Fail Safe and Restructurable Flight Control System

K.Kanai and Y.Ochi

Dept. of Aerospace Engineering

The National Defense Academy

1-10-20 Hashirimizu, Yokosuka 239, JAPAN

Introduction

Importance of in-flight failure accommodation for impaired aircraft through the use of restructurable or reconfigurable flight control systems (RFCS) is widely recognized. The underlying idea of the RFCS is to take full advantage of the remaining controls to compensate for the functions which are lost due to failures. It is called functional redundancy[1]. Many papers have been published in this area since 1983[2,3,4]. Among them, a simple and practicable approach, referred to as self repairing flight control systems, is evaluated by not only numerical simulation[5] but also motion-based simulation[6] and flight tests on the F-15[7]. From the point of view of controller design, however, most of these studies are based on linearized mathematical models and linear control theories. Although linear controllers are known to be useful in practice even in some nonlinear environments, failures often cause severe nonlinearity due to large motions apart from the nominal trim point. In that situation, the linearized models can no longer describe the impaired aircraft dynamics well, so that the RFCS may fail to accommodate failures.

To meet such nonlinear situations, the authors proposed an RFCS[8,9] using the feedback linearization method, a nonlinear control method[8,9]. The first feature of the RFCS is that failures are identified as parameter changes in the six-degree-of-freedom(6-DOF) nonlinear equations of motion by a recursive least squares algorithm, an on-line version of the least squares method. Using the algorithm, estimated parameters are updated at every sampling time, and converge to constant values. Control parameters are updated using the latest estimated parameters. Thus the RFCS is a kind of adaptive control system and has the potential to accommodate failures which considerably change the characteristics of the aircraft dynamics, including the nonlinear ones. The second feature is that the parameters are identified using generic inputs which are introduced to decrease the number of parameters to be identified. The control law is also determined for generic inputs, and then they are distributed to actual ones. Generally it is desirable to have many independent control effectors to accommodate failures. For example, the right and left elevators are wanted to deflect independently, so that they can be used to control the rolling motion as well as the pitching motion. However, increasing the number of inputs leads to the increase of the number of parameters to be identified and makes the control law more complicated. Using the generic inputs, these problems are removed. Performance of the RFCS was demonstrated through computer simulation using the nonlinear model of an F-14 class fighter aircraft with half of the right wing missing[9].

In this paper, the RFCS has been extended to include a feedforward of stabilator and engine controls to counteract actuator failures. The method is applied to an airliner (Boeing 747) to recover from control failures such as partial loss of control surfaces, actuator jam, etc. The proposed RFCS consists of feedforward control and the feedback control based on the feedback linearization method and parameter identification. The former gives control inputs for the slow control effector and the latter for the fast control surfaces. Both controls cooperate to accommodate control surface jam and other airframe failures.

Let us explain the feedforward control more in detail. Generally, large transport aircraft are different from fighter aircraft in the following points:

1) Control surfaces of airliners produce smaller forces and moments for the scale of the airframe than those of fighter aircraft.
2) Control effectors such as stabilators, flaps, engines, etc., have much larger time constants than other surfaces such as elevators, ailerons or rudders.

The first point makes it more difficult for the aircraft to recover from failures than for fighters. The second point makes it difficult to use the slow control effectors along with the fast ones in transient attitude control. In fact, using the slow ones may cause the fast control effectors to cause disturbances because of their rate or position limits.

However, since some slow effectors can produce large forces or moments, it is important to take advantage of them from the first stage of recovery before the aircraft motions grow too large. The above observation leads to the idea of using the slow effectors in a feedforward manner to counteract the disturbances caused by stuck control surfaces, which can be a fatal failure for large transport aircraft. The proposed method has three features: First although the original RFCS is an adaptive control system, the feedforward inputs are determined using the nominal parameters instead of estimated parameters. Thereby the inputs can be acceptable ones for most jam failures. Secondly they are applied stepwise on detecting jams. Using the nominal parameters makes it possible to apply the step inputs immediately, since parameter estimation is not required. If the nominal parameters change because of other airframe failures, they do not give a correct trim solution, which may cause a serious problem. The error of the trim point, however, can be accommodated by the adaptive feedback control, unless the parameter errors are too large, or the airframe failures are too severe. Thus, in the proposed RFCS the feedforward and feedback controls cooperate. This is the third feature. To illustrate the performance of the RFCS, computer simulation has been conducted using a 6-DOF nonlinear aircraft model of the Boeing 747 aircraft. Failures considered include elevator and rudder jams. As a more practical example, the RFCS is applied to the accident of the El Al freighter near Schiphol Airport in 1992. In this paper, sensors and computers are assumed to be normal, and sensor noises are not taken into account.

