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Quantitative Model-Free Method for Aircraft Control System Failure Detection

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Abstract. The problem of the failure detection in the aircraft control system in the presence of disturbance is considered. A history based model-free nonstatistical method using the aircraft control and state data measurements only is proposed. The method needs no a priori information about the model of an aircraft, solving the prediction, identification and training problems.

1 Introduction

Faults in the aircraft control system are the most dangerous and can lead to an accident. In the event of such faults aerodynamic coefficients of the aircraft and moment characteristics of the control surfaces are changed. An important problem is to detect the abnormal dynamics of the aircraft as fast as possible.

As a rule, for the control system fault detection we use methods implying the existence of any priori information about aircraft model parameters. These methods employ three different approaches for the fault detection. The first approach is based on determining some model invariants, the second is based on solving the prediction problem, and the third is based on analytical redundancy [1–3].

In such model-based methods the parameter errors in aircraft models inevitably increase the threshold values of the fault detection criteria, thus increasing the time of the fault detection and decreasing the accuracy of determining the time the fault occurred. The derivation of error-free aircraft models proves to be practically a very hard problem [4].

The methods that do not use any priori information about the model may be qualitative or quantitative. Qualitative model-free methods are subjective analyzing the behavior of processes or employing expert systems.

Quantitative ones can be subdivided into statistical and nonstatistical methods. The statistical methods, which themselves are subject to inevitable errors, include principal component methods, partial least square methods, and methods based on classification algorithms. Determination of accurate and reliable solutions using statistical algorithms requires a large amount of data. They are characterized by high computational costs and response times.

The well-known nonstatistical quantitative model-free methods include methods based on artificial neural networks and genetic algorithms only. They require preliminary training/tuning for a particular aircraft.

The nonstatistical quantitative model-free method that does not need training is described in [5]. It uses only the control signals and data measuring of aircraft motion parameters. It’s needed no a priori information about aircraft parameters and is based on an algebraic solvability condition for the problem of identifying the aircraft mathematical model. The main disadvantage of this method is its low reliability under disturbances. The paper develops this method to make it valid in case of external bounded disturbances.

2 Problem formulations

Let the model of the nonfaulted aircraft be represented in the state space as [5]:

$$x_{i+1} = Ax_i + Bu_i + m_o , \quad (1)$$

where A, B are the matrices of eigen dynamics and control efficiency; x is the state vector of length n_x; u is the output signal of the control system that, if no faults occurred, coincides with the control deflection vector of length n_u; $m_o = Bu_o$ is the vector of the constant coefficients that depend on the trim deflections of the controls; u_o is the vector of the trim deflections corresponding to the equilibrium state of the aircraft; $i = \overline{0, l-1}$ is the discrete time before the occurrence of fault; and l is the instant a fault occurs.

When fault occurs in the control system, the model of the aircraft is rewritten as

$$x_{j+1}^f = Ax_j^f + Bu_j^f + m_o , \quad (2)$$

where $j = l, l+1, \ldots$ is the discrete time after a fault occurs and x^f is the state vector of the faulted aircraft whose control deflection is described by the expression

$$u_j^f = Fu_j + (I - F)u_o^f , \quad (3)$$

where F is the matrix of faults (loss of efficiency) of the control system

$$F = \operatorname{diag}\left[f(1) \cdots f(k) \cdots f(n_u) \right],$$

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u_o^f is the vector of control jamming in the case of fault

$$u_o^f = \left[u_o^f(1) \cdots u_o^f(k) \cdots u_o^f(n_u) \right]^{\mathrm{T}}.$$

Let us substitute (3) into (2) and write the model of the aircraft with the faulted control system as

$$x_{j+1}^f = A x_j^f + B_f u_j + m_o^f, \qquad (4)$$

where $B_f = BF$ is the matrix of control efficiency for the faulted aircraft and $m_o^f = B(I-F)u_o^f + m_o$ is the constant vector characterizing the combined control deflection in the case of fault. It is required, based only on the measurements of control signals and states, to detect faults in the control system of the dynamic aircraft.

3 The deterministic problem solution

Assume that the aircraft is observed over a certain period of time. Then the aircraft models in nonfaulted (1) and faulted (4) states are written in a matrix form as

$X_{i+1} = AX_i + BU_i + m_o e$, $X_{j+1}^f = AX_j^f + B_f U_j + m_o^f e$,

where $e = [1...1]$, $X_i = [x_i ... x_{i+h}]$, $X_j^f = [x_j^f ... x_{j+h^f}^f]$, $U_i = [u_i ... u_{i+h}]$, $U_j = [u_j ... u_{j+h^f}]$; h, h^f are the numbers of the observation steps for the nonfaulted and faulted states, respectively.

