Active Vibration Control for Aircraft Wing*

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In active vibration control, gust load alleviation and flutter suppression method are studied for aircraft. For this purpose, an aero-servoelastic analysis code was developed using the finite element code MSC/NASTRAN. Moreover, a control law synthesis method was developed and applied to the LQG (linear quadratic Gaussian) method for reduced-order control law design. To validate both aeroservoelastic analysis and the control law synthesis code, the wind tunnel test of gust load alleviation was conducted in a 2 m low-speed wind tunnel of Nagoya Aerospace Systems Works using an aeroelastic wind model equipped with an actively controlled trailing edge surface. Analytical results were compared with the results from tests, and good agreement was obtained for wing acceleration. To estimate the effects of gust load alleviation and flutter suppression, they were applied to a 150-passenger twin-engine commercial airplane. For gust load alleviation, a 46.9% wing bending moment reduction was caused, as well as a flutter suppression of 30.7% flutter speed augmentation.

Key Words: Active Vibration Control, Gust load Alleviation, Flutter Suppression, Linear Quadratic Gaussian Method, Wind Tunnel Test

1. Introduction

Active vibration control is being applied to aircraft for the purpose of increasing performance, extending the operational region and decreasing structural weight⁽¹⁾. In active control technology, gust load alleviation and flutter suppression methods are classified as controllers of structural vibration. These vibration control methods are for practical uses according to the development of control technology.

In the second and third sections, the analysis and control law synthesis methods for actively controlled aircraft are described. In the next section, the wind tunnel test results are described to verify our method, and in the final section, effects of active vibration control are discussed, as applied to a 150-passenger twin-engine commercial airplane.

2. Analytical Method for Aeroservoelastic Systems

A computational code (Fig. 1) combined with the

FEM code MSC/NASTRAN was developed to analyze the aeroservoelastic system such as an aircraft with active vibration control. In this code, the structural vibration model and unsteady aerodynamic model are generated from aircraft shape, weight and rigidity data. In unsteady aerodynamics calculation the DLM (doublet lattice method) is used as the frequency domain expression. In the next step, generalized mass, stiffness and aerodynamic matrices are calculated from these models and the aerodynamic matrices are augmented from the frequency domain to the Laplace domain to express arbitrary aircraft motion. In order to adapt further analysis and control law synthesis, finite-dimensional aerodynamics approximation, the Roger method, is used^{(2),(3)}.

$$[Q] = [A_0] + [A_1]S \cdot \frac{C}{2V} + [A_2]S^2 \left(\frac{C}{2V} \right)^2 + \sum_{m=3}^L \frac{[A_m]S \cdot \frac{C}{2V}}{S \cdot \frac{C}{2V} + \beta_{m-2}} \quad (1)$$

Here, β_{m-2} ($m=3 \sim L$) is aerodynamic time lag, $[Q]$ is generalized aerodynamic matrix, $[A_m]$ is aerodynamic coefficient matrix ($m=0 \sim L$) and L is the

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degree of freedom where $L=4$ to 6 is ordinarily used. A typical approximation example is described in Fig. 2. In Fig. 2, (1,1) and (2,2) elements of $[Q]$ are plotted in the complex domain, and the DLM aerodynamic force and the finite dimensional aerodynamic force are equal.

Next, the generalized mass, stiffness and aerodynamic matrix are converted to the state space form in order to enable use of a control law design. In this process, control surface actuators and the gust dynamics are modeled.

$$\{\dot{X}\} = [F]\{X\} + [G_u]\{u\} + [G_w]\{\eta_w\} \quad (2)$$

Here, $\{X\}$ is state variable, $\{u\}$ is control command, $\{\eta_w\}$ is gust input force and $[F]$, $[G_u]$ and $[G_w]$ are coefficient matrices corresponding to those variables. The stability analysis and response analyses, such the as gust response and control surface response analyses, of the aeroservoelastic system are conducted.

3. Control Law Synthesis Method

To design aircraft vibration control laws such as gust load alleviation and flutter suppression, the LQG

Fig. 1 Outline of analysis flow

Fig. 2 Finite dimensional approximation of aerodynamic

method is used for reduced-order control law synthesis⁽⁴⁾. The control law design procedure of this method is shown in Fig. 3. The mathematical model for control law design is the state space form that includes structure, aerodynamic force, actuator and gust dynamic models described in Eq. (2). However, the mathematical model for control design is simplified to neglect higher structural vibration modes that are used for stability and response calculation. The optimal control gain and Kalman gain of full states are used for the control law initial value. Next, the dimension of a control law is reduced. In this step, reduced states $\{X_c\}$, called key states, are selected from full states $\{X\}$ using

$$\{X_c\} = [R]\{X_s\} \quad (3)$$

Here, $[R]$ is a state transform Boolean matrix. In the case of vibration control, $\{X_c\}$ is a state corresponding to structural vibration modes. The dimension of $\{X_c\}$ is equal to the control law dimension. The control law is described by a dynamic compensator in the form

$$\begin{aligned} \{\dot{X}_c\} &= [A]\{X_c\} + [B]\{Y\} \\ \{u\} &= [C]\{X_c\} + \{\eta_u\} \end{aligned} \quad (4)$$