Design of RFCS

The RFCS is composed of an adaptive feedback control and a feedforward control. The former is applied to the fast control surfaces and the latter to the slow effectors.

RFCS by Nonlinear Feedback Control

This subsection gives a brief outline of design of the RFCS[9] which the proposed RFCS is based on. The state and output equations are assumed to be given by

$$\dot{X} = A(X) + B(X)U \tag{1}$$

$$Y = C(X) \tag{2}$$

where $X \in R^n$ is a state vector, $U \in R^m$ a control vector, and $Y \in R^k$ an output vector to be controlled . Elements of $A(X) \in R^n$ and $B(X) \in R^{n \times m}$ are assumed to have the form of linear combinations of constant parameters and known functions of X. $C(X) \in R^k$ is a vector of nonlinear functions. The control objective is to make Y track the desired reference outputs, Y^*. The control law is derived by the feedback linearization method.

A large transport aircraft has many control surfaces. Let us assume that in the aircraft considered in this paper all the surfaces can be driven independently. As stated above, to design the RFCS for such an aircraft, generic inputs are useful in parameter identification and determination of the control law. Accordingly we introduce a generic input vector, $U_G \in R^{m'}$, which is defined by

$$U = PU_G \tag{3}$$

where $P \in R^{m \times m'}$ is a constant matrix. P is called the control distributor(CD) matrix. The control law is determined for the generic inputs, which are distributed to actual ones, U, by Eq. (3). Substituting Eq. (3) into Eq. (1) yields

$$\dot{X} = A(X) + B_G(X)U_G \tag{4}$$

where $B_G(X) = B(X)P$. Effects of failures on the aircraft dynamics are identified by estimating the parameters of $A(X)$ and $B_G(X)$ in Eq.(4). The identification algorithm uses a recursive least squares method. The generic inputs, instead of the actual inputs, are used in the identification. The control parameters are modified using the estimated plant parameters. Although the 6-DOF aircraft model describes various failures including those losing symmetry with respect to the X-Z plane, there may be failures that cannot be modeled by the equations, and it is difficult to accommodate such failures by the RFCS. In this paper, we do not consider such failures and let us assume that failures can be modeled by Eq. (1).

Feedforward Control using Slow Effectors

Let us consider a large transport aircraft, Boeing 747. In Eq.(1), state variables and control variables are defined as follows.

$X = [u, w, q, \theta, v, r, p, \phi]^T$, where u=forward speed (m/sec), w=downward speed (m/sec), q=pitch rate (rad/sec), θ=pitch angle (rad), v=sideward speed (m/sec), r=yaw rate (rad/sec), p=roll rate (rad/sec), ϕ=roll angle (rad).

$U=[\delta_{fst}^T, \delta_{slw}^T, T_h^T]^T$, where $\delta_{fst}=[\delta_{eiL}, \delta_{eiR}, \delta_{eoL}, \delta_{eoR}, \delta_{aiL}, \delta_{aiR}, \delta_{aoL}, \delta_{aoR}, \delta_{rup}, \delta_{rlw}]^T$, $\delta_{slw}=[\delta_{stL}, \delta_{stR}]^T$, $\delta_t=[\delta_{t1}, \delta_{t2}, \delta_{t3}, \delta_{t4}]^T$, and the subscripts denote e=elevator, a=aileron, r=rudder, st=stabilator, i=inboard, o=outboard, L=left side, R=right side, up=upper side, lw=lower side. For example, δ_{eiL} denotes the left side inboard elevator angle. δ_{fst} and δ_{slw} include the fast and slow control surfaces respectively. δ_{ti}(i=1,..,4) indicates the ith engine output. Positive surface angles are defined as deflecting the trailing edge down for the elevators ailerons and stabilators, and the trailing edge left for the rudders.

The actuator and engine dynamics are assumed to be described by the first order systems,

$$\dot{U} = \Lambda(-U + U_C) \tag{5}$$

where $\Lambda=\mathrm{diag}\{1/T_i\}$ and T_i is the time constant of the actuators and engines. U_c indicates the command input vector.

The second term of the right-hand side in Eq. (1) can be rewritten as

$$B(X)U = B_{fst}(X)\delta_{fst} + B_{slw}(X)\delta_{slw} + B_{th}\delta_t \tag{6}$$

where $B_{fst}(X)\in R^{8\times 10}$, $B_{slw}(X)\in R^{8\times 2}$, and $B_{th}\in R^{8\times 4}$(a constant matrix).

