NONLINEAR MODEL FOLLOWING RECONFIGURABLE FLIGHT CONTROL SYSTEM

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Abstract

The method for reconfiguration of an aircraft flight control system was developed based on a model following algorithm with a nonlinear aircraft reference model. A fuzzy logic method was applied to calculate the control degradation coefficient used to produce control signals for preventing the consequences of the failures. The method was implemented in the aircraft autopilot system with a six degrees of freedom rigid airplane nonlinear reference model. The simulations of selected failures demonstrated the effective alleviation of the failure influence of the aircraft control performance. The method is general and may be used in a variety of failure cases.

1 Introduction

Failures within the flight control system may significantly influence aircraft safety and lead to safety critical situations. The reliability of a control system is assured by a proper design of its structure, by comprehensive testing during development and certification and by regular maintenance during operation. In aircraft systems a design redundancy is usually obtained by increasing a number of hardware and software elements. These measures make the failure of the aircraft control system very unlikely, but they do not necessarily allow for the effects of hostile actions, which may lead to damage not predicted during the design of control system. There is therefore an increasing interest in methods of flight control reconfiguration, which potentially can cope with the majority of the "unpredicted" situations.
Airplanes are usually controlled by primary control surfaces like ailerons, elevators and rudders, supported by additional control surfaces: stabilizer, spoilers, flaps and by control of the engine thrust. Aircraft motion for various degrees of freedom is controlled by single or multiple control surfaces (usually roll by ailerons, pitch by elevators, yaw by rudder). The conventional operation of control surfaces has some design limitations. For instance ailerons usually deflect anti-symmetrically and in the event of component failure it may be desirable to have more flexibility in operating the remaining functioning surfaces, for example each aileron surface or each elevator half could deflect separately. This approach to control system design may be named structural redundancy [1]. Before incorporating structural redundancy, the effects of individual control surface deflection have to be investigated through simulation and such approach to control system design may be evaluated.

The objective of the research presented in this paper is to develop and analyze the methods for controlling an aircraft after severe damage of a control system. The method should be general enough to cover several failure cases. Design limitations of deflection of control surfaces and actuators dynamics should be included in the analysis [2]. Since the majority of commercial airplane pilots use autopilots in all phases of flight, the reconfiguration of the control algorithm should be included in the fully automatic, reconfigurable flight control system.

Several methods and algorithms are available for the implementing reconfiguration of flight control system. These algorithms use FDI (fault detection and isolation) [3,4] or an adaptive control approach [5-8]. The adaptive reconfigurable control systems employ the model following method [9,10], combined with eigenstructure assignment, sliding mode control [11], fuzzy logic or neural network [12] methods.

In this research a model following control method is used, with a nonlinear aircraft reference model and a fuzzy logic approach to obtain the control signals for a damaged aircraft. The signals are calculated by the minimization of control system deflections with a fuzzy logic algorithm to keep the control values within the design limitations of the control system. The nonlinear airplane model ensures better airplane performance after system reconfiguration, due to more accurate description of the degradation of airplane performance after control system failures.

Although the results presented in this paper were obtained for a Business Jet airplane, the method is general and may be applied to various types of aircraft.

2 Aircraft Simulation Model

The nonlinear model of a multi jet engine passenger airplane was used to investigate the reconfiguration of the flight control system. In this chapter the general structure of the model is described.

The airplane equations of motion were obtained by summing up inertia, gravity, aerodynamic, and propulsion loads (forces and moments). This approach allows to build a modular model, which may be extended to represent more sophisticated cases in which the parts of the model influenced by the aircraft damage or system failures may be modified.

The position and attitude of the aircraft (Fig.1) are described by the vector $\mathbf{y}=\begin{bmatrix}x_1 & y_1 & z_1 & \phi & \theta & \psi\end{bmatrix}^T$ composed of the aircraft position vector $\mathbf{r}_1=\begin{bmatrix}x_1 & y_1 & z_1\end{bmatrix}^T$ in the ground system of co-ordinates $0x_1y_1z_1$, and angles of roll ϕ, pitch θ and yaw ψ describing the aircraft attitude. The state vector $\mathbf{x}=\begin{bmatrix}\mathbf{v} & \boldsymbol{\omega}\end{bmatrix}^T$ of the aircraft is composed of two vectors: linear velocity $\mathbf{v}=\begin{bmatrix}U & V & W\end{bmatrix}^T$ and angular rate $\boldsymbol{\omega}=\begin{bmatrix}P & Q & R\end{bmatrix}^T$.

Aircraft state vector and vector of position and attitude are related by the kinematics equation:

$$\dot{\mathbf{y}}=\mathbf{T}\mathbf{x}, \tag{1}$$

where T is the transformation matrix from body co-ordinate system 0xyz to inertial co-ordinate system $0_1x_1y_1z_1$.

