

# Active Flutter Suppression for a Wing Model

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**The paper presents an experimental verification of a method to design active control flutter suppression systems that allows eigenstructure assignment directly within a  $p$ - $k$  flutter approximation. By using a wing model, it is shown how different and simple direct feedback control laws can be effective in producing a substantial improvement of the flutter speed and of the overall damping below the critical speed. Difficulties encountered in correlating designs to test by using a "Maximum Likelihood" identification method are also addressed.**

## Introduction

THE advantages offered by active flutter suppression in terms of improved overall structural efficiency and weight saving are well known, and the feasibility of different design methods and implementations have been demonstrated by flight and model tests, both for commercial and combat aircraft.<sup>1-5</sup> These design methods can be grouped into two main categories.

The first, and more largely used, adopts general purpose design techniques from control theory. Of these, classical control methods are readily adaptable to the frequency domain modeling typical of the well known and established flutter analysis techniques generally used by aeroelastic analysts. Despite this, their application is somewhat awkward for multi-input-output flutter suppression systems due to the many trial designs required before a good result is obtained. Nevertheless, when used by experienced designers in a computer aided approach, these many trials are often helpful in building a strong physical understanding of the system behavior and lead to robust designs.

Modern control design methods in state space seem more compatible with general multi-input-output systems, and many techniques, e.g., optimal Linear Quadratic Gaussian (LQG), pole and partial or complete eigenstructure assignment, can be used.<sup>6-8</sup> None of these seems to demonstrate a particular advantage over the others, and generally the trials required by the classical design methods are here substituted by sensitivity analyses on cost weights and pole-eigenstructure localization, as these determine the robustness of the control laws. When applied to active-flutter suppression, all of them suffer from the need of greatly augmenting the states of the model in order to recast the aerodynamic frequency response matrices into the state space. These added states are fictitious, unmeasurable, and very sensitive to the modeling approximation implied by the aerodynamic formulation adopted. It must be also remarked that even if a good aerodynamic formulation could be used, the aerodynamic forces vary greatly within the flight envelope. Then, since state reconstruction is needed,

even the simplest flutter model is strongly affected by this state augmentation, and it is very difficult to ensure robustness in the control laws so designed.

Different forms of robustness recovery, e.g., singular value analyses of the return difference transfer function and process noise addition,<sup>6-8</sup> have been used either interactively or in an automated way in order to try to alleviate these problems. Nonetheless, control laws design by modern control methods are relatively complex, and, in general, would be difficult to test and tune in flight if some form of gain scheduling is required to cover the different flight regimes.

Attempts have been made to adopt suboptimal control methods<sup>9</sup> that allow arbitrary constraints to be imposed in order to simplify the controller structure. Moreover, since in the suboptimal approach, the design is carried out by numerical optimization, both multimodel stabilization and/or the previously cited techniques can be included as constraints in the optimization process in order to ensure an adequate robustness recovery. A method is available to help in the design of constrained stabilizing controllers for multimodel systems that can profitably be used for suboptimal designs.<sup>10</sup> Because of the numerical optimization used in the design process, suboptimal methods are more costly in terms of computer usage, but, if simplified and effective control laws can be obtained, the added cost is worthwhile.

Not all of the design methods previously addressed have been verified by tests, and, even if comparisons of some design techniques appeared in the literature,<sup>1</sup> no extensive and comprehensive cross checks are yet available to definitely assess the pros and cons of the different methods.

A radical departure from the previous techniques is shown by the second group of design methods, which is more strictly bound to the peculiarities of the flutter formulation generally adopted in aeroelastic analyses. These tend to regard active control laws as alternatives to the addition of masses, damping, and stiffness to the structure without actually using structural components. Notable examples of this approach are the aerodynamic energy<sup>11</sup> and the Identical Location of Accelerometer and Force (ILAF)<sup>12</sup> concepts.

The former is somewhat constrained in that it is readily usable only for simple models of the servomechanical part of the aeroelastic system and therefore does not permit complex models for sensors, actuators, and compensators. It can be extended to treat simultaneously many flight conditions, and thus it can ensure some robustness in the produced control laws.

The ILAF concept cleverly tries to build grounded dampers without any connection to ground and thus ensures energy dissipation regardless of the flight condition and configuration provided that a good tandem compensation can be established to account for variation in the aerodynamic response of the actuated control surface.

Presented as Paper 88-1.2.2 at the 16th Congress of the International Council of the Aeronautical Sciences, Jerusalem, Israel, Aug. 28-Sept. 2, 1988; received Nov. 1, 1988; revision received Aug. 17, 1989. Copyright © 1989 American Institute of Aeronautics and Astronautics, Inc. All rights reserved.

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The method of Refs. 13 and 14 tries to bridge the different points of view by developing a method of eigenstructure assignment that can take into account passive and active design parameters and use a  $p$ - $k$ <sup>15,16</sup> approximation for the flutter analysis. In this way the controller can be assigned the simplest structure capable of satisfying the design objectives and recover robustness by simultaneously designing for different flight conditions.

This paper is aimed at showing how the latter approach can be used to design an active flutter suppression system that is, as proven by a series of flutter tests, capable of extending the operational envelope of a wing model by using a single, trailing-edge aileron and very simple control laws.

### Problem Formulation and Control Modeling

The design process presented in this paper is aimed at increasing the flutter speed and the overall damping below critical speed of the built-in wing model with a heavy wing-tip mass that is shown in Fig. 1. The model is made up of a composite spar to which aerodynamically shaped wing profiles are connected by means of metallic ribs screwed to the spar itself. The design goal is an increase of the flutter speed to be achieved through an active flutter suppression system, which should also be capable of obtaining an overall qualitative improvement of the damping below the critical speed. The goal should be achieved with the simplest possible control law by using an available trailing-edge outboard aileron driven by an electro-mechanical actuator. The actuator is made up by a dc torque motor with its shaft constrained to the support by mechanical springs and dampers in order to simulate a realistic transfer function of the type

$$\frac{\theta_a}{i_m} = \frac{\omega_a^2}{s^2 + 2\xi_a\omega_a s + \omega_a^2} \quad (1)$$

in which  $\theta_a$  is the absolute physical aileron rotation and  $i_m$  is the current driving the torque motor.

