EXTENDING THE USEFUL FREQUENCY OF "RIGID" WIND TUNNEL MODELS WITH ACTIVE CONTROL

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Abstract

Unsteady pressure measurements on model wings are commonly used for unsteady aerodynamics code validation and for buffet predictions. It is important that the motion of the wing is well defined and that all points on the structure are in phase. Types of motion that are commonly imparted onto the model by means of hydraulic actuators are pitching about a fixed axis or flapping. Because no model is perfectly rigid, the model will deform elastically due to mass coupling between the imparted motion and the structural modes, the aerodynamic forces generated by the imparted motion and wind tunnel turbulence. This elastic deformation limits the useful speed and frequency range of the model. It has been suggested that active control can be used to suppress the elastic response and increase the useful range of the model. In this study the effect of control by a trailing edge control surface on the pressure distribution was investigated. It was found that the additional pressure distribution caused by the control surface motion is of the same order of magnitude as the undesirable pressure distribution caused by the elastic deformation. It is concluded that the possibilities of internal structural control by means of piezo-electrics should be investigated as an alternate means of control.

Introduction

Unsteady pressure measurements on model wings are commonly used for unsteady aerodynamics code validation and for buffet predictions. It is important that the motion of the wing is well defined and that all points on the structure are in phase. Types of motion that are commonly imparted onto the model by means of hydraulic actuators are pitching about a fixed axis or flapping. Because no model is perfectly rigid, the model will deform elastically due to mass coupling between the imparted motion and the structural modes, the aerodynamic forces generated by the imparted motion and wind tunnel turbulence. This elastic deformation limits the useful speed and frequency range of the model. Destuynder and Barreau [1] suggested that active control through the actuator or a control surface can be used to suppress the elastic response and increase the useful range of the model. Control through the actuator will be effective in controlling the elastic response to turbulence or separated flow at the modal frequencies as long as they are not to close to the frequency at which the model is oscillated. If the frequency at which control is applied is too close to the frequency of the imparted motion, the control law will adversely affect the desired imparted motion. Control through a control surface will not affect the desired imparted motion, but may affect the pressure distribution. In this study the effect of control by a trailing edge control surface on the pressure distribution was investigated. It was found that the additional pressure distribution caused by the control surface motion is of the same order of magnitude as the undesirable pressure distribution caused by the elastic deformation.

Nomenclature

q_1	Generalized coordinate, first structural mode
q_2	Second structural mode
q_3	Third structural mode
q_4	Control surface mode
q_5	Actuator mode
α	Actuator pitch angle about axis of rotation
β	Control surface angle measured in constant-y plane

Analysis

The model used for this investigation is a fighter-type wing built for unsteady pressure measurements [2]. The model can be pitched about the 40%-chord line by a hydraulic actuator as shown in figure 1. The model was built from carbon fibre to have a high first natural frequency. The actual wing does not have an aileron, but an imaginary one was added over the outer 35% of the span and 25% chord. The modal data used in this analysis was the first three modes, shown in figures 2 to 4, obtained from a GVT.

Generalized forces were calculated for the three structural modes, the actuator mode and the aileron mode using the doublet lattice method [3]. A flutter analysis was performed using the three structural modes to determine the frequency of the first mode at Mach 0.8 at sea level. Thereafter a response analysis was performed over a range of frequencies covering the first modal frequency. The maximum response of mode 1 to actuator excitation corresponded closely to the frequency as determined by the flutter analysis. This frequency, 49 Hz, was used as one of two frequencies for which pressure distributions were calculated, the other being 20 Hz which is half the first natural frequency and a commonly used limit. The response of the wing to aileron excitation was also determined to derive the required aileron motion to suppress the response of mode 1. The results are shown in tables 1 to 4. The required aileron movement is given by

$$\beta = (-2.98 - i\ 1.18)\ \alpha \qquad (1)$$

at 49 Hz and

$$\beta = (-2.70 - i\ 0.50)\ \alpha \qquad (2)$$

at 20 Hz. The resulting response of the wing with aileron control is presented in tables 5 and 6. The response from mode 1 is not exactly zero because the control ratio was derived from a four digit printout of the response. Pressure distributions for the five modes were calculated at the corresponding reduced frequencies and the pressure distributions for the total response of the wing calculated as linear combinations thereof.

