ICAS-88-1.2.2

ACTIVE FLUTTER SUPPRESSION FOR A WING MODEL

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ABSTRACT

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 aircrafts [1-5]. 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 modelling typical of the well known and estabilished 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. Nonetheless, 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 behaviour and lead to robust designs.

Modern control design methods in state space seem more at ease with general multi input-output systems, and many techniques, e.g. optimal LQ, pole and partial or complete eigenstructure assignement, can be used [6-8]. 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 for 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 modelling 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 its very difficult to ensure robustness in the control laws so designed. Different form of robustness recovery, e.g singular value analyses of the return difference transfer function and process noise addition [6-8], have been used either interactively or in an automated way, in order to try to alleviate these problems. Nonetheless control laws designed by modern control methods are generally relatively complex and 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 [9] that allow arbitrary constraints to be posed in order to simplify the controller structure. Moreover, since in the suboptimal approach the design is carried out by numerical optimization, both multi-model 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 multi-model that can profitably be used for suboptimal designs [10]. Because of the numerical optimization used in the design process, suboptimal methods are more costly in term of computer usage, but, if simplified and effective control laws can be obtained, the added cost is worth its spending.

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 [1], no extensive and comprehensive cross checks are yet available to definitely assess the pros and contras 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 and tend to regard active control laws as aliases 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 [11] and the ILAF [12] concepts.

The former is somewhat constrained in that it is readily usable only for simple models of the servostructural part of the aeroelastic system and so it does not permit complex models for sensors, actuators and compensations. 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 brilliantly tries to build grounded dampers without any connection to ground, thus ensuring energy dissipation whatever the flight condition and configuration, provided that a good tandem compensation can be estabilished to account for variation in the aerodynamic response of the actuated control surface.

The method of Refs.[13,14] tries to bridge the different points of view by developing a method of eigenstructure assignement that can take into account passive and active design parameters while using a p-k [15,16] approximation for the flutter analysis. In this way the controller can be assigned the simplest structure capable of satisfaying the design objectives,recovering robustness by simultaneously designing for different flight conditions.

The 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 MODELLING

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 an heavy wing tip mass that is shown in Fig. 1 . The model is made up of a composite spar to which aerodynamically

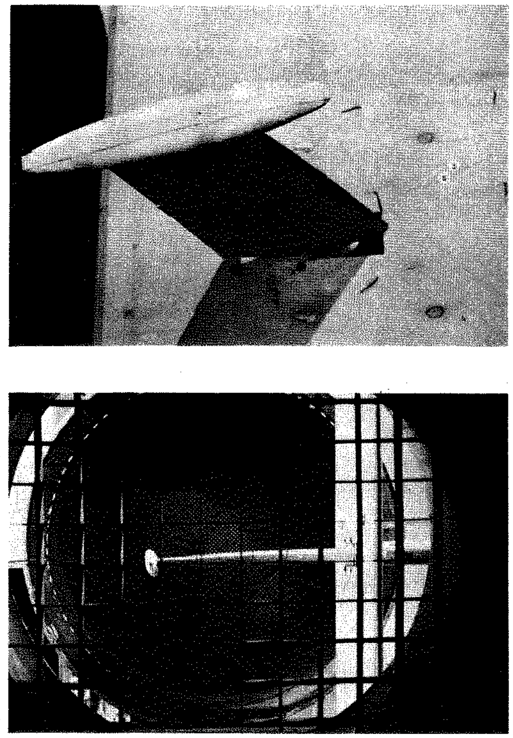

Fig.1 - Wing model

shaped wing profiles are connected by means of metallic ribs screwed to the spar itself. The design goal is an increase of the open loop 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 electromechanical 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:

$$\frac{\theta_\alpha}{i_m} = \frac{\omega_\alpha^2}{s^2 + 2\xi_\alpha \omega_\alpha s + \omega_\alpha^2} \qquad (1)$$

in which θ_α is the 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 rigid aileron rotation q_α. When combined with the aileron transfer function, i.e Eq. 1, the open loop aeroelastic model has the following form:

