

**EXPERIMENTAL INVESTIGATION OF THE AERODYNAMICS
OF FIXED AND RAPIDLY-ACTIVATED SPOILERS**

by

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ABSTRACT

Spoilers are control devices used to reduce the amount of lift and to provide lateral stability on flight vehicles. However, when a spoiler is rapidly activated, there is the formation of an initial vortex, which causes the lift to increase momentarily before the final "spoilt" lift is reached and this is known as the adverse lift effect. The present investigation looked into the aerodynamics of 2-D fixed and rapidly acting spoilers. Experiments were carried out with pressure and velocity measurement to study both the steady and unsteady characteristics.

For the fixed spoiler, forces acting on the aerofoil and the spoiler were measured separately using a pressure averaging technique. Measurements of surface pressure are used to determine features of the local flow field. Spectral analysis was carried out to study vortex shedding. It was found that the resulting flow field resembles that of an unsymmetrical bluff body, and is strongly dominated by the interaction between the top shear layer (TSL), which separates from the spoiler tip, and the bottom shear layer (BSL), which separates from the trailing edge of the lower aerofoil surface.

For the rapidly acting spoiler, transient forces were measured at different spoiler deployment times using the pressure averaging technique. In addition, transient pressure distributions were also measured at two deployment times. It was found that the adverse lift effect on the aerofoil is not directly related to the rate of spoiler deployment, but depends on the overshoot of the compression pressure ahead of the spoiler and the suction aft of the spoiler. The compression overshoot is caused by the acceleration of the spoiler and the suction is due to the passage of the initial vortex.

The initial vortex and its suction effect were further studied by performing velocity measurement in the transient wake. Vorticity was calculated from the velocity measurements for two spoiler deployment times. It was found that at a moderate deployment time when the adverse lift is still profound, more than one vortex is shed from the moving spoiler. The relationship between the suction aft of the spoiler and the velocity outside the wake due to the passage of the these vortices was also studied at different deployment times. The number of vortices shed from the moving spoiler is dependent on the spoiler deployment time, which dictates how and when the TSL and BSL interact. Therefore, as in the case of a fixed spoiler, the transient flow field is strongly influenced by the interaction between the TSL and BSL.

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LIST OF SYMBOLS

BSL	Bottom shear layer
$C_L(t)$	Lift coefficient as a function of time
C_{La}	Maximum adverse lift coefficient
C_N	Normal force coefficient
C_N'	Rms normal force coefficient
C_p	Pressure coefficient
C_p'	Rms pressure coefficient
C_{pb}	Base pressure coefficient
C_{px}	Spatial mean pressure coefficient on the spoiler front surface
C_{py}	Spatial mean pressure coefficient on the spoiler rear surface
C_f	Coefficient of the force acting on the spoiler
C_f'	Coefficient of the rms force acting on the spoiler
c	Aerofoil chord
f	Vortex shedding frequency
h	Spoiler height
H	Spoiler projection height
l_a	Afterbody length aft of the spoiler
S_c	Strouhal number with the chord length as the length parameter
S_H	Strouhal number with the spoiler projection height as the length parameter
t	Time from 10 ms prior to the spoiler activation
t_a	Time to adverse lift

t_d	Spoiler deployment time
t_e	Time following the onset of spoiler deployment
t_f	Time to achieve 95% of the final lift
t_s	Time at which the spoiler stops moving
t_v	Time at which peak C_{p16} is reached after the onset of deployment
T	Non-dimensional spoiler deployment time
TSL	Top shear layer
u	Velocity in the x direction
u'	Rms velocity in the x direction
u^*	u-velocity non-dimensionalised by U_o
U	Time mean velocity in the x direction
U_b	Boundary layer separation velocity
U_o	Free stream velocity
U_{tip}	Velocity of the spoiler tip
v	Velocity in the y direction
v'	Rms velocity in the y direction
v^*	v-velocity non-dimensionalised by U_o
V	Time mean velocity in the y direction
x	distance along the co-ordinate axis parallel to the chord
x_s	Spoiler location along the chord
y	distance along the co-ordinate axis perpendicular to the chord
α	Aerofoil incidence
δ	Spoiler angle

δ_a	Spoiler angle at which maximum adverse lift occurs
δ_{ss}	Final spoiler angle at steady state
Γ	Circulation
Γ^*	Non-dimensional circulation
ω	Vorticity
ω^*	Non-dimensional vorticity

1. INTRODUCTION

1.1. General Remarks

Spoilers are control devices used on flight vehicles to provide lateral stability. A simple spoiler is basically a flat plate mounted on the upper surface of a wing, and will cause the flow to separate when deflected, thereby "spoiling" the streamline flow and reducing the lift. Although the primary role of spoilers is to provide lateral stability, they are also being used to provide gust alleviation, or to provide air braking and lift dumping on landing.

The aerodynamics of spoilers is a complex subject. Besides reducing lift, it also involves other aerodynamic phenomena, both unsteady and transient. When a spoiler is very rapidly deployed to its desired position, there is a momentarily increase of lift and a subsequent delay in response time before the final "spoiled" lift is attained. As the spoiler is deflected to its static position, there is a flow separation from the tip of the spoiler, with alternate shedding of vortices from the tip of the spoiler and the trailing edge of the lower surface of the wing section, leaving a wake region aft of the spoiler where the fluctuating pressures may pose a severe phenomenon known as buffet. For this reason, a thorough understanding of the aerodynamics of spoilers is essential to ensure that they are designed and used successfully.

The idea of using spoilers to provide lateral control is not a new one and can date back to as early as 1930s, when NACA conducted extensive tests to ascertain the effectiveness of spoilers, for example, see Weick et al (1934) and Eastman & Sherman (1937). However, the nature of these tests was made towards the evaluation of the performance of spoilers, rather than towards the understanding of their aerodynamics. And it was due to the non-linear characteristics found in these tests that spoilers were not used for another decade or two.

With the advent of Active Control Technology (ACT), however, there is a revival of interest in the aerodynamics of spoilers. Moreover, in the recent search for means to control the super-maneuvrability of combat aircraft, there is an interest in the use of rapidly acting spoilers to provide dynamic lift.

Unfortunately, there exists very little work done in the area of the dynamic effect exhibited by spoilers, as noted in the review by Siddalingappa & Hancock (1980a).

1.2. Spoilers on a flat plate

1.2.1. Normal Spoilers on a Flat Plate

Naturally, a good beginning towards the understanding of the aerodynamics of spoilers is through the study of separated flow past a two-dimensional spoiler on a flat plate. This was the problem that Barnes (1966) investigated with both pressure measurements and flow visualization. When the flat plate length is very large, the only length scale that will affect the flow is associated with the boundary layer. Therefore, two important dimensionless parameters are identified, namely the non-dimensional boundary layer displacement thickness on the plate at the spoiler position in the absence of the spoiler, and the Reynolds Number defined with the boundary layer displacement thickness as the length scale. In Fig 1.1 where the pressure distribution on a flat plate due to the presence of a spoiler is shown, it can be seen that there is a compression ahead of the spoiler and C_p , the conventional pressure coefficient, reaches a maximum value on the spoiler; the flow separates from the tip of the spoiler, resulting in a drastic drop in C_p ; a nearly uniform base suction pressure immediately aft of the spoiler, indicating the presence of a recirculating flow region; a subsequent pressure recovery following the reattachment of the separated boundary layer. In fact, in Fig 1.2, which is also adopted from Barnes (1966), is a sketch of the flow pattern inferred from the flow visualization, which confirms that the flow separates ahead of the spoiler as a result of the compression, and the flow reattaches onto the spoiler surface, resulting in a recirculating bubble. A second recirculating bubble is formed when the flow separates again from the tip of the spoiler and reattaches onto the flat plate subsequently.

The process of flow separation and reattachment aft of the spoiler is not unlike those features associated with flow past a rearward facing step. Tani et al (1961) demonstrated that the process of reattachment for

the rearward facing step is governed largely by the entrainment of fluid from the separated region, and there is a balance between the pressure recovery and the stirring work done by the shear stress. It is argued by Tani et al (1961) that the cavity flow is maintained by the turbulent shear stress, which is set up in the mixing region and is independent of the step height and the approaching shear layer. Therefore, the pressure coefficient C_p is only a function of x/h , where x is the distance aft of the spoiler or the rearward facing step, and is independent of the displacement thickness of the boundary layer and this can clearly be seen in Fig 1.1.

1.2.2. Inclined Spoilers on a Flat Plate

The foregoing discussion is limited to the case where a spoiler is positioned normal to a flat plate, and no further work was carried out to investigate spoilers deflected at other angles on a flat plate until that due to Siddalingappa & Hancock (1980b). It can be seen in Fig 1.3 that the main features of flow past an inclined spoiler with no gap are similar to that of a normal spoiler except that there is no flow separation ahead of the inclined spoiler due to the "not so severe" compression set up by the spoiler at a not so severe deflection. The associated pressure distribution can be seen in Fig 1.4, which reveals a compression ahead of the spoiler, a flow separation from the tip of the spoiler, a constant base pressure region due to the separation bubble aft of the spoiler, and a pressure recovery to its static value after the reattachment of the separated boundary layer. Basically, the entrainment and the reattachment process is similar to that of a normal spoiler.

1.3. Statically Deflected Spoilers

1.3.1. Flow Patterns due to Statically Deflected Spoilers

Flow past a spoiler on an aerofoil exhibits much resemblance to that of a spoiler on a flat plate. However, in the case of a spoiler on an aerofoil, the separated flow from the tip of the spoiler may or may not reattach onto the aerofoil depending on the normal projection height of

the spoiler with respect to the flow direction, and the length of the aerofoil aft of the spoiler. And it is usually the case that the spoiler will be located and deflected such that the separated boundary layer does not reattach onto the aerofoil, but with alternate shedding of vortices from the tip of the spoiler and the trailing edge of the lower surface of the aerofoil.

Basically, there are three types of separation as hypothesized by Mack et al (1979) as shown in Fig 1.5. When the spoiler deflection is small, there is a separation bubble aft of the spoiler. As the deflection increases, instead of reattachment of the separated boundary layer onto the aerofoil, the separated boundary layers from the tip of the spoiler and the trailing edge interact with each other, giving rise to alternate shedding of vortices with a constant near wake region aft of the spoiler. At high, post stall angle of attack, the spoiler is completely immersed in the low velocity wake caused by the flow separation well ahead of the spoiler on the aerofoil, and thus would have little effect on the aerodynamic performance of the post stalled aerofoil.

However, these are not the complete pictures associated with a deflected spoiler. Lee & Bodapati (1988) reported at least four flow patterns as inferred from their base pressure measurements. As shown in Fig 1.6, the flow patterns are similar to those hypothesized by Mack et al (1979) except that there is a hinge bubble ahead of the spoiler at positive angles of attack. The hinge bubble is due to flow separation as a result of the compression set up by the presence of the spoiler. When the angle of attack decreases, the adverse pressure gradient ahead of the spoiler also decreases to the extent that it is no longer strong enough to cause the flow ahead of the spoiler to separate, and hence the hinge bubble disappears. As the angle of attack is decreased even further, the flow will start separating upstream of the trailing edge on the lower surface.

1.3.2. Spoiler Effectiveness

Typical results of spoiler effectiveness for a two-dimensional spoiler are given in Fig 1.7. Wentz et al (1981) investigated the effectiveness of spoilers on a two-dimensional Boeing research aerofoil

with a thickness of 11.3% chord. The spoilers used in the tests have a height equal to 15.7% chord with the hinge line located at 73.3% chord on the upper surface of the aerofoil. The spoiler effectiveness, which is expressed in terms of the incremental lift obtained with the spoiler compared to that without the spoiler, is plotted in Fig 1.7 as a variation of differential lift with the normal projection height of the spoiler. It can be seen that the differential lift increases as the deflection of the spoiler increases, where the deflection of the spoiler is expressed in terms of the normal projection height. The response is almost linear at low and moderate angles of attack but diminishes greatly at angles of attack near stall and is associated with a high degree of non-linearity. The same spoiler-aerofoil configuration was also investigated by McLachlan et al (1982), with different Reynolds Numbers. Instead of plotting against the normal projection height of the spoiler, the incremental lift is plotted directly against the deflection angle as in Fig 1.8. A slightly higher non-linearity is observed in this figure as compared to that of Mack et al (1981). This could be due to the different parameters used in the abscissa, as the normalized projection height is related to the deflection angle through a trigonometric term. This comparison might suggest that the normal projection height could be a more suitable parameter used to establish a linear correlation with the incremental lift. As a further example, Fig 1.9 shows the results due to Jones et al (1979). The results here were obtained on an aerofoil with a Fowler-flap and a spoiler. It can be seen that the response is almost linear for the case without flap deployment. However, the response becomes highly non-linear with the deployment of the flap and this demonstrated the complexity when designing the use of a spoiler together with other control surfaces.

The above discussion is limited to two-dimensional aerofoil with two-dimensional spoilers only and there appears little fundamental work done on the aerodynamic characteristics about finite spoilers. There exists, however, a considerable amount of data on the performance of finite spoilers. For instance, plotted in Fig 1.10 is the spoiler effectiveness of a finite spoiler on a two-dimensional aerofoil due to Wentz et al (1981). The figure shows that the incremental lift is almost a linear function of the span of the spoiler.

Jones et al (1979) also tested the case for a finite spoiler on a two-dimensional aerofoil as shown in Fig 1.11. It can be seen that, with an optimal selection of the spoiler span, a 20% increase in the spoiler effectiveness can be obtained. Jones et al (1979) investigated further into the effect of finite spoilers on a finite rectangular wing. This time the spoiler effectiveness is expressed in terms of the increase in the rolling moment, as shown in Fig 1.12, where high non-linearity prevails.

1.3.3. Pressure Distributions on Aerofoil with Spoilers

The pressure distribution on an aerofoil with a spoiler is not unlike that of a normal spoiler on a flat plate. However, the separated boundary layer from the tip of the spoiler does not reattach onto the aerofoil, and alternate vortex shedding takes place from the tip of the spoiler and the trailing edge of the lower surface of the aerofoil. This gives rise to a region of nearly constant pressure aft of the spoiler, where this constant pressure is known as the base pressure, as can be seen in Fig 1.13 due to Barnes (1966b) for the case of a normal spoiler on an aerofoil. The symbol E in the figure is the non-dimensional location of the spoiler. It can also be seen that the stagnation point is on the upper surface of the aerofoil and the flow is accelerated drastically to a maximum suction peak on the upper surface before a compression takes place ahead of the spoiler. It is then followed by a sudden drop in C_p due to flow separation at the spoiler tip, and a wake region of constant suction pressure aft of the spoiler. It can also be seen that the sudden drop in C_p becomes less drastic and the compression less severe as the spoiler moves further back towards the trailing edge.

McLachlan et al (1982) also measured the pressure distributions around an aerofoil with spoilers deflected at different angles with respect to the aerofoil chord, with different spoiler deployment angles. Shown in Fig 1.14 are the pressure distributions for the case of an aerofoil at two different angles of attack. Included in this figure are the pressure coefficients on the upper surface of the spoiler, which are plotted as three points immediately after the hinge line for each pressure distribution. Generally, when the spoiler deflection is large, the pressure distributions show the same features as discussed in the case for a normal spoiler.

However, as the deflection decreases, the compression ahead of the spoiler and the suction in the near wake also decreases accordingly.

A typical flow field around an aerofoil with a spoiler is shown in Fig 1.15. Bodapati & Karamcheti (1982) measured the main flow with a split film anemometer which is capable of measuring reverse flows. It can be seen from the figure that the reverse flow region ends in the wake some distance aft of the trailing edge. The flow field can be even more complicated when a flap is deployed together with a spoiler, as shown in Fig 1.16 for example, due to Mack et al (1979).

1.3.4. Time Lag and Adverse Lift

It is a well known fact that when a spoiler is rotating rapidly towards its intended position, the lift actually increases momentarily, and this is known as the adverse lift effect, as it is adverse to the final response. Subsequently, there will be some time before this transient effect settles down, resulting in a time lag. These phenomena are well documented even though little work has been done to understand the associated aerodynamics thoroughly. For example, Fig 1.17 due to Shortal (1961) shows the time lag in terms of chord lengths travelled and is plotted against the location of the spoiler on the aerofoil. The time lag increases as the location of the spoiler moves forward towards the leading edge of the aerofoil. A time history of the control motion of a spoiler together with its response is also given in Fig 1.18. The response due to an aileron actuated by the same control motion for the spoiler is also included in this figure for comparison. The dashed-line represents the control movement that was used to activate either the aileron or the spoiler. Each of the other two lines is the response of the wing in terms of rolling motion due to the deployment of either the aileron or the spoiler. Shortal (1961) noticed that the retractable spoiler causes the wing to roll initially in the opposite direction before rolling in the desired direction. This undesired motion, which is absent during the deployment of the aileron, is due to the effect of adverse lift mentioned earlier. It is also evident that the spoiler gives a much longer time lag than that due to the aileron, for the same amount of rolling motion.

The adverse lift effect was also observed by Hoerner (1975) in an experiment done in a water tunnel as shown in Fig 1.19, and the final lift here takes about the time required for the aerofoil to travel 12 chord lengths to settle down. The increase of lift is due to the formation of an initial vortex at the tip of the rapidly acting spoiler and this initial vortex causes a large suction pressure before it is convected downstream off the aerofoil. Further discussion on the aerodynamics of the transient effect associated with a rapidly acting spoiler will be expounded in a later section.

Hoerner (1975) also tested the effect of adverse lift and time lag with a rake type spoiler which is shaped like a rake. Hoerner (1975) found that both the adverse lift and the time lag are reduced considerably as illustrated in Fig 1.20. The idea of using non-solid spoilers, such as the rake type spoiler, is not a new one, as this had been tested extensively by Kramer et al (1951). Kramer et al (1951) evaluated the effect of time lag with various spoiler configurations, such as perforated spoilers, screen spoilers and rake spoilers. Fig 1.21 shows the measured time lags of various spoilers used at three lift coefficients. Clearly, it can be seen from the figure that the solid spoiler exhibits the highest time lag, whereas the time lag for an aileron is almost insignificant. Generally, when a spoiler is to be used, the performance can be increased by about 100% with a non-solid spoiler.

1.3.5. Unsteady Characteristics

As has already been explained in the previous section, there is a near wake formed aft of a spoiler, together with alternate shedding of vortices from the tip of the spoiler and the trailing edge of the lower surface of the aerofoil. The near wake is highly turbulent in nature, and a high level of pressure fluctuations is to be expected there. This unsteady phenomenon is not unlike that due to bluff bodies, where the near wake and alternate vortex shedding determine to a great extent the overall flow profile. Unfortunately, very little work has been done in this area, and there are only a few papers published on the unsteady flow characteristics. Ayoub et al (1982) recognized this shortcoming, and had performed experiments with pressure spectral analysis.

Fig 1.22a shows the aerofoil that was used by Ayoub et al (1982), together with the locations where the pressure transducers were mounted, for the case with a spoiler deflected at an angle of 60 degrees with respect to the aerofoil chord. Typical pressure spectra can be seen in Fig 1.22b, where strong spikes associated with the shedding frequency of 112.5 Hz are clearly identifiable. This shedding component can be seen at all locations, and its magnitude on the lower surface increases progressively towards the trailing edge, indicating the strong influence of vortex shedding near the trailing edge. On the upper surface of the aerofoil, the magnitude of the shedding component increases from locations 1 to 3, but remains more or less the same at locations 4, 5 and 6. A strong broad band signal is also present in each of the spectra for locations 4, 5 and 6, indicating turbulence associated with the hinge-bubble and the near wake. Ayoub et al (1982) also noticed the presence of the second harmonic of the shedding frequency, and these components are more clearly identified at the locations near the trailing edge where the vortices from the upper and lower sides of the wake have similar pressure signatures on the aerofoil surface.

McLachlan et al (1983) studied the unsteady flow field on the same spoiler-aerofoil configuration as that used by Ayoub et al (1982) more systematically. Fig 1.23 shows the surface pressure spectra, which exhibit the same features that have been discussed in the last paragraph. Clearly, there is a very high level of broad band fluctuations on the front surface of the spoiler due to the presence of the hinge bubble, and this is critical to the buffeting of the spoiler. Typical fluctuating pressure distributions expressed in terms of root mean square values are shown in Fig 1.24, and it confirms that the extent of pressure fluctuations is the strongest near the trailing edge. McLachlan et al (1983) also analyzed the degree of bluffness due to the deflected spoiler in terms of Strouhal number. The Strouhal number, S , is defined as fH/U_0 , where f is the vortex shedding frequency, H the normal projection height of the spoiler and U_0 the free stream velocity. The results are shown in Fig 1.25, together with the Strouhal numbers of some other bluff bodies. The Strouhal number analysis confirms the bluff body characteristics of the deflected spoiler.