Here, $\{Y\}$ is a measurement state, $[A]$, $[B]$, $[C]$ are coefficient matrices and $\{\eta_u\}$ is fictitious noise introduced for stability augmentation.

$$\{Y\} = [H]\{X_s\} + \{\eta_v\} \quad (5)$$

Fig. 3 Optimal reduced order control law design algorithm

Here, $[H]$ is a measurement matrix and $\{\eta_v\}$ is a measured noise. From Eqs.(2), (4) and (5), a closed loop equation is constructed:

$$\{\dot{X}_a\} = [F_a]\{X_a\} + [G_a]\{\eta\} \quad (6)$$

and $\{X_a\} = \{X, X_c\}^T$. Here,

$$[F_a] = \begin{bmatrix} F_s & G_u C \\ B H & A \end{bmatrix} \quad (7)$$

$$[G_a] = \begin{bmatrix} G_u & G_w & 0 \\ 0 & 0 & B \end{bmatrix} \quad (8)$$

$$\{\eta\} = \{\eta_u \ \eta_w \ \eta_v\}^T \quad (9)$$

When the system matrix $[F_a]$ is stable, $\{X_a\}$'s covariance matrix $[X_a]$ satisfies the Lyapunov equation described below.

$$[F_a][X_a] + [X_a][F_a]^T = -[G_a][R_a][G_a]^T \quad (10)$$

Here, $[R_a]$ is a covariance matrix of noise vector $\{\eta\}$. The control law is designed to minimize the cost function J .

$$J = E[\{X_a\}^T [Q_a] \{X_a\}] \\ = \text{tr} [[Q_a] [X_a]] \quad (11)$$

Here, $[Q_a]$ is a weighting matrix. A Lagrange multiplier matrix $[\Lambda]$ is introduced to construct an adjoined cost function $\bar{J}$.

$$\bar{J} = \text{tr} [Q_a x_a + \Lambda (F_a x_a + x_a F_a^T + G_a R_a G_a^T)] \quad (12)$$

When the cost function has the minimum value, it satisfies a dual Lyapunov equation.

$$[F_a]^T [\Lambda] + [\Lambda] [F_a] = -[Q_a] \quad (13)$$

In this study, a conjugate gradient method is used in order to obtain the control law, in other words, $[A]$, $[B]$, $[C]$ of Eq.(4). Then a numerical optimal solution, that is, a suboptimal solution is obtained with respect to the optimal control theory. From a Lyapunov equation the gradients of $[J]$ by $[A]$, $[B]$ and $[C]$ are obtained.

It should be noted that stability cannot be assured because the control law dimension is less than the system dimension, and the stability of the solution may depend on the key state selection. When a system has dipoles, a solution is not obtained. But in an actual situation, dipoles rarely exists in a system. If they do, they are slightly changed artificially in the control law design process, as described in Eq.(2).

4. Wind Tunnel Test Verification

The wind tunnel tests of an aircraft wing with gust load alleviation were conducted in the MHI's 2 m low-speed wind tunnel to verify aircraft vibration control technology proposed here. The general description of a wind tunnel wing model equipped with a control surface is presented in Table 1, and its main structure is shown in Fig. 4. A small electric motor generation system is stored in the wing model. The wing model is supported by coil springs to move it out of the horizontal plane direction, and the upper sur-

face of the wind tunnel is regarded as the aerodynamic center, as described in Fig. 5. As the gust response of the aircraft is short-period motion combined with pitching and plunging, where pitching motion is suppressed by a horizontal tail, plunging motion suppressed by a flap is simulated in this test. Hence the model can undergo plunging motion, and dummy weights are added to simulate short-period motion.

Gust load is generated by a set of wings installed on the floor of the wind tunnel upper stream position; and they are driven by hydraulic pressure. The gust load characteristics in this wind tunnel are plotted in Fig. 6. Almost uniform distribution of turbulence is obtained in the upper half of it where the aeroelastic wing model is installed. The test and analysis results of wing model vibration modes are shown in Fig. 7. At first, it was considered that the main contribution to

Table 1 Wind tunnel model characteristics

wing	aspect ratio	3.0
	elastic axis	31.8 °
	wing thickness ratio	6 %
structure	spar	aluminum
	wing element	8 elements
	support	movable
flap	type	plain flap
	position	0.52 ~ 0.83
	hinge line	80 %

Fig. 4 Main structure of wind tunnel model

Fig. 5 Support system of wind tunnel model

model rigidity was by the spar, but the total model rigidity exceeds the spar rigidity considerably because of the contribution of the generating system and flap rigidity.