The aircraft equations of motion resulting from the equilibrium of the inertia, gravity $\mathbf{f}_G$, aerodynamic $\mathbf{f}_A$ and propulsion $\mathbf{f}_T$ loads (forces and moments) have the general form [13]:

$$\mathbf{A}\dot{\mathbf{x}}+\mathbf{B}(\mathbf{x})\mathbf{x}=\mathbf{f}_A(\mathbf{x},\mathbf{y},\boldsymbol{\delta})+ \\ +\mathbf{f}_G(\mathbf{y})+\mathbf{f}_T(\mathbf{x},\mathbf{y},\boldsymbol{\delta}_T) \tag{2}$$

where $\boldsymbol{\delta}$ is the vector of control surface deflection, $\boldsymbol{\delta}_T$ is the vector of throttle lever position.

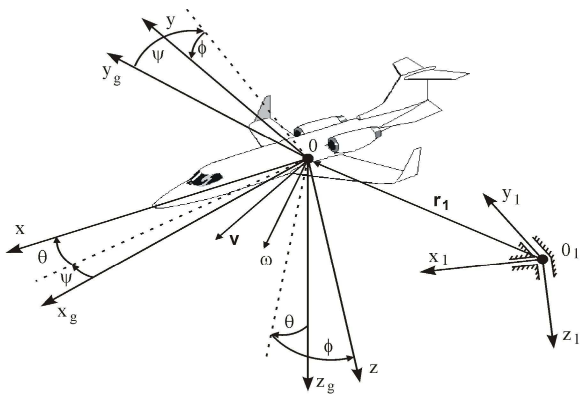

Fig. 1. Position and attitude diagram

Matrix **A** describes the mass properties of the aircraft and has form:

$$\mathbf{A}=\begin{bmatrix} m & 0 & 0 & 0 & S_z & -S_y \\ 0 & m & 0 & -S_z & 0 & S_x \\ 0 & 0 & m & S_y & -S_x & 0 \\ 0 & -S_z & S_y & I_x & -I_{xy} & -I_{xz} \\ S_z & 0 & -S_x & -I_{xy} & I_y & -I_{yz} \\ -S_y & S_x & 0 & -I_{xz} & -I_{yz} & I_z \end{bmatrix} \tag{3}$$

where m is the aircraft mass, S_x, S_y, S_z are static moments and I_x, I_y, I_z, I_{xy}, I_{xz}, I_{yz} are moments of inertia. Matrix $\mathbf{B(x)}$ is calculated as:

$$\mathbf{B}(\mathbf{x})=\boldsymbol{\Omega}(\mathbf{x})\mathbf{A} \tag{4}$$

where matrix $\boldsymbol{\Omega}(\mathbf{x})$ contains components of aircraft linear and angular velocities:

$$\mathbf{\Omega}(\mathbf{x})=\begin{bmatrix} 0 & -R & Q & 0 & 0 & 0 \\ R & 0 & -P & 0 & 0 & 0 \\ -Q & P & 0 & 0 & 0 & 0 \\ 0 & -W & V & 0 & -R & Q \\ W & 0 & -U & R & 0 & -P \\ -V & U & 0 & -Q & P & 0 \end{bmatrix} \tag{5}$$

It was assumed that the aerodynamic loads $\mathbf{f_A}$ may be written as the sum of two components:

$$\mathbf{f_A}(\mathbf{x},\mathbf{y},\boldsymbol{\delta})=\mathbf{f_{AS}}(\mathbf{x},\mathbf{y})+\mathbf{f_\delta}(\mathbf{x},\mathbf{y},\boldsymbol{\delta}) \tag{6}$$

In which the part $\mathbf{f}_\delta(\mathbf{x},\mathbf{y},\boldsymbol{\delta})$ depends on deflections of control surfaces and the part $\mathbf{f}_{AS}(\mathbf{x},\mathbf{y})$ does not.

Substituting Eq. (6) into (2), rearranging and multiplying by $\mathbf{A}^{-1}$, the nonlinear aircraft model can be formulated in general form as:

$$\dot{\mathbf{x}}=\mathbf{f_1}(\mathbf{x},\mathbf{y})+\mathbf{f_2}(\mathbf{x},\mathbf{y},\boldsymbol{\delta})+\mathbf{f_3}(\mathbf{x},\mathbf{y},\boldsymbol{\delta}_\mathbf{T}) \tag{7}$$

In Eq. (4), due to previous assumptions, the first component does not depend on aircraft control, the second component describes loads due to control surface deflection, and the third component describes airplane loads due to thrust control.