1.4. Time-dependent Aerodynamics of Spoilers

1.4.1. Transient Aerodynamics

It has been explained in the previous sections that a main feature associated with spoilers is the effect of adverse lift and time lag. The effect engenders the use of spoilers unattractive for gust alleviation, but then it may be regarded as favourable with respect to increasing manoeuvre lift in combat aircraft. Therefore, the understanding of rapidly acting spoilers will not only improve conventional designs for gust alleviation but also shed some light into ways of increasing dynamic lift.

The adverse lift effect and the time lag effect have long been recognized and work has been done such as those due to Shortal (1961) and Hoerner(1975) etc. However extensive these evaluations are, little fundamental work has been done on the causes and aerodynamics of adverse lift and time lag. The first systematic study of these effects is that due to Siddalingappa & Hancock (1980c). Typical time histories due to a rapidly acting spoiler are shown in Fig 1.26. It is thought that there is an initial vortex formed immediately after the spoiler is actuated. The initial vortex, which gives rise to the strong suction peaks in the pressure traces, takes about the time required for the aerofoil to travel one chord length to form and another one chord length time to convect downstream past location 56, resulting in a time lag before the steady value is reached.

One convenient way to express the time scale in the study of time-dependent aerodynamics is to non-dimensionalise the time scale by the time taken for the aerofoil to travel one chord length, i.e. $T = tc/U_0$, where t is the time scale, c the chord length and U_0 the free stream velocity, and T shall be known as the time parameter. The time parameter used by Siddalingappa & Hancock (1980c) to actuate the spoiler from its initial position to its final position is 1 T . This is equivalent to saying that the spoiler will reach its final position after the aerofoil has travelled a distance of one chord length. Hence, it can be

seen from Fig 1.26 that the initial vortex takes about 1 T to form and the pressure traces take about another 4 to 5 T to settled down.

Also included in Fig 1.27 are the time histories for the case of a forward spoiler mounted at 16% chord. Obviously, the initial vortex takes longer time, about 15 T, to convect downstream past the trailing edge as there is now a much longer afterbody length. However, care must be taken to interpret these results quantitatively as the pressure measurement system used by Siddalingappa & Hancock (1980c) has a resonant frequency of 30 Hz due to the long tubings used, and hence time lags between the actual and the measured quantities could be introduced.

The formation of the initial vortex is confirmed by the investigation of Finaish et al (1985) through flow visualization. Instead of using a rapidly acting spoiler on an aerofoil, the flow is accelerated at a constant acceleration past a normal spoiler on a flat plate. A series of pictures was taken, which shows that the flow starts separating from the tip of the spoiler, giving rise to a growing vortex together with other secondary vortices. The growing vortex is subsequently convected and reattaches on the flat plate somewhere downstream. It is exactly due to the formation of this initial vortex, which generates a high suction, that the adverse lift occurs. A sketch of the two-dimensional vortices that may develop is shown in Fig 1.28, as adopted from Finaish et al (1985). Of course, in the case of a finite plate such as an aerofoil, with insufficient afterbody length for the initial vortex to reattach onto, the vortex may convect past the trailing edge and the flow field will then be dominated by alternate vortex shedding from the tip of the spoiler and the trailing edge of the lower surface of the aerofoil.

Other investigations for the case with a rapidly acting spoiler are those due to Mabey et al (1982), Consigny et al (1984), Costes et al (1986), Kalligas (1986) and Myers (1990), of which the work due to Mabey et al (1982) deserves elaboration. Mabey et al (1982) conducted the experiments with a finite spoiler on a two-dimensional aerofoil. The time parameters used in the experiments to actuate the spoiler from its initial position to its final position were 2.5 and 5 T. Typical transient

lift measurements for the extension and retraction of the spoiler are shown in Fig 1.29. Three new parameters are defined, t_o the time for the onset of lift change immediately after the spoiler is activated, t_a the time to achieve the adverse lift and t_f the time taken to achieve the final lift. The typical pattern prevails for the case of the extension of the spoiler, with the effect of adverse lift and time lag. However it can be seen that t_o on the upper surface is earlier than that on the lower surface, but t_f is somewhat later than that on the lower surface. To use the words of Mabey et al (1982), "This difference in times reflects the long time constant associated with the establishment of the thick viscous shear layer on the upper surface compared to the short time constant associated with the relatively minor change in the thickness of the thin attached boundary layer on the lower surface". In the case where the spoiler is retracted rapidly, the lift just decreases monotonically, however, a longer t_o resulted and hence a longer t_f . This long delay is probably due to the fact that the near wake aft of the spoiler tends to maintain its general shape and position, despite the rapid retraction of the spoiler. This is also the case observed by Costes et al (1986) as shown in Fig 1.30. In addition, Costes et al (1986) also noticed that when the activation rate of the spoiler gets slower, the adverse lift effect will get smaller as well. A time history of the pressure traces due to Costes et al (1986) is also given in Fig 1.31, which shows that the response is only non-linear aft of the spoiler where the initial vortex prevails.

Recently, a series of test programmes has been conducted in Bristol University to look into the adverse lift effect caused by the rapidly acting spoiler, and the main findings were reported in Kalligas (1986) and Myers (1990). The emphasis of their investigation was placed on how the various spoiler configurations and activation parameters affect the adverse lift effect. Kalligas (1986) conducted experiments mainly on an aerofoil with a rearward spoiler, and one of the important tests was the activation of the spoiler with a multiple-step ramp motion. Some typical examples due to Kalligas (1986) are given in Fig 1.32. In Fig 1.32a and Fig 1.32b, the adverse lift is large where a rapid ramp motion is applied, and the adverse lift is much smaller when a two step ramp motion is used instead, as shown in Fig 1.32c and Fig 1.32d. Also shown in Fig 1.32e and Fig 1.32f is the response due to applying a rapid ramp motion on the spoiler of an aerofoil at an angle of

attack just below static stall, where the adverse lift is much smaller than at zero lift. Therefore, it was postulated by Mabey (1988), who reviewed the results of Kalligas (1986), that the potential improvement in dynamic lift is much smaller when the motion is initiated at high angle of attack than at zero angle of attack. Clearly, this is one of the areas which deserves further investigation.

Myers (1990), on the other hand, investigated mainly on the effect of spoiler positions on the adverse lift effect, with particular emphasis placed on the adverse lift effect due to a forward spoiler located at $x/c = 0.3$ on the aerofoil. It was found from the investigation that when the forward spoiler is retracted rapidly, there is an adverse effect instead of the normal monotonical decline towards the steady state, as can be seen from Fig 1.33.

However, the results of Kalligas (1986) and Myers (1990) were made with eleven pressure tappings distributed unevenly on each side of the aerofoil surface. Therefore the integrated lift might not reflect the actual magnitude and variation of the adverse lift. As a result, it is not easy to relate the local flow field, in terms of the transient pressure distribution, to the evolution of the adverse lift effect.

Nevertheless, the major drawback of their investigation lies with the technique they used to measure pressures. Due to space constraints posed by the size of the aerofoil, dynamic pressures were measured outside the aerofoil, where the pressure tappings were connected to the pressure transducers through long tubings with length of 630 mm each. To overcome the effect of frequency distortion, they introduced damping into the path of each of the pressure tubings in the form of restrictor which has a very small diameter relative to the tubing diameter. The design of their pressure transmission path was done via a prediction method reported by Bergh & Tijdeman (1965). According to this prediction method, the resulting pressure transmission path designed by Kalligas (1986) resembles a low pass filter with a cut-off frequency of 100 Hz, and with a near linear phase shift. Kalligas (1986) therefore concluded solely from the calculation that, since the measured signals did not contain significant fluctuations above 70 Hz, the pressure

transmission path was fit to be used. However, since no experimental calibration was carried out, the validity of their measurements is therefore in some doubt.

1.4.2. Oscillating Spoilers

The investigation into oscillating spoilers is not new, some earlier works were done in this area to investigate the behaviour of separated flows, where oscillating spoilers were used to provide a controlled form of unsteady separation. Some examples are those works due to Lang (1975) and Francis et al (1979) where oscillating spoilers were used to provide a fixed separation point. In recent times, it is demonstrated that locally introduced unsteady vorticity due to controlled separation is capable of manipulating the separation region and its reattachment characteristics, for example see Koga et al (1984) and Nelson et al (1990).

Francis et al (1979) investigated the aerodynamics of an unsteady flow by using a transversely oscillating spoiler. This simply means that the spoiler is oscillating on a plane normal to the chord of the aerofoil. The unsteady characteristics are described in terms of the time-varying mean value of the surface pressure and vorticity field. Basically, the vorticity field measurements reveal a vortex like structure, and this vortical structure either grows and washes off, or elongates and retracts towards the aerofoil, depending on the choice of oscillating frequency. It was also observed that the dimensionless characteristic growth rate of this vortex is directly proportional to the square root of the dimensionless frequency.

Siddalingappa & Hancock (1980c) also carried out unsteady pressure measurement for a spoiler undergoing angular oscillation about its hinge line, at frequencies of 1, 5 and 18 Hz. It was found that at the low frequency of 1 Hz, the pressure traces indicate a quasi-steady behaviour as shown in Fig 1.34. As the frequency is increased to 5 Hz, it can be seen from Fig 1.35 that there are large suction peaks downstream of the spoiler at locations 52 and 56, associated with the formation of the spoiler vortex as the spoiler angle increases. When the

frequency is increased to 18 Hz as shown in Fig 1.36, the pressures at locations 52 and 56 appear to be out of phase. Siddalingappa & Hancock (1980c) also observed that, at 18 Hz, there is a breakdown of the compression peak at stations 52 and 56, and it is thought that this is due to the interaction of the compression formed by the formation of the spoiler vortex as the spoiler angle increases with the compression formed as the spoiler angle decreases.

Consigny et al (1984) tested the oscillating spoiler by oscillating the spoiler at a small amplitude about a mean spoiler deflection, and the effects of Mach number, mean spoiler deflection, reduced frequency and oscillation amplitude were tested. The effects of spoiler deflection and Mach number on unsteady lift are illustrated in Fig 1.37. Generally, the lift increases to a maximum value as the spoiler deflection increases, it then falls gradually with further increases in the spoiler deflection. It can also be seen that the value of the maximum lift as well as the associated deflection, greatly depends on the free stream Mach number. Shown in Fig 1.38 is the unsteady lift coefficient plotted against the spoiler mean deflection, also due to Consigny et al (1984). This time, the spoiler effectiveness is strongly dependent on the spoiler oscillation frequency. As the frequency increases, the spoiler effectiveness reduces greatly, and at low frequencies, the spoiler effectiveness follows closely to that of a quasi-steady approximation.

Nevertheless, these measurements are confined to a quantitative description, with no reference to the associated flow field developed. Certainly, there exists a need to study the flow field more completely via flow visualization. However, there has not been any work done on the flow visualization of an oscillating spoiler on an aerofoil except that of an oscillating spoiler on a flat plate by Koga et al (1984). In essence, there is a local flow separation region formed when the spoiler starts to rise from the floor, and the subsequent behaviour of this separation region will depend on the corresponding oscillatory spoiler frequency. At low frequency, this separation region extends for a large distance downstream and is eventually swept away as the retracting spoiler allows the boundary layer flow to re-establish itself on the flat plate surface, exhibiting quasi-steady characteristics, as have been observed

quantitatively by Siddalingappa & Hancock (1980c) and Consigny et al (1984) mentioned earlier. As the frequency increases, the oscillatory motion begins to produce an energetic vortical structure, and the spacing between each shed vortex is dependent on the rate of oscillatory spoiler motion and the convective velocity of vortices. Clearly, when an oscillatory spoiler test is to be carried out on an aerofoil, the relation between the shed vortex spacing and the remaining length of the aerofoil aft of the spoiler will play an important role in the overall aerodynamic performance, and this relationship is yet to be explored.

1.5. Numerical Simulation of Flow past an Aerofoil with a Spoiler

Although spoilers have been tested extensively over the years, there has been little theoretical information about their performance characteristics. With the advent of computers, attempts have been made to predict the spoiler performance numerically. Earlier investigations have aimed to predict the main flow characteristic associated with spoilers. One of the recent investigations, which used a potential flow model, is due to Parkinson & Yeung (1987).

Parkinson & Yeung (1987) combined a conformal mapping, which maps the aerofoil and the spoiler into a circle, and a wake source method to simulate separated flow. However, this method requires the base pressure coefficient as an empirical input. In addition, any hinge bubble, which arises due to the adverse pressure gradient set up by the presence of the spoiler, was also not included in the model.

In a separate development to remove the base pressure coefficient as an empirical input, Tou & Hancock (1985) used a different method based on a vortex sheet representation of the wake. In this method, the surface of the aerofoil and the spoiler was modelled with panel elements of piecewise continuous vorticity. The separating thin shear layers from the spoiler tip and the trailing edge of the aerofoil were modelled similarly with panel elements of constant vorticity. However, there are now two empirical factors need to be specified, the length of the separated vortex sheets and the strengths of the two point vortices used to close the wake.

The above two methods not only require the input of empirical constants but are also limited to the modelling of stationary spoilers only. In order to overcome these limitations, a different approach was adopted by Kalkanis (1988), who simulates the separated flow generated by the spoiler using the discrete vortex method. Using a multi-stage analytical transformation, a Joukowski aerofoil with a spoiler was transformed into a circle. Discrete vortices were then made to shed continuously in the circular plane to simulate the shear layers separating at the tip of the spoiler and the trailing edge of the aerofoil. The velocity field and the vortex positions were calculated in the circular plane as a sum of attached potential flow about the body and the interaction between the moving point vortices. The lift force induced on the body was also calculated using the momentum integration method. The mean flow characteristics calculated from this approach, without any empirical input, agree well with the experimental results.

To illustrate the transient effect calculated by Kalkanis (1988), a typical numerical flow visualization is shown in Fig 1.39, where the separated shear layers from the tip of the spoiler and the trailing edge of the aerofoil are represented by two rows of discrete vortices. The test case shown here is for the spoiler being deployed from an initial spoiler angle of 10° to a final spoiler angle of 50° in a non-dimensional deployment time of $T = 0.18$. It is clear from this figure that as the spoiler moves from its initial position, the shear layer separating from the tip of the spoiler begins to roll up into a clockwise rotating vortex, which is also known as the initial vortex. On the other hand, the shear layer separating from the trailing edge of the aerofoil extends steadily downstream. As the spoiler continues to move, the initial vortex grows in size and moves downstream. When this vortex approaches the trailing edge, by virtue of its suction, it induces the shear layer separating at the trailing edge of the aerofoil to roll up into a secondary vortex. While this secondary vortex continues to roll up, it moves into the upper surface of the aerofoil near the trailing edge, causing a momentarily reverse flow. The resulting variation of lift is also shown in Fig 1.40 which, with the occurrence of the adverse lift, concurs well with the experimental trend.

1.6. Present Investigation

The present investigation looks into the aerodynamics of 2-D fixed and rapidly acting spoilers. It can be gathered from the previous sections that although extensive works have been done on the aerodynamics associated with spoilers, there are some areas which need further study. In the case of a fixed spoiler, the present study places the emphasis on the unsteady flow field generated by the spoiler. In addition, the study of aerodynamic loading on the spoiler is also made. In the case of a rapidly acting spoiler, the present emphasis is placed on the effect of the initial vortex, and to relate how this vortex affects the transient aerodynamics.

This thesis, naturally, sums up the experimental findings from the investigation. The thesis is organized into six main chapters. In each of the chapters from chapter 3 to chapter 5, the experimental results and their discussions are presented together.

Chapter 2 describes in detail the various components of the experimental set-up, the models, the experimental procedures, the instrumentation system and the various measurement techniques. It also includes a description of the data reduction techniques and the analysis.

Chapter 3 presents and discusses the aerodynamics of an aerofoil with a stationary deployed spoiler at different operating conditions. Measurements of the steady and unsteady pressure distributions, the steady and unsteady normal forces, are presented and analysed. Measurements of the predominant frequency generated by a stationary spoiler are also presented in the form of Strouhal number.

Chapter 4 presents measurements of the transient aerodynamics of the aerofoil with a rapidly moving spoiler, for different experimental parameters. The adverse lift effect and its causes are studied with the aid of the measured transient forces and surface pressures.

Chapter 5 discusses the transient flow field obtained in the wake of the aerofoil with the rapidly moving spoiler using hot-wire anemometry. The distributions of vorticity are also calculated in order to study the vortices associated with the deployment of the spoiler. Attempts are also made to relate the velocity measurements with the pressure measurements as presented in Chapter 4 to provide a clear picture of the transient flow field.

Chapter 6 presents the overall conclusions of the investigation and makes recommendations for future work.

2. EXPERIMENTAL ARRANGEMENT AND EQUIPMENT DETAILS

2.1. Wind Tunnels

The experiments were conducted in two phases. In phase one, measurement was carried out on an aerofoil with a stationary spoiler, whereas in phase two, measurement was carried out with a rapidly acting spoiler. Three different wind tunnels were used throughout the investigation.

In phase one, the experiments were conducted in two wind tunnels in the Department of Aeronautics, Imperial College of Science, Technology & Medicine. The reason for using two different wind tunnels is due to their availability. The first wind tunnel has a working section of 1.37 m by 1.22 m and is capable of providing a maximum wind speed of 50 m/s. It is of closed circuit design and has a turbulence level of about 0.1%. The second wind tunnel is of dimensions 3 m by 1.52 m and is capable of providing a maximum wind speed of about 35 m/s. It is also of closed circuit design and has a turbulence level of about 0.1%.

In phase two, the span of the aerofoil used is 500 mm due to design consideration, and so a wind tunnel with a smaller width was used. The wind tunnel used has a working section of 0.6 m by 1 m, and a maximum wind speed of 50 m/s. It is also of closed circuit design and has a turbulence level of 0.1%.

2.2. Models and their Installation

Two different basic models were constructed as the experiments were conducted in two phases. Nevertheless, the basic aerofoil profiles used were the same.

2.2.1. Static Model and its Installation

The basic model essentially consists of two distinct parts: the plain aerofoil and the spoilers. The aerofoil selected is of a NACA 0018 section with a chord length of 250 mm. It is made of wood and

has a span of 1.32 m. The aerofoil is wholly solid except at its mid span, where a hollow section was improvised so as to accommodate pressure transducers and pressure averagers, which will be discussed later. A total of 32 pressure tappings of internal diameter 1.6 mm were mounted equispaced from $x/c = 1/32$ to $31/32$ in steps of $2/32$ on both the upper and lower surface of the aerofoil around the mid span section to provide both static and dynamic pressure measurements. The profile of the aerofoil section together with the locations of the pressure tappings are given in Fig 2.1a. Also shown in Fig 2.1b is the pressure distribution on the plain aerofoil at $\alpha = 0^\circ$, and it can clearly be seen that the aerofoil flow is reasonably symmetrical.

The spoilers are made of Dural and all have the same thickness of 3 mm. Three heights of spoilers were used, these are 37.5 mm, 31.25 mm and 25 mm, corresponding to 15%, 12.5% and 10% of the aerofoil chord respectively. At any one time, only one spoiler would be mounted onto the aerofoil in an orientation normal to the aerofoil chord. The mounting was done via some brackets, which were fabricated to be mounted flushed onto the aerofoil at any desired positions.

Altogether three different sets of brackets were made to mount the spoilers in three different chordwise positions on the aerofoil surface. The chordwise positions chosen in the experiments were at 75 mm, 125 mm and 175 mm, corresponding to 30%, 50% and 70% chord length respectively. A typical spoiler-aerofoil configuration is given in Fig 2.2, where the spoiler with a height equivalent to 15% chord length is mounted at a location equivalent to 70% chord length.

Later, when the aerodynamic loading on the spoiler was studied separately, a wooden spoiler of $h/c = 0.15$ and thickness of 9 mm was fabricated. A row of pressure tappings was mounted on the mid span of the spoiler, with 4 pressure tappings on the front surface and 2 on the rear surface. The pressure tappings on the front surface are located at positions $x/h = 1/8, 3/8, 5/8$ and $7/8$, whereas those on the rear surface are located at $x/h = 1/4$ and $3/4$.

Two identical end-plates with dimensions shown in Fig 2.3 were made and were fastened onto each end of the aerofoil during the experiments. This was done to minimise any three dimensional effects that may have been introduced by the ends of the aerofoil and spoiler.

When used in the 1.37 m by 1.22 m tunnel, the model was mounted in a horizontal position, leaving a gap of about 1" from each side of the tunnel wall to the end-plate. The blockage ratio, with the installation of the spoiler, was less than 5% in this tunnel. In the 3 m by 1.52 m tunnel, however, the model was mounted vertically, giving a blockage ratio of less than 2%.