Let deflection angles of the stuck surfaces be $\delta_f\in R^{i\times 1}$ (i is the number of the stuck control surfaces) the nominal coefficient matrix for the stuck surfaces be $B_f(X)\in R^{8\times i}$, and the deviation of the stuck surfaces from the nominal trim position be $\Delta\delta_f$, which is assumed to be known. On the assumption that sensors and computers are normal it will not be difficult to identify the stuck surfaces and their deflection angles using potentiometers and an FDI (failure detection and identification) algorithm. Then in order to counteract the disturbances caused by the stuck surfaces using the stabilators or engines, the following equation must be satisfied for X at a trim point.

$$B_f(X)\Delta\delta_f + [B_{slw}(X), B_{th}][\Delta\delta_{slw}{}^T, \Delta\delta_t{}^T]^T = 0 \tag{7}$$

where $\Delta\delta_{slw}$ and $\Delta\delta_t$ are the stabilator angles and engine outputs to be added to the nominal trim angles and feedback thrust inputs, respectively. Solving Eq. (7) for $[\Delta\delta_{slw}^T, \Delta\delta_t^T]^T$ yields

$$[\Delta\delta_{slw}{}^T, \Delta\delta_t{}^T]^T = -[B_{slw}(X), B_{th}]^+ B_f(X)\Delta\delta_f \tag{8}$$

where '+' denotes pseudo-inverse.

Thus, disturbances caused by the surfaces can be rejected by the modification of the stabilator angles and engine outputs. However, the use of engines is to be limited to yawing control in the case of rudder jam. The differential thrust should be kept less than a certain percentage of the maximum thrust. The reasons for the restrictions are:

1) The engines have little effect on the aircraft motions except for the linear one along the X-axis and the angular one about the Z-axis.
2) When airspeed is high or stuck rudder angles are large, counteracting the resulting yawing moment can require too much differential thrust. Since thrust is used for airspeed control, producing large yawing moment by thrust can affect airspeed control.

Consequently, the differential thrust can be considered to have effect on the yawing motion only and the stabilators have little effect on it, so that Eq. (7) can be split into Eqs. (9) and (10), i.e.,

$$B_{fo}(X)\Delta\delta_f + B_{slw}(X)\Delta\delta_{slw} = 0 \tag{9}$$

$$B_{f_r}(X)\Delta\delta_{f_r} + B_{thr}\Delta\delta_t = 0 \tag{10}$$

where $B_{fo}(X)\in R^{7\times i}$ is a matrix obtained by removing the row concerning the yaw motion from $B_f(X)$, and $B_{fr}(X)\in R^{1\times 2}$ and $B_{thr}\in R^{1\times 4}$ are given by the rows concerning the yawing motion in $B_{fr}(X)$ and B_{thr}, respectively. $\Delta\delta_{fr}$ is the displacement of the stuck rudder. Then $\Delta\delta_{slw}$ can be determined from Eq. (9).

$$\Delta\delta_{slw} = -B_{slw}(X)^+ B_{fo}(X)\Delta\delta_f \tag{11}$$

Assuming that $\Delta\delta_{t1}=\Delta\delta_{t2}=-\Delta\delta_{t3}=-\Delta\delta_{t4}$, Eq. (10) can be solved uniquely for $\Delta\delta_{t1}$, i.e.,

$$\Delta\delta_{t1} = -(1/2)B_{f_r}(X)\Delta\delta_{f_r} / (B_{thr}(1,1) + B_{thr}(1,2)) \tag{12}$$

$\Delta\delta_t$ is added to δ_{tcl}^*, which are thrust command inputs given by the feedback control, and $\Delta\delta_{slw}$ to δ_{slwtr}, which are the nominal trim angles of the slow surfaces, to make command inputs to the actuators of the stabilators.

Figure 1 shows a block diagram of the RFCS. It is composed of two parts: the feedforward and feedback ones. The feedback part is further broken down into three parts: the generator of the feedback linearization controls, the discrete-time servo controller, and the parameter identifier. The controls, δ_{Gfst}^* and δ_t^*, serve as reference inputs for δ_{Gfst} and δ_t, respectively, in the servo controller. Control parameters of the control law are modified using the parameters estimated by the identifier. On the other hand, the feedforward part gives control inputs for the slow effectors to accommodate control surface jam.

Let us make three remarks about the control laws, Eqs. (11) and (12).