Results

Pressure distributions were calculated at two spanwise positions, viz. at 37% and 83% span. The first position being inboard of the aileron and the other within the spanwise limits of the aileron. The ideal would be to have the pressure distribution due to the actuator mode only. This distribution is shown together with the pressure distributions for the elastically responding wing with and without control. Figures 5 to 8 show the pressure distributions at 49 Hz at the inboard (station 1) and outboard (station 2) positions. The Cp–values are for a 1 radian rotation of the actuator. Figures 9 to 12 show the pressure distributions at 20 Hz. The deviation of the pressure distributions from the ideal is due to the aileron movement as well as the elastic response of modes 2 and 3.

Discussion

The control law was chosen to completely suppress the response of mode 1. This is reflected in the pressure distributions at station 1 where the values with control are close to the desired values (the values for the actuator mode only). Even over the leading portion of the wing at station 2, there is an improvement, but over the trailing portion the values are much further from the desired values with control than without control. The greatest improvement seems to be in the pressure distribution over the leading portion at station 2 at 49 Hz. Although the elastic response to actuator excitation of the wing is much less at 20 Hz than at 49 Hz, the aileron deflection required to suppress the modal response at 20 Hz is almost the same as at 49 Hz. (This will be the case over a wide range of speed and frequency.) The additional pressure distribution caused by the control law is therefore almost the same as at 49 Hz, while the pressure distribution caused by the elastic response without control is much smaller. The nett improvement in pressure distribution is therefore much less at 20 Hz than at 49 Hz.

Conclusion

From the results presented, it is evident that suppression of modal response by a control surface for the purpose of unsteady pressure measurements is not the ideal solution. The control activity disturbs the pressure field in the vicinity of the control surface as much as the elastic response of the wing does. It is also evident that the suppression of modal response does improve the pressure distribution away form the immediate vicinity of the control surface to such an extent that a model could be used at its modal frequencies.

Considering the state of the art in smart structures, it seems possible to devise some form of internal structural active control that would not disturb the pressure distribution. To obtain an approximate indication of the forces involved, the bending and torsion moments generated by the actuator mode at 49 Hz was calculated and are shown in figures 13 and 14. For buffet investigations, small amplitudes are dictated by other considerations [1], so that the root bending moment of 282 Nm/deg may not be excessive in the near future provided the piezo–electrics can withstand the strain of static deformation, which may be much greater than the dynamic strain.

Where conventional control surfaces are used, the type of analysis presented here should be performed to assess the effect of the control law on the pressure distribution.

References

[1] Destuynder, R. and Barreau, R., "Nouvelle Methode de Determination des Forces de Tremblement en Soufflerie", 27 éme Colloque d'Aerodynamique Appliquee, Marseille, 15–17 October 1990.

[2] Marillier, C.A.R., "Design, Construction and Ground Vibration Testing of a Rigid Pressure Model", NIAST Report 88/70, March 1988.

[3] Rodden, W.P. , Giesing, J.P. and Kalman, T.P., "New developments and applications of the subsonic doublet lattice method for non–planar configurations", AGARD CP–80–71, Part 2, No. 4, 1971.

Tables 1 to 6

Table 1 Response to actuator excitation at 49 Hz.

	Real	Imaginary
q_1	-29.44	99.44
q_2	3.38	-4.43
q_3	4.66	-3.16
q_4	0.00	0.00
q_5	1.00	0.00

Table 2 Response to aileron excitation at 49 Hz.

	Real	Imaginary
q_1	2.98	32.18
q_2	-1.38	-1.32
q_3	1.90	0.65
q_4	1.00	0.00
q_5	0.00	0.00

Table 3 Response to actuator excitation at 20 Hz.

	Real	Imaginary
q_1	-36.45	0.07
q_2	-0.60	-0.09
q_3	1.82	-1.64
q_4	0.00	0.00
q_5	1.00	0.00

Table 4 Response to aileron excitation at 20 Hz.

	Real	Imaginary
q_1	-13.07	2.44
q_2	-1.20	0.10
q_3	0.42	0.13
q_4	1.00	0.00
q_5	0.00	0.00

Table 5 Response to excitation with control at 49 Hz.