By using NASTRAN a discrete analytical aeroelastic model is obtained adopting the first five vibration modes, which are the first bending  $q_{b1}$ , the first torsion  $q_{t1}$ , other two deformation modes, i.e.,  $q_{b2}$  and  $q_{t2}$ , and the aileron vibration mode  $q_a$ . When combined with the aileron transfer function, i.e., Eq. (1) with  $\theta_a$  dependent from  $i_m$ , the open loop aeroelastic model has the following form<sup>17</sup>

$$\left\{ s^2[M] + s[C] + [K] - \bar{q} \left[ A \left( \frac{sc}{2V} \right) \right] \right\} \{q\} = [B]i_m \quad (2)$$

where  $\{q\}^T = [q_{b1} \ q_{t1} \ q_{b2} \ q_{t2} \ q_a]$ ,  $s$  is the Laplace transform variable,  $[M]$ ,  $[C]$  and  $[K]$  are the mass, damping, and stiffness matrices,  $[B]$  the input influence matrix,  $[A]$  the aerodynamic transfer function matrix,  $\bar{q}$  the dynamic pressure for asymptotic speed  $V$ , and  $c$  an aerodynamic reference length.

Note that the matrices  $[M]$ ,  $[C]$ , and  $[K]$  are not the diagonal modal matrices because when Eq. (1) is taken into account, they become completely full since  $\theta_a$  is coupled with all of the vibration modes as the aileron is unbalanced. For the same reason the matrix  $[B]$  is also completely full. Nonetheless these couplings are not too strong, and the absolute physical rotation  $\theta_a$  can be confused with  $q_a$  and vice versa.

A simple approach to stabilize Eq. (2) is to assume two independent measurements of displacements, velocities, and accelerations at two points at the wing tip are available. Calling them  $\{m\}$ ,  $\{\dot{m}\}$ ,  $\{\ddot{m}\}$ , a stabilization of Eq. (2) could be obtained by direct feedback of appropriate combinations of the available measurements, i.e., by taking

$$i_m = (s^2[G_a] + s[G_v] + [G_d])\{m\} \quad (3)$$

where  $[G_a]$ ,  $[G_v]$ , and  $[G_d]$  are one by two feedback gain

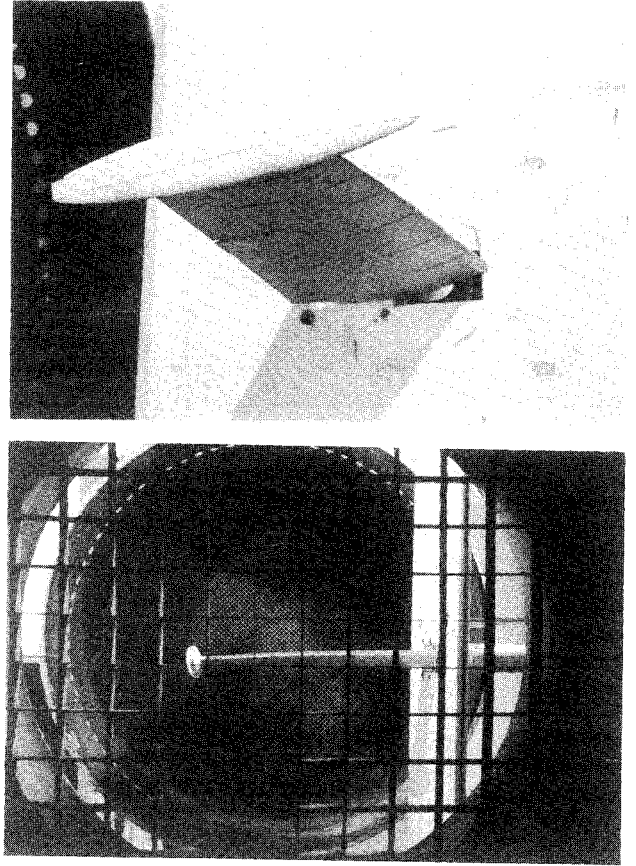


Fig. 1 Wing model.

matrices to be determined in order to satisfy the design specifications. If  $[N_m]$  is the matrix of the modal displacements at the sensors' locations, we have

$$\{m\} = [N_m]\{q\} \quad (4a)$$

$$\{\dot{m}\} = [N_m]\{\dot{q}\} \quad (4b)$$

$$\{\ddot{m}\} = [N_m]\{\ddot{q}\} \quad (4c)$$

Closing the loop in such a way is equivalent to changing  $[M]$ ,  $[C]$ , and  $[K]$  with the addition of the following corresponding matrices:

$$[M_c] = [B][G_a][N_m] \quad (5a)$$

$$[C_c] = [B][G_v][N_m] \quad (5b)$$

$$[K_c] = [B][G_d][N_m] \quad (5c)$$

Since it is not possible to directly measure  $\{m\}$  and  $\{\dot{m}\}$ , they are obtained by a double integration of  $\{\ddot{m}\}$  as measured by two piezoelectric accelerometers located at the leading and trailing edges of the wing tip. Since these integrations are carried out analogically, it is important to avoid long-term drift and saturation of the integrators. Thus, assuming an acceptably high frequency for the lowest frequency vibration mode ( $q_{b1}$ ), each integration is performed as a band pass filtering, i.e., by using as integrator the following transfer function:

$$I(s) = \frac{s}{s^2 + 2\xi_f\omega_f s + \omega_f^2} \quad (6)$$

The previous approximation behaves like  $1/s$  if  $\omega_f$  and  $\xi_f$  are appropriately chosen and has the added advantage of making the flutter suppression system uncoupled from the aircraft

free-body motions provided that enough separation exists between  $q_{b1}$  and the short period motions of the aircraft. The two functions can be achieved within the same integrated circuit by using components commercially available at low cost but of good quality.<sup>18</sup> The inclusion of the integrators and high pass filters requires the augmentation of Eq. (2) from 5 to 9 deg of freedom:

$$\begin{pmatrix} s^2 \left[ \begin{array}{c|c} [M] & 0 \\ \hline 0 & [I_f] \end{array} \right] + s \left[ \begin{array}{c|c} [C] & 0 \\ \hline 0 & 2\xi_f \omega_f [I_f] \end{array} \right] + \left[ \begin{array}{c|c} 0 & 0 \\ \hline 0 & \omega_f^2 [I_f] \end{array} \right] \\ - \bar{q} \left[ \begin{array}{c|c} [A] & 0 \\ \hline 0 & 0 \end{array} \right] \end{pmatrix} \begin{Bmatrix} \{q\} \\ \{q_f\} \end{Bmatrix} = \begin{Bmatrix} [B] & 0 \\ \hline 0 & [I_f] \end{Bmatrix} \begin{Bmatrix} i_m \\ \{i_f\} \end{Bmatrix} \quad (7)$$

in which  $[I_f]$  is an identity matrix of fourth order, and  $\omega_f$  and  $\xi_f$  are assumed equal for all of the filters.