R1: There are two reasons why the estimated parameters cannot be used to compute the slow controls. But before giving the reasons, let us assume that the actuators of the slow effectors should be given constant command inputs. This assumption will be accepted, because the stuck surface angles are constant and the slow effectors are too sluggish to be used for transient motion control. To give constant command inputs, the parameters in Eqs. (11) and (12) should not be updated. Keeping the above things in mind, let us give the following reasons: First it takes the estimation algorithm some time to converge the estimates, so that the inputs cannot be obtained until the estimates converge. Considering that the effectors are slow, the inputs should be applied as soon as possible after detecting the jams. Secondly the estimated parameters are not true in general. In addition, the trim solution given by Eqs. (11) and (12) is very sensitive to the parameters of the equations. Consequently estimated parameters cannot be used, and we have no choice but to use the nominal parameters. On the other hand, ,there are two advantages of using the nominal parameters. One is that since the parameters are known, the trim solution can be computed and applied on detecting the jams. The other is that using the nominal parameters, an unacceptable trim solution does not occur for most cases of control surface jam. Since the control surfaces do not produce so much force and moment as the slow effectors, if the jam is not too severe, we can expect that the nominal parameters will not give an inadequate trim solution.

R2: Besides the control surface jam, other airframe failures may occur at the same time. Then the nominal model will be inadequate, so that the trim solution will be incorrect. But this does not become a serious problem, because the adaptive feedback control can accommodate the error of the trim solution using the fast control surfaces. Thus, in the proposed RFCS, basically the feedforward control counteracts the forces and moments produced by the stuck surfaces, and the feedback control accommodates airframe failures. If both types of failure occur at the same time, both controls cooperate.

R3: In the case of rudder jam, the stabilator angles and thrusts are independently determined by Eqs. (11) and (12), respectively. Although the control law is based on the idea of using the stabilators for pitching/rolling control and the differential thrust for yawing control, the stabilators may cause the yawing motion and the thrust the pitching/rolling motion. Hence there may be a concern that the stabilators and thrust might cause cross disturbances, if both effectors are used at the same time. However, since the stabilators can produce little yawing moment, they do not significantly affect the yawing control by the thrust settings. On the other hand, the thrusts have little effect on the pitching/rolling control by the stabilators. Therefore no problem exists in using the inputs from Eqs. (11) and (12).

Simulation

Performance of the RFCS is evaluated through computer simulation. The mathematical model used is the 6-DOF nonlinear equations of motion of the Boeing 747 transport. The equations have the same form as those used in Ref. 9 except for the terms of control effectors, which include the stabilator and differential thrust in the transport model. The values of parameters such as non-dimensional aerodynamic force coefficients, moment coefficients, etc. are from Ref. 10. The identification model also is the same as that used in Ref. 9 except for the control effector terms.

The control objective is to trim the aircraft at a desired attitude and airspeed. The outputs to be controlled are selected as $Y=[u, \alpha, \theta, \beta, \phi]^T$ (α =angle of attack and β=sideslip angle). Corresponding to Y, the reference outputs, $Y^*=[u^*, \alpha^*, \theta^*, \beta^*, \phi^*]^T$, are given by the time functions, $Y^*(t)=Y^*(\infty)+\mathrm{diag}\{\exp(-.5t), \exp(-t), \exp(-.5t), \exp(-.5t), \exp(-.5t)\}(Y(0)-Y^*(\infty))$, where $Y^*(\infty)$ is a reference output vector at the desired trim point.

Generic inputs are chosen as

$$U=[\delta_{Gfst}^T, \delta_{Gslw}^T, \delta_{Gt}^T]^T \tag{13}$$

where $\delta_{Gfst}=[\delta_{h1}, \delta_{h2}, \delta_a, \delta_r]^T$, $\delta_{Gslw}=[\delta_{stcl}, \delta_{stdf}]^T$, $\delta_{Gt}=[\delta_{tcl}, \delta_{tdf}]^T$. δ_{Gfst}, δ_{Gslw}, and δ_{Gt} are generic inputs for δ_{fst}, δ_{slw}, and δ_t, respectively, i.e.,

$$\delta_{fst} = P_{fst}\delta_{Gfst} \tag{14}$$
$$\delta_{slw} = P_{slw}\delta_{Gslw} \tag{15}$$
$$\delta_t = P_{th}\delta_{Gt} \tag{16}$$

The elements of the generic input vectors have no meaning in general, but some meaning can be given by properly choosing the CD matrices. Let us give the CD matrices as follows:

$$P_{fst} = \begin{bmatrix} .5236 & .2618 & .2618 & .0 \\ .5236 & .2618 & -.2618 & .0 \\ .5236 & .5236 & .5236 & .0 \\ .5236 & .5236 & -.5236 & .0 \\ .2618 & .5236 & .5236 & .0 \\ .2618 & .5236 & -.5236 & .0 \\ .2618 & .2618 & .2618 & .0 \\ .2618 & .2618 & -.2618 & .0 \\ .0 & .0 & .0 & .5236 \\ .0 & .0 & .0 & .5236 \end{bmatrix}, P_{slw} = \begin{bmatrix} 1 & 1 \\ 1 & -1 \end{bmatrix}, P_{th} = \begin{bmatrix} 1 & 1 \\ 1 & 1 \\ 1 & -1 \\ 1 & -1 \end{bmatrix}$$

With the matrices, δ_{h1}, δ_{h2} and δ_{stcl} correspond to the control surfaces for the pitching motion control such as the elevator, δ_a and δ_{stdf} for the rolling motion control such as the aileron, and δ_r for the yawing motion control such as the rudder. δ_{tcl}, and δ_{tdf} mean the collective and differential thrust, respectively.

