Unsteady Pressure Distribution of a Wing-Body Configuration. MSC / NASTRAN results compared with Wind Tunnel Tests.

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ABSTRACT

Although the doublet-lattice method implemented in MSC/NASTRAN give good results for the aeroelastic analysis with SOL 75, there are configurations where test data is required because of the very important safety criterion of flutter stability for an airplane design.

In the present paper, a new method of unsteady pressure measurement was adopted. With this method, the surface of the model is covered by a special coating which changes the reflection properties of a laser beam with pressure. With this method, no internal pressure tubes are necessary, which facilitates the pressure measurements considerably.

The measured pressure distribution is compared to the results of MSC/NASTRAN using the doublet lattice method combined with slender body and interference elements.

1. Introduction

For an aeroelastic analysis of an airplane, the modal analysis and the determination of the unsteady aerodynamic forces are the most important steps.

The modal analysis is usually verified by the ground vibration test performed on the prototype. Normally, the theoretical model is updated in accordance with the test results. However, with MSC/NASTRAN, also the test results can be used for the flutter analysis using some minor RF-Alters and Fortran programs, writing the geometry and the mode shapes to OUTPUT2-files.

The verification of the unsteady aerodynamic forces involves wind tunnel testing of a dynamic similar model, or at least with a model being able to represent the downwash distribution of the critical modes. In addition to the difficulties with the requirement of dynamic similarity, the measurement of the dynamic pressure distribution requires a large number of pressure transducers and tubes inside the model, which leads to a very expensive wind tunnel model.

Therefore, in most cases, the flutter calculation relies upon the theoretical values calculated; for example, by the doublet lattice method for subsonic flow. There are, however, cases where theory cannot predict reliable aerodynamic coefficients. Examples are flow at high angles of attack, where dynamic stall will introduce non-linearities, and coplanar motion of T-tails. For control surfaces with tabs, the theoretical aerodynamic coefficients are very sensitive to the cross section and to the flow through the slot. In this case, also the boundary layer, the effective axis of control surface and tab hinge and the effective aerodynamic balance cannot be predicted accurately without experimental data

The doublet lattice method does not take the thickness of the airfoil into account. Furthermore, the aerodynamic modeling itself influences the results more than what is the case for the structural model.

Transonic effects, including interaction with the boundary layer and the shock wave motions are not yet state of the art for aerodynamic codes. Also for this case, measurements are necessary.

With a new measurement technique, based upon optical measurement of the pressure acting on a special coating, even the dynamic pressure can be measured externally at any point (of the model), without any complication of the model.

The main purpose of this paper is to demonstrate how this new measurement method, applied to a very simple rigid model oscillating at different amplitudes in subsonic flow, can be related to the pressure distributions calculated by MSC/NASTRAN with the doublet lattice method. With a Fortran program, the experimental generalized aerodynamic influence coefficients can be calculated and compared to the coefficients found with MSC/NASTRAN. With a DMAP Alter, these coefficients can be used for the flutter ananlysis in SOL 75.

2. The Model.

Until now, the wind tunnel model shown in fig. 1 has been tested at a Mach number of 0.88, oscillating with an amplitude of 1.5 degrees about a quasi steady angle of attack of 0 degrees.

Further tests at Mach 0.6 and for a quasi steady angle of attack of 0 and 10 degrees will be performed.

The pressure was measured along the chord lines of the panel distribution shown in fig. 2, which is a cosine distribution in chord and span directions.

Pitch motions about an axis of 30% chord have been measured until now.

The results of the low frequency oscillations can be used for flight dynamic analysis. This means that phugoid, Dutch roll, etc. can be studied. Also these calculations can be performed using MSC/NASTRAN. For the rigid airplane, only the rigid body modes need be accounted for and the generalized aerodynamic coefficients are the flight dynamic derivatives. In this case, the influence of the fuselage is important.

For the first tests, the frequency was 10 Hz. Using the chord as reference length, the reduced frequency was 0.02, which is a rather low value.

Actually, the test frequency is limited to the speed of the data aquisition system. This system is now being further developed in order to be able to scan up to 60000 external measuring points 5 times per cycle at frequencies up to 50 to 100 Hz. However, in order to obtain higher reduced frequencies, the model could also be scaled up and tested in a larger tunnel.