The loop is now closed by making

$$\begin{Bmatrix} i_m \\ \{i_f\} \end{Bmatrix} = \left( s^2 \left[ \begin{array}{c|c} 0 & [G_{hf}] \\ \hline [N_{af}] & 0 \end{array} \right] + s \left[ \begin{array}{c|c} 0 & [G_{bf}] \\ \hline 0 & [N_{bf}] \end{array} \right] \right) \begin{Bmatrix} \{q\} \\ \{q_f\} \end{Bmatrix} \quad (8)$$

in which  $[N_{af}]$  is  $[N_m]$  with zero rows included in correspondence to acceleration integrators. The  $[N_{bf}]$  routes the output of the acceleration integrators, i.e., the velocities to the inputs of the velocity integrators. The  $[G_{hf}]$  sets the two gains for the high-passed accelerations and  $[G_{bf}]$  the four velocity and displacement gains available from the corresponding bandpass output of the filters.

The structure of the closed-loop system equations remains the same both for the augmented and unaugmented systems, and the designer has to choose appropriate values of  $\omega_f$ ,  $\xi_f$  and for the six nonzero terms of the gain matrices in order to make it possible to satisfy the design requirements.

Note that even more simplified versions are also possible by discarding some feedback loop and the corresponding gain matrices and filter connections.

This approach leads to a constrained structure of the controller that cannot be handled by many of the design methods briefly presented in the introduction. It should also be noted that, provided that a design could be carried out that is robust enough to achieve a satisfactory flutter suppression for a large part of the flight envelope with a single control law, the simplicity of the controller structure can make possible a safe and reliable tuning of the gain scheduling policies eventually required for an operative flutter suppression system.

### Control Law Design and Experimental Verification

The general problem of control-law design can then be reduced to the determination of the elements of the matrices of Eqs. (7) and (8). This process involves the determination of sensor and actuator locations, i.e., the matrices  $[N_m]$  and  $[B]$ , the controller structure, and the evaluation of the related gains.

For their computation the imposition of a set of eigenvalues  $s_i^*$  and all or part of the corresponding eigenvector  $\{q_i^*\}$  leads to the solution of a nonlinear system of equations corresponding to Eq. (7), which is synthetically written as

$$[F_i(s_i^*, [N], [B], [G])] \{q_i^*\} = 0 \quad (9a)$$

with the added relationship

$$\frac{1}{2} \{q_i^*\}^T \{q_i^*\} = 1 \quad (9b)$$

in order to ensure a solution of  $[\{q\}^T \quad \{q_f\}^T]^T$  different from zero.

This solution can be obtained as indicated in Ref. 19 by assigning an appropriate number of eigenstructures. Very of-

ten in practical designs, a control strategy is adopted that makes use of a number of feedback loops greater than the minimum required by Eq. (7) in order to increase the reliability and the robustness of the controller exploiting the unknowns in excess to build into the controller some degree of adaptivity to different flight conditions. In these cases it is necessary to resort to the minimization of an adequate objective function with Eqs. (9) imposed as equality constraints. A candidate merit function could be of the type

$$F = \frac{1}{2} \{a\}^T [W] \{a\} \quad (10)$$

where  $\{a\}$  is the vector of the unknown control parameters, and  $[W]$  is a diagonal matrix of positive weights used for scaling and for limiting the actuator activity through the reduction of critical unknowns.

It is worthwhile to stress that the vector of design variables  $\{a\}$  may in general include the characteristic parameters and locations of sensors and actuators, provided that realistic technological constraints have been established. This feature has not been exploited in the present work in which both the transfer function and the location of the actuator and sensors and the integrator dynamics are assigned. So only the gain matrices  $[G_f]$  are taken as unknown in the present application and determined by imposing only some eigenvalues.

Then using Lagrange multipliers, the function to be minimized assumes the following form:

$$\begin{aligned} \mathcal{L} = & \frac{1}{2} \{a\}^T [W] \{a\} + \{\lambda_i\}^T [F_i] \{q_i^*\} \\ & + \{\underline{\lambda}_i\}^T \left( \frac{1}{2} \{q_i^*\}^T \{q_i^*\} - 1 \right) \end{aligned} \quad (11)$$

where  $\{\lambda_i\}$  and  $\{\underline{\lambda}_i\}$  are the Lagrange multipliers associated with the constraints. The minimization of  $\mathcal{L}$  leads to a set of nonlinear equations of the type

$$\frac{\partial \mathcal{L}}{\partial \{q_i\}} = [F_i]^T \{\lambda_i\} + \underline{\lambda}_i \{q_i\} = 0 \quad (12a)$$

$$\frac{\partial \mathcal{L}}{\partial \{\lambda_i\}} = [F_i] \{q_i\} = 0 \quad (12b)$$

$$\frac{\partial \mathcal{L}}{\partial \{\underline{\lambda}_i\}} = \left\{ \frac{1}{2} \{q_i\}^T \{q_i\} - 1 \right\} = 0 \quad (12c)$$

$$\frac{\partial \mathcal{L}}{\partial \{a\}} = [W] \{a\} + \left\{ \{\lambda_i\}^T \left[ \frac{\partial F_i}{\partial a} \right] \{q_i\} \right\} = 0 \quad (12d)$$

that can be solved in continuation form on an assigned path that leads from the old eigenvalues to the desired new ones. Possibly a predictor-corrector method of the type presented in Ref. 13 can be used. A detailed description of the numerical procedure is reported in Ref. 16, and all of the control laws presented in this paper have been obtained by this method.