The flap angle changes in accordance with angle

Fig. 6 Gust distribution in wind tunnel

Fig. 7 Vibration modes of wind tunnel model

input changes are measured to reveal the basic characteristics of the generating system. For a high angle, marked gain decrease and phase lag are observed. But within the level of flap angle 5° and frequency 15 Hz, which were adopted in this test, the dynamics of the generating system can be modeled with the second degree of freedom. Before the wind tunnel tests to verify the control effect, the wind model frequency response according to flap angle change is measured, as shown in Fig. 8. This figure shows that the analytical results approximate simulate the test results.

From these test data, the aeroelastic wing mathematical model is constructed and the gust load alleviation control law is determined. In this case, the control law is 4-dimensional to suppress rigid body and first elastic modes of the wing. The accelerometer attached to the spar is used for control, as shown in Fig. 4. The system block diagram of the gust load alleviation test is shown in Fig. 9. The control is carried out based on digital control method and sampling frequency is selected to be 200 Hz. The test and analytical results for when the control system is on or off are shown in Fig. 10. In the control system of the on case, 1 Hz and 3 Hz load peaks of the wing excited

Fig. 8 Control surface response characteristics

Fig. 9 System block diagram of wind tunnel

by gust load load are measured. These peaks indicate rigid body and first elastic modes. These two peaks indicate suppressed by active control and therefore gust load decreases. The analytical results approximate the test results; verification of the analysis and control law synthesis method is successfully obtained.

5. Application to a Commercial Airplane

The effects of aircraft vibration control technology such as gust load alleviation and flutter suppression on a commercial airplane are considered. The application model is a 150-passenger twin-engine middle-class commercial airplane. The general outline of the model, its main characteristics and vibration modes of wings are shown, respectively, in Fig. 11, Table 2 and Fig. 12. The control surfaces are outboard ailerons and ledgers (leading edge flaps). Also, elevators are operated in gust load alleviation mission. The outboard ailerons and ledgers of the wing are used as control surfaces in order to control both wing bending and torsional modes. Two accelerometers on the wing and an accelerometer and pitch rate gyro in the fuselage are used as control sensors. The sensors in the fuselage are used to control the rigid body mode in the gust load alleviation mission.

First, the gust load control law effect is consid-

ered. The typical control effect is shown in Fig. 13. Input load is a (1-cosine)-type discrete gust that is used for load specification. The main gust load is caused by the wing first bending mode, and wing bending moment is reduced 46.9% for gust load alleviation control. Next, the flutter suppression control law effect is considered in the same manner as gust load alleviation. The flutter root loci when the control system is on or off are shown in Fig. 14. In the control system of the off case, the flutter bending mode is the wing torsional mode interacting with the third bending mode, where flutter speed $V_F = 319.9$ m/sec. In the control system of the on case, $V_F = 418.1$ m/sec and 30.7% flutter speed augmentation is obtained. In this case, the wing torsional mode (flutter mode) is mainly controlled.

In this analysis, the gust load alleviation law and

Fig. 11 Aircraft model

aircraft parameter	dimension
hight	24500 ft
velocity	.81 Mach
weight	5596.9 kgs ² /m
inertia moment(IY)	3.388x10 ⁵ kgms ²
wing area	120 m ²
MAC	3.707 m

Fig. 10 Comparison of gust load alleviation effect

Fig. 12 Wing vibration mode

Fig. 13 Gust load alleviation effect

Fig. 14 Flutter suppression effect

the flutter suppression control law are determined independently because the frequencies of the two active control laws of the mainly controlled modes are separated. When both control laws operate at the same time, so interference between the two control laws exists. In order to regulate interference, control gain corresponding to the second bending mode is multiplied by 0.9. In the vibration control problem, the actuator frequency characteristic is the main problem in the hardware system. Mathematical simulations are conducted to determine actuator frequency characteristics. Frequency greater than 10 Hz is needed in gust load alleviation and greater than 20 Hz in flutter suppression. But this result is valid for first

wing bending mode control in gust load alleviation and wing torsional mode control in flutter suppression. Higher frequency characteristics are needed control higher modes.

6. Conclusions

In this paper, aircraft active vibration control such as gust load alleviation and flutter suppression were considered. Results obtained are described below.

(1) The analysis and control law synthesis methods of aeroservoelasticity were established.

(2) In gust load alleviation, wind tunnel tests were conducted in a 2 m low-speed wind tunnel to verify the analysis and control law synthesis methods.

(3) For a 150-passenger twin-engine middle-class commercial airplane, the effects of gust load alleviation and flutter suppression were considered. The load alleviation effect of 46.97% in the gust load alleviation case and 30.7% flutter speed argumentation effect in flutter suppression were obtained.

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