3. Wind Tunnel Measurements.

The model was tested in a pressurized wind tunnel with a cross section of 1.25 by 1.25 meter.

The span of the model was 0.45 meter so the blockage was minimal. The measurement section has slotted walls in order to minimize the interference.

The model was oscillating free about the pitch axis, and the motion was registered by means of accelerometers in order to obtain a basis to the pressure measurements. In this way, the real and imaginary pressure could be calculated. In additon to the pressure, the normal force was measured by means of a strain gauge balance.

In Fig. 3, the angle of attack, the normal force and one pressure measurement point are shown. In the pressure curve, the scattering is relatively large so a smoothing tecnique has to be used. Fig. 4 shows the distribution of the measured pressure coefficients of the lower and upper side of the wing at 80% span.

4. Aerodynamic Force calculated with MSC / NASTRAN

The analytical model with panel distribution is shown in fig. 2. The panels of the wings are cosine distributed. The fuselage was modeled by slender body elements. Interference elements were placed around the cylindrical part of the fuselage. A typical pressure distribution of the wings without fuselage, assuming the wings to be extended to the center line, is shown in fig. 5 for Mach 0.88 and a reduced frequency of 0.02.

Fig. 6 shows the pressure coefficients for the two different panel divisions.

Using larger panels near the leading edge, the pressure peak has decreased.

5. Comparison between Theory and Experiment..

A comparison of the pressure coefficients at the 80% spanwise station of the wing is shown in fig. 7. The solid line represents the results of MSC/NASTRAN, the dashed line shows the results of VARDOB, a doublet lattice program developed by the Dutch Aerospace Laboratory (NLR). The dotted line respresents the experimental data obtained by the optical measurement method.

The theory is based on the flow around an oscillating flat plate. The profile used was NACA 64 006 with 6% thickness at 40% chord as shown in fig. 8. At Mach number 0.88, the local Mach number at 40% chord is 0.96 for 0 angle of attack. When the section is oscillating with an amplitude of 1.5 degrees, there will probably be weak shock waves and boundary layer interactions around the maximum profile thickness and expansion lines further aft. This can explain the difference in the curves between leading edge and half chord, and the pressure drop further downstream. Near the trailing edge, the boundary layer may have some influence on the pressure coefficient.

The most significant difference is found close to the leading edge. In the theory, there is a singularity at the leading edge, which leads to an infinite pressure. The experimental pressure distribution shows only a slight increase near the leading edge but no singularity. This is probably due to the leading edge radius of curvature (0.256% of chord) of the real profile which cannot be modeled with the doublet lattice theory.

Further tests will be made for lower Mach numbers in order to avoid transonic effects.

6. Associated Programs and DMAP Alters

Special programs and DMAP-Alters were used for the following purposes:

- Plots of MSC/NASTRAN aeroelastic panels and pressure distribution.

- Plots of earodynamic influence matrices.

- Interface program and DMAP-Alter for using experimental aerodynamic influence matrices in SOL 75.

7. Conclusion.

A new rational method for pressure measurement of wind tunnel models has been applied to a wing body configuration oscillating in pitch. The test results are compared with the theoretical pressure distribution obtained with MSC/NASTRAN using the doublet lattice method and interference elements. In this first comparision the Mach number was 0.88, which means that transonic effects were signifcant.

With this measurement method, it is possible to check the unsteady aerodynamic pressure distribution of models and details where the theory may be insufficient.