The experimental setup used in wind-tunnel tests is sketched in Figs. 2 and 3. A personal computer controlled the test by exciting the model with a 20-s frequency sweep between 3 and 10 Hz acquiring the acceleration signals at a sampling frequency of 50 Hz and implementing the control laws at 100 Hz.

The experimental identification of the aeroelastic system eigenvalues has been carried out by using a maximum likelihood (ML) technique<sup>20,21</sup> and using a continuous excitation through the aileron. A problem emerged during the tests on closed-loop configurations in which the high damping introduced by the control law made it impossible to identify the aeroelastic mode with the highest damping, typically the first torsional mode  $q_{t1}$  even by means of narrow-band sweeps of long duration centered around the frequency of interest.

In the following discussion, the main results of many design trials with different control strategies are presented together with the experimental implementations of the control laws and their verification in the wind tunnel.

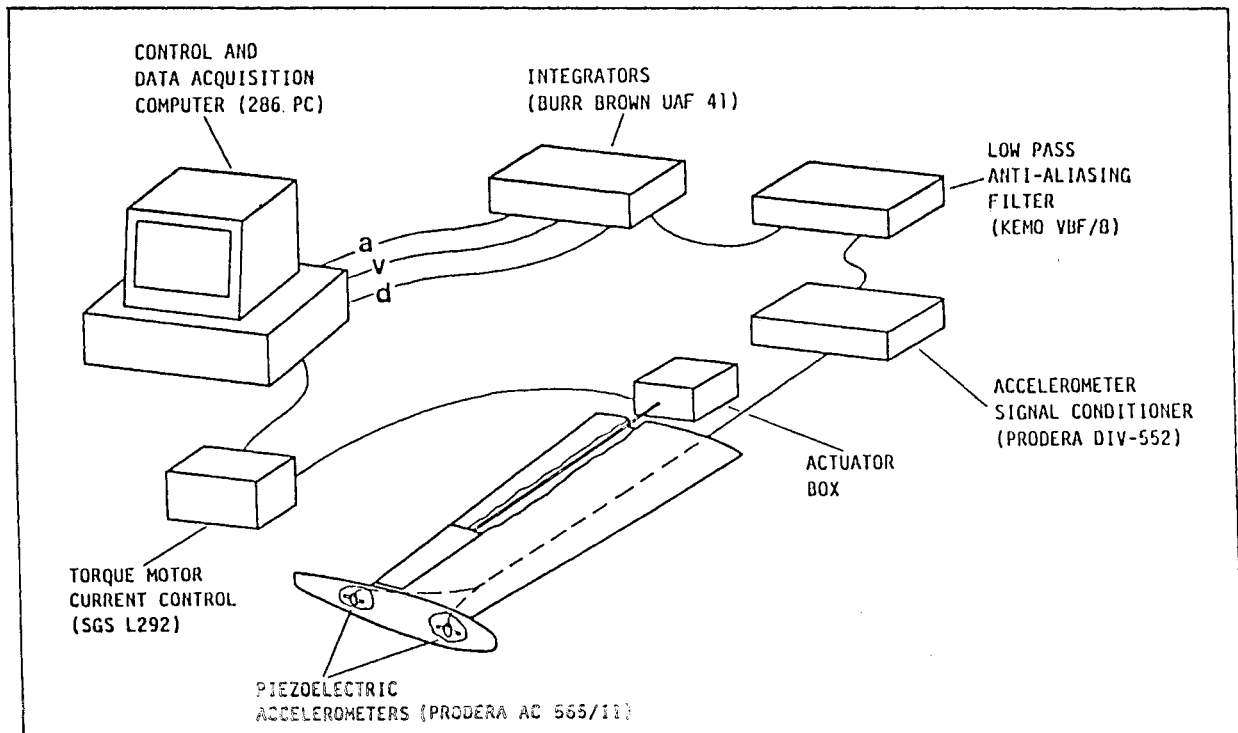


Fig. 2 Experimental setup.

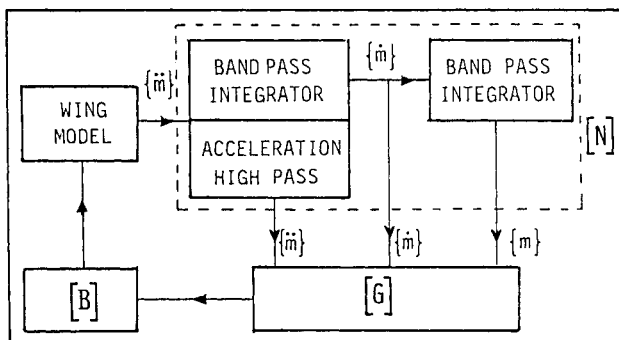


Fig. 3 Active flutter suppression system block diagram.

The response of the wing model, both uncontrolled and controlled, is presented by  $V$ - $g$  plots, both predicted and experimentally identified and by the amplitude of the transfer function of the trailing-edge accelerometer response to an aileron input at a wind speed of 45.5 m/s, i.e., just below the open-loop flutter speed. All the transfer functions are normalized to the maximum amplitude of the open-loop system transfer function  $|H_{OL}|_{\max}$  to give an immediate qualitative idea of the response reduction with respect to open loop.

Different control strategies have been evaluated. The same sensors' location at the wing tip has been assumed. The two transducers were actually piezoelectric accelerometers, and when velocity and/or displacement measurements were required, these quantities have been obtained by filtering and integrating accelerometer output as explained in the previous paragraph. All the integrators had a transfer function of the type of Eq. (6) with  $\xi_f = .5$  and  $\omega_f = 2\pi$ : these parameters have been chosen as a compromise between a fast transient response of the filter and its accuracy as an integrator at frequencies close to  $\omega_f$ . As a general rule, when the characteristic parameters of the integrators are well apart from the natural modes of the aeroelastic system, the inclusion of the integrators into the model does not strongly affect the model. This is true in the