3 Control System Model

It was assumed that the primary control surfaces (ailerons, elevator and ruder) were driven by hydraulic actuators, so the control surface actuators were modeled as linear second order systems. In the state variables the equations of motion of a single primary control surface have the form:

$$\dot{\mathbf{z}}_\mathbf{i}=\mathbf{G_i z_i}+\mathbf{h_i u_i} \tag{8}$$

where $\mathbf{z_i}=\begin{bmatrix}\delta_i & \dot{\delta}_i\end{bmatrix}^T$ is the state vector of a single control surface, composed of a deflection angle δ and a deflection rate $\dot{\delta}$. The command signal for the single surface is denoted by $\mathbf{u_i}$.

It is assumed that a secondary control surface, stabilizer, is driven by an electric actuators. Its motion was modeled as a nonlinear first order system, with a dead band type nonlinearity for small values of control signal.

The equation of the stabilizer motion has the form:

$$\dot{\delta}_{ST}=g_{ST}(\delta_{ST})\cdot\delta_{ST}+h_{ST}(\delta_{ST})\cdot u_{ST} \tag{9}$$

Combined equations of motion of all control surfaces in the state variables have the form:

$$\dot{\mathbf{z}}=\mathbf{Gz}+\mathbf{Hu_S} \tag{10}$$

$$\boldsymbol{\delta}=\mathbf{Kz} \tag{11}$$

In the above, the state vector of the system actuating the control surfaces was defined as:

$$\mathbf{z}=\begin{bmatrix}\delta_{HL} & \dot{\delta}_{HL} & \delta_{HR} & \dot{\delta}_{HR} & \delta_{LR} \\ \dot{\delta}_{LR} & \delta_{LL} & \dot{\delta}_{LL} & \delta_{V} & \dot{\delta}_{V} & \delta_{ST}\end{bmatrix}^T \tag{12}$$

and the command vector of pilot / autopilot is:

$$\mathbf{u_S}=\begin{bmatrix}\mathrm{u_H} & \mathrm{u_L} & \mathrm{u_V} & \mathrm{u_{ST}}\end{bmatrix}^\mathrm{T} \tag{13}$$

The state matrix **G** and the output matrix **H** in (10) and (11) are defined as :

$$\mathbf{G}=\begin{bmatrix}\mathbf{G_{HL}} & \mathbf{0} & \mathbf{0} & \mathbf{0} & \mathbf{0} & \mathbf{0} \\ \mathbf{0} & \mathbf{G_{HR}} & \mathbf{0} & \mathbf{0} & \mathbf{0} & \mathbf{0} \\ \mathbf{0} & \mathbf{0} & \mathbf{G_{LR}} & \mathbf{0} & \mathbf{0} & \mathbf{0} \\ \mathbf{0} & \mathbf{0} & \mathbf{0} & \mathbf{G_{LL}} & \mathbf{0} & \mathbf{0} \\ \mathbf{0} & \mathbf{0} & \mathbf{0} & \mathbf{0} & \mathbf{G_V} & \mathbf{0} \\ 0 & 0 & 0 & 0 & 0 & g_{ST}\end{bmatrix} \tag{14a}$$

$$\mathbf{H}=\begin{bmatrix}\mathbf{h_{HL}} & \mathbf{0} & \mathbf{0} & \mathbf{0} \\ \mathbf{h_{HR}} & \mathbf{0} & \mathbf{0} & \mathbf{0} \\ \mathbf{0} & \mathbf{h_{LR}} & \mathbf{0} & \mathbf{0} \\ \mathbf{0} & -\mathbf{h_{LL}} & \mathbf{0} & \mathbf{0} \\ \mathbf{0} & \mathbf{0} & \mathbf{h_V} & \mathbf{0} \\ 0 & 0 & 0 & h_{ST}\end{bmatrix} \tag{24b}$$

The subscripts denote: *HL* – left elevator, *HR* – right elevator, *LR* – left aileron, *LL* – left aileron, *V* – ruder, *ST* – stabilizer.

4 Autopilot

An autopilot model was designed (Fig. 2) using classical feed-back control laws. The autopilot is composed of three main parts: autothrottle, longitudinal and lateral modules, with the auto trim block in the longitudinal module and the compensator of slip angle in the lateral autopilot module. The control signal $\mathbf{u_S}$ from the autopilot is transferred to the actuating system of the airplane control surfaces. The details of the autopilot design are given in Ref.14-16.

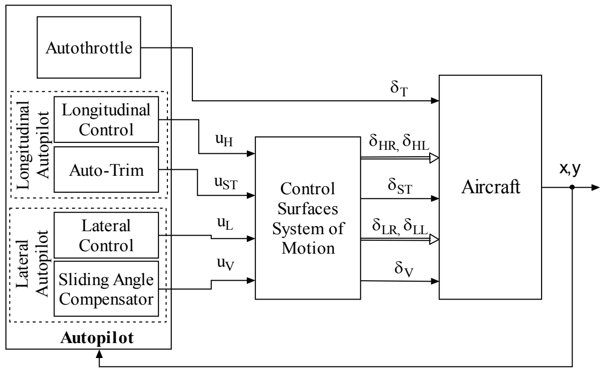

Fig. 2. Automatic flight control system

5 Reconfiguration Algorithm

The reconfiguration block of the flight control system is placed between the autopilot (or pilot) and the control surface actuation system (Fig.3). It allows to apply reconfiguration in both flight cases: a man-controlled by a pilot and an autonomous controlled by an autopilot.