2.2.2. Dynamic Model and its Installation

The dynamic model consists of four parts: the aerofoil, the end-plates, the spoiler and a spoiler shaft. The section and construction of the aerofoil are the same as that of the static model except that it now has a span of 500 mm. A semi-circular groove was cut across the aerofoil upper surface at location $x = 0.7c$, with a nominal diameter of 16 mm, so that the spoiler shaft could be accommodated.

As in the previous case, two end-plates were fitted onto each end of the aerofoil. On each of the end-plates at the position where the groove of the aerofoil meets the end-plate, a hole was bored and a housing with roller ball bearing was also installed. This was done to enable the spoiler shaft to pass through the end-plates, sit inside the groove across the span of the aerofoil and be supported by the ball bearings. The aerofoil mounted with the spoiler and the end plates is shown in Fig 2.4.

The spoiler shaft has a length of 1 m, and it also has an irregular cross section. Once the spoiler shaft was mounted, part of it runs across the surface of the aerofoil and the two end parts of it protrude from the ball bearing housings away from the end-plates. Basically the spoiler shaft has a circular section of diameter 15 mm except at the part which runs across the aerofoil surface, where it has a semi-circular section with

a nominal diameter of 15 mm. The diameter of the spoiler shaft was such that the stiffness of it was sufficient to prevent any torsional distortion at the quickest spoiler activation time. The shaft was fitted tightly within the ball bearings and is supported by them. Of the part of the spoiler shaft which runs across the aerofoil surface, it sits inside the groove but a clearance is maintained between them so that the spoiler shaft can be freely rotated without any frictional contact with the aerofoil surface. Of the two ends of the shaft, during the experiments, one end was connected to the shaft of an air cylinder through a linkage system, and the other end was connected to a rotary potentiometer.

The spoiler has a height of 37.5 mm or $0.15c$, a span of 500 mm, and a thickness of 3 mm. It was mounted onto the part of the spoiler shaft with semi-circular section. Once mounted, when the spoiler shaft is rotated, the spoiler follows the rotation of the shaft, and the spoiler can thus turn freely from its closed to its open position. At its closed position, the spoiler rests flatly on the aerofoil surface. At its open position, the spoiler inclines at an angle with the aerofoil chord, and the final spoiler angular position is determined by the linkage system. During the experiments, the final spoiler angle was either 30 or 90 degrees, with respect to the aerofoil chord. When the spoiler was fully extended at 90 degrees, the blockage ratio in the 0.6 m by 1 m tunnel was less than 5%.

2.3. Pneumatic Activation System

During the second phase of the investigation, the spoiler was required to turn rapidly from its closed to its open positions, so a spoiler activation system was designed. At the inception of the design process, various activation systems had been considered. The activation system was required to provide a high starting torque, so that the spoiler could complete its deployment in about 10 ms. Four systems were thus identified, namely, a servo-motor system, a hydraulic system, a mechanical-spring system and a pneumatic system. Finally, it was decided that the pneumatic system would be a cost-effective means.

The basic pneumatic activation system consists of five parts: the pneumatic cylinder, the solenoid valve, the flow control valve, the valve

controller and the air supply. Some of the basic components and the installation of the system can be seen from Fig 2.5. The air cylinder has a bore of 25 mm and a stroke length of 50 mm. Its air inlet and outlet were connected to the solenoid valve and a flow control valve. The solenoid valve is a four way-two direction valve, and can be activated manually through a valve controller. By manually switching the valve controller, it not only activates the air cylinder, but also sends a signal to trigger the data acquisition system, as will be discussed later. The air supply was maintained at 10 bars throughout the experiments. The deployment time of the spoiler, or rather the deployment time of the air cylinder, was then regulated through the flow control valve, in a range from 11 to 110 ms.

The interface between the pneumatic activation system and the spoiler shaft is through a linkage system. The linkage system consists of two metal bars which are pivoted freely together. One end of the spoiler shaft was fitted with the linkage system, and the linkage system was in turns pivoted with the shaft of the air cylinder. When the air cylinder is fully extended, the spoiler is at its open position, and the spoiler is at its closed position when the spoiler is fully retracted. During the experiments, the two final angular positions of the spoiler, i.e. 30 and 90 degrees, were obtained by changing the lengths of the metal bars of the linkage system

2.4. Pressure Instrumentation and Measurement

Two types of pressure measurement were carried out: static and unsteady pressure measurement. In both types of measurement, pressure transducers manufactured by Druck were used. The model chosen is Type PDCR 42 and has an operating pressure range of 75 mbar. The output from the pressure transducer is in terms of differential electrical voltage with a range of 15 mV. Therefore, differential amplifiers were also used to provide a gain of 500. The pressure transducers were calibrated before each series of tests and the calibration constants were found to remain practically unchanged.

When used in measuring the static pressures only, a scanivalve was used. The scanivalve, which was controlled through a computer, has provisions to connect each of the 32 pressure tappings in turn to one pressure transducer. In

this way, all 32 static pressures could be obtained with the use of only one pressure transducer.

In measurement involving fluctuating quantities, either the pressure tapings were connected directly to the pressure transducers, or the pressure tapings were connected via pressure averagers to the pressure transducers to provide spatially averaged pressure values. Each of these methods will be discussed in detail in the following sections.

In the case where unsteady pressure distributions were required, the pressure tapings were connected to the pressure transducers directly. Since there were just four pressure transducers available, four pressure tapings could only be measured at any one time. In order to measure all 32 pressure tapings, experiments were repeated at the same operating conditions while the pressure tapings were measured four by four in turn. Therefore, a total of 8 runs was required to obtain an unsteady pressure distribution around the aerofoil.

It is a well known fact that if a pressure transducer is not mounted flush onto the point of interest, and is instead connected via tubing, there arises the problem of frequency response. In order to eliminate any distortion due to frequency response, the length of the connecting tubing must be maintained as short as possible, with a large internal diameter. Preliminary tests were conducted to ascertain the distortion due to frequency response, and to evaluate the permissible length of the tubings to be used. It was found that, if the distance between the diaphragm of the pressure transducer and the pressure tapping, connected via tubing with 1.6 mm diameter, is maintained below 100 mm, there will be no appreciable distortion in the range of frequency of interest. For instance, with a tubing length of 100 mm, the gain practically remains flat up to about 200 Hz, whereas the frequency of interest in the experiments was in the range of less than 200 Hz. Therefore, tubings of less than 100 mm were used in the measurement of unsteady pressure distribution.

In order to obtain the fluctuating forces due to pressure fluctuations, it is ideal to measure all 32 pressure tapings simultaneously where the fluctuating forces can then be obtained through numerical integration. However, as only four pressure transducers were available, a pneumatically averaging technique was used in the experiments. This technique has been investigated by Surry &

Stathopoulos (1977) and Holmes & Lewis (1987), by whom favourable results were reported. A sketch of one of the pressure averagers used in the present experiments is given in Fig 2.6. Basically it consists of 8 inputs, and these 8 inputs are all interconnected to one output. In this way, the output from the pressure averager will be equal to the arithmetic mean of the 8 inputs.

To obtain the fluctuating force, the pressureappings around the aerofoil were grouped into 4 groups of 8appings each. Each group of pressureappings was connected in turn to a pressure averager, and four spatial mean pressures were then obtained. The four spatial mean pressures are known as P_A , P_B , P_C and P_D , which are the averaged pressures of pressureappings no. 25 to 32, 17 to 24, 9 to 16 and 1 to 8 respectively, as shown in Fig 2.7. In all groups, the distance between each pressure tapping and the diaphragm of the pressure transducer was maintained at 100 mm. Again, preliminary tests were conducted to determine its frequency response and it was found that the response practically remains undistorted up to about 200 Hz, which is well beyond the range of frequency of interest. With this pressure averaging technique, four averaged pressures can be obtained simultaneously, as will be explained in a latter section. Subsequently, the normal force per unit span can be obtained by evaluating the mathematical expression of

$$\left[\frac{P_A + P_B}{2} - \frac{P_C + P_D}{2} \right] \cdot c$$

To ascertain the reliability of the pressure averaging technique, a comparison between the values of C_N , which is the normal force coefficient, obtained through the pressure averaging technique and individual pressure measurement is made in Fig 2.8. Shown in Fig 2.8 is the variation of C_N obtained for a spoiler deployed to the final angle of 30 degrees in 10 ms, with $\alpha = 0^\circ$. The results of C_N obtained through individual pressure measurement and the pressure averaging technique are shown in Fig 2.8a and Fig 2.8b respectively. Clearly, the two variations agree well both in their magnitudes and phases.

2.5. Hot-Wire Anemometry and its Calibration

Hot-wire anemometry was employed in the experiments to measure velocity quantities in the wake. The wake generated by the rapidly acting spoiler on an aerofoil was two-dimensional and had components in both x and y directions. Therefore, cross wire probes were used in the experiments. During the experiment, only one hot-wire probe was used at any one time. It was connected to an anemometer with an overheat ratio of 1.72, to provide the desired output in terms of voltage. The hot-wire probes used in the investigation are DANTECH 55P51, X-array gold-plated tungsten wire probes. Each of the wires is made of platinum-plated tungsten wire, with a diameter of 5 μm .

Before each experiment, calibration was carried out to determine the empirical constants in the calibration equation. The calibration equation used is as follows:

$$E^2 = A + B U_e^n$$

where E is the output voltage, U_e is the effective velocity, with A, B and n as the empirical constants.

For a single inclined hot-wire probe with an effective angle of α_e , the effective velocity will be:

$$U_e = u \cos \alpha_e - v \sin \alpha_e$$

where u and v are the velocity components in the x and y directions respectively. Hence,

$$E^2 = A + B (u \cos \alpha_e - v \sin \alpha_e)^n \quad 2.1$$

To obtain the empirical constants, the hot-wire probe is first calibrated with $v = 0$ and with the probe axis aligned in the direction of u. By varying u, constants of A, n and $B \cos \alpha_e$ can then be determined. The next step in the

calibration procedure is to determine the effective angle, α_e . This is done by yaw-calibration, which involves changing the yaw angle. If the change in the yaw angle is denoted by φ , then the calibration equation (2.1) can be re-written as:

$$E^2 = A + B (u \cos (\alpha_e + \varphi) - v \sin (\alpha_e + \varphi))^n$$

To determine the effective angle, v is again set to zero, and u is set to a constant value of U_o . Then, the calibration equation can be re-arranged into:

$$\left(\frac{E^2 - A}{E_o^2 - A} \right)^{1/n} - \cos \varphi = - \tan \alpha_e \cdot \sin \varphi \quad 2.2$$

where E_o is the voltage at which $u = U_o$ and $\varphi = 0$.

Thus by changing the yaw angles in small steps, the effective angle can be determined through equation (2.2). The hot-wire probes used in the experiments were all calibrated using the above procedure. Each of the hot-wire probes consists of two wires, namely wire 1 and wire 2. Wire 1 inclines at an effective angle of α_{e1} and wire 2 at α_{e2} . Thus for each hot-wire probe, calibration procedures were carried out separately for wire 1 and 2. Once the hot-wire probes were all calibrated, the velocity components could then be re-arranged in terms of the calibration constants as follows:

$$\frac{u}{U_o} = \frac{K1 \tan \alpha_{e2} + K2 \tan \alpha_{e1}}{\tan \alpha_{e1} + \tan \alpha_{e2}} \quad 2.3a$$

$$\frac{v}{U_o} = \frac{K1 + K2}{\tan \alpha_{e1} + \tan \alpha_{e2}} \quad 2.3b$$

where,

$$K1 = \left(\frac{E_1^2 - A_1}{E_{o1}^2 - A_1} \right)^{1/n_1} \text{ and } K2 = \left(\frac{E_2^2 - A_2}{E_{o2}^2 - A_2} \right)^{1/n_2}$$

In the above expressions, the effective angles for the wires of the probe are denoted by α_{e1} and α_{e2} . The calibration constants with subscript 1 and subscript 2 were obtained for wires with α_{e1} and α_{e2} respectively. E_{o1} and E_{o2} were the voltage values obtained from wire 1 and wire 2 for $u = U_0$ and $\varphi = 0$.

2.6. Experimental Procedure, Data Acquisition and Reduction

As unsteady quantities were required in the experiments, there was a need to collect data over a period of time. The data which was sampled with a pre-selected frequency, can then be analyzed either in the time or frequency domain, as the case may be.

The basic blocks of the data acquisition system consist of several parts : the computer, the interface bus, the analog/digital converters, the storage media, the controlling softwares and the signal conditioners. Each of these items will be discussed in turn in the following paragraphs.

The computer, which is the heart of the system, is a Meta-Dyne AT compatible. It has a CPU of Intel 80286, equipped with a maths co-processor 80287. The computer acted as an interactive interface between human operator and the system hardware.

To provide communication between the computer and its peripheral devices, an IEEE 488 interface bus was used. The IEEE 488 interface card used in the experiments was sourced from Brain Boxes, and was installed into the computer in the form of a plug-in interface board. A device driver to use the interface board was also provided together with the interface board.

Through the IEEE 488 interface bus, the computer can communicate with the data acquisition system through binary instructions. The data acquisition system consists of the time base unit TB, and five analog/digital converters CH12B, made by MicroLink. The time base unit is to provide the desired sampling frequency that the analog/digital converters will operate. Five CH12B were available to capture 5 signals simultaneously. Each CH12B can be set into the desired input voltage range, and it will start capturing the input signal according to the sampling frequency provided by the time base, either

through software triggering from the computer or hardware triggering from an external signal input connected to the TB unit. The CH12B unit is of 12 bit resolution, which means that it can discretize the selected input voltage range into 4096 intervals. It is also capable of capturing up to 4096 data values for each run. The captured data will stay resident in the buffer RAM (Random Access Memory) of each CH12B, and the data can then be transferred to the computer upon software instruction.

As has been discussed earlier, the output from the pressure transducers is in terms of differential electrical voltage in the range of 15 mV. For this reason, it is necessary to pre-condition the output voltages from the transducers before being presented to the data acquisition units. The output voltage from each of the pressure transducers was first connected to a differential amplifier, which provides a gain value of 500. The amplified signal was then presented to a low pass filter which was set with a cut off frequency equal to the Nyquist frequency.

To control the operations of data acquisition and perform data reduction, several pieces of software were written, using programming language compilers of Microsoft Fortran Version 4.0 and Microsoft QuickBASIC Version 4.5. Programs were written to perform data acquisition, evaluation of statistical quantities, calculation of fast fourier transform, evaluation of power spectral density, generation of graphical output of the spectra on monitor, generation of hardcopy of the spectra through HP plotter etc. All of these programs were developed on an ad hoc basis, and were optimised for the specific need in the present application.

2.6.1. Experimental Procedure for Phase One

In phase one, only pressure measurement was carried out on the aerofoil with a stationary spoiler, either using the pressure averaging or individual pressure measurement techniques. Whichever the case, four pressure transducers were used. They were either connected to selected pressure tappings directly or connected to the pressure averagers. The electrical output signals from the pressure transducers were connected to the signal conditioners for amplification and filtering, before they were connected to four CH12B units. The gains of the amplifiers were set at

500 and the cut-off frequencies of the low-pass filters were set at 500 Hz.

Upon command from the computer through the IEEE interface, the data acquisition system was activated with a sampling rate of 1000 Hz. When data acquisition was completed, the samples from each of the CH12B units were transferred into the memory of the computer before storing onto floppy discs. The raw data could then be processed into the desired statistical quantities for analysis at a later time.

In order to analyse the unsteady characteristics, power spectra were calculated using a fast fourier transform (FFT) method. The program that performs the fast fourier transform uses an algorithm adopted from Newland (1975). It was found that by performing FFT over one sample of 4096 data values, the result contains a lot of noise interference. For this reason, it was decided to perform ensemble average over a number of spectral samples. This technique has been discussed in Bendat & Piersol (1971) as Ensemble Smoothing procedure. The spectra with ensemble smoothing of 4, 8 and 16 samples, together with the spectrum with no smoothing, are shown in Fig 2.9. Clearly, the noise components in the spectra were smoothed out with ensemble smoothing, and it can be seen that there is no or little improvement for smoothing of more than 8 samples. Therefore, all spectra that will be discussed later were obtained with ensemble smoothing of 8 samples.

Also noted is that the experiments were all conducted with the blockage ratio of less than 5%. If the drags and the base pressures acting on the aerofoil are measured, it is possible to correct the blockage effect for the results obtained with the stationary spoiler using a correction method proposed by Maskell (1963). However, as no correction method is available for the transient results obtained for the rapidly acting spoiler, it was decided not to carry out any blockage correction in order to facilitate the comparison of transient and steady state results.

Finally, it should also be noted that the measurements of C_N and C_L , which are the normal force and lift coefficients respectively, are the forces acting on the aerofoil only and do not include any forces acting on

the spoiler. This is because the usual spoiler is only a flat plate with a small thickness which renders the installation of pressure tapings or pressure transducers impossible. This limitation has also been encountered by all other researchers studying the aerodynamics of spoilers.

2.6.2. Experimental Procedure for Phase Two

In phase two, measurement of both pressure on the aerofoil and velocity in the wake was carried out. The procedure was more complicated than in phase one as the activation of the spoiler was involved.

For the measurement of pressure on the aerofoil, four CH12B units were connected to the electrical signals from the pressure transducers, preconditioned by the amplifiers and filters. A fifth CH12B unit was connected to the rotary potentiometer, which was fixed to one end of the shaft of the spoiler to monitor the movement of the spoiler. To synchronise the activation of the spoiler with the commencement of data acquisition, hardware triggering was required. For this purpose, a signal from the solenoid valve controller was used as an external triggering signal, and was connected to the TB unit.

The activation of the spoiler was done by manual switching the solenoid valve controller, which then activated the pneumatic cylinder. At the same time, a signal was sent from the controller to hardware trigger the data acquisition system, which then began sampling at a rate of 2000 Hz. Upon completion of data acquisition, data was transferred from the CH12B units to the computer for storage. A diagram showing the relationship between various components is given in Fig 2.10.

The measurement of velocity at a location $0.5c$ downstream from the trailing edge of the aerofoil was carried out using hot-wire anemometry. Altogether measurement was performed at 51 positions across the wake in steps of 5 mm. The hot-wire probe was mounted through a traverse system which, with the aid of a manual controller,

could position the probe at any desired positions in the y direction, where y was in a direction perpendicular to the span and chord axes of the aerofoil, as shown in Fig 2.11.

Basically, the set-up of the spoiler activation and data acquisition system was the same as in phase one, though, one hot wire probe was used instead of the four pressure transducers. There were two electrical signals from each of the wires of the X-array probe, and they were connected to a TSI anemometer for signal conditioning. The output signals from the anemometer were then connected to two CH12B units, with a third CH12B unit connected to the rotary potentiometer. Upon activation of the solenoid valve, the data acquisition was hardware triggered and data was acquired at a sampling rate of 5000 Hz. After that, data was transferred to the computer for storage. The sampling rate of 5000 Hz was chosen so that sufficient data was available for the calculation of vorticity values.

Again, eight runs were conducted for each of the experimental conditions. As all runs were synchronised with the activation of the spoiler, ensemble average of the time variation of pressure coefficient, lift coefficient and velocity were obtained.

2.7. Vorticity Calculation

To relate the adverse lift effect on the aerofoil with the vortices generated by the moving spoiler, the resulting flow field is best discussed in terms of vorticity. Although velocity measurement was performed in a t-y plane as shown in Fig 2.12, where t is the time and y is the distance defined in Fig 2.11, it is possible to estimate the vorticity distribution through Taylor's hypothesis. Taylor (1938) hypothesized that the time history of the flow signal from a stationary probe can be regarded as that due to advection of a 'frozen' spatial pattern of turbulence past the probe with the mean speed U_0 . Although the origin of Taylor's hypothesis lies in the measurement of grid turbulence, subsequent researchers have demonstrated its validity in the measurement of large-scale coherent structures, e.g. Zaman & Hussain (1981) and Takamoto (1986) etc.

The vorticity, ω , in the x-y plane is defined as,

$$\omega = \frac{\partial v}{\partial x} - \frac{\partial u}{\partial y} \quad 2.4$$

and it is assumed from Taylor's hypothesis that the rate of change of v in the x direction can be related to the rate of change of v with respect to time through the following expression:

$$\frac{\partial v}{\partial x} = - \frac{\partial v}{\partial t} \cdot \frac{\partial t}{\partial x} = - \frac{\partial v}{\partial t} \cdot \frac{1}{u} \quad 2.5$$

where u is the convection velocity of the vortex centres. However, in the present study, the vortices generated by the rapidly moving spoiler have different strength and hence different convection velocities. It is therefore decided to use the local u velocity as the convection velocity of vorticity, instead of a single vortex convection velocity. The validity and accuracy of this assumption was discussed elaborately in Takamoto (1986), who concluded that it was unlikely that any significant error would arise. However, this assumption is not strictly correct and the actual error may depend on the true convection velocities and the degree to which the potential field of the aerofoil is involved.