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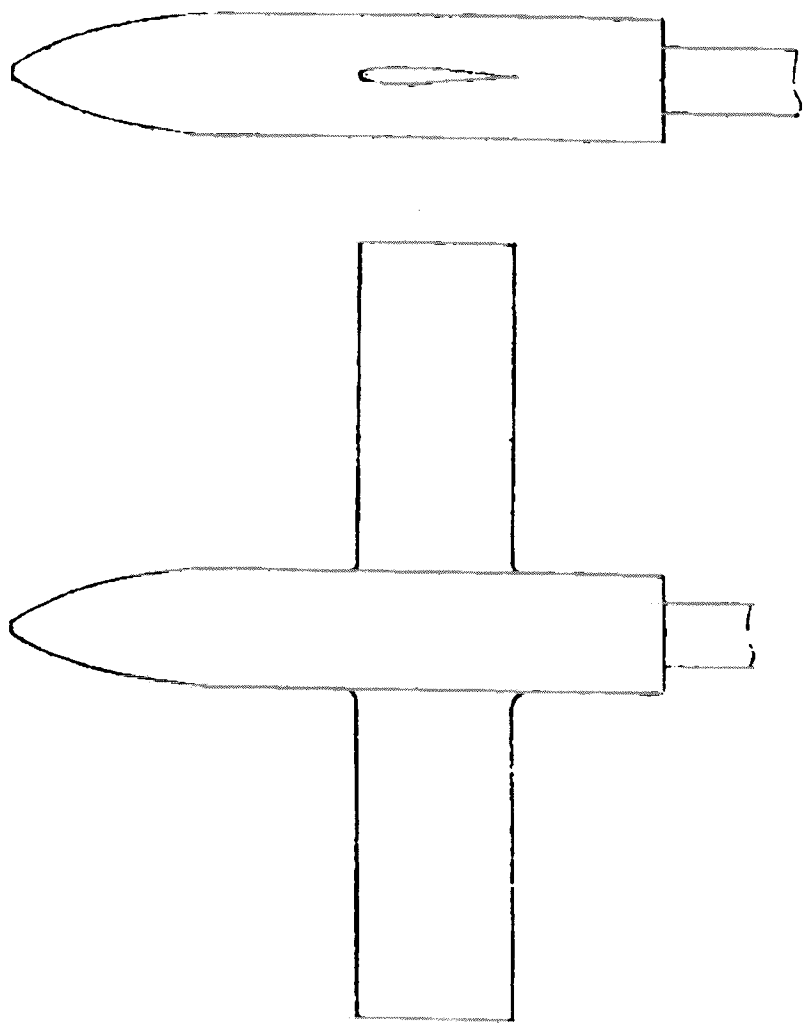