The rationale behind the reconfiguration method developed during this research is the requirement, that the airplane should be guided by the control system along the demanded flight trajectory also after failure.

When a failure occurs the efficiency (ability to fly the required flight phase) of the control system degrades and usually it would be very difficult to control the aircraft to exactly follow the assumed flight path. The ability to maintain the assumed flight path and to preserve aircraft controllability depends on the severity of the failure. In the method developed here it was assumed that, in case of the failure, the main task of the control reconfiguration would be to maintain the ability at least to continue the steady flight. If the aircraft can be controlled in a more effective way, the reconfigured flight control system will keep the aircraft flight trajectory demanded by the autopilot or the pilot.

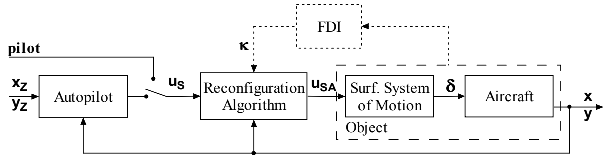

Fig. 3. Structure of reconfigurable flight control system

The reconfiguration algorithm generates the signals $\mathbf{u_{SA}}$, which are inputs to the actuators of the airplane control surfaces. During the failure-free flight, the reconfiguration algorithm does not change the signals generated by the autopilot (or pilot) $\mathbf{u_{SA}} = \mathbf{u_S}$. In case of a failure the output signal $\mathbf{u_{SA}}$, from the reconfiguration block is calculated using comparison of the control loads in the damaged (real) and undamaged (model) aircraft.

The nonlinear aircraft model with control surface dynamics is composed of aircraft equations of motion Eq. (7), kinematics equations, and actuator dynamics described by Eq. (10) and Eq. (11). When a failure occurs the aircraft model can be written as:

$$\dot{\mathbf{x}}_A = \mathbf{f}_1(\mathbf{x}_A, \mathbf{y}_A) + \mathbf{f}_{2A}(\mathbf{x}_A, \mathbf{y}_A, \boldsymbol{\delta}_A) + \mathbf{f}_3(\mathbf{x}_A, \mathbf{y}_A, \boldsymbol{\delta}_{TA}) \tag{35}$$

$$\dot{\mathbf{y}}_A = \mathbf{T}(\mathbf{x}_A) \cdot \mathbf{x}_A \tag{46}$$

$$\dot{\mathbf{z}}_A = \mathbf{G}_A \mathbf{z}_A + \mathbf{H}_A \mathbf{u}_{SA} \tag{57}$$

$$\boldsymbol{\delta}_A = \mathbf{K}_A \mathbf{z}_A \tag{68}$$

where subscript „A” refers to the equations’ components, parameters and variables which are dependent on the failure.

To preserve the demanded trajectory after a failure the aircraft control system should generate control loads proportional to the loads in the failure-free case. The control loads with and without a failure would be different. These differences would depend on the aircraft ability to generate sufficient control loads after a particular failure case. The behavior of a damaged aircraft would be similar to the behavior of a failure-free aircraft, when the following condition is satisfied:

$$\mathbf{f}_{2A}\left(\mathbf{x}_A,\mathbf{y}_A,\boldsymbol{\delta}_A\right)=\mathbf{N}\cdot\mathbf{f}_2\left(\mathbf{x},\mathbf{y},\boldsymbol{\delta}\right) \qquad (79)$$

where **N** is the coefficient of the control system degradation.

In a general case the coefficient of the control system degradation **N** is a matrix with coefficient values in the range $\mathbf{N}_{ij}\in\langle 0,1\rangle$.

The matrix of control system degradation coefficients reflects the coupling between various control loads acting on the aircraft.