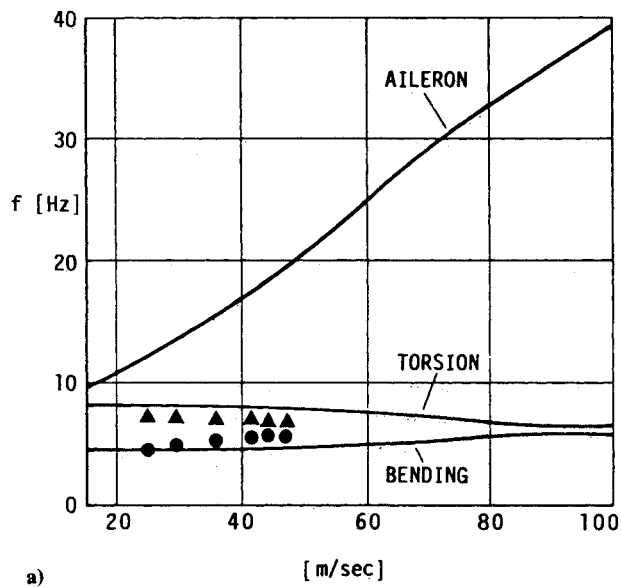
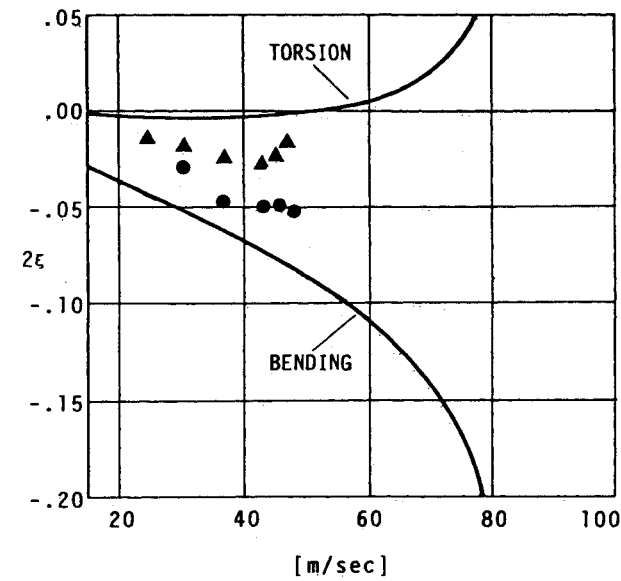
present case since  $\omega_f$  is about one-fourth of the bending frequency of the wing.

The dc motor driving the aileron was current controlled by a pulse width modulated (PWM) integrated power amplifier: the switching frequency and the parameters of the control network have been designed in order to have a transfer function of the type of Eq. (1).

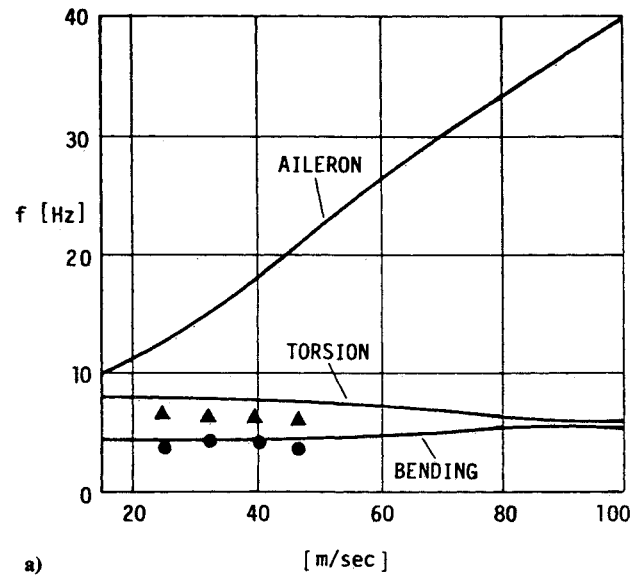
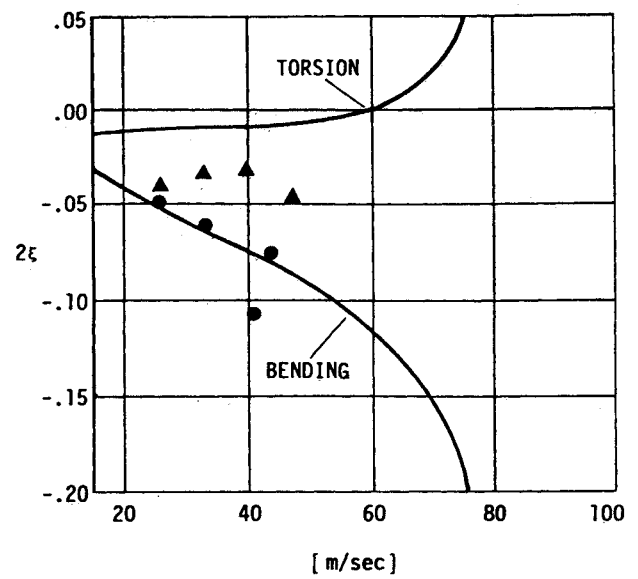
The first experimental tests were directed toward the identification of the open loop behavior of the wing model equipped with the aileron electromechanical actuator. The flutter speed was very near to the predicted one, and the flutter mechanism was of the classical bending-torsion type. Even if, from the analysis of Figs. 4, some differences between analytical and measured frequencies were revealed, it was decided not to modify the analytical model considering that this situation reproduced well the real design situation.

The simplest control strategy in the example under examination was to measure only the accelerations at the wing tip and feed them back to the aileron: the first trend designs have shown a little increase in flutter speed and a sufficient damping over the speed range of interest. Fig. 5a is representative of the typical results obtained. They are quite insensitive to the number and location of assigned poles so that the final gains have been obtained fixing only one eigenvalue. The wind tunnel tests have confirmed the numerical results as shown in Fig. 5a. It is worthwhile to note that the first implementation of the computed control law gave rise, in the high-speed range, to a sort of limit cycle that induced low amplitude-high frequency motions of the aileron. Nevertheless the system was well damped, leading to the retention of this law with the addition of an intervention threshold below which the control was inactive. This first design has proven its efficiency in stabilizing the wing model with a very simple control law, which shows the tendency to highly dampen the torsional motion (see Fig. 5b) without any appreciable change on the flexural response of the first bending mode.