δ_{Gfst} and δ_{tcl}, are given by the feedback linearization control law. δ_{Gslw} is determined by Eq. (11) and $\delta_{tdf}(=\Delta\delta_{tl})$ by Eq. (12). In addition, the following conditions are assumed. The time constants are 2 sec for the stabilators, 0.1 sec for the other surfaces, and 5 sec for the engines. The deflection limits are ± 20 deg for the stabilators and ± 30 deg for the other surfaces. The available thrust range of each engine is $0 \leqq \delta_{ti} \leqq 7.1\times10^4$ N. The differential thrust is limited to $-2\times10^4 \leqq \Delta\delta_{ti} \leqq 2\times10^4$ N. The updating intervals are 0.02 sec for the inputs to the actuators, 1 sec for those to the engines, 0.05 sec for the estimated parameters, 0.05 sec for the control parameters.

The control objective is to trim the aircraft at a nominal trim point, $Y^*(\infty)$, in all simulation cases except Case 8 in Simulation #1.

<u>Simulation #1</u> The flight condition chosen had an altitude of 6080 m and airspeed of 205 m/sec. The nominal trim variables are u= 205 m/sec, $\alpha = 2.5$ deg, $\theta = 2.5$ deg, $\beta = 0$ deg, $\phi = 0$ deg, $\delta_{ti}\times 4 = 1.56\times10^5$ N, $\delta_{stL} = \delta_{stR} = 0.5$ deg, and other control surface angles = 0 deg. The initial states are $X(0)=[210, 21, 0, 0.1, 1, 0, 0, 0.1]^T$. The following four failure cases are considered.

Failure A : the inboard right elevator and the outboard right and left elevators are stuck at 0.3 rad .
Failure A': in addition to Failure A, the effectiveness of the left stabilator is reduced by 40 %.
Failure B : the upper rudder is stuck at -0.2 rad and the effectiveness of the lower rudder is reduced by 50 %.
Failure B': in Failure B, the effectiveness of the lower rudder becomes 0.

The investigated cases are summarized in Table 1.

Table 1 Simulation Cases

Case	Failure	$\Delta\delta_{stL}$(deg)	$\Delta\delta_{stR}$(deg)	$\Delta\delta_{ti}$(N)
1	A	.0	.0	0
2	A	-4.24	-4.35	0
3	A'	-4.24	-4.35	0
4	B	.858	-.858	0
5	B	.858	-.858	-2×10^4
6	B'	.858	-.858	-2×10^4
7 & 8	B'	-11.46	11.46	-2×10^4

Case 1 : In this case, the flight control system is restructured using the fast control surfaces only. The slow effectors, i.e., the stabilators, are not used. Figure 2 shows the results. The aircraft dives, rolling as much as -180 deg. It gets out of control, and eventually crashes to the ground around 35 sec after the failure.

Case 2 : Figure 3 indicates that in contrast with Case 1 the aircraft keeps the nominal level flight with the feedforward control utilizing the stabilators, which counteract the disturbance caused by the stuck elevators.

Case 3 : In this case, the nominal parameters in Eq. (11) or (12) have changed due to the failure on the left stabilator. The results shown in Fig. 4 indicate that the effects of the parameter change can be accommodated by the feedback control using the fast effectors. Though it takes more time to trim the aircraft than in Case 2, a level flight is attained in spite of using the wrong parameters.

Case 4 and Case 5 : While in Case 4 the stabilators only are used to counteract the rudder jam, in Case 5 both stabilators and differential thrust are used. In either case, as Fig. 5 and Fig. 6 show, the aircraft recovers to level flight. However, convergence of the outputs in Case 4 is slower than that in Case 5. Besides, while the sideslip angle does not converge to $\beta^*(\infty)=0$ deg in Case 4, it does in Case 5. Moreover, the deflection angles of the control surfaces in Case 5 are smaller than those in Case 4. This means that in Case 5 more control power is left after the failure accommodation. In Failure B, displacement of the stuck rudder is so large that the desired differential thrust exceeds the limit, -2×10^4 N. However, the actual differential thrust, although less than desired, improves the performance of the RFCS anyway.