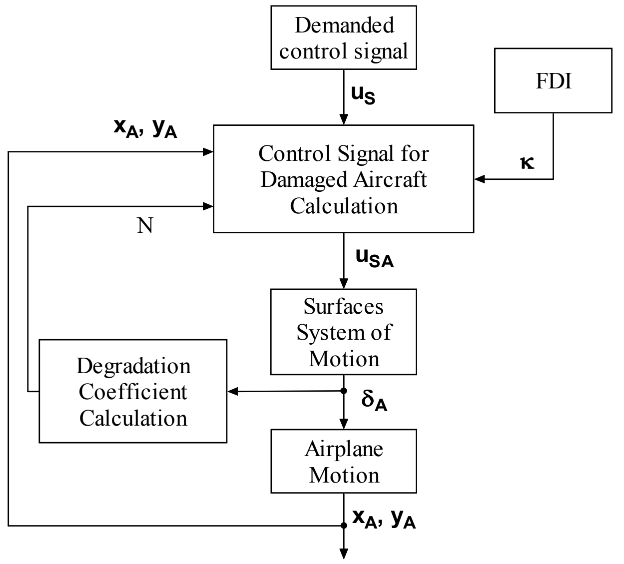

Fig. 4. Reconfiguration algorithm scheme

The block diagram of the reconfiguration algorithm is shown in Fig. 4. The Fault Detection and Identification system is not considered within this study. Coefficient **κ** denotes information from the FDI system, which in the simulation is realized by proper assumptions concerning the model data.

The output signal from the reconfiguration module is calculated by comparison of control loads for damaged and failure-free airplanes using the coefficient **N** of control system degradation. The signal $\mathbf{u}_{SA}$ from the reconfiguration block depends on a signal from the autopilot (or pilot) $\mathbf{u}_S$ and actual state variables $\mathbf{x}_A$ of a damaged airplane, i.e. $\mathbf{u}_{SA}=f\left(\mathbf{u}_S,\mathbf{x}_A,\mathbf{y}_A\right)$.

The output from the reconfiguration algorithm results from the requirement of minimizing the deflection of control surfaces to satisfy Eq. (18). This task is solved using an optimization method, with the objective function in the form:

$$f\left(\boldsymbol{\delta}_A\right)=\sum_i\left(\delta_{Ai}\right)^2 \qquad (20)$$

The Equations (15-19) form the optimization constrains for the objective function Eq. (20).

To keep the deflection of the control surfaces within design constraints the coefficient of control system degradation **N** is calculated using Fuzzy Logic (Fig. 5) method

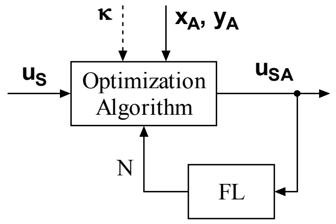

Fig. 5. Structure of reconfiguration algorithm

6 Sample Simulation Results

The first part of the simulation results presented in this paper concerns the sole (without reconfiguration) autopilot efficiency in the case of a failure. The left elevator was blocked at a deflection of -2 deg and the right aileron was blocked in 0, 5, 12, 15 deg (15 deg is the maximum aileron deflection) positions. The results are presented in Fig. 6 for a commanded altitude change from 5000 ft to 4000 ft and a heading change from 0 deg to -30 deg at 200 kts airspeed.

These results reveal that the aileron failure influences both lateral and longitudinal control efficiency. Blocking the aileron deflection causes the greatest disturbance to the descending path (Fig. 6a). The autopilot loses airplane control for higher values of blocked aileron deflection.

The efficiency of the reconfiguration algorithm is illustrated also for the same failure case as above. The aircraft was commanded to change simultaneously altitude from 5000 to 6000 ft and heading from 0 to -30 deg. The control system failure was: right aileron blocked at 5 deg and left elevator blocked at -2 deg.

The research undertaken in this study included also the option of using the stabilizer as an additional control surface in the failure case and two ways of calculating the control degradation matrix **N**. The first case was the single value of N for the all control loads (diagonal matrix **N** with the same values of all elements Eq. (21a)). In the second case two values were used: N_1 for longitudinal control and N_2 for lateral control, which formed a control system degradation coefficient matrix, Eq. (21b):

$$\mathbf{N}=\begin{bmatrix} N & 0 & 0 & 0 & 0 & 0 \\ 0 & N & 0 & 0 & 0 & 0 \\ 0 & 0 & N & 0 & 0 & 0 \\ 0 & 0 & 0 & N & 0 & 0 \\ 0 & 0 & 0 & 0 & N & 0 \\ 0 & 0 & 0 & 0 & 0 & N \end{bmatrix} \qquad (21a)$$

$$\mathbf{N}=\begin{bmatrix} N_1 & 0 & 0 & 0 & 0 & 0 \\ 0 & N_2 & 0 & 0 & 0 & 0 \\ 0 & 0 & N_1 & 0 & 0 & 0 \\ 0 & 0 & 0 & N_2 & 0 & 0 \\ 0 & 0 & 0 & 0 & N_1 & 0 \\ 0 & 0 & 0 & 0 & 0 & N_2 \end{bmatrix} \qquad (21b)$$

In Fig. 7 and Fig. 8 the trajectories of aircraft are shown for the cases: no failure, reconfiguration using primary control surfaces (ailerons, elevators and ruder) and reconfiguration using primary control surfaces and stabilizer to control aircraft after damage.
The flight trajectory, shown in Figure 7, was obtained for the single coefficient of control system degradation, Eq. (21a). The system with the reconfiguration module works efficiently and for both reconfigurable algorithms (with and without stabilizer) the airplane is controlled

