. The results reported in this paper show detection delays comparable to our values. This active method, however, requires hardware redundancy and may destabilize the system with the additional inputs involved. On the contrary, the missile involved in our case study is equipped with non-redundant devices and leaves the control law unchanged.

A complementary comparison strategy, which is being carried out but requires much more time, is to implement all the approaches to be compared on the same significant test case. The system used in the present paper is part of a generic test-case for in-flight FDI, which has classical dynamics, sensors, actuators and faults scenarios (cf Marzat et al. (2009)). The results of the candidate approaches should be evaluated with respect to detection delays, but also to false-alarm and non-detection rates, and computational complexity. The development of a systematic comparison method between diagnosis approaches is seeked for. Such a method should tune each diagnosis procedure to its best before comparing them on the basis of performance indices such as those mentioned above. The design of this methodology is beyond the scope of this paper but at the heart of our future work.

Nevertheless, the way our method is built already gives us some information on its usefulness. Model-based methods will require more knowledge of the system, more parameters to be tuned and more computational complexity than the one proposed here, which should be particularly interesting when no model is available, or when the model turns out to be too complex to be employed.

5. CONCLUSIONS

A model-free fault diagnosis method based on the study of closed-loop control signals has been proposed. This strategy has been applied to a realistic aeronautical benchmark composed of an aircraft with typical non-redundant sensors and actuators, described by its nonlinear longitudinal model. These first results are encouraging, but further investigations are needed to enhance the decision process (with, e.g., statistical considerations). The tuning of the thresholds should not only consider detection delay but also other performance indices such as false-alarm and non-detection rates. The effect of possible disturbances (wind) may be taken into account. Note that these robustness issues stand true for every diagnosis method and no simple solution is available. To clarify the situation, the design of a systematic comparison method between diagnosis approaches is to be addressed in the immediate future.

More generally, the new approach we have described has attractive properties compared to those described in Section 1. Active fault diagnosis may cause stability or controllability problems, due to unknown dynamics induced by additional inputs. Our method exploits the guidance constraints that form the basis of the elaboration of the control law. Therefore no performance reduction is observed, contrary to methods that try to combine control and diagnosis. If the control law is correctly designed according to the mission and physical constraints, fault diagnosis can be performed easily within the described framework. The model-free nature of the approach allows its application to every kind of model and not linear ones only.

Extension to the three-dimensional case should be straightforward. We will then have three different residuals that should allow detection of faults on each rudder (and partial isolation, due to coupling), along with the propulsion residual to identify thrust trouble.

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