Case 6 and Case 7 : In the above two cases, it is because the lower rudder, whose effectiveness decreases by half, is available that the level flight can be recovered. However, in Case 6 where the effectiveness becomes 0, the results indicate that the yawing moment caused by the stuck rudder cannot be canceled by any means. In fact, Fig. 7 shows that the aircraft loses control and crashes to the ground at about 47 sec after the failure. Thus, the control law given by Eqs. (11) and (12), which is used in Case 6, cannot save the aircraft. Considering the aircraft dynamics, first the stuck rudder directly produces a positive yawing moment and a negative rolling moment. But the yawing motion generates the large positive rolling moment resulting from the lift difference between the left and right wings. In Case 6, while the sideslip angle is not so large, the roll angle increases to about 150 deg. Note that Eq. (11) gives the stabilator angle to counteract the negative rolling moment caused by the rudder jam, but unfortunately it results in accelerating the positive rolling. As seen from the results, the effect of the rudder jam is more serious for rolling stability. Therefore when failures produce large yawing moment that cannot be canceled, some way to counteract the rolling moment must be taken. From this observation, in Case 7, the stabilator angles are given not by Eq. (11) as in Case 6, but appropriately so that the negative rolling moment can be produced. A control law giving the angles have not been obtained yet. As seen from Fig. 8, the aircraft motions do not diverge, though they are not settled perfectly.

Case 8 : The responses can be improved by changing the desired trim point. A trim point with a negative sideslip angle and a negative roll angle will counteract the rudder jam by the vertical tail. In fact, in this case, choosing $\beta^*(\infty)=-0.05$ rad and $\phi^*(\infty)=-0.2$ rad , the better time responses, which are shown in Fig. 9, are obtained. Since the modified trim point is an uncoordinated one, it may look as if the aircraft were flown in a crosswind approach.

Simulation #2 Let us consider a case similar to the accident of the El Al Boeing 747 freighter on Oct. 4, 1992. In that accident, engine #4 dropped and a fire started in engine #3, 7 minutes after taking off. In order to return to the airport, the aircraft was going around to descend. However, it crashed into an apartment building soon after flap trouble happened.

In the simulation, the flight condition is that altitude is 500 m and airspeed is 85 m/sec . The aircraft is in a -3-degree landing approach. The nominal trim variables are u = 85 m/sec, α = 5.7 deg, θ = 2.7 deg, β = 0 deg, ϕ = 0 deg, $\delta_{ti} \times 4 = 0.986 \times 10^5$ N, the stabilator angles are $\delta_{stL} = \delta_{stR} = 0.5$ deg, and other control surface angles are 0 deg. The initial states are $X(0)=[80, 8, 0, 0.05, 5, 0, 0, 0.2]^T$. As failures, engine #3 is shut down, engine #4 drops, and the inboard and outboard ailerons are stuck at -0.943 deg . In this example, the flap trouble with the El Al freighter is replaced by an outboard aileron jam.

Figure 10 shows the flights with and without the feedforward control by the stabilators. In Fig. 10, the aircraft is drawn every 2.5 sec. Without using the feedforward control, roll angle reaches as much as 60 deg, and the airspeed, pitch angle, and angle of attack are not controlled at all. The aircraft started to roll largely at 2.5 sec and crashes to the ground at 24 sec. On the other hand, using the feedforward control, the outputs are controlled to the desired values, and the -3-degree approach is recovered successfully. Although the altitude and direction have deviated from the nominal ones because of the initial disturbances, it would not be difficult for pilots to modify them, once a stable flight has been recovered.

Conclusions

This paper presents a method to accommodate failures that affect aircraft dynamical characteristics, especially control surface jams on a large transport aircraft. The approach is to use the slow effectors, such as the stabilators or engines, in the feedforward manner. The simulation results indicate the performance of the RFCS. In some cases of control surface jam, the aircraft cannot recover without using the stabilators. Although the inputs to the slow effectors are determined using the nominal parameters, the effects of parameter change can be compensated by adjusting the control parameters for the fast surfaces. In the case of rudder jam, if the remaining control surfaces and the differential thrust cancel the moments produced by the stuck rudder, using the engine control improves time responses and reduces deflection angles of the control surfaces. If not, however, the aircraft starts a large rolling motion following a yawing motion. In that case, the stabilators should be used to damp the induced rolling motion, instead of trying to directly cancel the moments caused by the stuck rudder. Unfortunately, the proposed control law for the stabilators does not give such inputs, because it does not take into account the *dynamical* effects which stuck surfaces have on the aircraft motions. However, we have shown through simulation that the aircraft can be recovered by giving the stabilators the control inputs that counteract the induced rolling moment. Besides, the method has also been shown through simulation to be effective in maintaining control during a situation similar to an actual accident. Finally let us mention a problem with the RFCS. As stated above, we have not established a method to select a trim point which can be reached as easily as possible using the remaining control effectors. In fact, recovery performance considerably depends on the trim states. As pointed out in Ref. 11, finding the best trim point for impaired aircraft will be one of the most difficult questions in RFCS design.