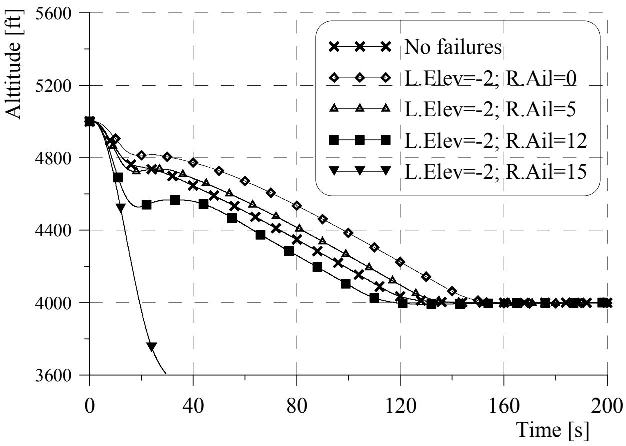

Fig. 6a. Altitude

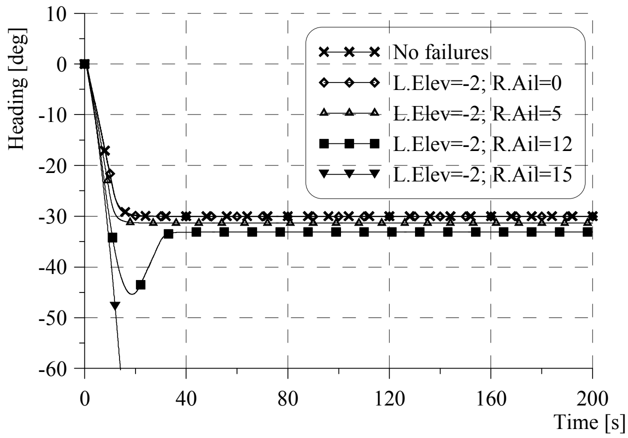

Fig. 6c. Heading

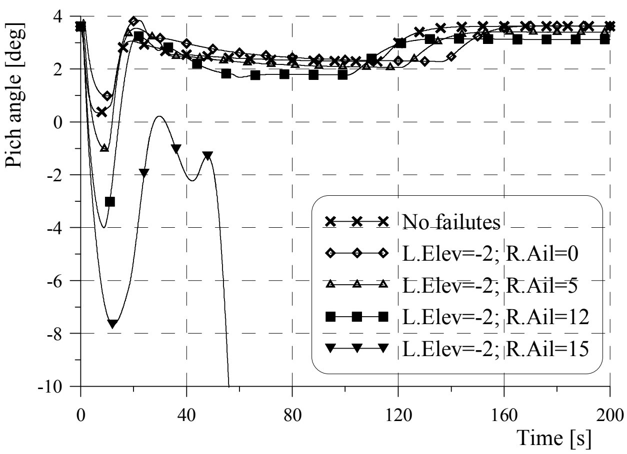

Fig. 6b. Pitch angle

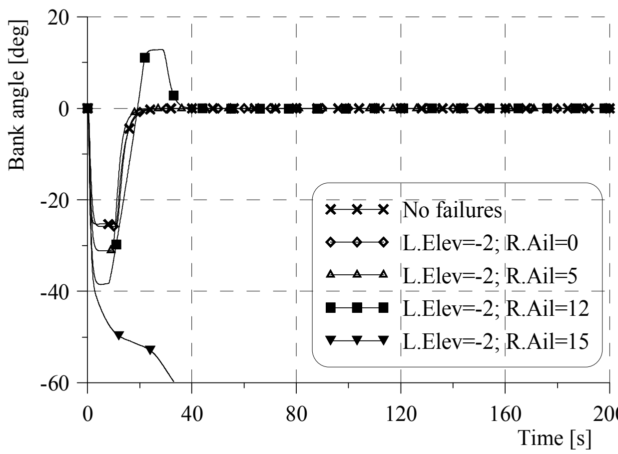

Fig. 6d. Bank angle

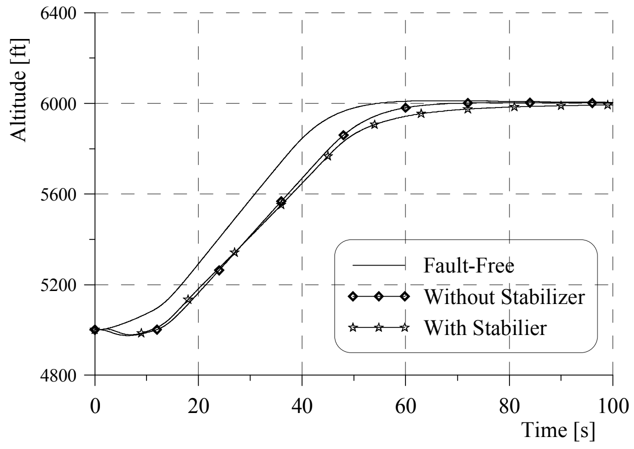

Fig. 7a. Altitude

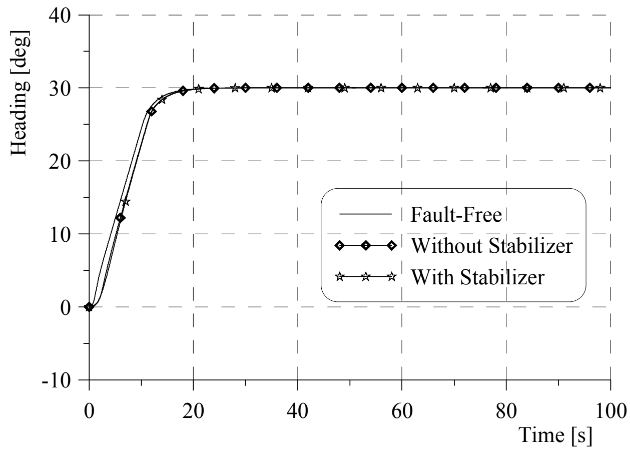

Fig. 7c. Heading

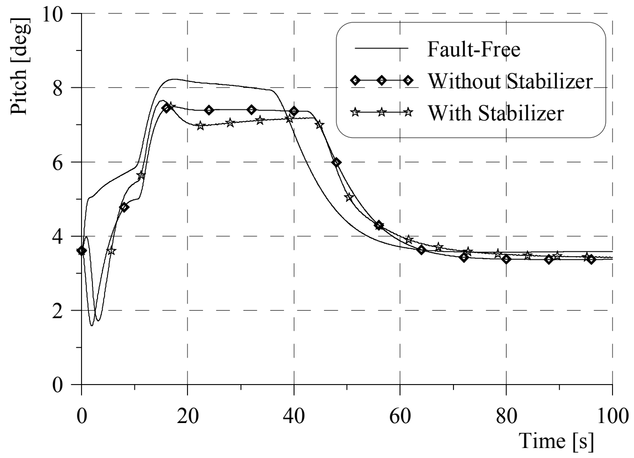

Fig. 7b. Pitch angle

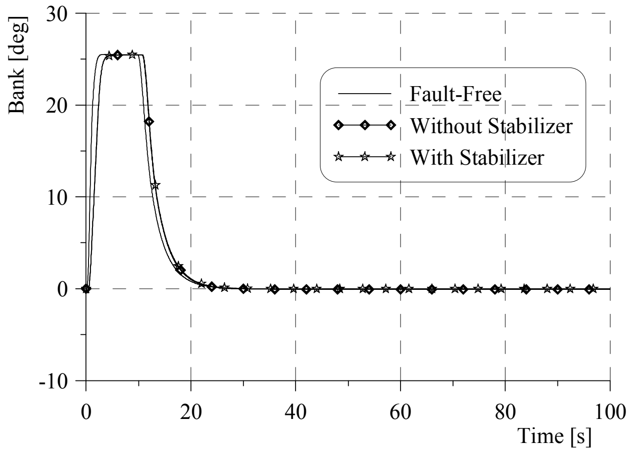

Fig. 7d. Bank angle

effectively. In the case of stabilizer control, the longitudinal control of the aircraft is performed mainly as due to its deflection. Such reaction of the algorithm may be attributed to using the sum of the squares of control surface deflections Eq. (20) as an objective function in the optimization procedure. A stabilizer is usually driven by an electrical actuator and its response time is longer that the response time of hydraulic actuators which are used here to drive primary surfaces. Therefore the time to reach the required altitude is shorter, when the stabilizer is not used. Although using the stabilizer takes a longer time to reach the desired flight phase, generally the application of the secondary control surfaces increases the redundancy of the control system. This may be important in the case of more serious and complex failures.

For the case of a single value of degradation coefficient, the airplane was descending, when performing simultaneously a climb and a turn. The ailerons reached maximum deflection during the turn, which caused the lowest value of aileron control degradation coefficient N. It means that the control signals for all control surfaces were reduced to minimum values, according to equation (19).

The results of simulation for the case of using two coefficients of degradation of control system, Eq. (21b) are shown in Fig. 8. Assuming different degradation coefficients for lateral and longitudinal control allows reduction of the airplane rate of descent (sink) during the turn.

