Dynamic response analysis of aircraft with actuation system failure: open-loop and closed-loop cases

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Abstract. On flight control implementation, the control surface actuator system plays one of the key roles that will significantly affect the closed-loop performance of the aircraft. The actuators must provide the required effort for moving the control surface so that the desired control action can be reached. Any failure on the actuator system will limit or degrade the operation of the control surface, so that it will affect the performance of the system in controlling the aircraft, or even worse, may bring the aircraft into an unsafe condition. In this project, the effect of some control surface problems due to actuator failures are modeled, simulated, and analyzed. To reconstruct the situation, the control device failure model is integrated into the flight dynamic model of an aircraft. The numerical model of the equation then is used for simulating the aircraft response under some failure conditions. It can be seen from the simulation results that control surface failure can significantly affect the trim condition of the aircraft, since the failures can change the balance of forces and moments that work on the aircraft. The simulation results also show that for some control surface failures may initiate an asymmetrical response from the aircraft dynamics. On closed-loop simulations, where some type of flight control systems for stabilization and maneuvering tasks are implemented, control surface failures may affect the performance of the control system. The results of the study suggest that any possibilities of control surface failure must be anticipated, by using a robust control scheme or other reconfigurable schemes that can cope with the change in the dynamic characteristics due to the failures.

Keywords: actuation system failure, dynamic response, stability, control, flight control system

1. Introduction
Failure in the actuator will change the dynamic characteristics of the aircraft itself. So it is very necessary to have an input that can re-adjust these dynamic characteristics so that it does not allow LOC (Loss Of Control) to occur. LOCs can endanger passengers on aircraft due to uncontrollable aircraft.

To analyze further, it is necessary to look at the effect of this actuator failure if it is implemented in a flight control system either to see its stabilization or to see the response to achieve the desired variable value. As the initial hypothesis where in case of failure there will be the inaccuracy of the aircraft output issued by the flight control system when compared with the given input. This is due to an error when reading the response of the deflection of the control surface failure. Therefore, there is a deflection of the other control surface to replace the performance of the control surface failure.
For modeling, it is necessary to make an aircraft flight control system design that can match the failure of the actuator. Therefore, to develop the existing AFCS system, response data from the actual condition of the aircraft due to actuator failure are needed which will later be used in the aircraft control system model. Based on the explanation above, this research will focus on designing an aircraft control system in normal conditions and various failure cases. Then, it will be used to analyze the dynamics and control of the aircraft with the hope that in the future the results of this research can help pilots in controlling the aircraft.

The model to be used in this study is the F4 phantom by modeling all control fields independently with a stick-fixed assumption so the actuator and control surface as one integrated system, then forming a non-linear model which is then linearized so that a linear model can be used to perform failure analysis with various desired cases.

2. Modeling of Aircraft

In this research, the F4 phantom is used as an airplane model and the specification contains the properties of the aircraft can be seen in the reference [1, 2]. In addition to the data, the F4 phantom also has several limitations of its control surface deflection which are presented in Table 1 below.

| Table 1. Control Surface Deflection Limit [2] |
|---------------------------------------------|
| Control surface deflection                  |
| $-21^\circ \leq \delta_e \leq 7^\circ$     |
| $-30^\circ \leq \delta_r \leq 30^\circ$   |
| $-15^\circ \leq \delta_a \leq 15^\circ$   |
| $0 \leq \delta_{TL} \leq 1$               |

The non-linear mathematical equation of motion described the aircraft motion in the body-fixed reference frame, it consists of 12 equations [3]. In each equation, there are the non-dimensional aerodynamics force and moment coefficients for the F4 model [2] relative to the mean aerodynamics center and the wingspan. In this research, the aerodynamic coefficient applies to fly with a maximum angle of attack of $15^\circ$ [2]. In this research, there will be a failure model also. Failure that occurs in actuators will induce other unwanted movements. The effects of these failures are summarized in table 2 below [1].

| Table 2. Actuator Failure Effect [1] |
|-------------------------------------|
| Aircraft motion | Left Elevator | Right Elevator | Rudder | Left Aileron | Right Aileron | Left Engine | Right Engine |
| Surge             | V             | V              |        |              |              | V           | V           |
| Sway              |                |                |        |              |              | V           | V           |
| Heave             | V             | V              |        |              |              | V           | V           |
| Roll              | V             | V              | V      | V            | V            | V           | V           |
| Pitch             | V             | V              | V      | V            | V            | V           | V           |
| Yaw               | V             | V              | V      | V            | V            | V           | V           |

Failure cases and their effects are then modeled as follows

$$\delta_e = \frac{1}{2}(\delta_{eL} + \delta_{eR})$$ (1)
\[ \delta_a = \frac{1}{2} (\delta_{aL} + \delta_{aR}) \]  
\[ T = (T_L + T_R) \]  

Due to the above failure modeling, the control surface deflection limit is also affected, shown in Table 3.