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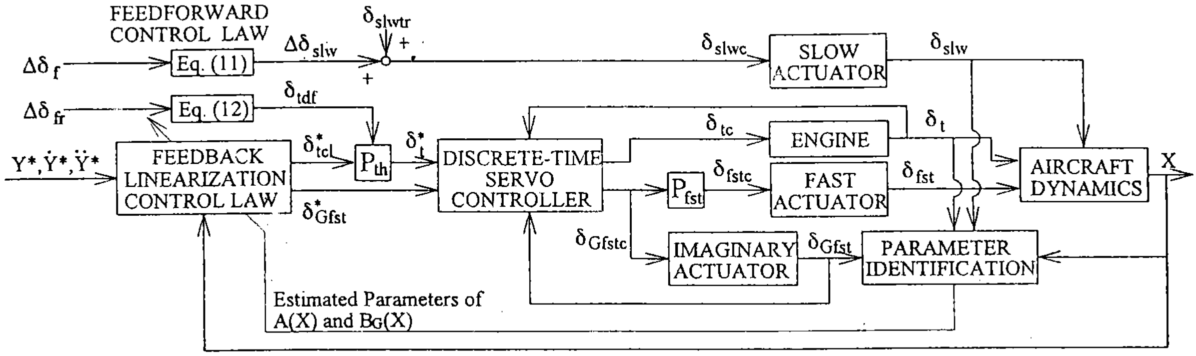

Fig. 1 Block diagram of the RFCS

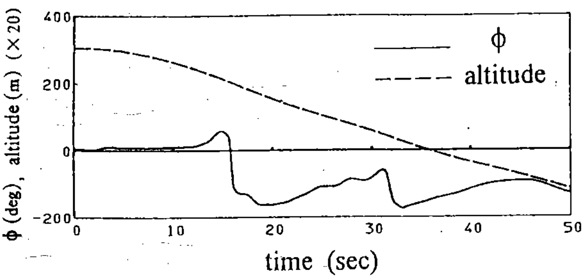

Fig. 2 Time responses without using the stabilators
(Case 1: Failure A)

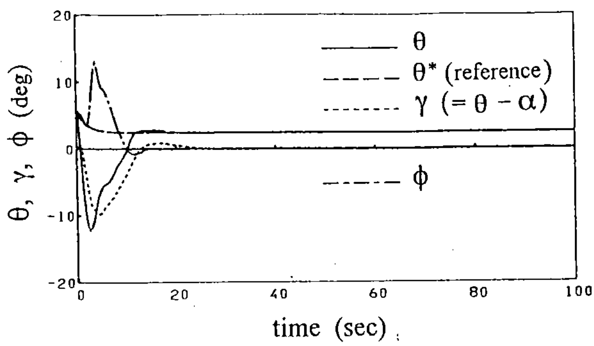

Fig. 3 Time responses using the stabilators
(Case 2: Failure A)

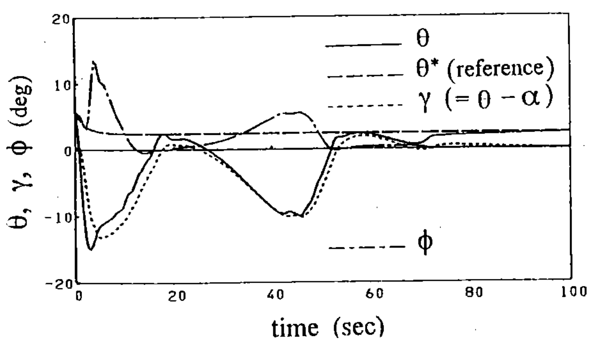

Fig. 4 Time responses using the stabilators
(Case 3: Failure A')

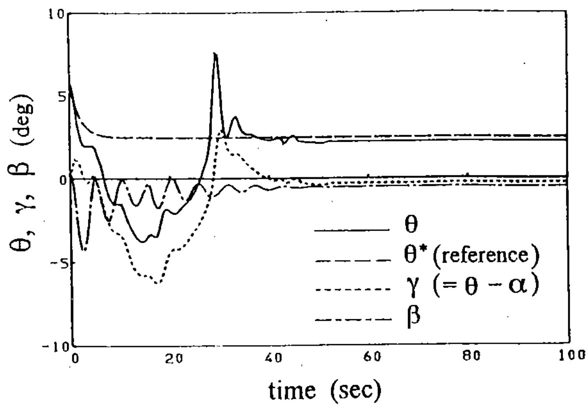

Fig. 5 Time responses using the stabilators
(Case 4: Failure B)