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

with $\{q\}^T=[q_{b1}\ q_{t1}\ q_{b2}\ q_{t2}\ q_\alpha]$, s is the Laplace transform operator, [M],[C] and [K] are the mass, damping and stiffness matrices, [B] the input influence matrix, [A] the aerodynamic transfer function, 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 θ_α is coupled with all of the vibration modes as the aileron is unbalanced. For the same reason also the matrix [B] is completely full. Nonetheless these coupling are not so strong to change the qualitative appearence of the mode and also θ_α can be confused with q_α and viceversa.

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

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

where $[G_\alpha]$,$[G_v]$ and $[G_d]$ are one by two feedback gain 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 change [M], [C] and [K] with the addition of the following corresponding matrices:

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

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

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

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 ω_f and ξ_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_{f1} and the short period motions of the aircraft. For this reason also $\{\ddot m\}$ is high passed with the same ω_f and ξ_f value of the bandpass integrators filters. The two functions can be achieved within the same integrated circuit by using commercially available low cost but of good quality components [17]. The inclusion of the integrators and high pass filters requires the augmentation of Eq.2 from 5 to 9 degrees of freedom :

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

in which $[I_f]$ is a unitary matrix of fourth order, while ω_f and ξ_f are assumed equal for all of the filters.

The loop is now closed by making:

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

in which $[N_{af}]$ is $[N_m]$ with zero rows included in correspondence to accelerations integrators, $[N_{bf}]$ routes the output of the acceleration integrators, i.e. the velocities, to the inputs of the velocities integrators, $[G_{hf}]$ sets the two gains for the high-passed accelerations and $[G_{bf}]$ the four velocity and displacement gains available from the corresponding band-pass 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 for ω_f, ξ_f and for the six non zero 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 breafly 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 it possible a safe and reliable tuning of the gain scheduling policies, eventually required for an operative flutter suppression system.

CONTROL LAWS 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 sensors and actuators 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 correspondig to Eq. 7 which is sinthetically written as:

$$\left[F_i(s^*_i,[N],[B],[G])\right] \{q^*_i\} = 0 \qquad (9a)$$

with the added relationship:

$$\frac{1}{2} \{q^*_i\}^T \{q^*_i\} = 1 \qquad (9b)$$

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

This solution can be obtained as indicated in Ref. [18] by assigning an appropriate number of eigenstructures.

Very often 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 adaptativity 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:

$$F = \{a\}^T [W] \{a\} \qquad (10)$$

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 out that the vector of design variables $\{a\}$ may in general include also 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 integrators dynamics are assigned. So only the gain matrices $[G_{.f}]$ are taken as unknown in the present application and determined by imposing only some eigenvalue.

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

$$\mathfrak{L} = \frac{1}{2} \{a\}^T [W] \{a\} + \{\lambda_i\}^T [F] \{q^*_i\} + \{\underline{\lambda}_i\}^T \left[\frac{1}{2} \{q^*_i\}^T \{q^*_i\} - 1\right] \qquad (11)$$

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

$$\frac{\partial \mathfrak{L}}{\partial \{q_i\}} = [F_i]^T\{\lambda_i\} + \underline{\lambda}_i\{q_i\} = 0$$

$$\frac{\partial \mathfrak{L}}{\partial \{\lambda_i\}} = [F_i]\{q_i\} = 0$$

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

$$\frac{\partial \mathfrak{L}}{\partial \{a\}} = [W]\{a\} + \left\{\{\lambda_i\}^T \left[\frac{\partial F_i}{\partial a}\right]\{q_i\}\right\} = 0$$

that can be solved in continuation form on an assigned path, that leads from the old eigenvalues to the desired new ones, possibly using a predictor corrector method of the type presented in Ref. [13]. 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 set-up used in wind tunnel tests is sketched in Figs. 2,3 : 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