**Table 3. Control Surface Deflection Limit due to Actuator Failure.**

| Control surface deflection |  
|\(-21^\circ \leq \delta_{eL} \leq 7^\circ\) |  
|\(-21^\circ \leq \delta_{eR} \leq 7^\circ\) |  
|\(-30^\circ \leq \delta_r \leq 30^\circ\) |  
|\(-15.5^\circ \leq \delta_{aL} \leq 15.5^\circ\) |  
|\(-15.5^\circ \leq \delta_{aR} \leq 15.5^\circ\) |  
| 0 \leq \delta_{T_L} \leq 1 |  
| 0 \leq \delta_{T_R} \leq 1 |  

Due to the adjustment of the control surface resulting in adjustments to the equation of motion of the aircraft and its aerodynamic coefficient. Some of them shown as follows

\[ \dot{u} = -qw + vr + g \sin \phi + \frac{F_{AX} + T_L + T_R}{m} \]  
\[ \dot{\psi} = \frac{q}{I_{ZX}}(l_Y - l_Z - I_X) + \frac{pq}{I_{ZX}^2}(I_{ZX}^2 + I_X^2 - I_Y l_Y) + \bar{q} S_b(C_s I_{ZX} + C_n I_X) + T_L(Y_{\delta_e}) - T_R(Y_{\delta_e}) \]

\[ C_X = -0.0434 + 2.39 \times 10^{-3} \alpha + 2.53 \times 10^{-5} \beta - 1.07 \times 10^{-6} \alpha \beta^2 \]  
\[- 4.75 \times 10^{-4} \delta_{eL} \]  
\[- 4.75 \times 10^{-4} \delta_{eR} \beta^2 \]  
\[ + \left( \frac{180 q \bar{c}}{\pi 2 V_t} \right) \left( 8.73 \times 10^{-3} + 0.001 \alpha - 1.75 \times 10^{-4} \alpha^2 \right) \]  
\[- 1.695 \times 10^{-5} \delta_{aL} + 1.695 \times 10^{-5} \delta_{aR} \]

\[ C_Y = -0.012 \beta + 1.55 \times 10^{-3} \delta_r - 8 \times 10^{-6} \delta_r \alpha \]  
\[ + \left( \frac{180 b}{\pi 2 V_t} \right) \left( 2.25 \times 10^{-3} \beta + 0.0117 \beta - 3.67 \times 10^{-4} \beta \alpha \right) \]  
\[- 0.875 \times 10^{-4} \beta \delta_{eL} + 0.875 \times 10^{-4} \beta \delta_{eR} \]

\[ C_Z = -0.131 - 0.0538 \alpha - 2.38 \times 10^{-3} \delta_{eL} - 2.38 \times 10^{-3} \delta_{eR} \]  
\[- 1.65 \times 10^{-5} \delta_{eL} \alpha - 1.65 \times 10^{-5} \delta_{eR} \alpha - 3.75 \times 10^{-5} \delta_{aL} \beta^2 \]  
\[- 7.5 \times 10^{-5} \delta_{aL} \beta \delta_{aR} - 3.75 \times 10^{-5} \delta_{aR} \beta^2 \]  
\[- \left( \frac{180 q \bar{c}}{\pi 2 V_t} \right) \left( -0.111 + 5.17 \times 10^{-3} \alpha - 1.1 \times 10^{-3} \alpha^2 \right) \]  
\[ + 0.735 \times 10^{-3} \delta_{aL} + 0.735 \times 10^{-3} \delta_{aR} \]

After the analytical model, the author builds a numerical model of the aircraft with failure conditions that are already included. The numerical model of the aircraft is designed using Simulink-Matlab software as shown in Figure 1.
3. Controller design

3.1 Trimming and Linearization

In the trimming process [4], it is set to maintain the velocity value on the XB-Axis and the altitude of the aircraft, which means we maintain the aircraft in a cruise condition. Besides that, the deflection limit of the control plane and the angle of attack value is set according to the reference and the cases. Trimming will be carried out in several cases presented in Table 4.