The problems of identifying the model parameters of the aircraft are described by the linear right-hand matrix equations in the unknown A, B, m_o, B_f, m_o^f:

$$\begin{bmatrix} A & B & m_o \end{bmatrix} \begin{bmatrix} X_i \\ U_i \\ e \end{bmatrix} = X_{i+1}, \quad \begin{bmatrix} A & B_f & m_o^f \end{bmatrix} \begin{bmatrix} X_j^f \\ U_j \\ e \end{bmatrix} = X_{j+1}^f.$$

These equations are solvable when and only when the following conditions are satisfied [5]:

$$X_{i+1} \overline{\begin{bmatrix} X_i \\ U_i \\ e \end{bmatrix}}^R = 0, \quad X_{j+1}^f \overline{\begin{bmatrix} X_j^f \\ U_j \\ e \end{bmatrix}}^R = 0, \qquad (5)$$

where

$$\begin{bmatrix} X_i \\ U_i \\ e \end{bmatrix} \overline{\begin{bmatrix} X_i \\ U_i \\ e \end{bmatrix}}^R = 0, \quad \begin{bmatrix} X_j^f \\ U_j \\ e \end{bmatrix} \overline{\begin{bmatrix} X_j^f \\ U_j \\ e \end{bmatrix}}^R = 0.$$

Expressions (5) show that the problem of aircraft linear model identification is solvable both before and after the occurrence of fault. However, at the instant of fault occurrence behavior of the aircraft cannot be described by a single linear model. This fact is used in [5] to detect fault by a criterion that characterizes the identification problem solution accuracy:

$$\varepsilon = \left\| \begin{bmatrix} X_{i+1} & X_{j+1}^f \end{bmatrix} \overline{\begin{bmatrix} X_i & X_j^f \\ U_i & U_j \\ e & e \end{bmatrix}}^R \right\|_2, \qquad (6)$$

where matrix zero divisor of the input and output data

$$\begin{bmatrix} X_i & X_j^f \\ U_i & U_j \\ e & e \end{bmatrix} \overline{\begin{bmatrix} X_i & X_j^f \\ U_i & U_j \\ e & e \end{bmatrix}}^R = 0 \qquad (7)$$

has an orthogonal form

$$\left(\overline{\begin{bmatrix} X_i & X_j^f \\ U_i & U_j \\ e & e \end{bmatrix}}^R \right)^{\mathrm{T}} \overline{\begin{bmatrix} X_i & X_j^f \\ U_i & U_j \\ e & e \end{bmatrix}}^R = I.$$

The criterion (6) does not require a priori information about the aircraft model, solving the problems of identification and prediction while using only the measurement data and state control vectors. However, this method has a serious shortcoming. As it is based on exact equality (7), even the smallest system disturbances can lead to the essential change of structure of zero divisor. This eventually leads to low reliability procedure for detecting faults in practice.

4 The disturbed problem solution

To increase the reliability of detection of the fact and the time of occurrence of the fault in the aircraft control system in the presence of disturbance instead of an exact zero divisor (7) we will find its approximate value, a so-called numerical zero divisor. For this purpose we will write down the equation for numerical right zero divisor calculation of the some matrix C:

$$CZ \approx 0 = \Theta_{m \times s}. \qquad (8)$$

The degree of the equation (8) solution approximation can be defined by a finite small value, which characterizes the permissible level of disturbances in the system, which can be evaluated with the help of Frobenius norm, also known as the Hilbert-Schmidt or Shura norm:

$$\delta = \|\Theta\|_2 = \sqrt{\sum_{i=1}^{\min\{m,s\}} \sigma_i^2},$$

where σ_i are the singular values of the matrix Θ.

For ensuring the given norm we use the singular value decomposition of a matrix C:

$$C = C^{LT} \Sigma_C C^{RT} = \begin{bmatrix} C^{LT} & \overline{\overline{C}}^{LT} \end{bmatrix} \begin{bmatrix} \Sigma_C^{\max} & 0 \\ 0 & \Sigma_C^{\min} \end{bmatrix} \begin{bmatrix} C^{RT} \\ \overline{\overline{C}}^{RT} \end{bmatrix}, \qquad (9)$$

where C^L, C^R are the matrices of left and right singular vectors satisfying the orthogonality conditions $C^{LT} C^L = I$, $C^R C^{RT} = I$; $\Sigma_C^{\min}$ is the diagonal matrix of the minimum singular values satisfying a condition $\|\Sigma_C^{\min}\| \le \delta$; $\Sigma_C^{\max}$ is the diagonal matrix of the maximum singular values; C^L, C^R, $\overline{\overline{C}}^L$, $\overline{\overline{C}}^R$ are the matrix of left and right singular vectors corresponding to the maximum and minimum singular value.

Substituting (9) into equation (8)

$$\begin{bmatrix} \tilde{C}^{LT} & \overline{\overline{C}}^{LT} \end{bmatrix} \begin{bmatrix} \Sigma_C^{\max} & 0 \\ 0 & \hat{\Sigma}_C^{\min} \end{bmatrix} \begin{bmatrix} C^{RT} \\ \overline{\overline{C}}^{RT} \end{bmatrix} Z = \Theta$$

and premultiplying the resulting expression by the matrix of left singular vectors we do not change the norm of the right side of the equation:

$$\begin{bmatrix} \Sigma_C^{\max} & 0 \\ 0 & \Sigma_C^{\min} \end{bmatrix} \begin{bmatrix} C^{RT} \\ \overline{\overline{C}}^{RT} \end{bmatrix} Z = \begin{bmatrix} C^{L} \\ \overline{\overline{C}}^{L} \end{bmatrix} \Theta = \Theta^* . \quad (10)$$