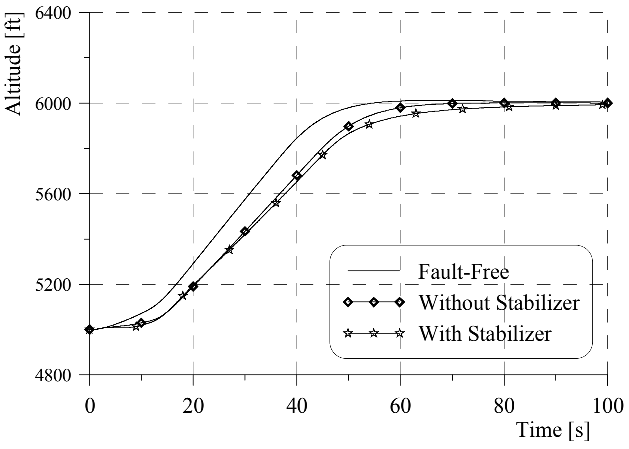

Fig. 8a. Altitude

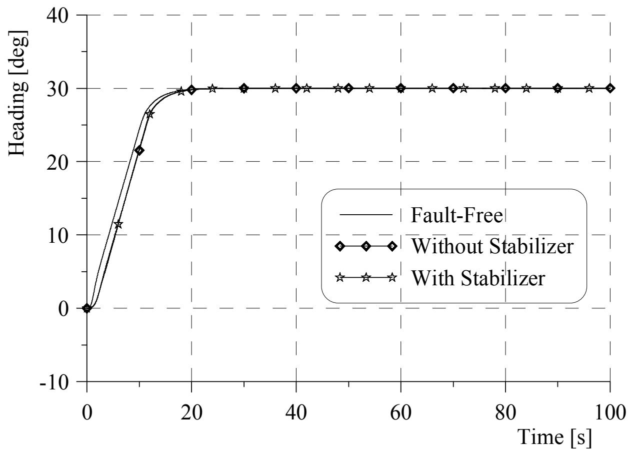

Fig. 8c. Heading

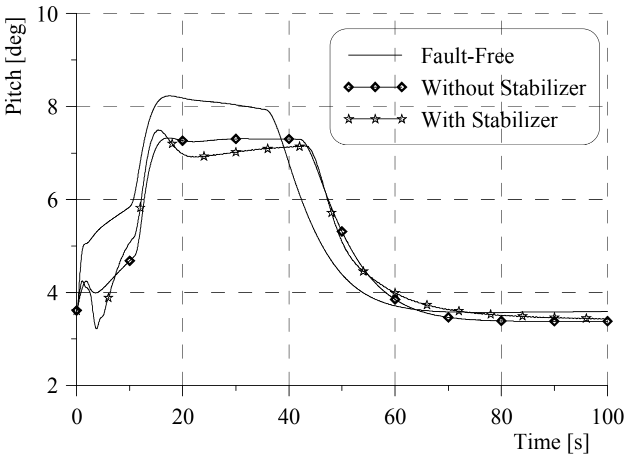

Fig. 8b. Pitch angle

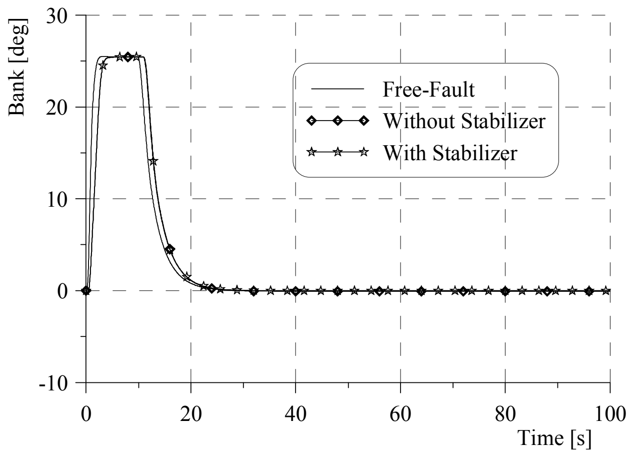

Fig. 8d. Bank angle

7 Conclusion

The new method of reconfiguration of aircraft control system was developed and tested in computer simulations. The method applies a nonlinear aircraft dynamic model with the nonlinearity in dynamics of control surface actuators. The nonlinear airplane model allows the simulation and investigation of control system failures such as control surface blockage, losing the part of the control surfaces, actuator failures etc.

The method of reconfiguration of control system is based on evaluation of the efficiency of a control system after failure and calculating new control signals for improvement. The design constrains of control surfaces deflections are taken into account using a fuzzy logic method for the calculation the control system degradation coefficient matrix. The matrix of coefficients of control system degradation in a quantitative manner describes the ability to control the airplane after failure.

The simulation results proved the efficiency of the reconfiguration method. The possibility of controlling aircraft after failure depends on several factors, in which the scope and severity of the damage is the most crucial factor. The efficiency of the reconfiguration method depends mainly on the number of control surfaces available after aircraft damage. In the method developed in this research the efficiency of the control system reconfiguration depends also on the objective function taken into account in the optimization procedure and of the method of calculation the coefficients of control system degradation. These factors will be investigated in further studies.

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