Although the vorticity at any t and y could be calculated discretely with the following equation,

$$\omega_{i,j} = - \frac{v_{i,j} - v_{i-1,j}}{\Delta t u_{i,j}} - \frac{u_{i,j} - u_{i,j-1}}{\Delta y} \quad 2.6$$

where the subscript i represents any instant of t and j any location of y , it was decided to use a different approach in order to minimise the error caused by the random fluctuations due to the small-scale turbulence in the wake. Therefore $\omega_{i,j}$ was calculated indirectly using a least square method, as will be outlined in the following sections.

First, values of u and v were to be fitted by the following second order polynomial equations.

$$u = \alpha_1 t^2 + \alpha_2 ty + \alpha_3 y^2 + \alpha_4 t + \alpha_5 y + \alpha_6 \quad 2.7$$

$$v = \beta_1 t^2 + \beta_2 ty + \beta_3 y^2 + \beta_4 t + \beta_5 y + \beta_6 \quad 2.8$$

By using the above equations, the vorticity equation (2.4) is recasted into the following form:

$$\omega = -\frac{1}{u} (2\beta_1 t + \beta_2 y + \beta_4) - (\alpha_2 t + 2\alpha_3 y + \alpha_5) \quad 2.9$$

To determine the values of the coefficients in equation (2.7) and (2.8), a least square error method was employed. At any i and j , original values of $u_{i,j}$ and $v_{i,j}$ and the 24 samples of u and v surrounding location (i,j) were used to obtained least square fitted values of $u_{i,j}$ and $v_{i,j}$. One such estimation region is shown in Fig 2.12. This process was carried out for the entire range of samples.

The square error equations are given by,

$$e_u^2 = \sum_{q=j-2}^{j+2} \sum_{p=i-2}^{i+2} (\alpha_1 t_p^2 + \alpha_2 t_p y_q + \alpha_3 y_q^2 + \alpha_4 t_p + \alpha_5 y_q + \alpha_6 - u_{p,q}) \quad 2.10$$

$$e_v^2 = \sum_{q=j-2}^{j+2} \sum_{p=i-2}^{i+2} (\beta_1 t_p^2 + \beta_2 t_p y_q + \beta_3 y_q^2 + \beta_4 t_p + \beta_5 y_q + \beta_6 - v_{p,q}) \quad 2.11$$

Then by setting:

$$\frac{\partial e_u^2}{\partial \alpha_i} = 0, \quad \frac{\partial e_v^2}{\partial \beta_i} = 0 \quad \text{for } i \text{ and } j = 1 \text{ to } 6 \quad 2.12$$

and after re-arrangement of the resulting equations, the following matrix equations are obtained:

$$\mathbf{X} \cdot \mathbf{A} = \mathbf{Z}_u \quad \text{and} \quad \mathbf{X} \cdot \mathbf{B} = \mathbf{Z}_v \quad 2.13$$

$$\therefore \mathbf{A} = \mathbf{X}^{-1} \cdot \mathbf{Z}_u \quad \text{and} \quad \mathbf{B} = \mathbf{X}^{-1} \cdot \mathbf{Z}_v \quad 2.14$$

where,

$$\mathbf{A} = \begin{pmatrix} \alpha_1 \\ \alpha_2 \\ \alpha_3 \\ \alpha_4 \\ \alpha_5 \\ \alpha_6 \end{pmatrix} \quad \mathbf{B} = \begin{pmatrix} \beta_1 \\ \beta_2 \\ \beta_3 \\ \beta_4 \\ \beta_5 \\ \beta_6 \end{pmatrix}$$

$$\mathbf{Z}_u = \begin{pmatrix} \sum \sum t^2 u \\ \sum \sum tyu \\ \sum \sum y^2 u \\ \sum \sum tu \\ \sum \sum yu \\ \sum \sum u \end{pmatrix} \quad \mathbf{Z}_v = \begin{pmatrix} \sum \sum t^2 v \\ \sum \sum tyv \\ \sum \sum y^2 v \\ \sum \sum tv \\ \sum \sum yv \\ \sum \sum v \end{pmatrix}$$

$$\mathbf{X} = \begin{pmatrix} \sum \sum t^4 & \sum \sum t^3 y & \sum \sum t^2 y^2 & \sum \sum t^3 & \sum \sum t^2 y & \sum \sum t^2 \\ \sum \sum t^3 y & \sum \sum t^2 y^2 & \sum \sum ty^3 & \sum \sum t^2 y & \sum \sum ty^2 & \sum \sum ty \\ \sum \sum t^2 y^2 & \sum \sum ty^3 & \sum \sum y^4 & \sum \sum ty^2 & \sum \sum y^3 & \sum \sum y^2 \\ \sum \sum t^3 & \sum \sum t^2 y & \sum \sum ty^2 & \sum \sum t^2 & \sum \sum ty & \sum \sum t \\ \sum \sum t^2 y & \sum \sum ty^2 & \sum \sum y^3 & \sum \sum ty & \sum \sum y^2 & \sum \sum y \\ \sum \sum t^2 & \sum \sum ty & \sum \sum y^2 & \sum \sum t & \sum \sum y & \sum \sum 1 \end{pmatrix}$$

where,

$$\sum \sum t^k y^l u^m = \sum_{q=j-2}^{j+2} \sum_{p=i-2}^{i+2} t_p^k y_q^l t_{pq}^m$$

Once the values of the coefficients in equations (2.7) and (2.8) were calculated for each (i,j), vorticity values were then calculated by the following equation, which is re-arranged from equation (2.9).

$$\omega_{i,j} = \left[-\frac{2\beta_1}{u_{i,j}} - \alpha_2 \right] t_i + \left[-\frac{\beta_2}{u_{i,j}} - 2\alpha_3 \right] y_j - \frac{\beta_4}{u_{i,j}} - \alpha_5$$

3. NORMAL SPOILER DEFLECTED STATICALLY

3.1. General Remarks

Although the time mean characteristics for the stationary spoiler have been investigated extensively, there are still some aspects which deserve further investigation. One of these is the unsteady characteristics associated with the stationary spoiler, where the unsteady flow field in terms of pressure fluctuations and vortex shedding will be studied. The other aspect concerns the steady and unsteady aerodynamic loading on the spoiler. This is one area which has never been investigated before and will also be studied here. All the results reported here are for a normal spoiler of $h/c = 0.15$ located at $x_s/c = 0.7$ unless specified otherwise.

3.2. Time Mean Values on the Aerofoil

3.2.1. Pressure Distribution

The variation of the pressure coefficient, C_p on the aerofoil at $\alpha = 0^\circ$ is shown in Figure 3.1, where C_{pu} denotes the pressure coefficient on the aerofoil upper surface and C_{pl} for the pressure coefficient on the aerofoil lower surface. The experiments were conducted at a Reynolds number of 4.2×10^5 , and this Reynolds number was used throughout all the experiments unless specified otherwise. The pressure distribution exhibits the same features as those reported by other researchers. First the flow ahead of the spoiler experiences a compression, then followed by separation of the boundary layer from the tip of the spoiler, resulting in an abrupt drop in the value of C_p and the formation of a near wake aft of the spoiler with constant base pressure. On the lower surface as x/c progresses, C_p generally increases from a high suction value towards the C_p value of the near wake. To illustrate the effect due to the presence of a spoiler, Fig 3.2 shows the pressure distribution for the case of an aerofoil with and without a spoiler. Clearly, it can be seen that the change in C_p is very drastic due to the presence of the spoiler.

3.2.2. Effect of Reynolds Number

The effect of Reynolds number will be illustrated in terms of variations of the normal force coefficient (C_N), the Strouhal number (S), and the root mean square value of the fluctuating normal force coefficient (C_N') at $\alpha = 0^\circ$. Although it is not the intention to discuss the variation of the unsteady quantities here, the variations of the Strouhal number and the fluctuating normal force coefficient are included to provide a complete picture of the Reynolds number effect. Fig 3.3 shows the variation of the Strouhal number with the Reynolds number. It is known from the studies of bluff bodies that in the case of a sharp edge separation, the flow should be free of Reynolds number effect, so it appears that the Strouhal number in the present case should remain constant. However, it can be seen from Fig 3.3 that the Strouhal number reduces slightly as the Reynolds number is increased, and the variation in the Strouhal number is of a magnitude of the order of about 4%. This variation is thought to be due to the change in the boundary layer thickness and the hinge bubble size, as the Reynolds number is changed. The boundary layer and the hinge bubble then in turn modify the effective shape of the aerofoil, resulting in a lower Strouhal number.

In the case of C_N , it is known from Barnes (1966a) that the pressure distributions upstream of spoilers are dependent on the boundary layer thickness, and hence the Reynolds number. As C_N is obtained through integrated C_p , the variation of pressure distributions as a result of increasing Reynolds number will give rise to a corresponding variation in C_N as shown in Fig 3.4. However, whether the base pressure will vary with the Reynolds number and contributes towards the variation in C_N is unknown, as no measurement was carried out.

The variation of C_N' with Re as shown in Fig 3.5, however, presents a much more complicated phenomenon. It is believed that the hinge bubble will shrink in size as the Reynolds number is increased, and hence resulting in a smaller area being exposed to the turbulence associated with the bubble. But then the contribution due to the turbulence from the hinge bubble and also from the near wake is not as significant as the component associated with vortex shedding, as will be

illustrated in a later discussion. As such, there must be some relationship between the shedding component and the Reynolds number.

3.2.3. Effect of Boundary Layer Transition

To investigate the effect of boundary layer transition on the performance of the aerofoil with a spoiler, transition strips, which were made of coarse sand paper with width of 10 mm and particle size of 0.5 mm, were fixed onto the aerofoil surface, across its span on both the upper and lower surfaces, at location $x/c = 0.2$. The Reynolds number based on the particle length scale is about 900, well above the recommended minimum value of 600, as quoted in Pankhurst & Holder (1948). The results reported here were obtained with a spoiler height of $h/c = 0.15$ located at $x_s/c = 0.7$, at a wind speed of 25 m/s giving $Re = 4.2 \times 10^5$.

Fig 3.6 shows the variation of C_N with α for the cases with and without transition strips fitted. It can be seen from the graph that the two curves generally follow the same trend, albeit with some small deviations. As the two curves almost coincide with each other, it is therefore conjectured that transition in the aerofoil boundary layers has little effect on the overall flow field around the aerofoil. This is to be expected as the main flow characteristics here are dominated by flow separation at the tip of the spoiler. The only effect of transition, then, is to change the boundary layer thickness, which in turn modifies slightly the effective shape of the aerofoil, resulting in the deviations in C_N .

It can be seen from Fig 3.7, in which the variation of C_N' with the angle of attack is plotted, that at high angles of attack, the agreement between the two sets of values is good. However, at angles of attack less than -10 degrees, the deviation between the two sets of C_N' starts to grow and reaches a maximum difference of about 0.01, which is equivalent to a change of 15%.

The transition strips have little effect on Strouhal number, as shown in Fig 3.8. This reflects the fact that for a typical bluff body

with fixed separation points, the state of the boundary layers contributes little towards the instability process in the near wake. To illustrate this further, a typical set of spectra is shown in Fig 3.9, where the angle of attack is -2 degrees. As expected, the two spectra look alike, with the same intensity in the shedding components and the same background fluctuation level.

Since the boundary layer transition has little effect on the measured quantities, especially at $\alpha = 0^\circ$, it was decided that no transition strips to be fitted for the rest of the investigation.

3.2.4. Normal Force Measurements

The variation of C_N with angle of attack for two spoiler configurations is shown in Fig 3.10. It can be seen that, in each case, C_N generally follows more or less a linear relationship. At very high angles of attack when the hinge bubble no longer reattaches onto the spoiler, the spoiler can be considered as enclosed in the separation region and hence plays little or no part on the overall performance of the aerofoil. This is in fact similar to the phenomenon of deep-stalled aerofoil. Similarly at very low angles of attack, flow separation takes place on the lower surface of the aerofoil ahead of the trailing edge, resulting in an increase in C_N . As an illustration of how C_N is affected by the presence of a spoiler, the variation of C_N for the case of a spoiler of $h/c = 0.15$ located at $0.7c$ is shown in Fig 3.11 together with the case of a plain aerofoil. Clearly, it can be seen that, as a result of the presence of the spoiler, the whole curve is translated downward, and it follows almost the same gradient as that of the original curve, where the gradient of the curve corresponding to the plain aerofoil is 5.21 and that corresponding to the aerofoil with the spoiler is 4.89. The other way of looking into the effect of the spoiler is to express the results in terms of differential normal force coefficients caused by the presence of the spoiler, as shown in Fig 3.12. But the presentation of results here is by no means the best way to describe the performance of the spoiler as the effectiveness is best expressed in terms of the change in lift, which was not obtained in the present case as only normal force measurement was carried out.

3.2.5. Effect of Spoiler Geometric Parameters

To determine the effect of different spoiler heights and spoiler locations, three spoilers of $h/c = 0.1, 0.125$ and 0.15 were in turn mounted normal to the chord at three locations of $x_s/c = 0.3, 0.5$ and 0.7 consecutively. Fig 3.13a shows the variation of C_N with x_s/c for different h/c . Generally C_N decreases as the spoiler is moved backward towards the trailing edge of the aerofoil. As the spoiler is moved backward, less area on the aerofoil is exposed to the base suction aft of the spoiler and hence a reduction in C_N . It can also be seen from Fig 3.9a that C_N tends to increase as the spoiler height is decreased, and this can be seen more clearly in Fig 3.13b where the variation of C_N with h/c for different x_s/c is shown. It is known that the longer the spoiler height, the bluffer the section, resulting in a larger vortex formation region with a lower base pressure. The changes in the base pressure will then alter the entire flow field around the aerofoil. It is obvious from Fig 3.1 that the contribution of the base pressure towards C_N is not as great as that due to the pressures ahead of the spoiler. Therefore the reduction in the base pressure due to a longer spoiler will be offset by a much higher negative differential pressure loading ahead of the spoiler, resulting in the overall decrease in C_N .

3.2.6. Base Pressure Measurements

The base pressure is important in a sense that as a result of flow separation from the tip of the spoiler, the base pressure arises due to the formation of the near wake region aft of the spoiler. The base pressure, or rather the separation region in particular, will then modify the entire flow field around the aerofoil. Shown in Fig 3.14 is the variation of base pressure with angle of attack. The indication here reveals that the higher the angle of attack, the bluffer the section and hence a lower base pressure. This trend breaks down at about 13 degrees when the flow separation ahead of the spoiler in terms of the hinge bubble is so severe that the spoiler is immersed inside the separated region, and hence the spoiler has little or no influence on the overall performance. It is believed that as the angle of attack is changed, the vortex formation region might be exposed to the influence of the free stream, and hence

the velocity at the tip of the spoiler where separation occurs may change, and this will in turn change the base pressure as the base pressure is related to the velocity at the tip of the spoiler.

3.3. Unsteady Characteristics

3.3.1. Vortex Shedding Frequency Measurements

Before proceeding into the discussion proper, it is necessary to define some parameters which will influence the phenomenon of vortex shedding. Shown in Fig 3.15 is a diagram of an aerofoil with a spoiler with all the important parameters listed. Clearly, there are two separated shear layers. The one which separates from the tip of the spoiler shall be known as the top shear layer, abbreviated as TSL, whereas the one separates from the trailing edge on the lower surface of the aerofoil shall be known as the bottom shear layer, abbreviated as BSL. In addition two physical dimensions are defined. First, the distance between the spoiler tip and the trailing edge projected onto a plane normal to the free stream direction, and this is known as the normal projection height of the spoiler, represented symbolically by H . The second dimension is the length of the afterbody aft of the spoiler projected onto a plane parallel to the free stream direction and is represented by l_a . As will be discussed later, these parameters play an important role in the shedding phenomenon.

The vortex shedding frequencies, f_s , were estimated from the power spectra of normal force coefficients obtained through the pressure averaging measurements. The frequencies are presented here in terms of Strouhal number, $S = f_s l / U_0$, where l is any convenient length scale and U_0 the free stream velocity. One obvious choice of the length scale is to use the physical dimension which is directly responsible for the phenomenon of vortex shedding. McLachlan et al (1981) adopted a length scale which is equal to the projection height of the spoiler normal to the free stream, that is the dimension H defined earlier. They found that if the frequencies are reduced in this way, Strouhal numbers with comparable values to those due to bluff bodies can then be obtained. In the present case, if the shedding frequencies are reduced with the same

length scale as that used by McLachlan et al (1981), Strouhal numbers in the range of 0.18 can be obtained, which are in good agreement to the typical values for bluff bodies. However, as different spoiler heights at different locations were used in the present experiments, there is a need to use a constant length scale throughout so as to facilitate comparison. Naturally, it was decided that the aerofoil chord, c , would be an appropriate length for the purpose of comparison. Therefore, in addition to the Strouhal number defined with H , which shall be known as S_H , a Strouhal number defined with c will also be used and shall be known as S_c .

The variation of the Strouhal number, S_c , with the aerofoil incidence is shown in Fig 3.16, for two spoiler configurations. Plotted in Fig 3.16a is for the case with a spoiler of height $h/c = 0.15$ and located at $x_s/c = 0.7$. Basically the variation can be categorized into three regimes. At angles of attack from -10° to 15° , S_c appears to be nearly constant, with a mean value of about 0.88. At high angles of attack, S_c drops monotonically because the spoiler is now immersed inside the separation region due to flow separation ahead of the spoiler. At a negative angle of attack of about -12° , S_c drops drastically to a value of about 0.3, and this is due to flow separation on the lower surface of the aerofoil ahead of the trailing edge. The nearly constant trend of Strouhal number obtained in this case, which corresponds to the spoiler effective regime, appears to indicate that the vortex shedding frequency is insensitive to the variation of angle of attack, but this is not to say that the near wake characteristics are invariant to the angle of attack. As the base pressure measurements suggest, the angle of attack will, to some extent, determine the base pressure. Therefore, it seems only right to say that the rate of vortex formation is nearly constant, but that the base cavity in fact changes in size, resulting in the variation of C_{pb} . However, as will be discussed later for other values of h/c and x_s/c , the Strouhal number is not quite constant, and it does follow a certain trend.

For the case with a spoiler of $h/c = 0.1$ located at $x_s/c = 0.3$, the Strouhal number S_c no longer appears to be constant, and in fact some non-linear relationship can be obtained, as shown in Fig 3.16b. If a

careful comparison between Fig 3.16a and Fig 3.16b is made, it can be realized that in fact both curves follow the same trend, which looks something like an inverted "S". This trend is thought to be a function of the location of the spoiler, where the trend damps down as the spoiler is moved further back towards the trailing edge, probably due to the effect of shorter l_a . Also noted in Fig 3.16b is that there is a regime between angles of attack 18° to 22° where no Strouhal number is plotted. From the corresponding spectra at the angles of attack 18° and 20° , there is exhibited a broad band characteristic, with no apparent shedding component. This phenomenon will be discussed in a later section.

In order to understand the present problem better, it is instructive at this juncture to look into the mechanism of vortex formation for symmetrical bluff bodies. It was proposed by Gerrard (1963) that vortices are formed due to the rolling up of one separated shear layer as induced by the opposite vorticity from the other shear layer. And the approach of this opposite vorticity, in sufficient strength, will cut the growing vortex from its feeding shear layer, and hence a vortex is shed. Clearly, the vortex formation depends much on how the two separated shear layers interact with each other. To this Roshko (1954) recognized a few important parameters, as shown in Fig 3.17. The shedding frequency depends on the velocity of the fluid at the point of separation, U_b , and the wake width d' . In addition, Roshko (1954) has shown that the shedding frequency can be related to the length of the formation region, l_f . He has shown that if l_f is reduced, interaction of the shear layers is facilitated and the periodic time is shortened, and hence the shedding frequency f_s increases accordingly. Although the foregoing discussion is for the case of a symmetrical bluff body, many similar features are exhibited in the present case. In the present case, the BSL always separates from the trailing edge at $x/c = 1$, whereas the TSL separates in a different x/c plane depending on the location of the spoiler and the aerofoil incidence at which it is separating. There is also an afterbody length l_a aft of the spoiler which will affect the formation length l_f , and this length is dependent on both the angle of attack of the aerofoil and the position of the spoiler. In addition, at different angles of attack, H changes accordingly and thus will affect d' across which the shear layers will interact. Therefore, at each angle of attack, the spoiler-

aerofoil configuration can be viewed as a different bluff body with unsymmetrical vortex shedding.