Fig. 1 Wind Tunnel Model

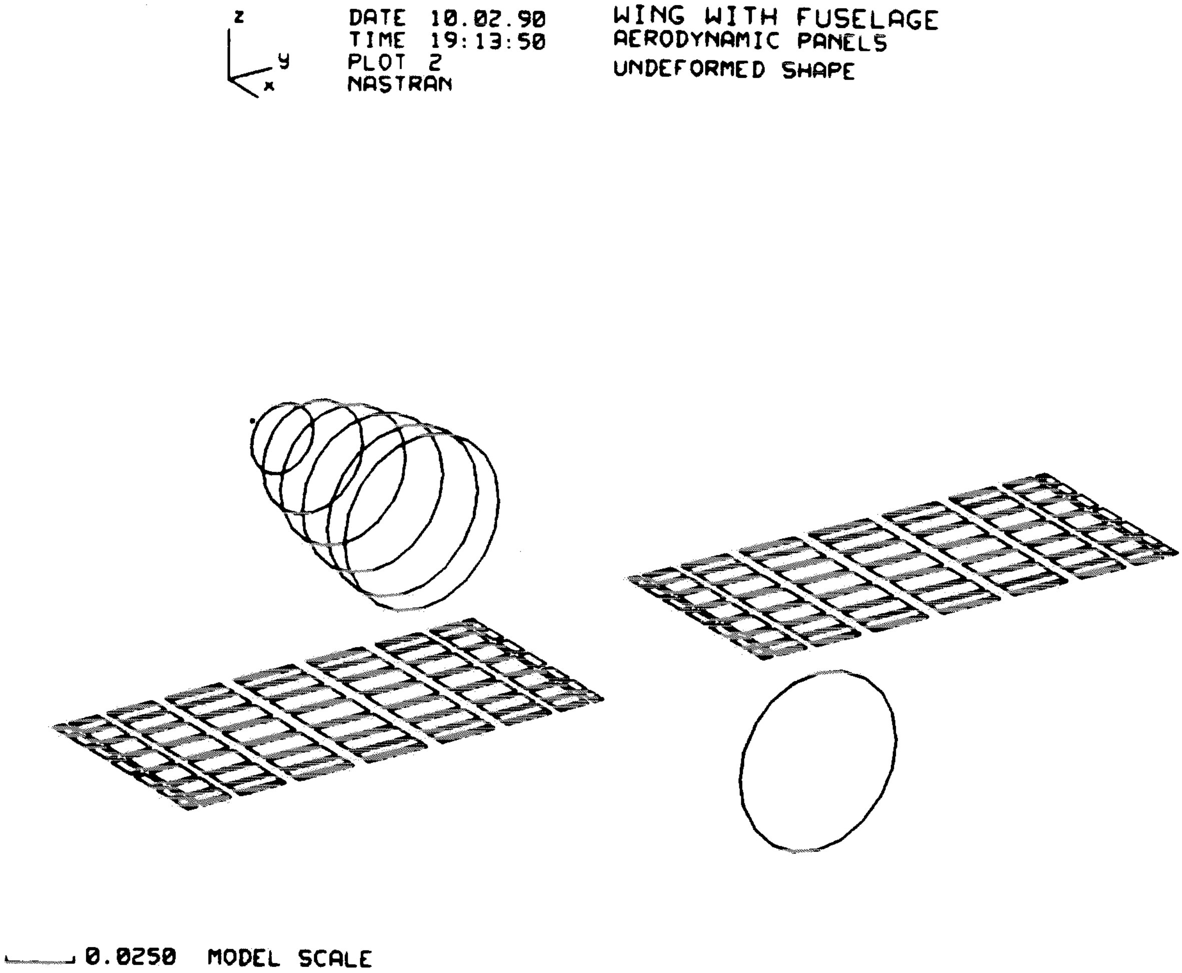

Fig. 2 Panel distribution, NASTRAN model

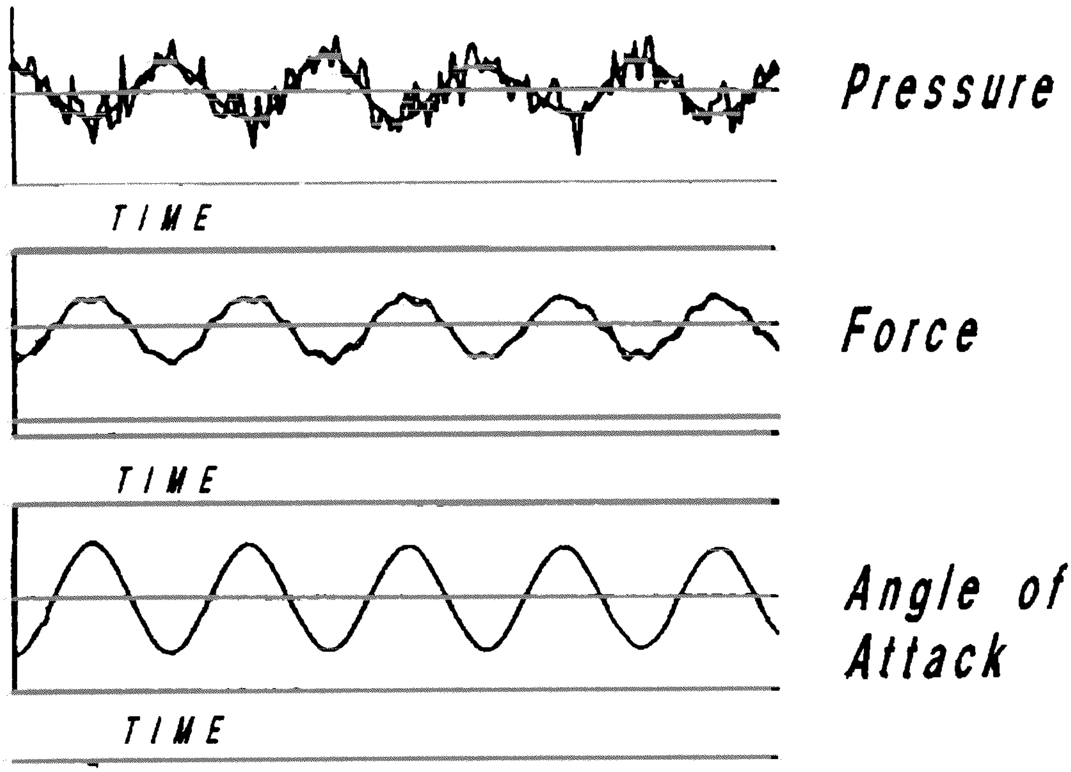