Table 4. Simulation Cases

| Failure Case | Failure Description |
|--------------|---------------------|
| Case 1       | Normal Case, all the actuator/control surface are healthy and can be used normally |
| Case 2       | Elevator Failure means the right elevator of the airplane cannot be deflected |
| Case 3       | Aileron Failure which means the right aileron of the airplane cannot be deflected |
| Case 4       | Elevator-Aileron Failure which means the left elevator and the right aileron cannot be deflected |
| Case 5       | Elevator-Rudder Failure which means the Left Elevator and the Rudder cannot be deflected |
| Case 6       | Aileron-Rudder Failure which means the left aileron and the rudder cannot be deflected |

The failure condition referred to above is a jamming failure condition where the control surface is stuck even though it has been given input so that before doing the trimming process the control surface deflection limit is adjusted to a value of 0 on the control surface/actuator that is a failure. Trimming is done with the value of operation/holding conditions at an altitude of 1000 m above sea level with a speed of 215 m / s at XB-axis and the maximum angle of attack is 15°. After that, the linearization process is carried out and produces a state-space model, there are 6 state-space models for each case.

In a state-space or linear model that produce by the linearization process. There are some variables such as A, B, C, D, x, and y, A is State matrix with size 8x8, B is input matrix with size 8x7, C is output matrix with size 8x8, and D is zero matrix. While x is a state vector with size 8x1 consist of u, w, θ, q, v, p, r, and φ. Y is output vector with size 8x1 consist of u, α, θ, q, β, p, r, and φ. And the last one u is the input vector with size 7x1 consist of control surface deflection that symbolized as in table 3. After
that, an open-loop analysis is carried out by providing input impulses to see the stability of the aircraft. When giving input still sees the deflection limit of the linear model control surface in table 3 with an adjustment for each case to value 0 as mention before. The location of poles for each of the linear models, all of them located in the stable area. so we just need to improve the stability of the system by design a stability augmentation system.

3.2. Closed-Loop Model
To perform closed-loop analysis, gain control is required. Determination of control gain using the Linear Quadratic Regulator (LQR) method [5]. LQR can be determined by solve optimization problem as follow

$$J = \int_0^\infty \bar{x}^T (Q + K^T R K) \bar{x} \, dt$$

(9)

where x is the state variable and u input variable, the input signal that fulfills the optimization problem (9)) can be defined as follows

$$u(t) = -Kx(t) = -R^{-1}B^TPx(t)$$

(10)

where P is a definite positive matrix solution of the Riccati equation defined as follow

$$A^T P + PA + Q - PB R^{-1}B^T P = 0$$

(11)

There are 7 gain controls obtained from each state space. The gain control obtained is used for closed-loop which will then be carried out the simulation process.

4. Results and Discussions
4.1 Closed-Loop Stability Analysis and Failure Cases
The closed-loop model is made using a linear model and the gain control has been get, the gain control will process the feedback signal [5] in the form of the open-loop system state variable value. the closed-loop model will follow Figure 2 below

![Figure 2. Closed-Loop Model](image)

By using the model in Figure 2, several simulations were carried out with a linear model (state space) and a different controller except for G_Norm (State Space Model in normal cases) the controller only from K_Norm (The gain is obtained by calculating using the state variable on G_Norm), but for the other, the closed-loop model will follow G_Failure(State Space Model in each failure cases, i.e elevator failure = G_Elev)+K_Norm, and G_Failure+K_Failure(The gain is obtained by calculating using the state variable on G_Failure, i.e. K_Elev from G_Elev).

There will be 15 simulations consist of 5 simulations of G_Norm+K_Norm with input for each control surface deflection and 10 simulations for each failure case (for every G_Elev simulation only get elevator deflection, for every G_Ail simulation will get aileron deflection), the input given is doublet signal. The result for each simulation with the same input will be compared, for example, G_Norm+K_Norm (elevator input) will be compared with G_Elev+K_Norm. after that G_Elev+K_Norm will be compared with G_Elev+K_Elev, the comparison will follow for case 2 until 6 in table 4.
From the simulation results, it can be seen that with normal gain control when an elevator failure occurs, the actuator cannot produce the same energy so that the control surface in this case the right elevator does not give any response due to the change in deflection. The output in the flight motion plane that corresponds to the control surface failure, in this case, the longitudinal flight motion plane being reduced from the normal response, so that the control surface that responds due to the feedback, in this case, the thrust produces a different response from the normal.