Since the spoiler-aerofoil configuration is presented as a varying bluff body to the flow when the angle of attack is changed, it is logical to discuss the parameters that are dependent on the angle of attack and that will influence the shedding phenomenon. The parameters that have been identified are U_b , H and l_a . U_b is the velocity at which the fluid separates from the tip of the spoiler or the trailing edge of the lower surface of the aerofoil. At each angle of attack, the aerofoil is presented as a different bluff body to the free stream, and the distribution of velocity around the body changes accordingly, hence resulting in a change in U_b as well. This change can be confirmed by observing the changes in the base pressure, C_{pb} , as U_b can be related to the base pressure C_{pb} by:

$$C_{pb} = 1 - \left(\frac{U_b}{U_o} \right)^2$$

where U_o is the undisturbed free stream velocity. It has already been shown in Fig 3.14 that C_{pb} is a function of the angle of attack, and so is U_b . On one hand, U_b governs the rate at which vorticity is fed to the separated shear layers, and hence the amount of time for each shear layer to be induced by the opposite vorticity of the other shear layer to roll up a vortex. On the other hand, C_{pb} is also related to the size of the base cavity. The larger the size of the base cavity the longer the periodic time the shear layers need to take to interact with each other for the formation of vortices, and hence f_s decreases.

The other parameter which has a direct influence is l_a , the afterbody length aft of the spoiler. As suggested previously, the shedding frequency is dependent on the length of the vortex formation region. But as the TSL can only be influenced directly by the BSL after it sweeps past l_a , and l_a is reduced by increasing the angle of attack, f_s is therefore expected to increase. However, this trend is not observed from Fig 3.16, because as the angle of attack increases, H increases as well, and the effect of increasing H is to prolong the time for the opposing shear layers to interact. Hence there must be a delicate balance between

the extent at which l_a and H change. To clarify the effect of l_a , there is a need to present the Strouhal number with H as the non-dimensionalising length scale.

The variation of S_H with angle of attack at two different spoiler configurations is shown in Fig 3.18. In Fig 3.18a, which is for the case of a spoiler $h/c = 0.15$ located at $x_s/c = 0.7$, it can be seen that S_H increases as the angle of attack increases, within the regime where the spoiler is effective. As l_a varies inversely with the angle of attack, the result can be re-interpreted as S_H increases when l_a is decreased. Therefore, this confirms the trend that as a result of the increase in l_a , the separated shear layers take longer periodic time to get interacted, and hence decrease in the shedding frequency. Similar trend can also be observed in Fig 3.18b for the case of a spoiler $h/c = 0.1$ located at $x_s/c = 0.3$. However, this time the marginal changes are more drastic than those in Fig 3.18a, as S_H here varies in the range from 0.02 to 0.26. The exact mechanism for this variation is yet to be known.

It is difficult to ascertain the exact mechanism determining the relationship between the Strouhal number, the angle of attack, and the location of the spoiler from the frequency measurements alone. The present case basically represents unsymmetrical vortex shedding, as opposed to symmetrical vortex shedding due to symmetrical bluff bodies as reported by many workers. There certainly exists an inter-relationship between U_b , l_a , H and the angle of attack. Therefore this is one of the areas where more work could be done to gain a better understanding, probably through a combination of flow visualization, surface velocity measurements and surface pressure measurements etc.

In order to look into the effect of l_a and H , experiments with different spoiler heights at different locations were also conducted and the results are plotted in Fig 3.19. Fig 3.19a shows the variation of S_c with x_s/c for different h/c at zero angle of attack. Since the angle of attack is zero, l_a is related to the position of the spoiler through $l_a = 1 - x_s/c$. It can be seen that as l_a decreases, that is also equivalent to x_s/c increases, S_c increases. This trend confirms the earlier postulation that the length of the vortex formation region is related to l_a , that is to say the

decrease in l_a facilitates the interaction between the two separated shear layers in the process of vortex formation, and hence the shedding frequency increases, corresponding to a higher S_c . It can also be seen in Fig 3.19a that S_c increases as h/c increases, and this trend can be illustrated more clearly in Fig 3.19b where S_c is plotted against h/c for different x_s/c . At zero angle of attack and at a particular x_s/c , H is related to h by $H = h + y_s/2$, where y_s is the thickness of the aerofoil section at x_s . Therefore, as h increases, H increases as well, and it can be seen that this increase resulted in a reduction in S_c . This also confirms the earlier discussion that as H increases, the separated shear layers take longer time to interact with each other and hence a corresponding reduction in the shedding frequency.

3.3.2. Unsteady Pressure Distribution

The unsteady flow characteristics around the aerofoil with a normal spoiler can be described in terms of the fluctuating pressure distribution. Fig 3.20 shows the variation of the fluctuating pressure with x/c for the case with a spoiler of $h/c = 0.15$ located at $x_s/c = 0.7$. The fluctuating pressure is expressed in terms of the root mean square value of the conventional pressure coefficient and is represented by C_p' . On the upper surface of the aerofoil, C_p' remains fairly constant up to $x/c = 0.4$, where C_p' jumps to a high value and this can be associated with the flow separation and the formation of a hinge bubble. Aft of the spoiler, C_p' increases towards the trailing edge and reaches a maximum value near the trailing edge, where the strongest effect of vortex shedding is felt. On the lower surface, C_p' increases towards the trailing edge, indicating the influence of vortex shedding at different x/c .

It is instructive at this point to analyze the fluctuating pressures more fully in terms of power spectrum. The study of pressure power spectra on the surface of the aerofoil has been carried out by Ayoub et al (1982) and McLachlan et al (1983) for some selective locations on the aerofoil surface. This study is critical as it will not only reveal the unsteady flow pattern around an aerofoil but will also indicate the level of fluctuations on different parts of the aerofoil.

The pressure power spectra at different measurement locations on the surface of the aerofoil are shown in Fig 3.21. For each of the spectra, the abscissa represents the reduced frequency, k , that is defined as fc/U_0 , where f is the frequency, and the ordinate is the non-dimensional power spectral density function of pressure. On the upper surface of the aerofoil at pressure measurement location 1, the spectrum contains a strong spike with a narrow band characteristics, which is associated with the vortex shedding frequency and will be referred to as the shedding component. Also apparent in the spectrum are random spikes associated with the interference noises, and the broad band background level remains of the order of 10^{-5} , compared to 10^{-3} for the shedding component. As the location moves towards the position of the spoiler, the spectrum exhibits more or less the same characteristics as that at location 1, with the magnitude of the shedding component increases progressively from an order of magnitude of 10^{-3} at location 1 to about 10^{-2} at location 6. Generally the broad band background remains at a level of the order of 10^{-5} from location 1 to location 6, indicating a small or near absence of turbulence level. Also obvious from these spectra is the presence of another narrow band signal centred at the second harmonic of the shedding frequency. This second harmonic increases from locations 1 to 6, to a level of the order of about 10^{-4} .

At location 7 the spectrum changes its appearance, with a much narrower shedding component as compared to those at locations 1 to 6, whereas the magnitude of the peak remains of the order of about 10^{-2} , comparable to those at locations 5 and 6. However, the narrow band component at this location is superimposed onto a strong broad band background level, which indicates the presence of strong turbulence. This is due to the formation of a separation bubble as a result of the adverse pressure gradient set up by the presence of the spoiler. The broad band turbulence is of the order of about 10^{-3} to 10^{-4} . As the location moves further towards the spoiler, the shedding component remains with the same narrow band characteristics, with its magnitude increases to the order of about 10^{-2} to 10^{-1} . There is also a concomitant decrease in the broad band turbulence, which is associated with the presence of the hinge bubble, to the order of about 10^{-4} . The second harmonic does not appear profoundly in these spectra, i.e. from locations

7 to 11, and even if it does, as at locations 9 to 11, its level is only slightly higher than the broad band fluctuations.

Aft of the spoiler, the shedding component remains with the same narrow band characteristics, and the bandwidth widens from locations 14 to 16. The magnitude of this shedding component increases from the order of about 10^{-2} to about 10^1 at location 16. Obviously, the magnitude of the shedding component on the upper surface of the aerofoil is the highest at location 16, where the effect of vortex shedding is the strongest, with a comparatively wider narrow band bandwidth. Generally, the level of broad band fluctuations, which are associated with the near wake aft of the spoiler, increases towards the trailing edge and reaches a level of 10^{-2} at location 16. This time, however, a stronger second harmonic can be seen in those spectra aft of the spoiler, with a magnitude of the order of about 10^{-2} . But this second harmonic appears absent at location 16, for it is buried in the background level, which has the same order of about 10^{-2} .

On the lower surface, as can be seen from Fig 3.21b, C_p' increases progressively towards the trailing edge, it is therefore logical to expect the same trend to be observed in the power spectral level. From locations 32 to 30, the power spectrum exhibits narrow band characteristics, with a peak of magnitude of the order of 10^{-2} corresponding to the shedding frequency. There is also a narrow band component corresponding to the second harmonic of the shedding frequency, with a magnitude of the order of 10^{-4} , together with a broad band level of the order of 10^{-5} to 10^{-6} . From location 27 onwards, the magnitude of the shedding component increases, from the order of 10^{-2} to the order of less than 10^1 at location 17. As the magnitude increases, the corresponding bandwidth of the shedding component also widens, and it can be seen that the bandwidth of the shedding component at location 17 is almost twice as wide as that at location 27. In much the same way, the second harmonic also increases its magnitude, from the order of 10^{-4} at location 32 to the order of 10^{-2} at location 17. Generally, the level of broad band fluctuation level remains nearly constant throughout, of the order of about 10^{-5} . This broad band level

increases slightly to the order of 10^{-4} at locations 17 and 18, due to the strong influence of the shed vortices, and the proximity of the near wake.

Clearly from the study of these spectra, the fluctuating level is the strongest near the trailing edge, aft of the spoiler in the near wake, and that the contribution towards C_p' comes mostly from the shedding component. Therefore, it is expected that the fluctuating normal force would have its predominant contribution coming from the shedding component, together with a secondary effect from the fluctuations associated with the hinge bubble and the near wake. This is critical as the shedding component is responsible for any vortex induced vibrations, whereas the turbulence is responsible for the phenomenon of buffeting of the aerofoil or any control surfaces that happen to be downstream.

3.3.3. Fluctuating Normal Force Measurements

In the previous sections, the unsteady pressure distribution on the aerofoil with a spoiler is discussed. It can be noticed that the contribution towards the unsteady characteristics comes mostly from the shedding component. In the following sections, the integrated unsteady normal force will be looked into. As defined in the previous section, the unsteady force will be represented by C_N' , where C_N' is the root mean square value of the fluctuating normal force coefficient.

First the variation of C_N' with angle of attack at two different spoiler configurations is given in Fig 3.22. Fig 3.22a shows the case with a spoiler of $h/c = 0.15$ located at $x_s/c = 0.7$. It can be seen that in this figure, there is a regime where C_N' remains more or less constant, and a regime where C_N' drifts capriciously. Although it could be argued with reference to the variation of Strouhal number with angle of attack that the sudden drift in C_N' is due to aerofoil stall, it is felt that a better understanding can be gained through a spectral analysis. The power spectra of C_N' for the case with a spoiler of $h/c = 0.15$ located at $x_s/c = 0.7$ are given in Fig 3.23 for different angles of attack.

At zero angle of attack a typical spectrum is obtained, from which a narrow band component associated with the vortex shedding frequency at $k = 0.88$ can be identified, together with another narrow spike associated with the second harmonic of the shedding frequency. The spectrum exhibits similar features to those obtained through the surface pressure power spectra. Also apparent in this spectrum is that the level of broad band fluctuations is of the order of about 10^{-3} to 10^{-4} , whereas the shedding component is of the order of 10^{-1} and the second harmonic is of the order of about 10^{-3} to 10^{-2} .

When the angle of attack is increased or decreased slightly away from zero angle of attack, the spectrum remains nearly the same. First the increase towards positive angles of attack will be discussed. It can be seen that the magnitude of the shedding component remains the same of the order of 10^{-1} as α increases, up to an angle of attack of about 14 degrees. The nearly constant level of shedding component in this regime corresponds to the relatively flat variation of C_N with the angle of attack. In this regime the flow around the aerofoil exhibits the same unsteady characteristics as at $\alpha = 0^\circ$, with no occurrence of aerofoil stalling. It is also apparent in these spectra that the second harmonic remains of the same order of magnitude. However, there is a slight increase in the broad band fluctuation level, possibly due to the lengthening of the hinge bubble.

At $\alpha = 16^\circ$, the shedding component drops to an order of magnitude of 10^{-2} . The second harmonic diminishes in this spectrum, possibly being buried in the background fluctuations. The most profound feature in this spectrum is that besides the shedding component, there also appears a relatively 'broad' narrow band component, of the order of 10^{-3} and centred at a reduced frequency of $k = 0.4$, corresponding to an actual frequency of about 40 Hz. This is due to the fact that the hinge bubble either reattaches onto the spoiler near its tip, which corresponds to $k = 0.88$, or encloses the spoiler completely with no downstream reattachment onto the aerofoil surface, which corresponds to $k = 0.4$. The flow here can then be viewed as undecided as to which shedding pattern to follow.

At $\alpha = 18^\circ$, the shedding component shifts to $k = 0.7$, indicating that the flow has now separated well ahead of the spoiler, and that the spoiler is completely immersed inside the resulting separated region with no reattachment occurring. At this juncture, the spoiler plays little or no part in the overall performance of the aerofoil. From there on, it can be noticed that the shedding component shifts gradually from $k = 0.7$ to $k = 0.4$, indicating the corresponding increase in the bluntness of the stalled aerofoil as the angle of attack is increased.

When the angle of attack is decreased from 0 degree to -10 degrees, the shedding component at about $k = 0.88$ remains of the order of about 10^{-1} . However the bandwidth of the shedding component increases as the angle of attack decreases, and there is also an overall decrease in the level of broad band fluctuations from the order of about 10^{-3} to the order of about 10^{-4} . The second harmonic is also apparent in each spectrum, and its magnitude remains of the order of about 10^{-3} . Therefore the flow characteristics do not change appreciably from the angle of attack of 0 degree to -10 degrees, at least in the spectra of C_N' .

At $\alpha = -12^\circ$, the shedding component diminishes completely, so does the second harmonic, and an emergence of a narrow band signal at $k = 0.4$, due to flow separation on the lower surface of the aerofoil. This also causes C_N' to change brusquely from 0.05 at -10 degree angle of attack, to a value of 0.14 at -12 degree angle of attack. In the spectrum at -12 degree angle of attack, the magnitude of the shedding component is not much greater than that of the background fluctuations, and the spectrum is actually not unlike that of a "white noise", or rather the so called "random signal".

At $\alpha = -14^\circ$, the signal is generally broad band in nature, with a spike of a magnitude of the order of 10^{-1} at $k = 0.2$ and its second harmonic at $k = 0.4$ with the same order of magnitude. However at -16 degree angle of attack, the intensity of the broad band fluctuations reduces, with a more prominent shedding component at $k = 0.38$.

Therefore, the flow can be classified into three regimes: from the angles of attack of -10 to 14 where the spoiler plays an effective role; for

angles of attack greater than 14 degrees where the spoiler is immersed inside the separated region; for angles of attack smaller than -10 degrees where flow separation occurs on the lower surface of the aerofoil. Some time variations of C_N at selected α are also given in Fig 3.24 for illustration.

After looking into the case for a rear spoiler located at $x_s/c = 0.7$, where there is little afterbody aft of the spoiler, it is logical to look into the case for a forward spoiler of $h/c = 0.1$ located at $x_s/c = 0.3$. Fig 3.22b shows the variation of C_N' with the angle of attack for this spoiler configuration. It can be seen that the value of C_N' remains almost constant from 2 degree angle of attack to 16 degree angle of attack. From 2 degree angle of attack onwards, C_N' increases as the angle of attack is decreased, until at about 8 degrees where C_N' drops first before picking up again. From 16 degree angle of attack onwards, C_N' increases slightly. Again, it is felt that the nature of the flow would be revealed more clearly through a study of the associated spectra.

Fig 3.25 shows the spectra for the case with a spoiler of $h/c = 0.1$ located at $x_s/c = 0.3$. At zero angle of attack, the spectrum shows similar features to those discussed earlier for a rear spoiler. The magnitude of the shedding component, which is centred at $k = 0.88$ is of the order of 10^{-1} whereas the second harmonic is of the order of 10^{-3} to 10^{-4} . As the angle of attack is increased towards 16 degrees, the shedding component declines in magnitude to the order of about 10^{-3} , together with a gradual reduction in its shedding frequency from $k = 0.9$ to $k = 0.7$. Similarly the second harmonic also declines in magnitude and becomes hardly discernible from 4 degree angle of attack onwards, probably being deeply buried in the broad band fluctuations. It can also be noticed that the shedding component decreases in magnitude, approaching that of the broad band fluctuations, and disappears altogether at 18 degree angle of attack, probably also being deeply buried in the broad band fluctuations of the same order of magnitude.

However, the decrease in the peak of the shedding component does not change the value of C_N' significantly and C_N' can be considered nearly constant as obvious from Fig 3.22b, for two reasons. First, the

reduction occurs mostly in the peak of the shedding component whereas the bandwidth does not alter much, hence the area enclosed by shedding component in these spectra remains more or less the same. Second, as the peak of the shedding component decreases in magnitude, the overall force is compensated by the corresponding increase in the spectral density at other frequencies associated with the broad band fluctuations, especially at reduced frequencies below $k = 1$. This second argument can be validated by a comparison between the spectra in Fig 3.25 for 16 and 18 degree angle of attack, where the level of the broad band fluctuations rises in magnitude. As the angle of attack is increased from 18 degrees to 20 degrees, there is a kink in the spectrum, indicating the re-emergence of the shedding component due to the phenomenon of aerofoil stalling. At 25 degree angle of attack, a shedding component of narrow band characteristics is apparent at $k = 0.15$, due to the fully established stalling phenomenon of aerofoil.

As the angle of attack is decreased towards negative values, the magnitude of the shedding component increases in magnitude, and can attain a high value with an order of magnitude of about 10^0 at -8 degree angle of attack. The second harmonic, however, declines quickly to the level of broad band background, after reaching a maximum level of the order of 10^{-3} at -6 degree angle of attack. As the angle of attack is decreased further from -8 degrees onwards, the bandwidth of the shedding component widens, and at the same time its magnitude decreases. At -14 degree angle of attack, the frequency of the shedding component shifts remarkably to a low k of about 0.2, indicating the onset of flow separation on the lower surface of the aerofoil ahead of the trailing edge. Again some time histories of C_N' at selected α are given in Fig 3.26.

Comparing the spectra for the two spoiler configurations, it can be noted that the variation of the magnitude of the shedding component with angle of attack is more drastic in the case for a front spoiler than for a rear spoiler. The exact mechanism for this occurrence is unknown, as no firm deduction can be made from normal force measurements alone. One possible reason could be the proximity of the stagnation point to the spoiler location in the case for a front spoiler. Furthermore, the hinge

bubble may or may not form at different angles of attack, and the bubble will lengthen or shorten when the angle of attack is changed, as the case may be.

To illustrate further this capricious feature, Fig 3.27 shows the variations of C_N' with different h/c and x_s/c . It can be noted that the strength of fluctuations is the strongest for the case with a spoiler of $h/c = 0.15$ located at $x_s/c = 0.7$, and is the weakest with $h/c = 0.1$ at $x_s/c = 0.5$. Basically at both $x_s/c = 0.7$ and 0.5 , the level of fluctuations increases with increasing h/c . But at $x_s/c = 0.3$, probably due to the large separation region involved aft of the spoiler, C_N' remains relatively insensitive to the choice of h/c , as the three data points almost coincide as shown in Fig 3.27a.

3.3.4. Comparison of C_N' with and without a spoiler

To determine the extent of the unsteady load induced on the aerofoil by the spoiler, a comparison is made with the values of C_N' obtained for a plain aerofoil. The results are plotted in Fig 3.28, where the filled square symbols represent the case with a spoiler of $h/c = 0.15$ located at $x/c = 0.7$, the triangular symbols for the case with a spoiler of $h/c = 0.1$ at $x/c = 0.3$, and the unfilled square symbols for the case of a plain aerofoil. For the plain aerofoil, only values for positive angles of attack are plotted as the aerofoil is symmetrical. The variation of C_N due to the presence of a spoiler has already been discussed in the earlier sections and will not be repeated here. For the plain aerofoil, C_N' remains at a fairly constant level of about 0.005 until the occurrence of aerofoil stalling at high angles of attack. After stalling, the magnitude of C_N' increases tremendously, to the same level as those obtained with the spoilers. Compared with the plain aerofoil, it is clear that the levels of fluctuations induced by the presence of the spoilers are high. For instance, at zero angle of attack, C_N' of the plain aerofoil is at about 0.005, but this value increases ten folds to about 0.05 when either of the spoilers was added.