Fig. 3 Time plots

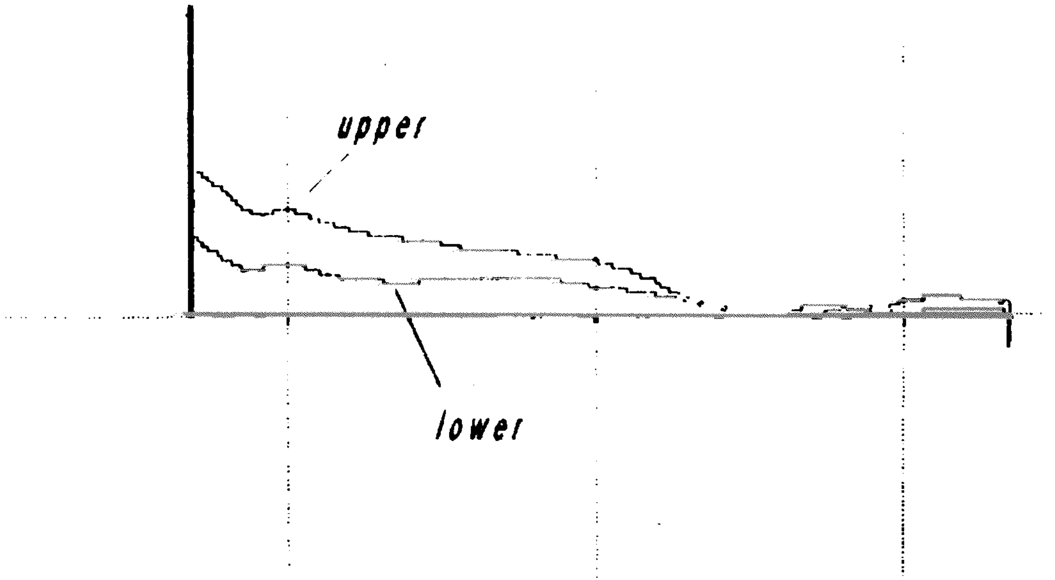

Fig. 4 Pressure coefficients on upper and lower side of wing station at 80%.