	Real	Imaginary
q_1	0.01	-0.01
q_2	2.21	0.43
q_3	-0.23	-7.57
q_4	-2.98	-1.18
q_5	1.00	0.00

Table 6 Response to excitation with control at 20 Hz.

	Real	Imaginary
q_1	-0.01	0.00
q_2	1.00	0.09
q_3	0.78	-2.28
q_4	-2.70	-0.50
q_5	1.00	0.00

Figures 1 to 14

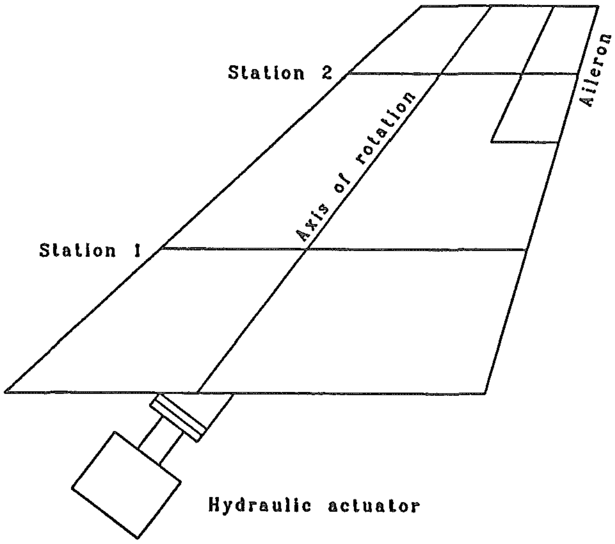

Figure 1 Planform of wing model showing actuator, aileron and stations where pressure distributions were calculated.