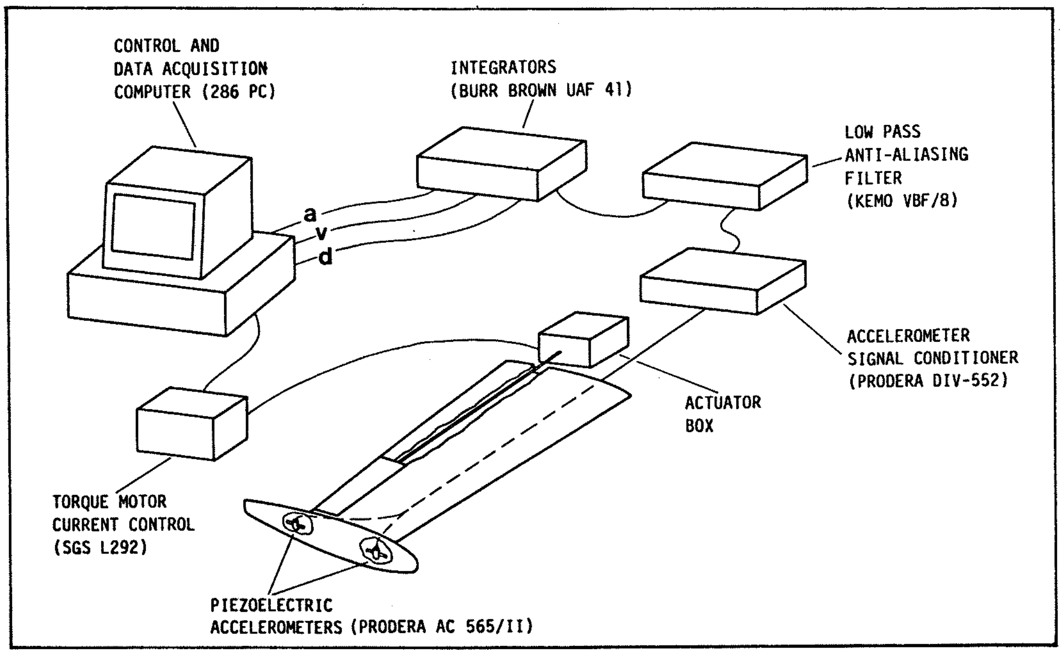

Fig.2 - Experimental set-up

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 technique [19,20], using a continuous excitation through the aileron. A problem emerged during the tests on close 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 torsional mode, even by means of narrow band sweeps of long duration centered around the frequency of interest.

In the following 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.

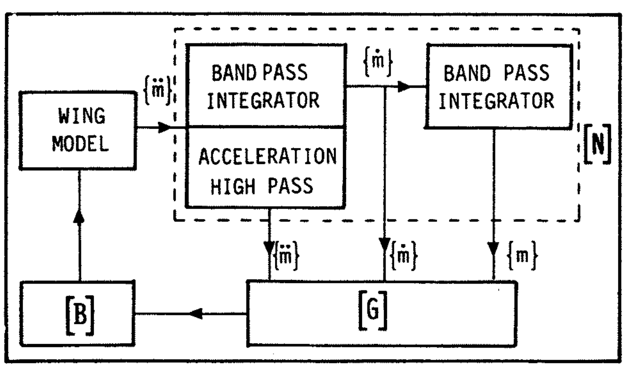

Fig.3 - Active flutter control block diagram

The response of the wing model, both uncontrolled and controlled, is presented in form of classical V-g plots, both predicted and experimentally identified, and in form of time response of trailing edge accelerometer at a wind speed of 45.5 m/s, together with the amplitude of the corresponding transfer function.

Different control strategies have been evaluated, always assuming the same sensors' location at the wing tip. The two transducers were actually piezoeletric accelerometers, and when velocity and/or displacements measurements were required, these quantities have been obtained by filtering and integrating accelerometers 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 ω_f. As a general rule, when the characteristics 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 response and this is our case since ω_f is about one fourth of the bending frequency of wing.

The DC motor driving the aileron was current controlled by a 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.