It is revealed in the spectrum for $\alpha = 0^\circ$ in Fig 3.29a that, without the spoilers, the level of fluctuations is of the order of magnitude

of about 10^{-6} . The spikes in the spectrum are due to electrical noise at 50 Hz and its harmonics. With the presence of the rear spoiler, however, it is clear from Fig 3.29b that the increase in C_N' arises not only from the dominant shedding component, but also from the general increase in the level of background fluctuations, from the order of magnitude of about 10^{-6} to 10^{-4} . This is probably due to the formation of the hinge bubble ahead of the spoiler and the near wake region aft of the spoiler. Therefore, with the deployment of spoilers, the aerofoil is not only subject to the possibility of vortex-induced vibration, but is also susceptible to buffeting due to broadband turbulence.

3.3.5. Reynolds Number Effect on C_N'

The variation of C_N' with the Reynolds number as illustrated in Fig 3.5 has already been discussed in section 3.2.2. It is argued that C_N' changes chiefly due to the changes in the shedding component with the Reynolds number. Fig 3.30 shows the pressure spectra at different Reynolds numbers, at $\alpha = 0^\circ$. Before any interpretation is made, it is recalled that to facilitate comparison, the abscissa in each spectrum is non-dimensionalised into reduced frequency k , and the conventional spectral density in the ordinate is also non-dimensionalised by a factor of U_o/c . The frequency range obtained from the Fast Fourier Transform is up to 500 Hz. But since the frequency range is normalized by c/U_o , this gives a range of k from 0 to 12.5 for the case where $U_o = 10$ m/s and a corresponding resolution of 0.0061 for the reduced frequency. In the case where $U_o = 30$ m/s, a range of k from 0 to 4.1667 with a reduced frequency resolution of 0.002 is obtained. To facilitate comparison, all spectra are shown in a reduced frequency range from 0 to 3. Therefore the coarse spectra obtained at low wind speeds is due to the comparatively large frequency resolution used.

In the Fig 3.30, it can also be seen that the shedding component in each of the spectra is of the same order of magnitude, which has an approximate value of 10^{-1} . The appearance of each spectrum does not differ much as well, though there is a change in the coarseness of the spectrum. It can also be seen from the spectrum for $U_o = 10$ m/s that there are components of a very narrow band nature, due to noise

interference, which dies away at high wind speeds. It is thought that this is due to the poor noise to signal ratio at low wind speeds. For this reason an artificially high value of C_N' in Fig 3.5 at $Re = 1.7 \times 10^5$ results. In fact, if the contribution from the noise interference is eliminated, a C_N' value of about 0.06 can be obtained. If the noise interference at low wind speeds is discounted, the appearances of the spectra look alike, which suggests that the normalized spectrum is indeed independent of the Reynolds number effect.

3.3.6. Pressure Measurements near Aerofoil Trailing Edge

In the steady case with attached flow over a plain aerofoil, the Kutta condition reflects the fact that the flows from the upper and the lower surfaces leave smoothly to the trailing edge, ensuring equality of pressure and velocity across the trailing edge. However in unsteady flow, which involves separation ahead of the trailing edge, the conventional Kutta condition at the trailing edge may no longer be valid. To determine the trailing edge condition when there is flow separation owing to the presence of the spoiler, pressure measurements were carried out near the trailing edge. Ideally, the measurement should be conducted just next to the trailing edge, but this is obviously impossible as space constraints dictate the positions for the mounting of pressure tappings. The positions nearest to the trailing edge where pressure tappings can be mounted are at $x/c = 0.97$, on both the upper and the lower surfaces of the aerofoil. The pressure coefficient thus obtained from the pressure tappings on the upper surface is known as C_{p16} and the one from the lower surface is known as C_{p17} . All measurement was carried out at a wind speed of 25 m/s, giving a $Re = 4.2 \times 10^5$, with a normal spoiler of $h/c = 0.15$ at $x/c = 0.7$ on the upper surface of the aerofoil.

Fig 3.31 shows the variations of the true mean values of C_{p16} and C_{p17} with angle of attack. In each case, there is a relationship between the pressure coefficient and the angle of attack. In the case of C_{p16} , it is expected to follow the variation of the base pressure, since it is inside the base cavity. On the other hand, the variation of C_{p17} is relatively gentle. At -4 degree angle of attack, C_{p17} is about -0.65 and it declines slowly towards a constant value of about -0.7 between the angles of attack from

0 to 4 degrees, before picking up again to -0.6 at 14 degree angle of attack.

To establish the relationship between C_{p16} and C_{p17} , phase differences between them at the shedding frequency were also obtained through cross-correlation. The phase difference, by which C_{p16} leads C_{p17} , is plotted in Fig 3.32, as a variation with angle of attack. At a glance, although C_{p16} always leads C_{p17} , the relationship is quite capricious, with two local maxima and two local minima. At $\alpha = -10^\circ$, the phase difference is about 20 degrees, and its value decreases to about 12.5 degrees at $\alpha = -6^\circ$. The phase difference increases slightly to 14 degrees at $\alpha = -2^\circ$ before falling off to an absolute minimum value of about 7.5 degrees between $\alpha = 4^\circ$ and 6° . As the angle of attack increases further, a phase difference of about 20 degrees is reached before falling off again. At any rate, the maximum value of phase difference attainable is about 20 degrees, and the variation is within the range of values between 20 and 7.5 degrees. The maximum value of 20 degrees is equivalent to a fraction of 1/18 or 5.5% of a cycle of vortex shedding. As the pressure fluctuations can be directly attributed to the interaction between the top shear layer and the bottom shear layer, it can be said that disturbances are generated in the near wake region and are propagated upstream. The fact that C_{p16} leads C_{p17} means the upstream propagation reaches location 16 before location 17, and that the delay is indeed very short.

Shown in Fig 3.33 are the variations of C_{p16}' and C_{p17}' with angle of attack. The value of C_{p16}' is about 0.2 between $\alpha = 0^\circ$ and 8° , and it falls to 0.15 as the angle of attack increases further from 0° to 14° . As for C_{p17}' , it declines gradually from a value of about 0.2 at $\alpha = -10^\circ$ to a value of 0.1 at $\alpha = 14^\circ$.

The above discussion does not suggest the breakdown of the Kutta condition. As x/c tends to 1, the measurements of pressures on the upper and lower surfaces should also approach the same value. The above measurements merely suggest that the change across or rather the pressure gradient near the trailing is very drastic. The amount of pressure gradient also varies largely with the aerofoil incidence. This

fact should be noted in modelling the flow numerically, especially when panel method is employed as in Tou & Hancock (1985), where the lengths of the panels next to the trailing edge are critical and are a function of α as suggested by the present measurements.

3.4. Aerodynamic Loading on the Spoiler

Surprisingly, hardly any work has ever been reported on the aerodynamic loading on spoilers. Of those works that have reported flow past a normal spoiler on a flat plate or an aerofoil, only measurements of C_p on the flat plate or the aerofoil were reported, for instance, Barnes (1966), Siddalingappa & Hancock (1980) and McLachlan & Karamcheti (1982) etc.

3.4.1. Distribution of C_p around the spoiler

Fig 3.34 shows the distribution of the time mean pressure coefficients around the spoiler at $\alpha = 0^\circ$. The abscissa is labelled in x/h , which is the normalised distance on the spoiler surface from the surface of the aerofoil; $x/h = 0$ is therefore the point at which the spoiler meets the aerofoil surface, and $x/h = 1$ is at the tip of the spoiler. The square symbols represent the values of C_p obtained on the front surface of the spoiler, at the respective positions indicated on the abscissa, whereas the triangular symbols correspond to the values of C_p obtained on the rear surface of the spoiler. On the front surface of the spoiler, C_p increases from 0.75 to about 0.85 as the flow moves from $x/h = 0.125$ to 0.625, it then drops to a value of 0.63 at $x/h = 0.875$. On the rear surface, C_p remains fairly constant at a value of about -0.7.

It has been discussed that as a result of the adverse pressure gradient set up by the spoiler, the boundary layer on the upper surface of the aerofoil separates ahead of the spoiler and reattaches subsequently onto the spoiler itself, leaving a hinge bubble of recirculating flow. Furthermore, it is expected that the pressure inside the hinge bubble will vary little. As part of the spoiler is exposed to the hinge bubble, the values of C_p of the spoiler in this region should not deviate much from the values obtained on the aerofoil surface just ahead of the spoiler. To

illustrate, plotted in Fig 3.35 are the values of C_p in the vicinity of the hinge line of the spoiler. The point "0" on the abscissa corresponds to the location where the spoiler meets the surface of the aerofoil; those on the right are the measurement points on the spoiler in terms of the spoiler height and those on the left are the points on the aerofoil surface ahead of the spoiler, in terms of the aerofoil chord. It can be seen from this figure that ahead of the spoiler inside the hinge bubble, C_p varies very little from 0.775 at 0.525c to 0.8 at 0.66c. On the spoiler, the values of C_p remain about the same level as those on the surface of the aerofoil just ahead of the spoiler. Then C_p drops quite abruptly to a value of about 0.65 at 0.88h. This is probably due to the fact that the separated boundary layer reattaches onto the spoiler somewhere before 0.88h.

3.4.2. Distribution of C_p' around the Spoiler

To pursue the pressure loading on the spoiler further, the distribution of fluctuating pressure coefficient is presented in Fig 3.36. It can be seen that, on the front surface, the value of C_p' increases almost linearly, from a value of 0.025 at 0.125h to a value of 0.09 at 0.875h. On the rear surface, C_p' changes from 0.055 at 0.25h to 0.06 at 0.75h, and the level here is comparable to the level of C_p' obtained on the surface of the aerofoil immediately behind the spoiler. To relate C_p' on the front surface of the spoiler to those obtained on the surface of the aerofoil just ahead of it, a plot similar to Fig 3.35 is presented in Fig 3.37. On the front surface of spoiler near to the hinge line, C_p' is of the same order of magnitude as those C_p' just ahead of the spoiler. Also it is noted that C_p' increases towards the tip of the spoiler, due to the occurrence of flow separation at the tip of the spoiler.

The spectra of C_p are given in Fig 3.38 and Fig 3.39, for the front and rear surfaces respectively. Fig 3.38a shows the power spectrum of the pressure coefficient obtained at location $x/h = 1/8$ on the front surface of the spoiler. As usual, the shedding frequency manifests itself as the most dominant component in the spectrum, with the background level of the order of magnitude of about 10^{-4} , and the second harmonic at about 10^{-3} . It can be noted further from Fig 3.38a to Fig 3.38d that as x/h increases, the shedding component remains of

the same order of magnitude, whereas the background level increases to the order of 10^{-3} to 10^{-2} . The increase in the background level is thought to be due to the proximity of either the reattachment of the boundary layer on the spoiler surface or the flow separation at the spoiler tip.

It is also interesting to note that, besides the shedding component, there is a substantial amount of energy associated with reduced frequencies less than 0.5 or frequencies less than 40 Hz. In line with turbulence theory, this indicates the presence of strong large-scale eddies in the recirculating bubble.

On the rear surface of the spoiler, the spectra as shown in Fig 3.39 for $x/h = 0.25$ and $x/h = 0.75$ resemble each other. They exhibit the same features as those spectra obtained on the aerofoil surface immediately aft of the spoiler.

3.4.3. Variation of Spatial Mean C_p

Spatial mean pressure coefficients were also measured using the pressure averaging technique that has been discussed in the previous chapter. The four pressure tapings on the front surface of the spoiler were connected with tubings, which were passed through an opening on the aerofoil skin so that the tubings were connected together via a pressure averager inside the aerofoil. The pneumatically averaged output could then be measured, the coefficient thus obtained is known as C_{px} . Similarly, the two pressure tapings on the rear surface were connected to an pressure averager to provide another averaged pressure coefficient, C_{py} . C_{px} is equivalent to the force coefficient acting on the front surface, and C_{py} the force coefficient of the rear surface. Therefore, the resultant force coefficient acting on the spoiler can be obtained by taking the difference between C_{px} and C_{py} , i.e. $C_F = C_{px} - C_{py}$, where C_F is the coefficient of force acting on the spoiler in a direction perpendicular to the spoiler surface, with the spoiler height as the non-dimensionalising length scale.

The variations of C_{px} and C_{py} with angle of attack are given in Fig 3.40. It can be seen that, for C_{px} , it first remains fairly constant at 0.7 as the angle of attack increases from -14 to 0 degree, it then falls abruptly in a linear manner to about 0.2 as the angle of attack increases further to 14 degrees. On the other hand, C_{py} follows generally the same trend as the base pressure.

To explain the variation of C_{px} , it is first instructive to consider the manner in which the spoiler is presented to the flow. At 0 degree angle of attack, it is similar to the case where a normal spoiler mounted on a flat plate, as illustrated in Fig 3.41a. At negative angles of attack, it then resembles the case where a spoiler is inclined towards the free stream as shown in Fig 3.41b. Similarly, at positive angles of attack, the spoiler-aerofoil configuration resembles the case where a spoiler is inclined away from the flow as shown in Fig 3.41c.

At $\alpha = 0^\circ$, since the spoiler-aerofoil configuration is presented to the flow as depicted in Fig 3.41a, there is a compression set up due to the presence of the spoiler, and the effect of this compression is reflected in a value of about 0.7 for C_{px} . As the angle of attack decreases to any negative values, the spoiler-aerofoil configuration is presented to the flow as in Fig 3.41b, where the compression is even more severe, and in fact is of the same order as that at zero degree angle of attack, as indicated in the measurements of C_{px} . However, as the angle of attack increases beyond zero degree, the configuration resembles that as in Fig 3.41c, with less obstruction to the flow and hence a lesser compression, which results in the decrease of C_{px} .

3.4.4. Variation of C_F

As has been mentioned earlier, the coefficient of the resultant force acting on the spoiler can be obtained simply by the numerical difference between C_{px} and C_{py} , and is known as C_F . The variation of C_F with angle of attack thus obtained is shown in Fig 3.42. In the range of angles of attack between -8 and 14 degrees, where no flow separation on the lower surface ahead of the trailing edge occurs, the maximum value of C_F is 1.5 at $\alpha = 0^\circ$. Basically at angles of attack greater than

zero degree, C_{py} remains relatively constant, and so C_F follows the variation of C_{px} . At negative angles of attack, where C_{px} remains constant, C_F follows the trend of C_{py} .

3.4.5. Variations of C_{px}' and C_{py}'

The variation of fluctuating pressure coefficient with the angle of attack, for both C_{px}' and C_{py}' , is shown in Fig 3.43. The highest value of C_{px}' attainable is 0.11 at 14 degree angle of attack. The value of C_{px}' then declines steadily as the angle of attack is decreased, and a minimum value of $C_{px}' = 0.04$ is reached at zero angle of attack. As the angle of attack decreases further, C_{px}' increases slightly. For C_{py}' , it varies little in the range of positive angles of attack, and remains at a level of about 0.06. However, at negative angles of attack, C_{py}' does increase to a high value of about 0.1.

The way in which C_{px}' and C_{py}' vary is best illustrated through their respective spectra. To start with, a typical set of spectra obtained at $\alpha = 0^\circ$ is given in Fig 3.44. In Fig 3.44a where the spectrum of C_{px} is plotted, a dominant shedding component of the order of magnitude of about 10^{-1} is evident. A very weak second harmonic of the shedding component can also be identified, with a magnitude not much higher than that of the background level. The background level is of the order of about 10^{-3} to 10^{-4} , except at $k < 0.4$ or $f < 40$ Hz, where the level is higher than 10^{-3} . The high energy content in this frequency range indicates the presence of strong large-scale eddies in the hinge bubble. On the rear surface the spectrum, as plotted in Fig 3.44b, shows that there is a very strong shedding component, of the order of magnitude of about 10^{-1} , and a second harmonic of the order of about 10^{-3} to 10^{-2} , and the broadband fluctuations roll off steadily from a level of about 10^{-2} at low reduced frequencies to a level of about 10^{-4} at high reduced frequencies.

As the angle of attack increases, the magnitude of the shedding component in the C_{px} spectrum remains generally of the same order of magnitude, as shown in Fig 3.45a, 3.46a and 3.47a, for angles of attack of 6, 8 and 12 degrees. There is an increase in the broadband level,

however, as the angle of attack increases. At 6 degree angle of attack as shown in Fig 3.45a, there is still a substantial amount of energy associated with $k < 0.4$, but the energy at other reduced frequencies also increases, to a level of 10^{-3} , where the second harmonic is effectively buried in the background fluctuations. At even higher angles of attack, the level of the broadband fluctuations generally increases, and attains a level near to 10^{-2} at 12 degree angle of attack. The spectrum at this angle of attack, as shown in Fig 3.47a, is almost similar to that for "white noise", with the superimposition of a slightly stronger shedding component. To explain the increase in the background level, it is recalled that the flow on the upper surface of the aerofoil first separates ahead of the spoiler, and reattaches onto the spoiler, giving rise to a hinge bubble. As the angle of attack increases, the adverse pressure gradient on the upper surface of the aerofoil becomes stronger, and hence the separation point on the aerofoil surface ahead of the spoiler moves upstream, which results in the elongation of the hinge bubble. Therefore the increase in the background level basically indicates the presence of a large hinge bubble with strong fluctuations.

Just as the background level of C_{px} increases as the angle of attack increases, so does the background level of C_{py} . From Fig 3.44b, Fig 3.45b, Fig 3.46b and Fig 3.47b, it is obvious that although the shedding component remains at a level of about 10^{-1} , the level of the broadband fluctuations increases in intensity, to the extent that the second harmonic is buried in them at $\alpha = 12^\circ$ onwards.

At negative angles of attack, the features first remain unchanged and are similar to the case at $\alpha = 0^\circ$. Fig 3.48 shows the spectra at -6 degree angle of attack, where the familiar features similar to those at $\alpha = 0^\circ$ are evident. However, these features degenerate as the angle of attack decreases further to -10 degrees. In Fig 3.49, where the angle of attack is -10 degrees, it can be seen that there is no well defined shedding frequency. On the front surface, as in Fig 3.49a, there is a hump at $k < 0.5$ in the spectrum, and the broadband level is also higher than those obtained at 0 and -6 degree angle of attack. The hump in Fig 3.49a is probably due to the influence of the large-scale eddies in the hinge bubble. On the rear surface, there is generally a roll off in the spectral

density as k increases, as is shown in Fig 3.49b, from a level between 10^{-2} to 10^{-1} , to a level of about 10^{-4} . It is believed that, at this angle of attack, the boundary layer on the lower surface starts separating ahead of the trailing edge, with the exact point of separation moving about, thus changing the shedding process. This can then be viewed as a transition in the shedding process, with a change from shedding at a discrete frequency to disorganised shedding over a range of frequencies.

At $\alpha = -12^\circ$ as shown in Fig 3.50, distinct peaks can now be identifiable. At this stage, the separation point on the lower surface of the aerofoil is well ahead of the trailing edge, and its position is also relatively more stable, a well defined shedding frequency can then be identified. On the rear surface of the spoiler, the spectrum in Fig 3.50b reveals two dominant peaks. The first peak, at about $k = 0.2$, is associated with the shedding frequency. The second peak, which is the second harmonic of the shedding frequency, is located at $k = 0.4$. The second harmonic is stronger than the shedding component because as the rear surface of the spoiler is exposed to the near wake, the influences of the rolling up of alternate vortices from both the TSL and BSL are felt equally on the rear surface, which results in the strong second harmonic. On the front surface, although peaks can also be identified at the reduced frequencies of 0.2 and 0.4, the appearances are not very distinct, probably due to the influence of the large-scale eddies in the hinge-bubble.

As the angle of attack decreases further to -14 degrees, it is clear from Fig 3.51 that the second harmonic diminishes, and a distinct peak with a rather wide bandwidth can be seen in Fig 3.51b. At this angle of attack, due to the geometry of the aerofoil-spoiler that is presented to the flow, the separation on the lower surface of the aerofoil is well ahead of the trailing edge, at least much more ahead than that at -12 degree angle of attack. It is thought that the geometry of the near wake changes in such a way that the influence of the rolling up of the BSL is not "felt" strongly on the surfaces of the spoiler, and hence the second harmonic in the spectra diminishes.

One final remark, the separation ahead of the trailing edge on the lower surface of the aerofoil commences at -10 degrees according to the measurements of loading on the spoiler. In previous measurements with a thin metallic spoiler, the separation ahead of the trailing edge on the lower surface of the aerofoil commences at -12 degrees. This difference is thought to be due to the fact that the wooden spoiler used in the present measurement is 9 mm thick. At high negative angles of attack, the afterbody of the spoiler is significant and may alter the interaction between the TSL and BSL, resulting in the discrepancy.