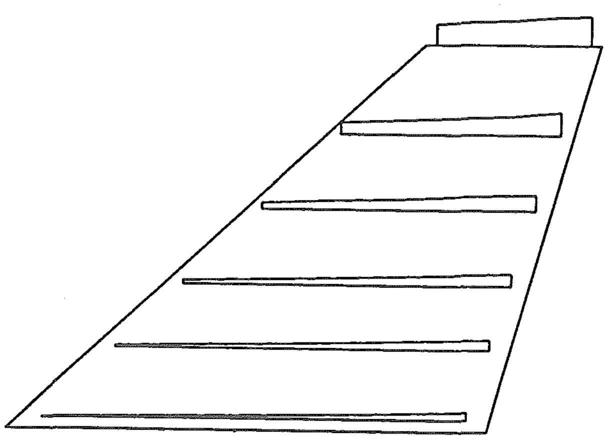

Figure 2 Natural mode 1, f = 39.28 Hz, M = 0.66 kg cm²

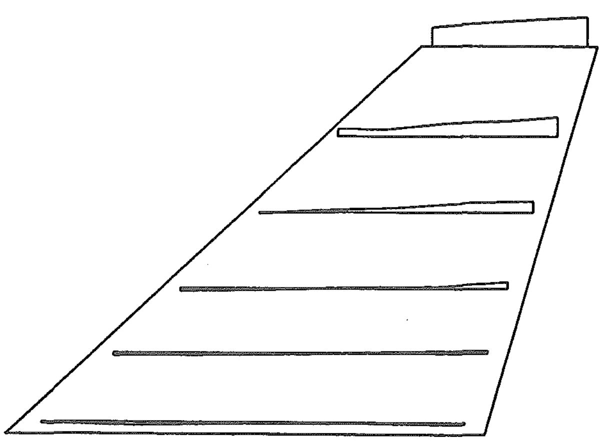

Figure 3 Natural mode 2, f = 66.57 Hz, M = 7.18 kg cm²

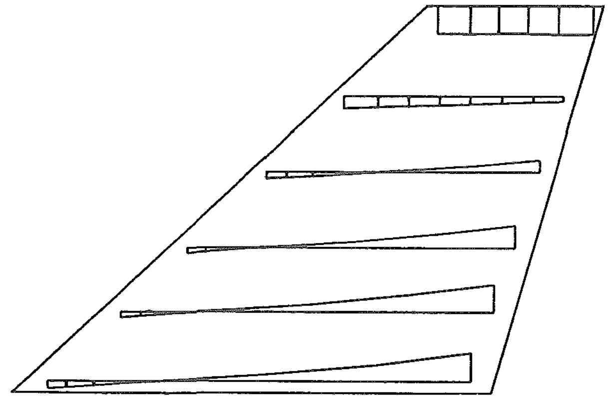

Figure 4 Natural mode 3, f = 92.28 Hz, M = 0.94 kg cm²

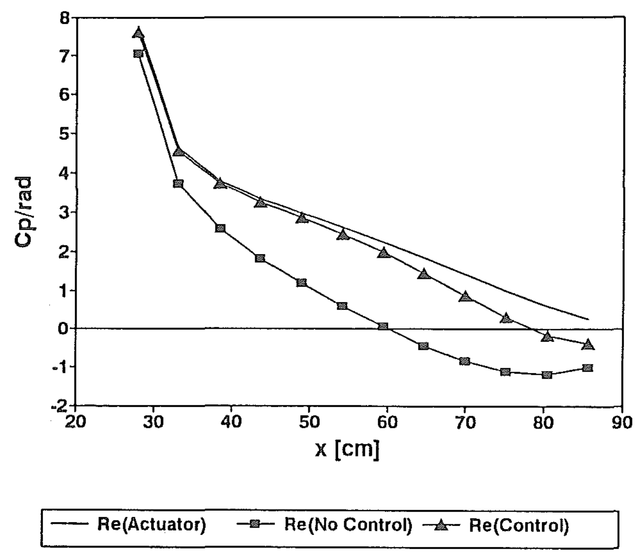

Figure 5 Pressure distribution at station 1 at 49 Hz, real part.

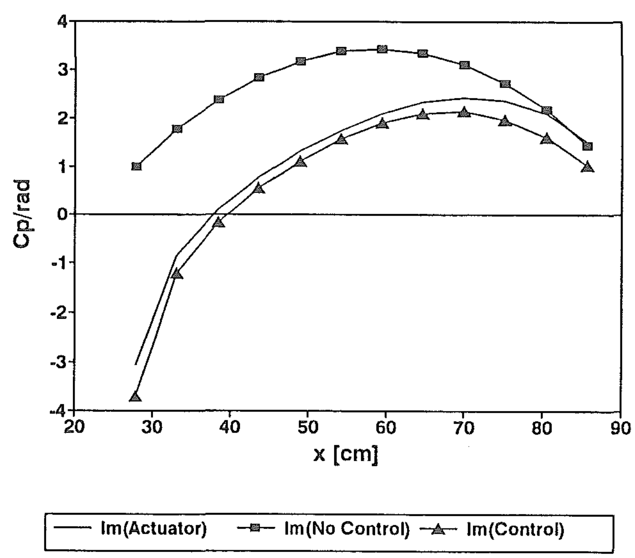

Figure 6 Pressure distribution at station 1 at 49 Hz, imaginary part.

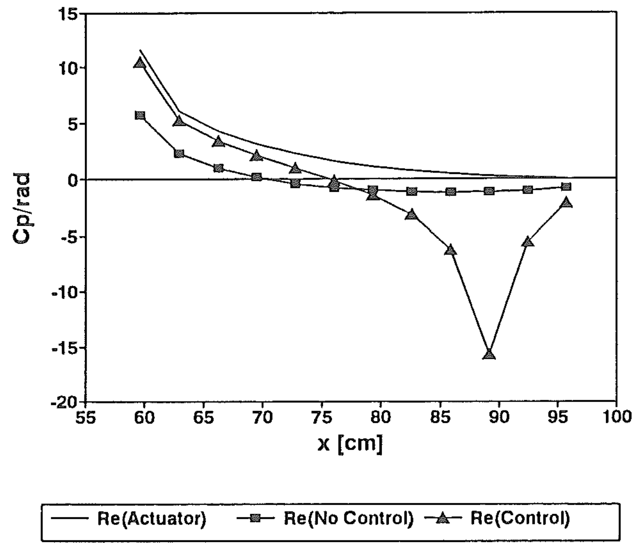

Figure 7 Pressure distribution at station 2 at 49 Hz, real part.