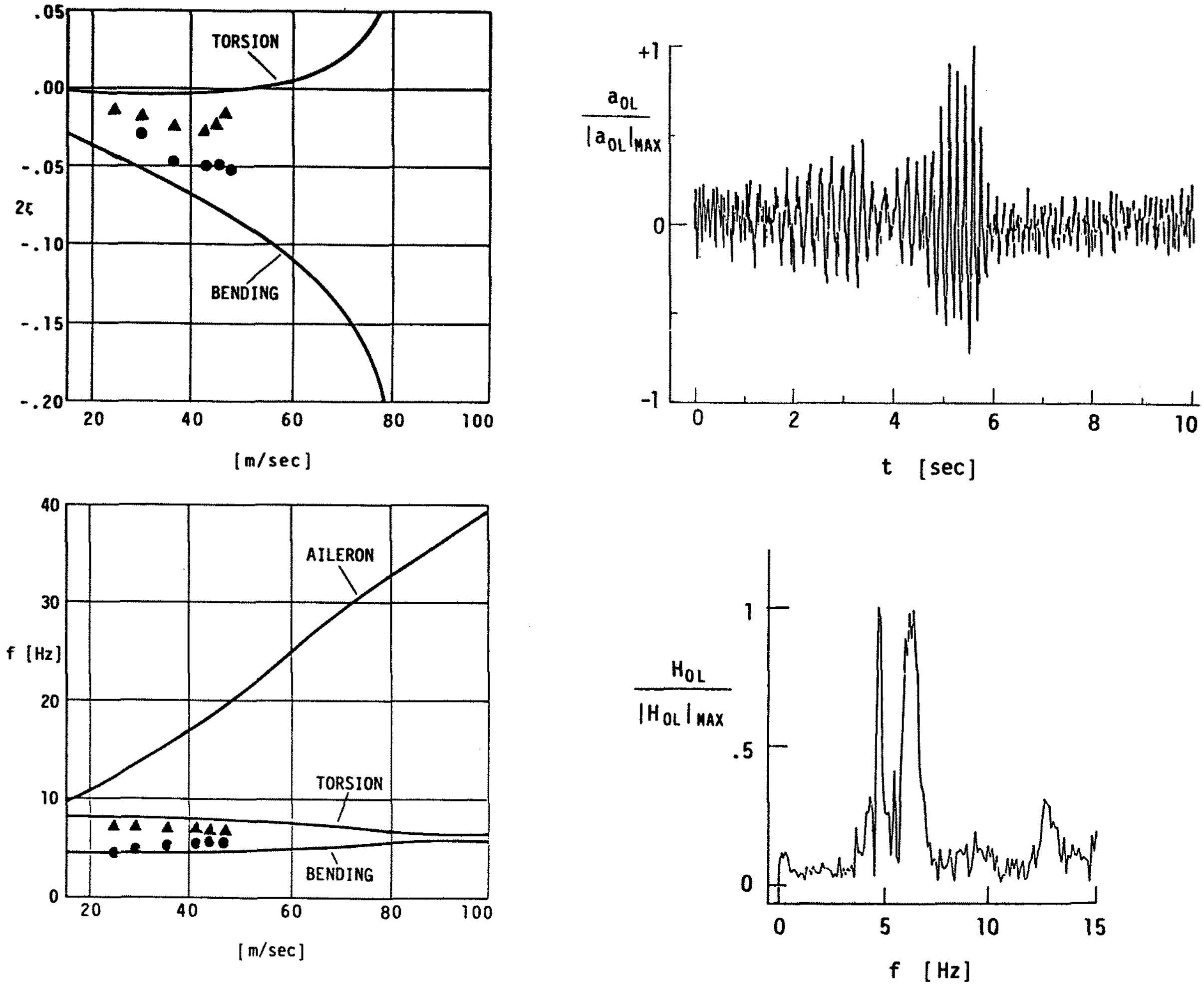

Fig.4 - Open loop system: a) V-g plot; b) Trailing edge accelerometer time response; c) System response transfer function amplitude

The first experimental tests were directed toward the identification of the open loop behaviour 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 Fig. 4 , some differences between analytical and measured frequencies were revealed, it was decided not to modify the analytical model, considering that this situation well reproduced the real design situation.