Then due to the failure, it causes an asymmetric condition so that the coupling effect occurs which makes the response to other flight motion planes, in this case, a lateral-directional flight motion plane.
Also because of the coupling condition, to keep the aircraft in a trim condition, the control surface that not compatible with the longitudinal flight motion plane, in this case, the rudder will be deflected. This condition also applies to compare simulations $G_{\text{Norm}}+K_{\text{Norm}}$ with $G_{\text{Ail}}+K_{\text{Norm}}$ or $G_{\text{EA}}+K_{\text{Norm}}$ or $G_{\text{ER}}+K_{\text{Norm}}$ or $G_{\text{AR}}+K_{\text{Norm}}$. while when compared between simulations for each failure linear model with different controllers for example $G_{\text{Elev}}+K_{\text{Norm}}$ with $G_{\text{Elev}}+K_{\text{Elev}}$, it is found that using a different controller in the linear model the same failure will not affect the output obtained.

4.2 Flight-Path Angle Hold and speed hold

To create a flight-path angle hold model and speed hold. The closed-loop model in figure 2 will be an inner closed loop and there will be 2 outer loops, so there will be 2 feedbacks, different from figure 2 this time the feedback comes from the output. The feedback value is the aircraft velocity obtained by transforming the variable $u$ (Velocity at XB-axis) by using equation (12), and the second is the flight path angle value obtained through the transformation of the variable values for pitch angle ($\theta$) and angle of attack ($\alpha$) by using equation (13).

Then the feedback variable is compared with the reference value. After that, the result of the comparison of the two values will be transformed using the PID controller into an input signal to the actuator so that there are 2 PID controllers for speed hold and flight-path angle hold. The flight path angle hold and speed hold model is shown in figure 5.

\begin{align}
V &= \frac{u}{\cos \alpha} \\
\gamma &= \theta - \alpha
\end{align}

![Figure 5. Closed-Loop Model for flight path angle hold and speed hold](image)

As mentioned before there will be 2 PID controllers, but because the PID for speed hold and flight-path hold is in the same closed-loop model design so that it is assumed to be 1 unit so that the names become 1 unit, and are used simultaneously. naming for the PID controller follows each case. After that, the simulation is carried out as in table 4, each linear model is paired with its PID for example, $G_{\text{elev}} + K_{\text{Norm}} + \text{PID}_{\text{elev}}$ will be simulated, then $G_{\text{elev}} + K_{\text{elev}} + \text{PID}_{\text{Elev}}$ and so on except for $G_{\text{Norm}}+K_{\text{Norm}}$ will only use $\text{PID}_{\text{Norm}}$.
Based on the result obtained, it can be seen that the normal model has the best performance so that a faster response is obtained to reach the desired reference value, according to the analysis in the previous section when there is a dynamic response failure there will be a decrease in performance and due to the failure of the control response, it produces an asymmetric effect that makes the response appear in the lateral-directional flight motion plane while the flight control system is designed for dynamics in a longitudinal flight motion plane.

According to the results obtained, it can be seen in the elevator-Aileron failure model the flight control system cannot be applied which has been designed using either normal gain or failure gain for the inner loop. This condition because of the PID controller that uses in the flight-path angle hold and speed hold mode. The PID controller is SISO while the model is MIMO, although the PID controller
for each flight-path angle hold and speed hold but each PID does not work as a unit so it does not care about other PID conditions which result in the desired response being difficult to obtain because each PID "works" to achieve the desired value.

5. Conclusions

Actuator failure that results in jamming conditions or the control surface is unable to deflect can be modeled using a normal controller or a controller from a model with failure condition included. In the failure condition, the aircraft can reach its trim condition even though there are limitations, therefore the linearization process can be calculated and the linear model can be obtained. By using the linear model and the controller that the aircraft has obtained, it can still be stabilized even though it creates unwanted dynamics (coupling effect) due to actuator failure. Besides, the aircraft can still be controlled to obtain the desired value, but there are 2 conditions when the failure of the elevator-aileron and aileron-rudder of the aircraft cannot be controlled. For future work, dynamic modeling of the actuator can be carried out (stick-free condition) so that the results are more accurate, besides that, analysis can also be carried out on other types of failure such as changes in the dynamic rate of control surface deflection, after that in the future work, automatic control modeling can be carried out which detects damage so that it can be anticipated to produce dynamic as desired and for the flight-path angle hold and speed hold can use another type of controller.

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