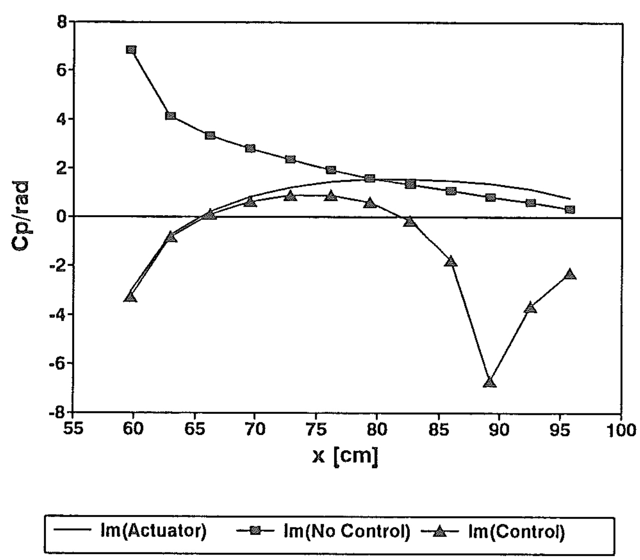

Figure 8 Pressure distribution at station 2 at 49 Hz, imaginary part.

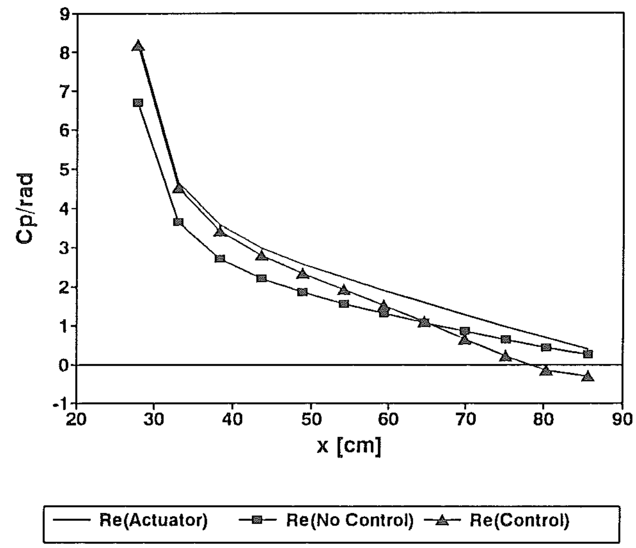

Figure 9 Pressure distribution at station 1 at 20 Hz, real part.

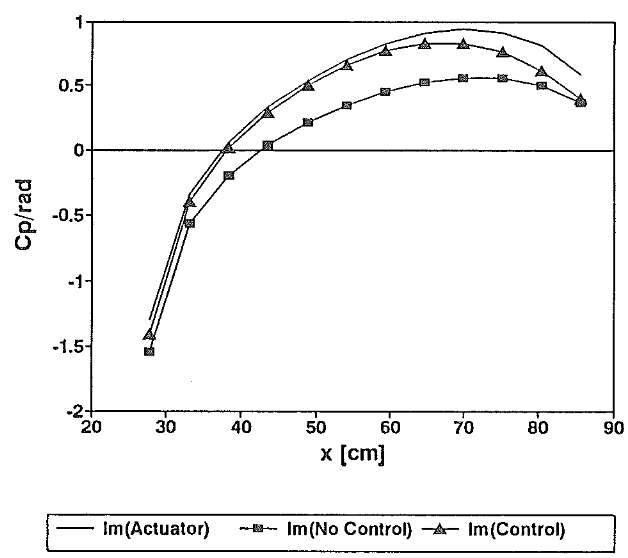

Figure 10 Pressure distribution at station 1 at 20 Hz, imaginary part.

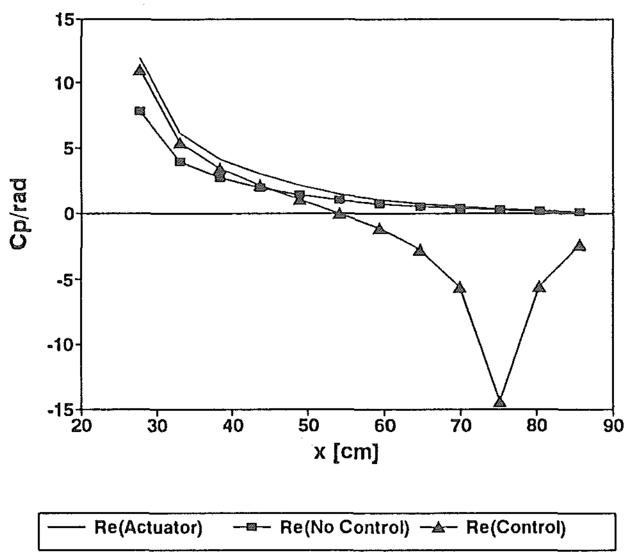

Figure 11 Pressure distribution at station 2 at 20 Hz, real part.