The simplest control strategy in the example under examination was to measure only the accelerations at the wing tip and feedback them 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 reprehesentative of the typical results attained, that 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 by the points in Fig.5a. It is worthwhile to note that the first implementation of the computed 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, showing the tendency to highly damp the torsional motion without any change on the flexural response.

The next series of tests made use of the accelerations and velocities at the wing tip, and was very easy to design in

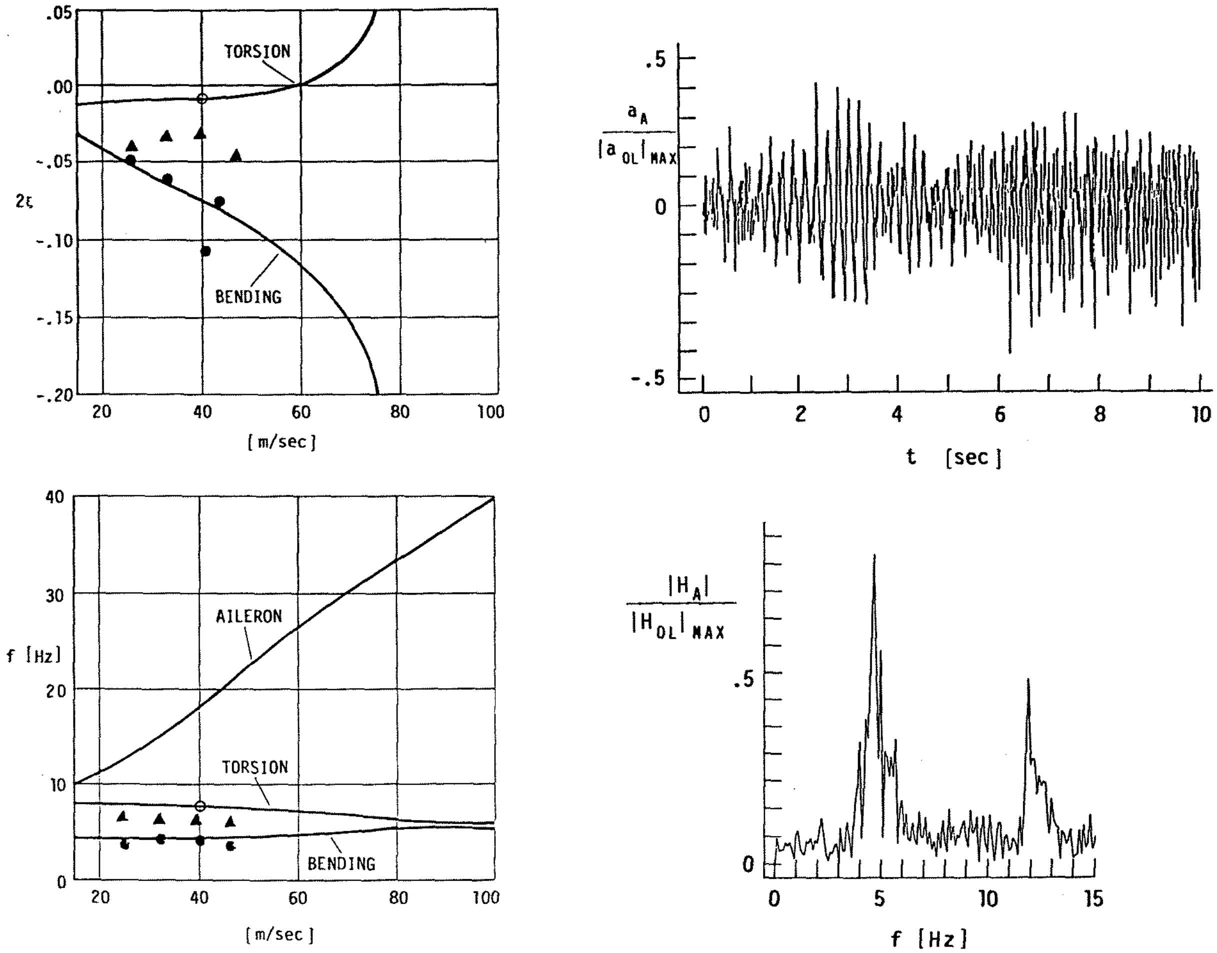

Fig.5 - Closed loop system (Acceleration feedback)): a) V-g plot, O assigned poles ; b) Trailing edge accelerometer time response (V=45.5m/s); c) System response transfer function amplitude

that it was able to achieve the desired flutter improvement, the only problems being related to high gains. In fact some control laws gave rise to a completely flutter free behaviour, but at an expense of excessively high control torques. Fig. 6a reports the final design choosen 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 on the whole speed range. The experimental flutter speed have not been obtained due to limitations in maximum speed attainable in the wind tunnel.

It is important to note that at a relatively high speed the torsional mode is so damped that is barely appearing in the response an the ML identification was unable to identify, which 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 preceeding one,and thus all the comments on control torque limitation are still valid. The response of the structure analitically predicted (Fig. 7a) is quite similar to that obtained using velocity and acceleration measurement. In the experimental realization of this law, velocity and displacement were directly acquired from the integrators, while the accelerations were numerically computed, due to lack of acquisition channels. The