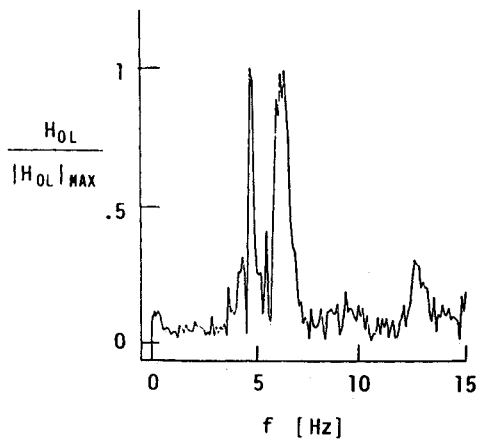
The next series of tests made use of the accelerations and velocities at the wing tip. It was very easy to design control laws that were able to achieve the desired flutter improvement. The only problems were related to high gains. In fact some control laws gave rise to a completely flutter-free behavior but



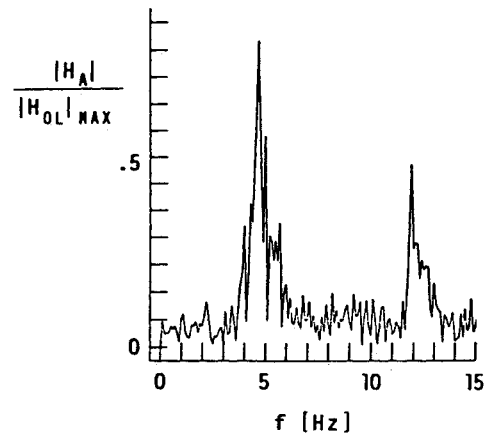
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a)



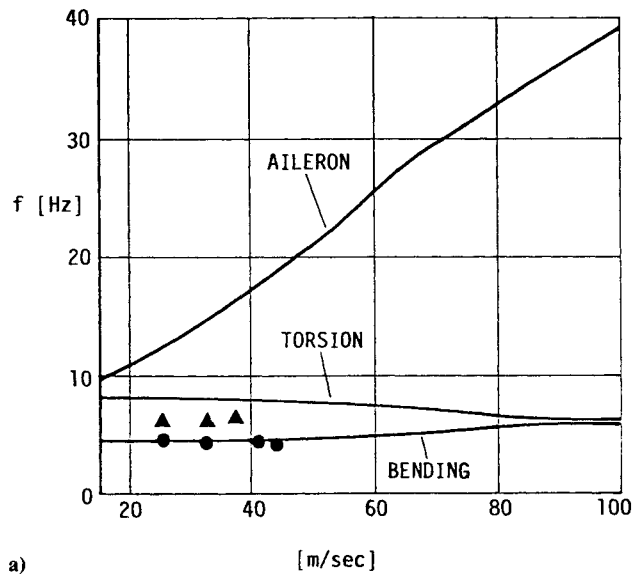
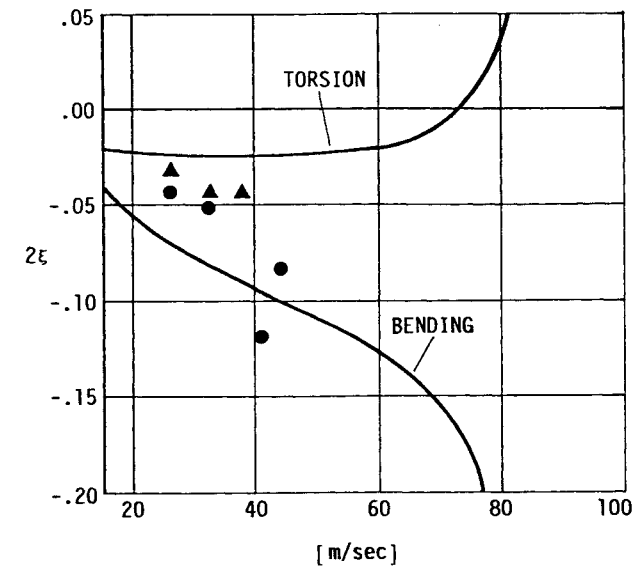
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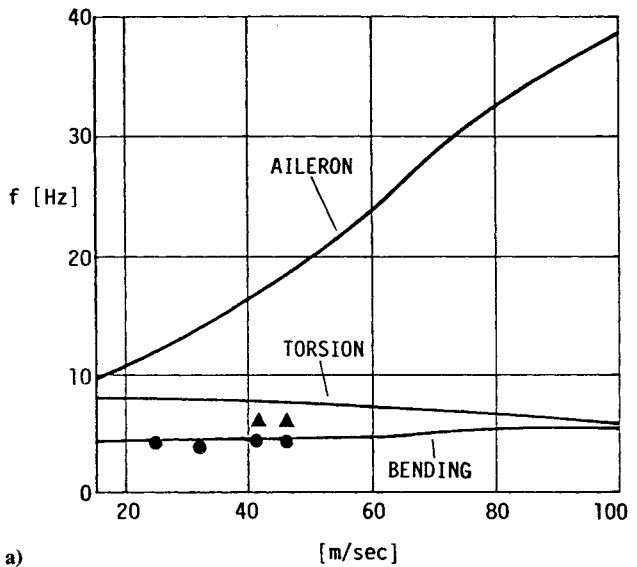
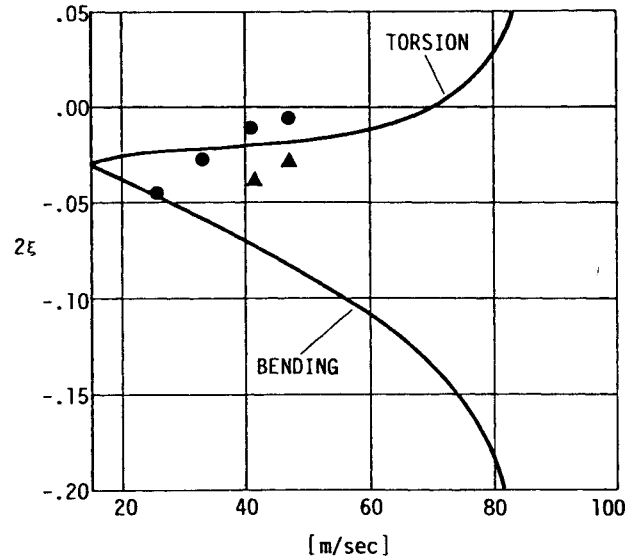
b)

Fig. 4 Open loop system: a) V-g plots ( $\bullet$  flexural mode,  $\blacktriangle$  torsional mode); b) amplitude of the transfer function of the trailing-edge accelerometer to an aileron input ( $V = 45.5$  m/s).