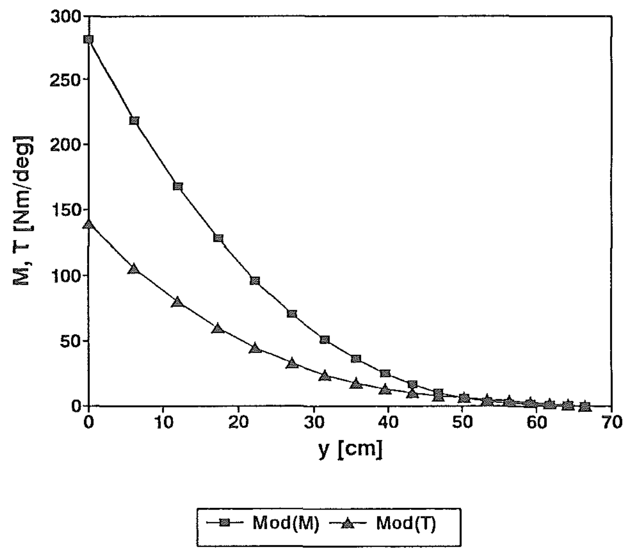

Figure 13 Bending and torsion moment along the span at Mach 0.8, magnitude.

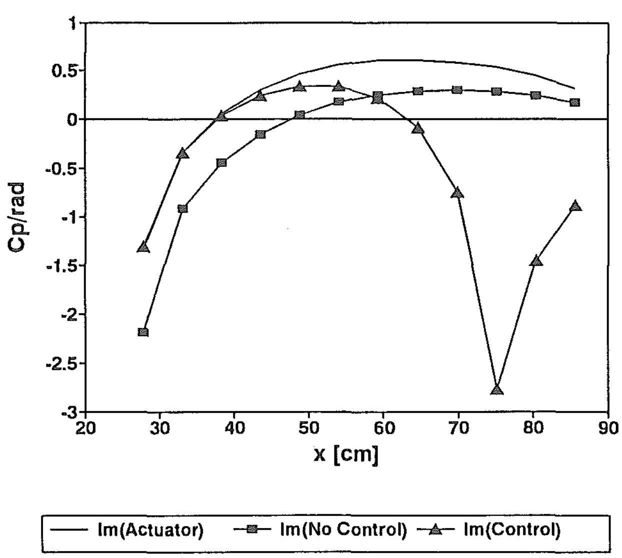

Figure 12 Pressure distribution at station 2 at 20 Hz, imaginary part.

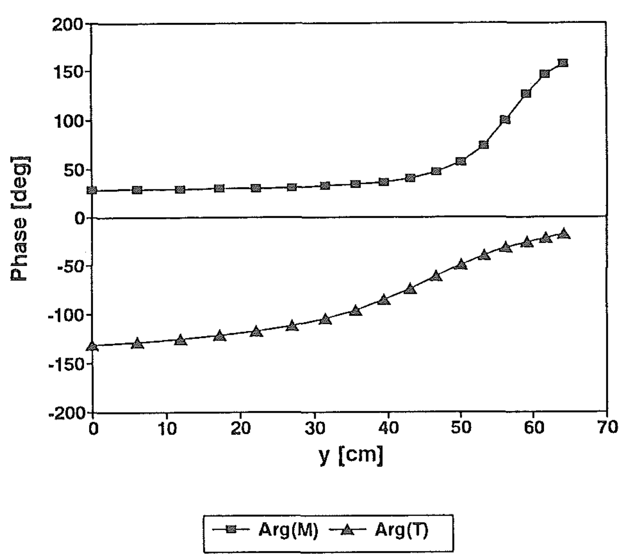

Figure 14 Bending and torsion moment along the span at Mach 0.8, phase.