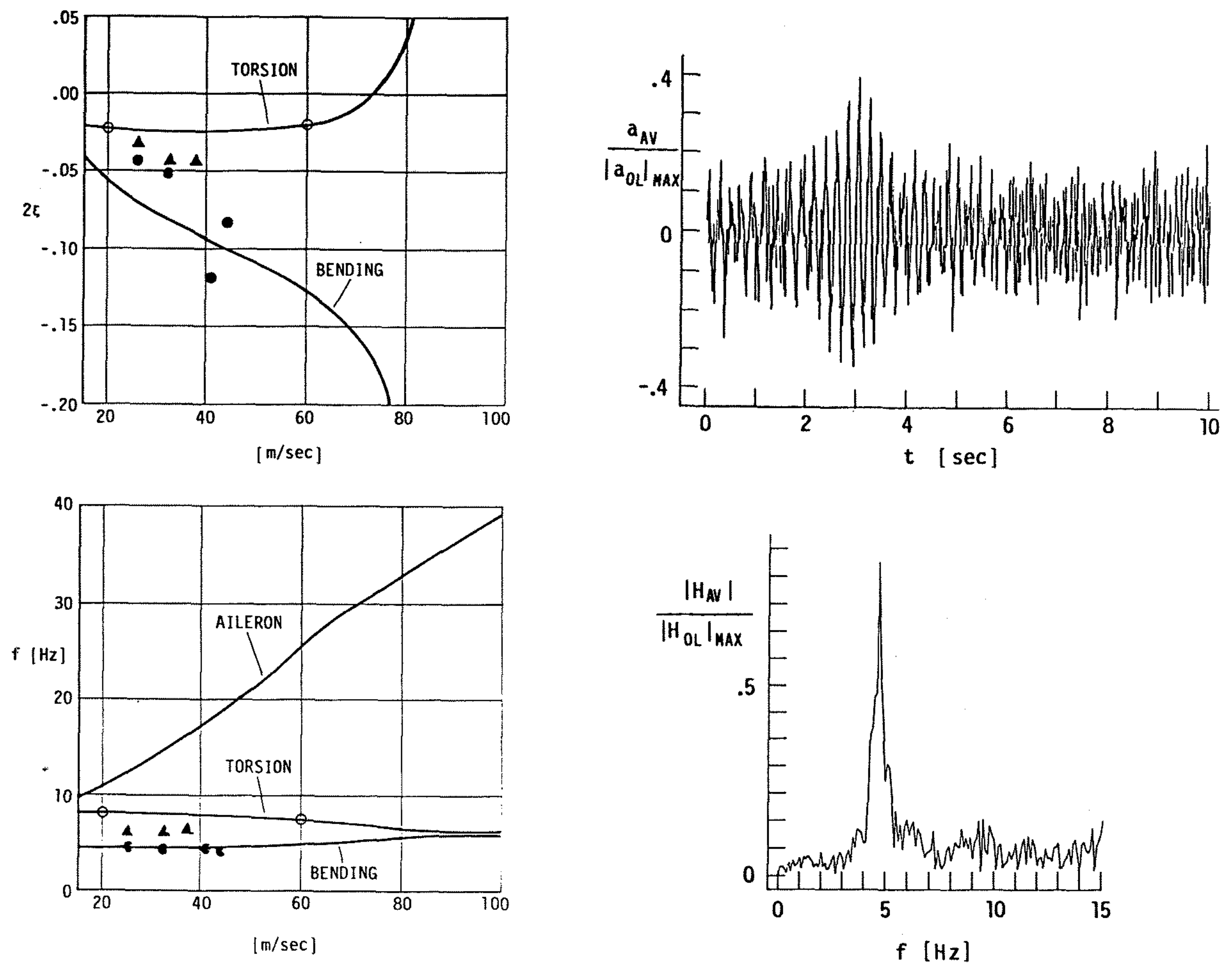

Fig.6 – Closed loop system (Acceleration-velocity feedback)): a) V-g plot, O assigned poles; b) Trailing edge accelerometer time response (V=45.5m/s); c) System response transfer function amplitude

lack of correlation between analysis and the wind tunnel identification remains unexplained, as qualitatively the system was well damped also at high speeds, as shown by Fig 7b.

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

The paper has proved that a simple and effective suppression of the flutter can be obtained by designing the control laws using an eigenvalue assignement 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 they make it easy to use redundant implementations in order to guarantee system reliability. Some of these control laws have proven there feasibility and effectivness in flutter tests on a wing model. The method used to design the control laws requires some tedious trials in order to understand the best location of the desidered eigenvalues 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 simoultaneously for different flight conditions can greatly improve control robustness to structured parameter uncertainties. The use of few