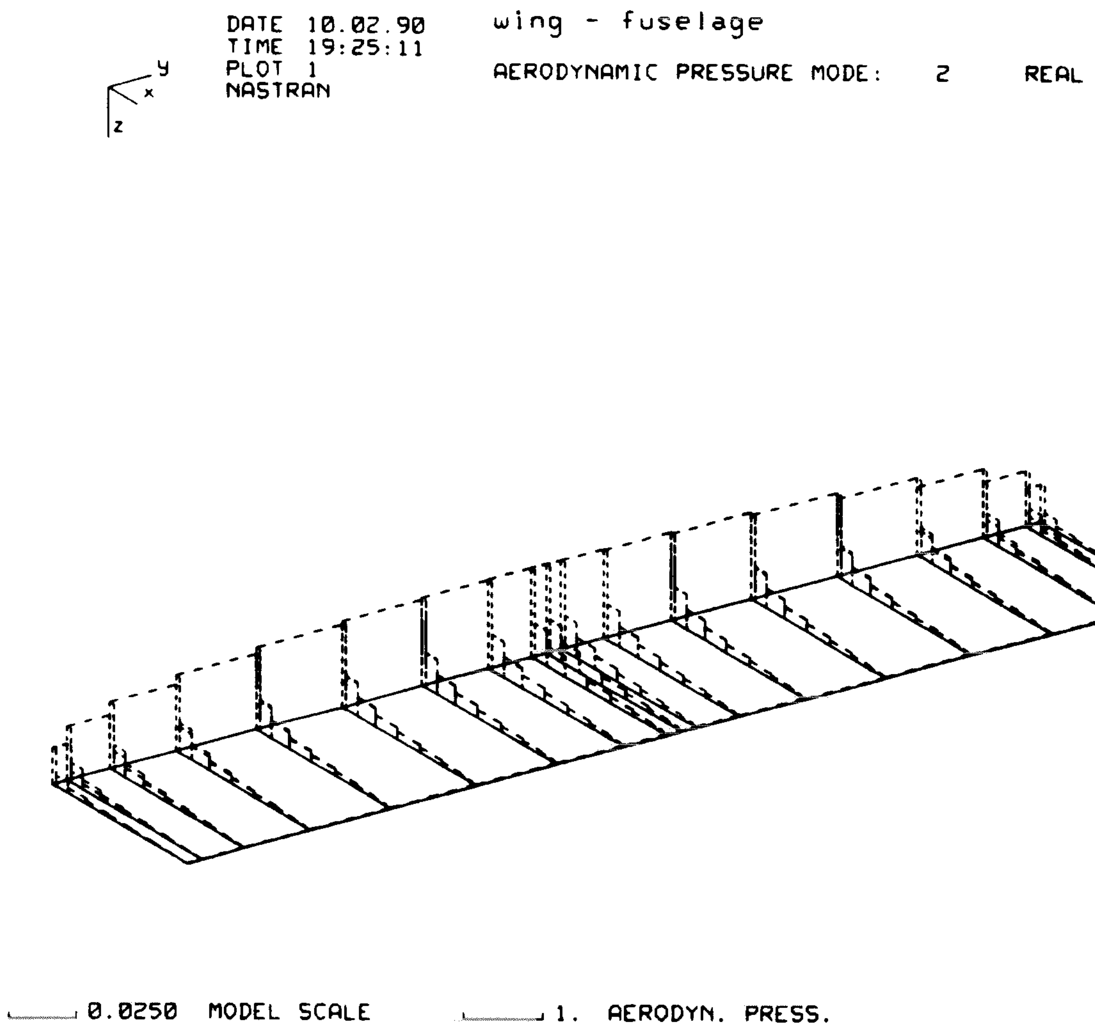

Fig. 5 Pressure distribution of NASTRAN model

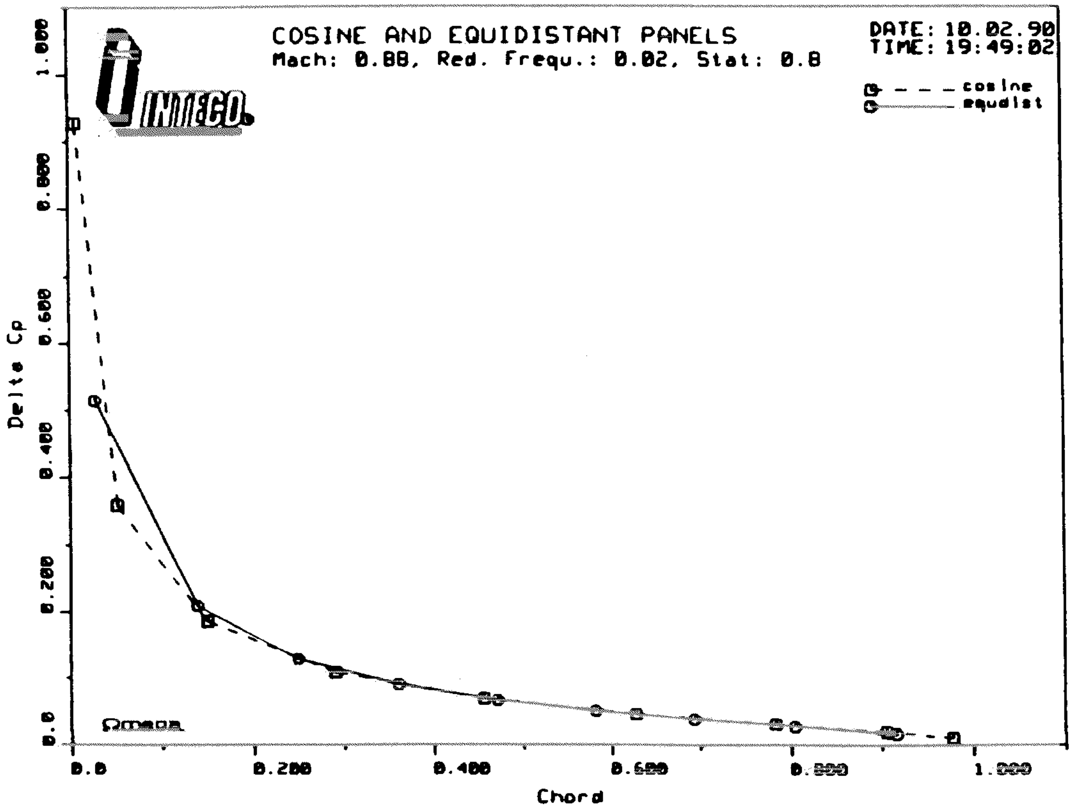

Fig. 6 Pressure coefficients of cosine and equidistant panels