3.4.6. Variation of C_F'

The variation of C_F' with angle of attack is plotted in Fig 3.52. The minimum C_F' occurs at zero angle of attack, with a value of about 0.06. As the angle of attack increases, C_F' increases as well, and a value of about 0.14 is reached at 14 degree angle of attack. At negative angles of attack, C_F' generally increases as the angle of attack decreases. Essentially the variation in C_F' can be explained in terms of the variations of C_{px}' and C_{py}' that have been discussed in the previous sections and will not be repeated here. To complete the discussion, a typical spectrum of C_F obtained at 0 degree angle of attack is also given in Fig 3.53.

4. PRESSURE MEASUREMENTS FOR RAPIDLY ACTING SPOILERS

4.1. General Remarks

Although it has been well observed that when a spoiler is activated rapidly a strong vortex is shed from the spoiler tip which, during its passage over the aerofoil surface, momentarily increases the lift, little work has been done in the detailed study of how this vortex affects the transient lift. Earlier works which have reported the adverse lift effect include:

Siddalingappa & Hancock (1980), which places emphasis on the measurement of surface pressures at selected positions on the aerofoil. However, due to the long pressure tubings employed in the measurement, which limit the frequency response, the scope of the work is limited to mainly a qualitative one.

Mabey et al (1982), which includes a limited amount of work in the study of C_{La} , t_f and t_a , at selected T , with no detailed study of the evolution of the initial vortex.

Consigny et al (1984) and Costes et al (1986), which include some brief accounts of the adverse lift effect.

More recently, experiments have been conducted at the University of Bristol to determine the effect of various configurations of spoilers on the adverse lift effect. These works are reported mainly in Kalligas (1986) and Myers (1990). However, there are some shortcomings in their measurement technique as has been discussed in Chapter 1, and therefore their results may not reflect accurately the actual magnitude and phase of the adverse lift.

The present investigation differs from those mentioned above in that emphasis has been placed on the effect of the initial vortex, on a fixed geometrical configuration of spoiler and aerofoil. All the results reported in this chapter are for a spoiler located at $x_s/c = 0.7$, with no gap between the spoiler and the aerofoil surface. The final deployment angle of the spoiler, δ_{ss} , is either

90° or 30° with respect to the aerofoil chord, with $\alpha = 0^\circ$ and $U_o = 25 \text{ m/s}$ ($Re = 4.2 \times 10^5$).

4.2. Time Histories of $C_L(t)$ at different t_d

To begin with, a typical set of results is shown in Fig 4.1 to define some common terms that will be used frequently in the subsequent discussion. The time history or trajectory of the spoiler angular movement is shown in Fig 4.1. The time axis is labelled as t , with $t = 0 \text{ s}$ corresponds to the instant 10 ms prior to the spoiler activation, i.e. activation of the spoiler occurs at $t = 10 \text{ ms}$. Also labelled in this figure are t_i , the time when the spoiler begins its movement; t_d , the length of the deployment time and t_e , the duration from the onset of the spoiler activation to any time t . The initial spoiler angle is -10° as the spoiler lies on the surface of the aerofoil at its closed position. Shown in Fig 4.1b is the time history of $C_L(t)$, where $C_L(t)$ is the lift coefficient at any t . Also included in this figure are t_f , C_{La} and t_a , where t_f is the duration from the onset of the spoiler activation to the instant when the lift falls to 95% of the average steady lift, C_{La} is the adverse lift coefficient defined as the value at the peak in the trace of $C_L(t)$ and t_a is the duration from t_i to the time when C_{La} is attained.

The time histories presented in Fig 4.1 are for the case of spoiler deployment to 90° with $t_d = 20 \text{ ms}$, or $T = 2$, where $T = U_o t_d / c$. Although the aerofoil is symmetrical and positioned at $\alpha = 0^\circ$, zero lift is not achieved before the spoiler is activated and C_L prior to the activation is -0.13 . This is due to the presence of disturbance caused by the spoiler. The lift does not respond until about 1 ms after the spoiler is activated. Soon after the change in lift, $C_L(t)$ increases until a peak is reached at $t_a = 10.4 \text{ ms}$, with $C_{La} = 0.26$. $C_L(t)$ decreases thereafter towards a steady mean level, at which alternate vortex shedding results in a superimposed regular pattern of lift oscillation. Also noted is that t_a is less than t_d , which suggests C_{La} is reached before the spoiler stops moving. The relationship between the spoiler deployment and C_{La} is a complex subject, and will be discussed in detail later in conjunction with the instantaneous pressure distributions. In the following sections, $C_L(t)$ for both the spoiler deployments to 30° and 90° at different T will be discussed.

4.2.1. Deployment to 90°

On deployment to 90°, different activation times were used, varying from $t_d = 11$ ms to $t_d = 110$ ms, giving a range of $T = 1.1$ to 11, and the corresponding time histories of $C_L(t)$ are given in Fig 4.2. At the quickest activation time, $t_d = 11$ ms or $T = 1.1$, it can be seen from Fig 4.2a that there is a highly complex pattern of $C_L(t)$. In fact it is difficult to identify the appropriate t_a , for there are two marked peaks, which differ only slightly in magnitude from each other. This is because in the course of the spoiler activation, there is an initial vortex which, during its passage over the surface of the aerofoil aft of the spoiler, exerts varying degrees of suction on the aerofoil. There is also an overshooting in compression ahead of the spoiler due to the inertial effect on the fluid as a result of the spoiler acceleration. The resultant effect of suction and compression, coupled in a very quick activation time, gives rise to the two peaks. This transient effect will be discussed in full in a later section.

To explain the variation of $C_L(t)$ after C_{La} is reached, it is vital to explain how the flow pattern around the aerofoil changes with time. When the spoiler starts moving, a strong initial vortex is formed from the spoiler tip. As the initial vortex moves along the surface of the aerofoil, it exerts a suction over that part of the aerofoil and hence an increase of lift to C_{La} . When the initial vortex moves near the trailing edge of the aerofoil, it induces a second vortex from the shear layer separated from the lower surface of the aerofoil. This second vortex changes the circulation around the aerofoil, with a resultant increase of lift, labelled as B in Fig 4.1a. Furthermore, this second vortex, which is induced by the strong initial vortex, has a higher circulation than those in the steady state, and causes a comparatively higher value at B. Thereafter, the circulation readjusts itself at each subsequent cycle of vortex shedding until the regular oscillation in the circulation is reached at steady state.

At a slower activation time of $t_d = 15$ ms or $T = 1.5$ as shown in Fig 4.2b, double peaks no longer exists. The magnitude of C_{La} is about the same as that for $t_d = 11$ ms. Here again the second peak B is still prominent, an indication that the second vortex, induced by the strong

initial vortex, is still strong in strength. Also noted at both $T = 1.1$ and $T = 1.5$, there are considerable ripples on the lift traces, probably due to the transient oscillation of the separation bubble ahead of the spoiler, as a result of the severe compression caused by the spoiler at quick t_d .

In Fig 4.2c for the case of $t_d = 20$ ms or $T = 2$, the change in $C_L(t)$ is much more gentle and smoother, with the second peak still apparent. Although t_d is slower than that in the previous cases, the magnitude of C_{La} increases somewhat. In fact C_{La} will continue to increase as t_d increases further until a maximum value of C_{La} is reached at $t_d = 28$ ms or $T = 2.8$ as in Fig 4.2f, thereafter C_{La} decreases. Apparently C_{La} is not directly related to t_d , and this will be further discussed in later sections. Although the second peak is still visible here, its strength decreases as t_d increases further, until it reaches a magnitude similar to that of the steady state at $t_d = 40$ ms or $T = 4$ as shown in Fig 4.2i. Although C_{La} is not directly related to t_d , it declines for t_d greater than 28 ms, and is barely distinguishable at $t_d = 110$ ms or $T = 11$ as in Fig 4.2l, when the lift history resembles that of a quasi-steady state. Finally, it should also be noted that at all t_d , C_{La} always occurs before the spoiler stops moving.

4.2.2. Deployment to 30°

In the previous section for spoiler deployment to 90° , it is observed that C_{La} is always reached before the spoiler stops moving. It is therefore presumed that by reducing the final spoiler angle, C_{La} might be reached at or near the instant when the spoiler completes its deployment. A final spoiler deployment angle of 30° is chosen for this purpose, and modification to the activation system was carried out to provide the desired movement. The results for this deployment angle are shown in Fig 4.3, with other experimental conditions similar to those of deployment to 90° .

It can be seen from Fig 4.3 that, for each activation time, $C_L(t)$ always responds immediately after the spoiler is activated. As in the case of spoiler deployment to 90° , double peaks can be observed at the quickest activation time, $t_d = 10$ ms or $T = 1$, as in Fig 4.3a. The

maximum lift, C_{La} , is again always reached before the completion of the spoiler deployment. The second peak, which is due to the shedding of the second vortex, is also prominent at each t_d . At steady state, both the oscillation period and the magnitude are smaller than that of deployment to 90° , because the spoiler-aerofoil configuration is less bluff at the final spoiler angle of 30° . Again, as in the case of $\delta_{ss} = 90^\circ$, C_{La} first increases until a threshold t_d , whence C_{La} decreases as t_d increases further.

4.2.3. Spoiler Retraction

It is known that in the course of activating a spoiler rapidly, the adverse lift effect, which is due to the formation of a strong initial vortex, is only associated with the extension of the spoiler but not with the retraction of the spoiler. It is for this reason that the present emphasis is placed solely on the extension of the spoiler, where the initial vortex and its effect are studied. Nevertheless, a sample of a very rapid retraction of the spoiler from its open to its closed position is included for the completeness of the present study.

Shown in Fig 4.4 are the results of a spoiler retraction from 90° to -10° , in a deployment time of 20 ms or $T = 2$. It can be seen that, following the retraction of the spoiler, $C_L(t)$ starts to respond and increases to its final value shortly after the spoiler stops moving. There is no significant feature on this deployment, except some transient ripples that are due to the re-establishment of an attached flow, with the near wake aft of the spoiler and the hinge bubble being swept away.

4.3. Time Histories of Pressures over the Aerofoil

The series of graphs shown in Fig 4.5 and Fig 4.6 (Fig 4.5 for the upper surface and Fig 4.6 for the lower surface) are pressure traces taken at different locations around the surface of the aerofoil, for a spoiler deployment to $\delta_{ss} = 90^\circ$ in $t_d = 23$ ms or $T = 2.3$. The abscissa of each graph is plotted in the same manner as that in the previous section, i.e. the onset of spoiler

deployment is at $t = 10$ ms, and the ordinate is plotted in terms of pressure coefficient, C_p .

Immediately ahead of the spoiler at $x/c = 0.66$ (location 11), it can be seen from Fig 4.5k that after a very short while from the onset of the spoiler activation, C_{p11} responds by increasing its magnitude, until a peak is reached before decreasing to a steady state value. This is because when the spoiler accelerates and moves, it causes a corresponding acceleration in the fluid ahead of it, which results in an additional inertial force in the fluid, causing the overshooting in C_p . The momentary overshoot raises C_p as high as 1.2. After the spoiler stops accelerating, the inertial force of the fluid diminishes, which explains the decrease of C_p to a steady state value. Also noted is that the value of C_p at steady state is higher than the value of C_p before t_i . This is due to the setting up of an adverse pressure gradient by the presence of the spoiler, which causes the formation of a hinge bubble. When the spoiler stops moving, the hinge bubble takes some time to settle down into its final size, and this transient oscillation of the bubble manifests itself in terms of the ripples observed in Fig 4.5k. Further ahead of the spoiler at $x/c = 0.53$ and $x/c = 0.6$, or locations 9 and 10 respectively, the peaks in overshoot of C_p are still clearly identifiable. At locations further away ahead of the spoiler, the pressure traces tend more to a quasi-steady state, where C_p increases steadily to its final value.

The graphs shown in Fig 4.5l to Fig 4.5p are all for C_p measured at the locations on the upper surface of the aerofoil aft of the spoiler. The prominent feature of these graphs is the indication of strong suction, due to the formation of the initial vortex. Of these five locations, locations 11 to 13 lie directly underneath the spoiler when the spoiler is at its closed position, and hence are subject to the suction exerted by the spoiler when it moves away at a high acceleration, and at a later time, they are also influenced by the initial vortex when the vortex starts to grow. As such, a kind of plateau can be observed in each of these pressure traces. At location 15, as a result of the passage of the initial vortex, a suction peak with a value of -3 occurs at about $t_e = 20$ ms, well before the spoiler stops moving. In fact, at the instant when the spoiler stops moving, C_p has already increased to -1, much nearer to the steady state level. At location 16, the effect of maximum suction is even more pronounced, with a maximum suction level of -7.5, again at a time before the spoiler stops moving. A computer simulation performed by Kalkanis (1988), based on a discrete

vortex method, of the flow generated by a rapidly activated spoiler, suggests that this low pressure is caused indirectly by the strong vortex shed from the spoiler tip, which induces a secondary vortex at the aerofoil trailing edge, and causes a momentary reverse flow from the lower surface of the aerofoil near the trailing edge into the region on the upper surface near the trailing edge, as shown in Fig 1.39. Finally, it should also be noted that the steady state level of all five pressure signals remains more or less the same at 0.8, indicating a near constant base pressure in the near wake.

On the lower surface of the aerofoil, it can be seen from the series of graphs in Fig 4.6, where selected pressure traces are shown, that the influence of the initial vortex is the strongest near the trailing edge at $x/c = 0.97$, as in Fig 4.6a, where a complex process of circulation re-adjustment takes place, and the steady state is attained only after the spoiler stops moving. After that the influence due to alternate vortex shedding persists. Far away from the trailing edge, C_p just decreases to a steady state value monotonically, as shown in Fig 4.6d to Fig 4.6h. One point should be noted here, however, that C_{p17} is in phase with C_{p16} , which is shown in Fig 4.5p, especially at steady state where alternate vortex shedding prevails, showing that the flow is attempting to satisfy the Kutta condition.

As these traces are all synchronised with the onset of the spoiler activation, their resultant should produce the transient lift and be the same as that obtained through the pressure averagers as has been shown earlier in the tests of the reliability of the pressure averagers. Therefore, the variation of the transient lift is best understood if it is discussed in conjunction with the pressure traces. It is noticed from the previous sections that, suction peaks on the aerofoil surface aft of the spoiler occur at different times, and there is also a compression overshoot ahead of the spoiler. Therefore, the instant for the occurrence of C_{La} may not relate directly to the strongest effect of the initial vortex. Rather C_{La} is related to the transient effect of individual pressure traces, which give rise to a resultant lift of complex profile. This relationship will be discussed again in a later section.

To further illustrate the effect of the activation of the spoiler, shown in Fig 4.7 and Fig 4.8 are some selected pressure traces of the upper and the lower

surface of the aerofoil respectively, for the case when the spoiler is activated to a final deployment angle of $\delta_{ss} = 30^\circ$ in $t_d = 10$ ms or $T = 1$.

Again, ahead of the spoiler at location 11 as shown in Fig 4.7i, there is an overshoot in compression following the activation of the spoiler, and a maximum value of 0.7 is reached just before the spoiler stops moving. This overshoot in compression ahead of the spoiler on the aerofoil slowly dies away as the location of measurement moves away from the spoiler towards the leading edge, as shown in the traces of C_{p10} to C_{p1} . Also prominent in these traces is the presence of severe ripples on the pressure traces just after the spoiler stops moving, due to the transient oscillation of the hinge bubble. Aft of the spoiler, the effect of the initial vortex is clearly identifiable in terms of strong suction peaks in Fig 4.7j to Fig 4.7n. The causes of the trend in these pressure traces have been discussed in the previous paragraph and will not be repeated here.

On the lower surface as shown in Fig 4.8, patterns similar to the case of deployment to 90° can be observed. Again, the causes of variation are similar to those of $\delta_{ss} = 90^\circ$, and hence will not be elaborated here.

4.4. Distributions of Transient Pressures

In another perspective, the influence of the rapidly acting spoiler is presented in terms of the pressure distribution around the aerofoil in one of the following two forms:

- I. Pressure distributions at different instants are presented in the same graph for either only the upper or lower surface of the aerofoil. This is to enable comparison between the pressure distributions at different instants to be carried out.
- II. Pressure distribution over the whole aerofoil at any one time is presented. This is to study how the pressure distribution contributes to the resultant lift.

The pressure distributions, as presented in the form described in I, for a spoiler deployment angle of 90° in 23 ms or $T = 2.3$, are shown in Fig 4.9a to

Fig 4.9d for the upper surface of the aerofoil, and Fig 4.10a to Fig 4.10d for the lower surface of the aerofoil. In Fig 4.9a are the pressure distributions of the upper surface from $t_e = 0$ ms to $t_e = 10$ ms. At time $t_e = 0$ ms, that is before the spoiler starts to move, the pressure distribution does not resemble that for a symmetrical aerofoil, and there is a slight discontinuity in the pressure distribution at $x/c = 0.7$ on the upper surface, due to the presence of the spoiler. Clearly from this figure, the pressures behind the spoiler are the first to respond when the spoiler commences moving, as at $t_e = 2.5$ ms, whereas those pressures ahead of the spoiler remain unchanged. At a later time at $t_e = 5$ ms, pressure signals in the region from $x/c = 0.7$ to $x/c = 0.9$ have changed considerably from their initial values, although the spoiler has completed only 22% of its deployment at this instant. This indicates the suction effect due to the moving away of the spoiler and the formation of the initial vortex originating from the tip of the spoiler. However, the effect of the initial vortex has not reached $x/c = 0.97$ yet. In fact, C_{p16} does not change much until after $t_e = 7.5$ ms, at which time the spoiler has already completed 33% of its deployment. For these pressures aft of the spoiler, their values continue to increase in magnitude as the spoiler moves. The initial vortex is already apparently strong at $t_e = 10$ ms or 44% of the spoiler deployment, as evident by the high suction value of C_{p15} . For those points ahead of the spoiler in this figure (Fig 4.9a), there is no appreciable change in their magnitude, as the spoiler has only completed 44% of its deployment by the end of $t_e = 10$ ms, at which time the inertial force of the fluid caused by the acceleration of the spoiler is yet to be felt in this region.

In Fig 4.9b are plotted the pressure distributions for the upper surface at times from $t_e = 10$ ms to 20 ms. The pressure distribution at 10 ms has been shown in Fig 4.9a and is included here for the purpose of comparison. As the spoiler moves on from 44% to 87% of its deployment, the pressure signals ahead of the spoiler begin to experience the compression set up by the acceleration of the spoiler, and increase their values, setting up an adverse pressure gradient between $x/c = 0.4$ and $x/c = 0.7$. Aft of the spoiler, the pressure signals near the trailing edge, i.e. C_{p15} and C_{p16} , respond dramatically to the passage of the initial vortex, and in fact a value of -7 is reached in C_{p16} at $t_e = 17.5$ ms, with a very severe pressure gradient set up near this region. At $t_e = 20$ ms C_{p16} drops to a value of -1, which is an indication that the most severe effect has in fact been reached before the spoiler completes 87% of its deployment.

The pressure distributions of the upper surface from $t_e = 20$ ms to 30 ms are given in Fig 4.9c. Ahead of the spoiler from $x/c = 0.5$ to $x/c = 0.66$, maximum values in C_p are attained at $t_e = 20$ ms, before the spoiler stops moving. The values of C_p then decrease steadily, indicating the subsidence of the overshoot in compression due to the slowing down of the spoiler. The initial vortex induced by the rapidly moving spoiler has a considerable effect on the lift, which manifests itself in the large pressure variation aft of the spoiler, as apparent in the figure from $x/c = 0.7$ to $x/c = 1$. The most drastic change occurs at $x/c = 0.97$, due to its proximity to the trailing edge.

Finally as t_e tends to large values, as shown in Fig 4.9d, a steady state is reached, as at $t_e = 100$ ms, when the only influence comes from alternate vortex shedding.

The corresponding pressure distributions of the lower surface from $t_e = 0$ ms to 10 ms are shown in Fig 4.10a. Clearly from this figure, there is not much change until after $t_e = 5$ ms. After that, the most appreciable change in the pressures occurs near the trailing edge, due to the re-adjustment of circulation as a result of flow separation there. From $t_e = 10$ ms to 20 ms as shown in Fig 4.10b, all pressure values increase in the suction level generally. Again the changes are the most prominent near the trailing edge, owing to the influence of flow separation and vortex interaction there. In fact a reverse flow into the upper surface round the trailing edge is expected here. As t_e increases further from 20 ms onwards, the pressure distribution changes gradually to that of a steady state, as evident in Fig 4.10c and Fig 4.10d.