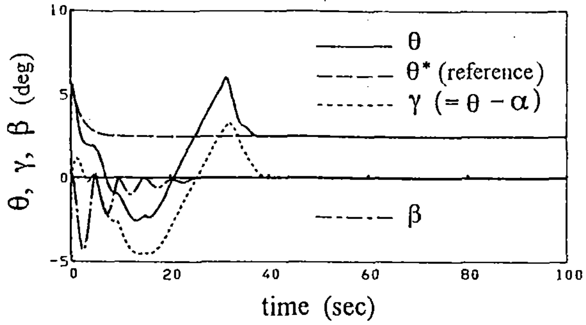

Fig. 6 Time responses using the stabilators and differential thrust(Case 5: Failure B)

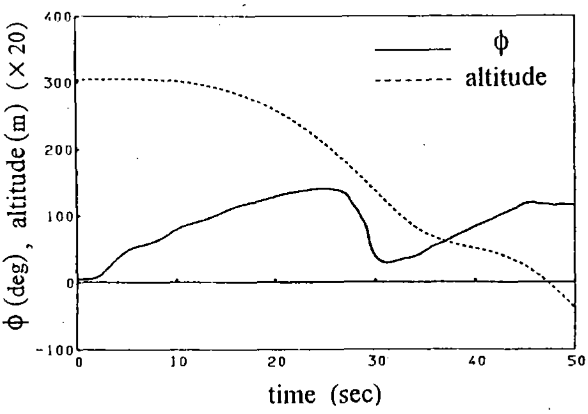

Fig. 7 Time responses using the stabilators and differential thrust(Case 6: Failure B')

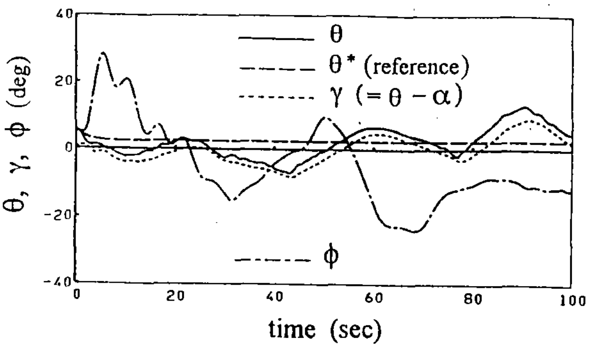

Fig. 8 Time responses using the stabilators and differential thrust(Case 7: Failure B', nominal trim point)

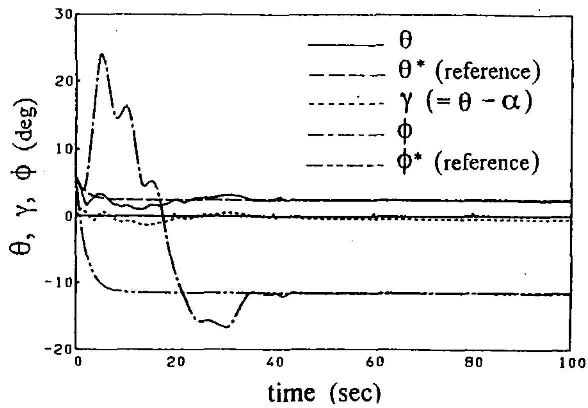

Fig. 9 Time responses using the stabilators and differential thrust(Case 8: Failure B', new trim point)

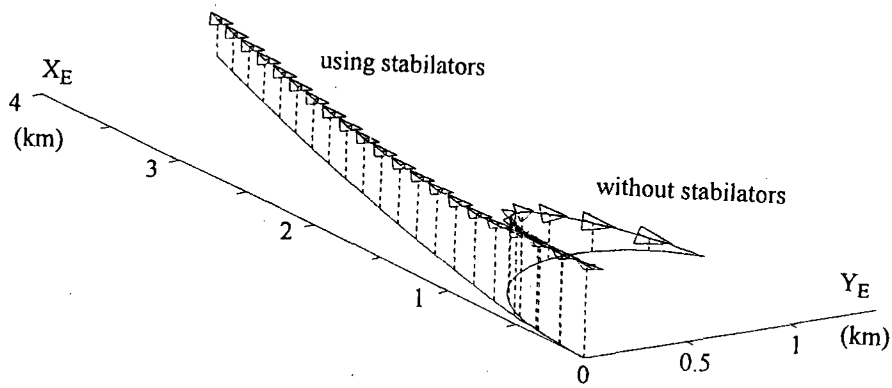

Fig. 10 Graphical representations of the flight with and without using stabilators