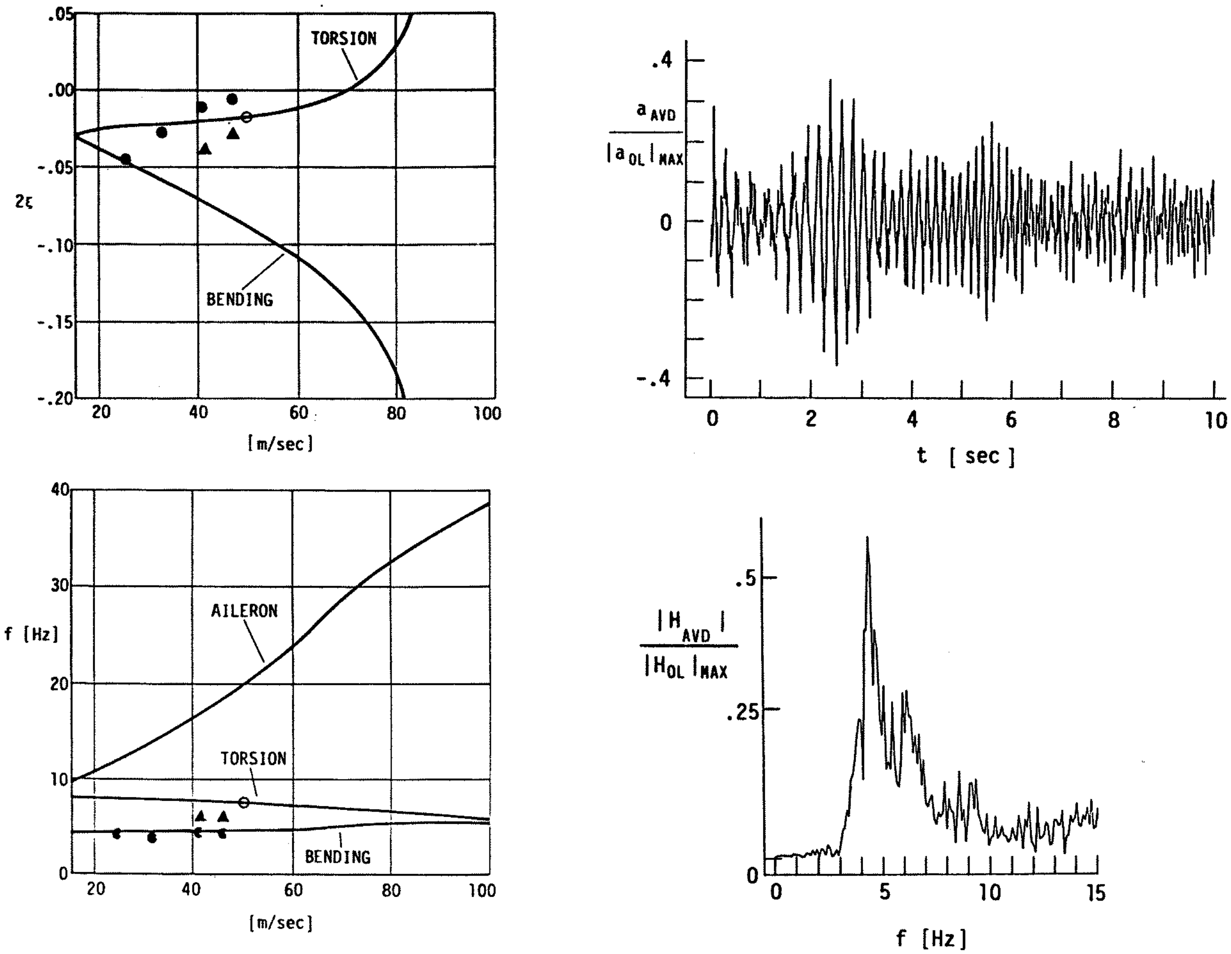

Fig.7 - Closed loop system (Acceleration-velocity and displacement feedback)): a) V-g plot, O assigned poles; b) Trailing edge accelerometer time response (V=45.5m/s); c) System response transfer function amplitude

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 identifications of the damping of aeroelastic systems are augmented by the presence of an effective active control system if the approaching flutter is of violent nature. In this case the flutter is close to the point of maximum damping and this is a condition