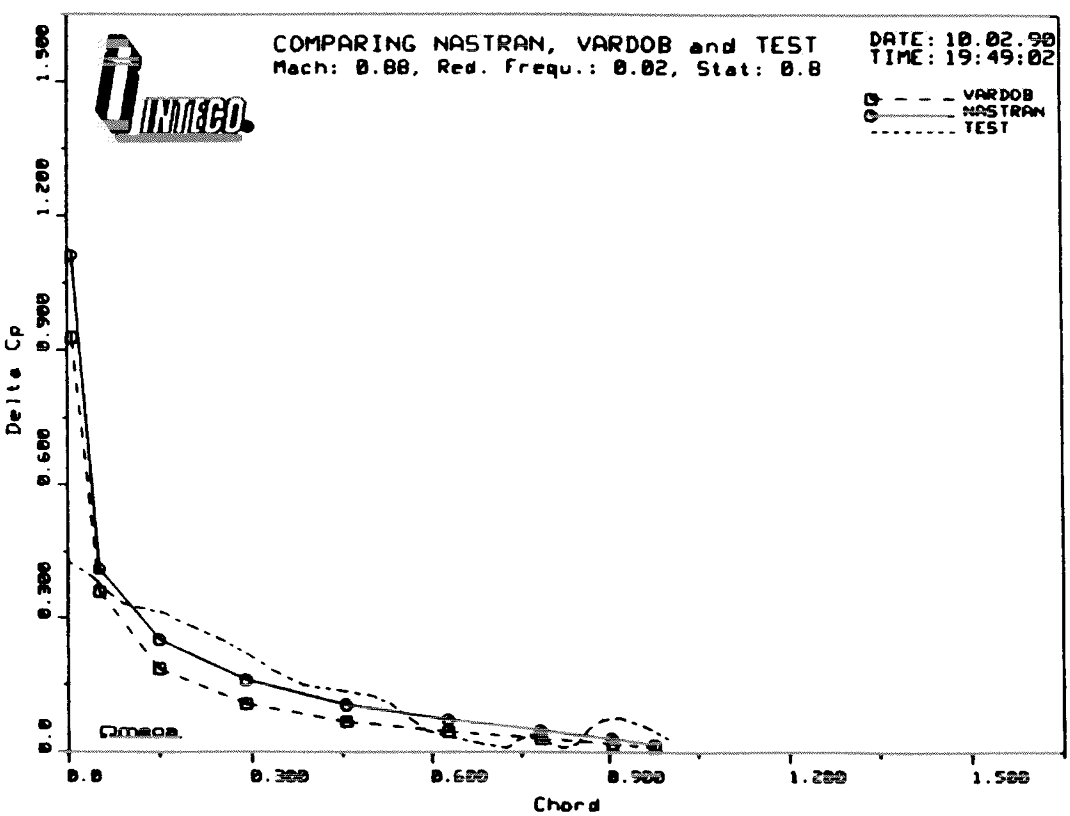

Fig. 7 Pressure coefficients of NASTRAN VARDOB and test

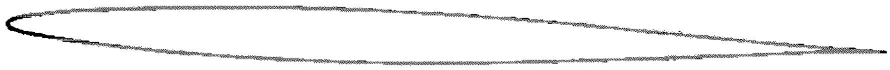

Fig. 8 NACA 64006 profile