Let us introduce an intermediate matrix

$$Z = \begin{bmatrix} C^R & \overline{\overline{C}}^R \end{bmatrix} \begin{bmatrix} \Upsilon \\ \Psi \end{bmatrix} \quad (11)$$

and substitute the expression (11) into (10):

$$\begin{bmatrix} \Sigma_C^{\max} & 0 \\ 0 & \Sigma_C^{\min} \end{bmatrix} \begin{bmatrix} C^{RT} \\ \overline{\overline{C}}^{RT} \end{bmatrix} \begin{bmatrix} C^R & \overline{\overline{C}}^R \end{bmatrix} \begin{bmatrix} \Upsilon \\ \Psi \end{bmatrix} = \begin{bmatrix} \Sigma_C^{\max} & 0 \\ 0 & \Sigma_C^{\min} \end{bmatrix} \begin{bmatrix} \Upsilon \\ \Psi \end{bmatrix} = \Theta^* .$$

This expression implies that the given accuracy of the solution is provided only when $\Upsilon = 0$:

$$Z = \overline{\overline{C}}^R \Psi ,$$

where Ψ is an arbitrary orthogonal matrix satisfying $\Psi^T \Psi = I$. Then the fault detection criterion in the presence of disturbances has a following form:

$$\varepsilon = \left\| \begin{bmatrix} X_{i+1} & X_{j+1}^f \end{bmatrix} \overline{\overline{\begin{bmatrix} X_i & X_j^f \\ U_i & U_j \\ e & e \end{bmatrix}}}^R \right\|_2 , \quad (12)$$

where the numerical zero divisor satisfies the expression

$$\left\| \begin{bmatrix} X_i & X_j^f \\ U_i & U_j \\ e & e \end{bmatrix} \overline{\overline{\begin{bmatrix} X_i & X_j^f \\ U_i & U_j \\ e & e \end{bmatrix}}}^R \right\|_2 \le \delta . \quad (13)$$

Thus, to detect the fact and the time of aircraft control system fault occurrence it is necessary at each time step to define the right singular vectors of data matrices (13), corresponding to the minimum singular values with given degree of accuracy and to check the excess of a certain threshold value criterion (12).

5 The fault detection example

Let us demonstrate the validity of the proposed criterion on the example of the fault detection in the right stabilizer actuator of a highly-maneuverable aircraft [5]. Assume external disturbances $\|w\| \le 0.1$ and a fault in the form of a right stabilizer actuator jamming in the neutral position occurs in the fifth second of the flight. Fig. 1 shows the values of fault detection criterion (12) for 10 s time period.

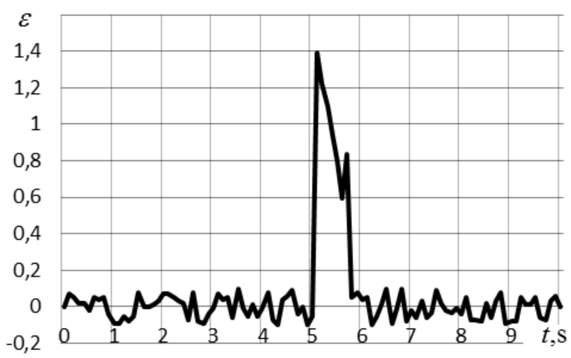

Fig. 1. The values of fault detection criterion.

It can be seen that before and after the occurrence of the fault, the values of criterion (13) is almost zero, while the fault itself is characterized by a spike the width of which corresponds to that of the identification window (h=h^f=8). Such a drastic change in the behavior of the plot makes it possible to accurately determine the time the fault occurred. Thus, similar to deterministic case [5] the fault is detected in the shortest possible time, corresponding to the integration step (0.1 s).

6 Conclusions

In this article the new quantitative model-free method for aircraft control system failure detection in the presence of external disturbances is proposed. It does not depend on the model parameters and is based only on the information about the observed signals involving no other auxiliary variables. This ensures its validity in the absence of any priori information without any training, identification or predicting procedures.

References

1. X. Qi, D. Theilliol, J. Qi, Y. Zhang, J. Han, D. Song, L. Wang, Y. Xia. Fault diagnosis and fault tolerant control methods for manned and unmanned helicopters: a literature review. *Proc. of the Conference on Control and Fault-Tolerant Systems SysTol*, pp. 132–139 (2013)
2. R.J. Patton, Fault-tolerant control: the 1997 situation. *Proc. of the IFAC Symposium on Fault Detection Supervision and Safety for Technical Processes*, pp. 1033–1055 (1997)
3. Y. Zhang, J. Jiang. Ann. Rev. Control. **32**(2), 229–252 (2008)
4. O.N. Korsun, *Mekhatronika, Avtomatizatsiya, Upravlenie*. **S6**, 2–7 (2008)
5. E.Yu. Zybin, V.V. Kosyanchuk. *Journal of Computer and Systems Sciences International*, **55**(4), 546–557 (2016)