that is difficult to ascertain as the mode driving the flutter can be so damped that is barely appearing in the response. This fact can be of major concern and an important issue in the validation and certification of really flying active flutter suppression systems.

ACKNOWLEDGEMENT

The authors would like to acknowledge O.Onori and P.Palmieri for their help and assistance during the set up and the execution of this work.

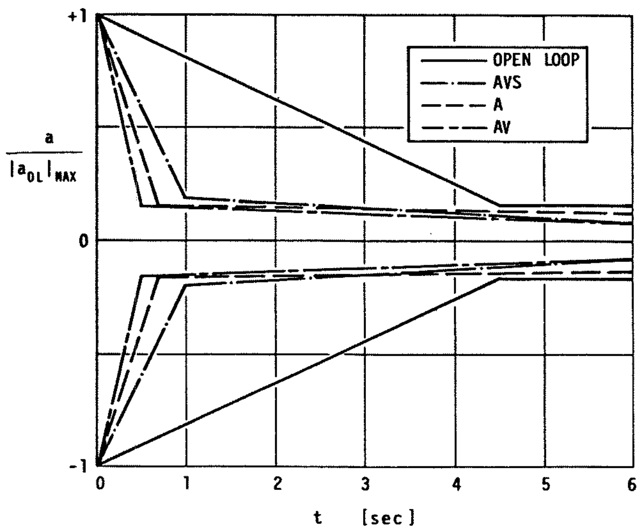

Fig.8 - Comparison of damping envelopes at 36 m/s

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