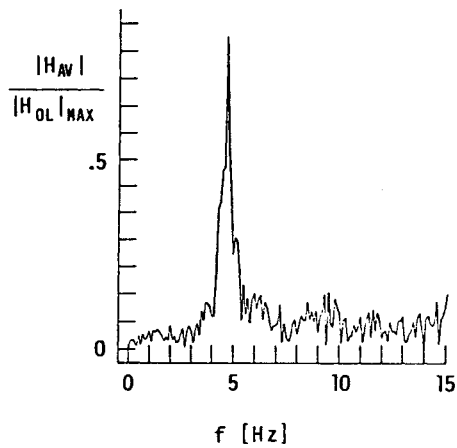
Fig. 5 Closed loop system (acceleration feedback): a) V-g plots ( $\bullet$  flexural mode,  $\blacktriangle$  torsional mode, assigned poles  $\circ$ ); b) amplitude of the transfer function of the trailing-edge accelerometer to an aileron input ( $V = 45.5$  m/s).



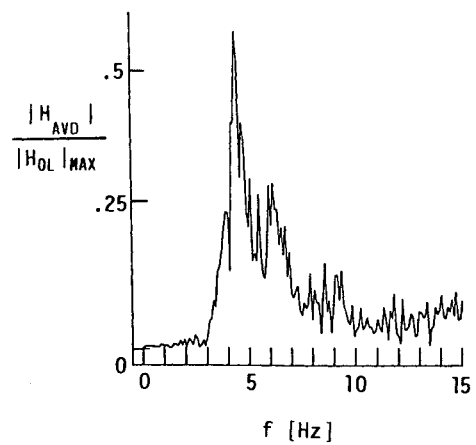
a)



a)



b)



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Fig. 6 Closed-loop system (acceleration-velocity feedback): a)  $V$ - $g$  plots (● flexural mode, ▲ torsional mode, assigned poles ○); b) amplitude of the transfer function of the trailing edge accelerometer to an aileron input ( $V = 45.5$  m/s).

Fig. 7 Closed loop system (acceleration-velocity and displacement feedback): a)  $V$ - $g$  plots (● flexural mode, ▲ torsional mode, assigned poles ○); b) amplitude of the transfer function of the trailing edge accelerometer to an aileron input ( $V = 45.5$  m/s).

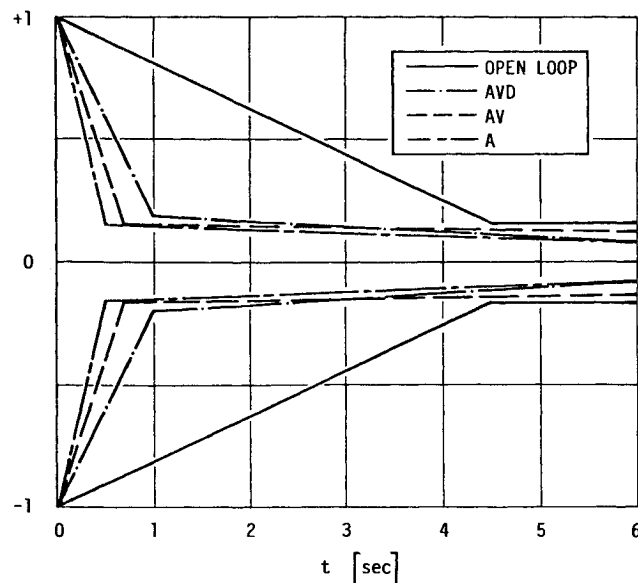


Fig. 8 Comparison of damping envelopes at 36 m/s; A-acceleration feedback; AV-acceleration and velocity feedback; AVD-acceleration-velocity and displacement feedback.

at the expense of excessively high control torques. Figure 6a reports the final design chosen together with the poles imposed for their attainment. The computed flutter speed is about 75 m/s, and a good level of damping is acquired over the whole speed range. The experimental flutter speed has not been obtained due to limitations in maximum speed attainable in the wind tunnel.

It is important to note that, as shown in Fig. 6b, at the relatively high speed of 45.5 m/s, the torsional mode is so damped that it barely appears in the response, and the ML identification was unable to identify it. This explains the lack of identified damping and frequencies at high speeds. This has been a source of trouble in some early designs that showed an unexpected flutter close to the speed at which an unidentifiable large damping was introduced by the active control.

The design using acceleration, velocity, and displacement feedback has shown characteristics very similar to the preceding one, and thus all the comments on control torque limitation are still valid. The analytically predicted flutter behavior, (see Fig. 7a) is quite similar to that obtained using velocity and acceleration measurements. In the experimental realization of this law, velocity and displacement were directly acquired from the integrators, and the accelerations were computed by finite backward differences of the velocity signals because of the lack of acquisition channels. The lack of correlation between analysis and the wind tunnel identification remains unexplained as qualitatively the system was well damped also at high speed as shown by Fig. 7b. A possible explanation could be related to the increased noise and time delay introduced by the computed accelerations.

Finally Fig. 8 shows, in synthetic form, the time response envelope of acceleration peaks of the different control laws after a 4-s frequency sweep between 5 and 7 Hz. As can be seen, all the control laws add substantial damping to the structure.

### Concluding Remarks

This paper has proved that a simple and effective suppression of flutter can be obtained by designing the control laws using an eigenvalue assignment method capable of working directly within a  $p$ - $k$  approximation of the flutter equations. The control laws make use of decentralized direct feedback and require a minimum of hardware and thus make it easy to use redundant implementations to guarantee system reliability.

Some of these control laws have proven their feasibility and effectiveness in flutter tests on a wing model. The method used to design the control laws requires some tedious trials to determine the best location of the desired eigenvalues that are capable of stabilizing unstable modes without producing instability in the stable ones. This work can nonetheless be worth the cost as it adds to the physical understanding of the system by the designer and can produce simple yet very effective control laws. Moreover the capability to design simultaneously for different flight conditions can greatly improve control robustness to structured parameter uncertainties. The use of few feedback gains makes it possible to rapidly produce scheduling policies for the widest flight envelope coverage allowing a safe verification and tuning of the gains in flight tests. In all cases the designed control laws required control surface activities well below saturation. A further point worth noting is that the control law could be implemented with a reduced amount of low-cost hardware.

A major critical point that has emerged from the experimental activity is related to the identification of the stability characteristics of the closed-loop system at subcritical speeds. It appears in fact from the experience gathered during the testing of the control laws that the well known difficulties in the identification of the damping of aeroelastic systems are augmented by the presence of an effective active control system if the approaching flutter is of a violent nature. In this case the flutter is close to the point of maximum damping. This is a condition that is difficult to ascertain as the mode driving the flutter can be so damped that it barely appears in the response. This fact can be of major concern and an important issue in the validation and certification of actual active flutter suppression systems.

### Acknowledgments

The authors would like to acknowledge O. Onori and P. Palmieri for their help and assistance during the setup and the execution of this work. This paper has been partially sponsored by the Consiglio Nazionale delle Ricerche (CNR) under contract N.86.00865.59.

### References

- <sup>1</sup>Noll, T. E., Hutsell, L. J., and Cooley, D. E., "Wing/Store Flutter Suppression Investigation," *Journal of Aircraft*, Vol. 18, No. 11, Nov. 1981, pp. 969-975.
- <sup>2</sup>Hoenlinger, H., "Active Flutter Suppression on an Airplane with Wing Mounted External Stores," AGARD CP-228, 1977.
- <sup>3</sup>Peloubet, R. P., Jr., Haller, R. L., and Bolding, R. M., "F-16 Flutter Suppression System Investigation Feasibility Study and Wind Tunnel Tests," *Journal of Aircraft*, Vol. 19, No. 2, Feb. 1982, pp. 169-179.
- <sup>4</sup>O'Connell, R. F., and Messina, A. F., "Development of an Active Flutter Margin Augmentation System for a Commercial Transport," *Journal of Guidance and Control*, Vol. 3, No. 4, July-Aug. 1980, pp. 352-360.
- <sup>5</sup>Redd, L. T., Gillman, J., Colley, D. E., and Stewart, F. D., "A Wind Tunnel Investigation of a B-52 Model Flutter Suppression System," *Journal of Aircraft*, Vol. 11, No. 11, Nov. 1974, pp. 659-663.
- <sup>6</sup>Newsom, J. R., Pototzky, A. S., and Abel, I., "Design of a Flutter Suppression System for an Experimental Drone Aircraft," *Journal of Aircraft*, Vol. 22, No. 5, May 1985, pp. 380-386.
- <sup>7</sup>Liebst, B. S., Garrard, W. L., and Adams, W. M., "Design of an Active Flutter Suppression System," *Journal of Guidance and Control*, Vol. 9, No. 1, Jan.-Feb. 1986, pp. 64-71.
- <sup>8</sup>Newsom, J. R., and Mukhopadhyay, V., "A Multiloop Robust Controller Design Study Using Singular Value Gradients," *Journal of Guidance and Control*, Vol. 8, No. 4, July-Aug. 1985, pp. 514-519.
- <sup>9</sup>Mukhopadhyay, V., Newsom, J. R., and Abel, I., "A Method for Obtaining Reduced-Order Control Laws for High-Order Systems Using Optimization Techniques," NASA TP-1876, Aug. 1981.
- <sup>10</sup>Mantegazza, P., "A Technique to Design Structurally Constrained Stabilizing Control Systems for Actively Controlled Aircrafts," *l'Aerotecnica Missili e Spazio*, Vol. 63, Dec. 1984, pp. 315-317.

<sup>11</sup>Nissim, E., "Flutter Suppression Using Active Control Based on the Concept of Aerodynamic Energy," NASA TN-D-6199, March 1971.

<sup>12</sup>Wikes, H. J., and Kordes, E. E., "Analytical and Flight Tests of a Modal Suppression System on the XB-70 Airplane," AGARD CP-46, 1970.

<sup>13</sup>Lanz, M., and Mantegazza, P., "An Improved Method for the Design of Flutter Suppression Systems by Eigenvalue Assignment," *L'Aerotecnica Missili e Spazio*, Vol. 60, May 1981, pp. 92-97.

<sup>14</sup>Lanz, M., and Mantegazza, P., "Design of Compensated Flutter Suppression Systems," International Council of Aeronautical Sciences, ICAS-82-3.2.1, 1982.

<sup>15</sup>Hassig, H. J., "An Approximate True Damping Solution of the Flutter Equation by Determinant Iteration," *Journal of Aircraft*, Vol. 8, No. 11, Nov. 1971, pp. 885-889.

<sup>16</sup>Cardani, C., and Mantegazza, P., "Continuation and Direct So-

lution of the Flutter Equation," *Computer and Structures*, Vol. 8, No. 2, 1978, pp. 185-192.

<sup>17</sup>MacNeal, R. H., *The NASTRAN Theoretical Manual*, The MacNeal-Schwendler Corporation, Los Angeles, CA, MSR-40, 1972.

<sup>18</sup>*Integrated Circuits Data Book*, Burr-Brown Corporation, June 1986.

<sup>19</sup>Lanz, M., and Mantegazza, P., "Design of Multi Input-Output Flutter Suppression System by Eigenvalue Assignment," *L'Aerotecnica Missili e Spazio*, Vol. 59, Sept.-Dec. 1980, pp. 303-313.

<sup>20</sup>Ghiringhelli, G. L., and Mantegazza, P., "An Approach to the Maximum Likelihood Identification of Flutter Eigenvalues," *L'Aerotecnica Missili e Spazio*, Vol. 65, Sept.-Dec. 1986, pp. 186-195.

<sup>21</sup>Ghiringhelli, G. L., Lanz, M., and Mantegazza, P., "A Comparison of Methods Used for the Identification of Flutter from Experimental Data," *Journal of Sound and Vibration*, Vol. 119, 1987, pp. 39-51.

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