However, the above discussion only looks at how the pressure distribution changes from one instant to the other. A better insight can be gained if the pressure distributions are presented in Form II which, as described above, reveals how the area enclosed by the pressure distribution curve gives rise to the transient lift. It is therefore proposed to study the transient pressure distribution at any one instant in close association with the overall lift curve. It is for this reason that the lift curve for activation to $\delta_{ss} = 90^\circ$ in $t_e = 23$ ms is re-plotted in Fig 4.11. Marked in this figure are the points labelled as A to G, the instants at which the corresponding pressure distributions, which are shown in Fig 4.12, will be studied.

Shown in Fig 4.12a is the pressure distribution at $t_e = 0$ ms, which corresponds to point A in Fig 4.11, when the spoiler commences its deployment, or rather the spoiler is still at its closed position. Clearly from Fig 4.12a the pressure distribution is unlike that of a symmetrical aerofoil, though the aerofoil itself is symmetrical. This is due to the presence of the spoiler, which changes the pressure distribution and causes a slight discontinuity at $x/c = 0.7$. This explains the cause of a negative lift before the activation of the spoiler.

In Fig 4.12b is shown the pressure distribution at $t_e = 5$ ms, when the changes occur mostly aft of the spoiler. The pressures ahead of the spoiler and those over most of the lower surface of the aerofoil remain virtually unchanged. By looking at the area enclosed by the curve, it is noticed that a small amount of positive lift is present, due mainly to the contribution from $x/c > 0.7$. The corresponding point B in Fig 4.11 does indeed indicate a small amount of positive lift.

As the spoiler progresses from 22% ($t_e = 5$ ms) to 33% ($t_e = 7.5$ ms) of its movement, it is observed from Fig 4.12c that the pressures aft of the spoiler respond quite drastically, especially at $x/c = 0.9$, except at $x/c = 0.97$ where C_p changes only very slightly. The considerable change in the region aft of the spoiler, as has been explained before, is due both to the suction created by the moving away of the spoiler, which has the strongest effect on C_{p12} and C_{p13} , and to the initial vortex, which has the strongest effect on C_{p14} and C_{p15} . At this instant, when the spoiler has only completed 33% of its deployment, the influence of the initial vortex has not reached $x/c = 0.97$ yet, which explains the slight variation in C_{p16} . As for the pressures ahead of the spoiler, there are little changes in their magnitude. And by examining the pressure distribution, one can estimate that there is a considerable degree of positive lift, which arises mainly from the contribution of the positive area enclosed by the distribution curve at $x/c > 0.7$. This can be confirmed in Fig 4.11 at point C, which reveals a value of 0.2 in $C_L(t)$.

Comparing Fig 4.12d and Fig 4.12c, it is observed that the pressure distribution at $t_e = 10$ ms remains little changed except at $x/c = 0.9$ and 0.97 , where there are increases in the suction levels, giving rise to an even higher lift, as is evident at point D in Fig 4.11. In fact the lift at this instant corresponds to

the maximum lift achievable, i.e. C_{La} , at the present rate of deployment. It is clear from Fig 4.12d that C_{La} occurs at a time when the pressures ahead of the spoiler and over most of the lower surface change little, but the suction influence is very strong aft of the spoiler.

At $t_e = 15$ ms, as shown in Fig 4.12e, the pressures ahead of the spoiler now start to increase due to the compression caused by the acceleration of the spoiler. Most of the pressures aft of the spoiler decrease in magnitude, i.e. less suction, except at $x/c = 0.97$ near the trailing edge, where C_p increases its suction level. At the same time, the pressures on the lower surface near the trailing edge from $x/c = 0.7$ to $x/c = 0.97$ also increase their suction levels. It is evident that although there is still a substantial positive area enclosed by the distribution curve after $x/c = 0.7$, this is offset by the negative area enclosed by the curve ahead of the spoiler, resulting in almost a zero lift, as indicated at point E in Fig 4.11.

At $t_e = 17.5$ ms, as shown in Fig 4.12f, the most prominent feature is the strong suction value at C_{p16} . This high suction value has already been presented in the trace of C_{p16} in Fig 4.5p. However high this suction level is, it has little influence in the overall lift, as the suction values of C_{p12} to C_{p15} have already decreased to a comparable level to those of the lower surface. Furthermore, the compression ahead of the spoiler is now relatively high. It is therefore clear from the area enclosed by the distribution curve in Fig 4.12f that a negative lift is expected, as indeed shown in Fig 4.11 at point F, where a value of -0.7 in $C_L(t)$ is recorded.

As t_e progresses further to 20 ms, C_{p16} decreases its suction level as shown in Fig 4.12g. Although the spoiler at this instant is still moving, it is already in the process of decelerating, and the influence of the suction is fading. Ahead of the spoiler, there is a propagation of the compression effect upstream, resulting in the general increase of C_p there. The corresponding point G in Fig 4.11 indicates that $C_L(t)$ decreases further, due to the general increase of compression ahead of the spoiler.

Also included in Fig 4.12h is the pressure distribution at steady state, at $t_e = 100$ ms, which is indicated as point H in Fig 4.11, where alternate vortex shedding prevails.

In Fig 4.13 are the pressure distributions at different instants for a spoiler deployment to $\delta_{ss} = 30^\circ$ in 10 ms. The resultant $C_L(t)$ is also shown in Fig 4.14, and those points marked A to D correspond with the instantaneous pressure distributions shown in Fig 4.13a to Fig 4.13d respectively. This is included as a further illustration of how the pressure distributions are related to the lift and will not be discussed in detail. Again it should be noted that C_{La} occurs at a time before C_{p16} reaches maximum suction.

4.5. Variation of C_{La} with T

Plotted in Fig 4.15 is the variation of C_{La} with T, where T is defined as $t_d U/c$, for two spoiler deployment angles. The square symbols are for the case of deployment to 90° and the diamond symbols for deployment to 30° . For the case of deployment to 90° , C_{La} first attains a value of about 0.25 at $T = 1.1$, which is the quickest activation time. As the activation time is increased, C_{La} first increases until at about $T = 2.5$, at which time the maximum C_{La} of 0.33 is reached. Thereafter C_{La} decreases as T increases, until a value slightly larger than the value of $C_L(t)$ before the spoiler activation is reached at $T = 11$. It is expected that as the activation rate decreases further, a value of C_{La} equal to $C_L(t)$ before the onset of the spoiler activation will be reached, in which case there is no longer any adverse lift effect. A similar trend can also be observed for the case of spoiler deployment to 30° . This time, however, C_{La} peaks at $T = 1.5$ with a value of 0.36. Thereafter, it declines quickly and reaches a value of -0.1 at $T = 4$. In this figure, it is clearly illustrated that, in both cases of spoiler deployment angles, C_{La} is not linearly related to the rate of spoiler deployment.

In order to explain the trend of C_{La} , it is expedient to recap here how C_{La} arises. As has been explained earlier, each position aft of the spoiler will experience the suction effect when the spoiler moves, and the maximum suction effect at each location occurs at different times. The adverse lift arises as a result of the average suction effect over the portion of the aerofoil aft of the spoiler. It occurs at a time before the considerable effect of compression is felt ahead of the spoiler, and before the maximum suction peak occurs near the trailing edge. Therefore, it is reasonable to argue that C_{La} occurs at an optimal instant when the resultant of the compression, which contributes to a negative

lift, and the suction, which contributes to a positive lift, yields the maximum lift. To put simply:

$$C_{La} = \text{average compression ahead of the spoiler} + \text{average suction aft of the spoiler}$$

The average compression in the above expression is related directly to t_d . The shorter the deployment rate, the higher the acceleration, and hence the higher the inertial effect on the fluid, giving rise to a higher compression. The spatially average suction aft of the spoiler is a correlation between different suction pressures at different locations. Therefore the maximum average suction effect does not necessarily occurs at the same instant when the local absolute maximum C_p aft of the spoiler occurs.

Although the previous graph shows the variation of C_{La} with T , it is not suitable to compare the results between different deployment angles, δ_{ss} . For instance at $T = 2$, although the deployment times are the same for both $\delta_{ss} = 90^\circ$ and 30° , the acceleration and velocity in the former case is much higher than the latter. It is for this reason that the variation of C_{La} is re-plotted with U_{tip}/U_o as the abscissa, as shown in Fig 4.16, where U_{tip} is the average velocity of the spoiler tip and U_o is the free stream velocity. It appears in the figure that there is a threshold U_{tip}/U_o of 0.08 where the two sets of curves crossed over. At $U_{tip}/U_o < 0.08$, C_{La} for $\delta_{ss} = 30^\circ$ is always greater than that for $\delta_{ss} = 90^\circ$, and vice versa for $U_{tip}/U_o > 0.08$. Of course if U_{tip}/U_o were to extend to 0, C_{La} for both δ_{ss} would converge at the same value as that of $C_L(t)$ before the spoiler activation.

4.6. Variation of t_a/t_d with T

One of the important parameters in the study of the adverse lift effect is the time taken to attain C_{La} . A non-dimensionalised parameter, t_a/t_d , is defined for this reason to indicate the extent of the relative spoiler movement. First the variation of t_a/t_d with T for two different spoiler deployment angles is presented in Fig 4.17. In both cases of δ_{ss} , it is noted that t_a/t_d is never greater than one, an indication that C_{La} is always attained before the spoiler stops moving. It is for this reason that the variation of t_a/t_d is also represented in Fig 4.18 as the

variation of δ_a with T , where δ_a is the instantaneous angular spoiler position when C_{La} is attained.

In the case of $\delta_{ss} = 90^\circ$, t_a/t_d does not exceed values higher than 0.55, indicating that C_{La} is always reached before the spoiler completes 55% of its deployment. This translates into a δ_a of 43° as shown in Fig 4.18. In both Fig 4.17 and Fig 4.18, the trend of either t_a/t_d or δ_a for $\delta_{ss} = 90^\circ$ reaches high values at low deployment rates, and falls away steadily as T increases. Similarly in the case of $\delta_{ss} = 30^\circ$, t_a/t_d and δ_a do not exceed 0.8 and 23° respectively. Also as T increases towards ∞ , C_{La} decreases until it can no longer be detected at low deployment rates.

As has been explained in a previous section, C_{La} occurs at a time when the contributions from the compression ahead of the spoiler and the suction aft of the spoiler provide the optimal result. t_a/t_d therefore does not necessarily indicate the maximum effect caused by the initial vortex. It should rather be considered in conjunction with the way the vortex grows and moves. The initial vortex is associated with a convection velocity, U_c , which is related to the free stream velocity U_o . Only when the initial vortex remains in the vicinity of the aerofoil surface can the adverse suction effect be felt. If U_c is assumed more or less constant, then the duration when the initial vortex remains in the vicinity of the aerofoil surface should be more or less the same. If it is assumed that the convection velocity of the initial vortex is the same for all T , then t_a will give an indication of the optimal position of the initial vortex when the adverse lift is reached through the relationship of $U_c \cdot t_a$. It is for this reason that the non-dimensionalised parameter $U_o t_a / c$ is plotted against T in Fig 4.19. In this figure it can be seen that there is a range of T between 2 and 4 where $U_o t_a / c$ remains fairly constant. At low T , the values of $U_o t_a / c$ deviate from the main trend not only because the vortex is fully formed before it is swept away but also because the compression ahead of the spoiler is achieved much quicker.

Finally, also included in the Fig 4.20 is the variation of δ_a with U_{tip}/U_o , for the purpose of comparison between different final spoiler deployment angles.

4.7. Variation of t_f/t_d with T

In Fig 4.21 is presented the variation of t_f/t_d with T for both $\delta_{ss} = 30^\circ$ and 90° . As has been defined earlier, t_f is the time taken from the onset of the spoiler activation to the instant when 95% of the average steady lift is reached, so t_f is a direct measure of the duration of the transient effect. Clearly t_f/t_d decreases as T increases, and approaches the value of one as T tends to ∞ . It is also seen from the figure that the values of t_f/t_d are high at low T , which indicates that the transient effect is directly related to the rate of deployment. For example at $T = 1.1$, it takes four times the value of t_d to reach steady state for $\delta_{ss} = 30^\circ$. Also presented in Fig 4.22 is the variation of t_f/t_d with U_{tip}/U_o . For the same U_{tip}/U_o , t_f/t_d in the case of $\delta_{ss} = 30^\circ$ is always higher than in the case of $\delta_{ss} = 90^\circ$.

Again t_f is also presented in Fig 4.23a, as a non-dimensional parameter, $U_o t_f/c$, versus T . As T tends to ∞ , $U_o t_f/c$ should be equal to T , so a straight line of gradient one is superimposed onto Fig 4.23a to indicate the range of quasi-steady state. At low T , it appears that values of $U_o t_f/c$ cluster in the region of 5 and 6 for $\delta_{ss} = 90^\circ$ and 30° respectively. This cluster is enlarged and plotted in Fig 4.23b, where it can be seen that values of $U_o t_f/c$ remain in narrow bands centred at 5 and 6 for $\delta_{ss} = 90^\circ$ and 30° respectively.

4.8. Selected Pressure Traces

Tests were conducted at selected pressure measurement locations to determine the effect of T on them. The pressure measurement locations which were identified as critical are locations 11, 16 and 17. C_{p11} is the pressure measured ahead of the spoiler at $x/c = 0.66$, giving some indication of how T affects the fluid inertial force. C_{p16} and C_{p17} are pressures measured near the trailing edge at $x/c = 0.97$ on the upper and lower surface of the aerofoil respectively, which enable the study of the trailing edge condition to be made. All tests were conducted for $\delta_{ss} = 90^\circ$ and the graphs are all plotted with the onset of the spoiler activation at $t = 10$ ms.

Shown in Fig 4.24 are the traces of C_{p11} , C_{p16} and C_{p17} for $t_d = 11$ ms or $T = 1.1$. In Fig 4.24a, a maximum compression of 1.75 in C_{p11} is reached

just before the spoiler stops moving, indicating the large fluid inertial pressure caused by the acceleration of the spoiler. After the spoiler stops moving, C_{p11} declines quickly to its steady state value of 0.7. In Fig 4.24b for C_{p16} , a very high suction level of -8.5 is reached after the spoiler stops moving. It is thought that this suction is due to the passage of the initial vortex, which in turn induces a secondary vortex from the bottom shear layer (BSL) that is separating at the trailing edge. The corresponding C_{p17} , which is shown in Fig 4.24c, also attains a maximum suction value of -1.4 at the same instant when C_{p16} is at its suction peak. Comparing the values of C_{p16} and C_{p17} at this instant, it is apparent that a strong pressure gradient is set up around the trailing edge, which causes a strong reverse flow in the form of the secondary vortex, a phenomenon which is demonstrated in the numerical simulation of Kalkanis (1988). This strong reverse flow is an indication of a possible temporary violation of the Kutta condition. Thereafter, the vortex shedding mechanism as described by Gerrard (1966) takes over, and a steady state of vortex shedding is reached after some transient re-adjustment of circulation. The transient re-adjustment occurs because the circulation shed by the initial vortex is not compatible with the circulation in the steady state. After the steady state is reached, it can be noticed that C_{p16} and C_{p17} appear to be equal both in magnitude and in phase, satisfying Kutta condition.

Before the discussion of C_p at other values of T is carried out, it is deemed necessary to digress into the mechanism of vortex formation. When the spoiler is fully extended, the spoiler-aerofoil configuration resembles that of an unsymmetrical bluff body. The vortex shedding mechanism is not unlike that described by Gerrard (1966), which attributes it to the interaction between the opposing separated shear layers. As has been explained previously in the discussion of vortex shedding from a stationary spoiler, one important parameter governing the vortex shedding is the projection height (H) of the spoiler, the smaller the H , the shorter it takes for the opposing shear layers to interact. In the course of a rapidly acting spoiler, H is not fixed and is a variable of t , i.e $H(t)$. The rate of change of $H(t)$, $\frac{dH}{dt}$, is dependent on T . Therefore, at a very large T , which gives a very low $\frac{dH}{dt}$, before the top shear layer (TSL) from the tip of the spoiler has time to roll up into an initial vortex, it already interacts with the BSL, and hence will not result in any adverse lift effect. On the other extreme, when T is very small, $\frac{dH}{dt}$ is high and there is hardly any time for the TSL and the BSL to interact before the TSL grows into an initial vortex. The roll up of the TSL into an initial vortex is due to the very large suction created

by the rapid moving away of the spoiler. Therefore at a T of 1.1, there is a low suction value in C_p due to the presence of the initial vortex. Whereas at $T = 4$, as will be shown later, there is a less severe suction level due to the interaction of the opposing shear layers at low $\frac{dH}{dt}$.

Shown in Fig 4.25 are the pressure traces taken at $T = 1.5$. The compression (due to the fluid inertia force) is still strong and reaches a maximum of 1.6 just before the spoiler stops moving, as shown in Fig 4.25a. Thereafter, C_{p11} declines quickly to its steady state value. The traces of C_{p16} and C_{p17} are shown in Fig 4.25b and Fig 4.25c respectively. At the present rate of deployment, although it is slower than the previous case when $T = 1.1$, a higher suction peak is observed in Fig 4.25b with a value of -9.2, whereas there is a slight decrease in C_{p17} as compared with the previous case.

As T increases further, the magnitudes of the compression peak in C_{p11} and the suction peaks in C_{p16} and C_{p17} decrease, as is apparent in Fig 4.26, Fig 4.27, Fig 4.28 and Fig 4.29 for T equal to 2, 2.5, 2.8 and 4 respectively. Clearly, at the slowest activation rate with $T = 4$, it can be seen from Fig 4.28b and Fig 4.28c that there are already oscillations in C_p before the spoiler stops moving. This is due to the interaction of the opposing shear layers at low $\frac{dH}{dt}$, and as a result multiple vortices are shed while the spoiler is still moving. The occurrence of the strong suction peak in this case is due to the fact that part of the TSL is able to roll up into a strong vortex which, while it traverses across the aerofoil surface, exerts a considerable suction. More on the relationship between the deployment rate and the level of suction pressure will be discussed later in conjunction with the velocity measurements.

To illustrate how fast the first vortex reaches location 16 at $x/c = 0.97$, a new parameter t_v is defined as the time taken from the onset of the spoiler deployment to the instant when the first suction peak, which is due to the passage of the first vortex, occurs in C_{p16} . The variation of $U_o t_v/c$ with T is shown in Fig 4.30 and is similar to the variation of $U_o t_a/c$ with T as shown previously in Fig 4.19. It can be seen that for a range of T from 1.5 to 2.5, $U_o t_v/c$ remains more or less constant at a level of 1.6, which shows that the first vortex takes about the same time to reach location 16 for this range of T . It also seems that as T increases towards large values, $U_o t_v/c$ falls gradually. The

parameter t_v is also plotted as a variation of t_v/t_d against T in Fig 4.31. In this figure, t_v/t_d decreases monotonically as T increases.

5. RESULTS OF HOT WIRE MEASUREMENTS

5.1. General Remarks

It has been stated in the earlier chapters that, up to the present time, relatively little work has been done on the effect on the aerofoil of the initial vortex shed as a result of a rapidly moving spoiler. On the fundamental study of the initial vortex itself, no experimental work has ever been carried out to study its evolution and the associated flow field. There are, though, some other works on the study of the flow field of an aerofoil with a stationary spoiler, e.g. Lee & Bodapati (1987), and of a moving spoiler on a flat plate, e.g. Nelson et al (1990). The present investigation aims to establish the relationship between the initial vortex and the adverse lift effect. For this reason, measurement was carried out in the wake of an aerofoil with a rapidly acting spoiler to obtain ensemble averaged descriptions of the initial flow field. The results reported here were obtained by using hot-wire anemometry, which has already been discussed in Chapter 2. All measurement was conducted at a location of $x/c = 0.5$ or 125 mm downstream from the trailing edge of the aerofoil, and the Reynolds number is the same as in pressure measurement, that is, $Re = 4.17 \times 10^5$.

5.2. Time Mean Values for a Statically Deployed Spoiler

The mean velocity quantities in the wake behind an aerofoil with a stationary spoiler have been investigated by Lee & Bodapati (1987). The results obtained in the present study are similar in trend to those reported in Lee & Bodapati (1987). They are included here for the completeness of the present investigation, and therefore will not be discussed in great detail.

Shown in Fig 5.1 are the mean velocity profiles of u and v across the wake at $x/c = 0.5$ downstream from the trailing edge of the aerofoil. In Fig 5.1a, the parameter U^* , which is the mean u velocity non-dimensionalised by the free stream velocity U_o , is plotted against y/c , where y is the normal distance measured away from a line coincident with an extension of the aerofoil chord. The x - y co-ordinate system with respect to the aerofoil has been illustrated in Fig 2.11. The variation of U^* resembles a typical wake pattern,

with a minimum value of about 0.7 at $y/c = 0.2$, and increases towards 1 at either side of the wake. It can also be gathered from this profile that the width of the wake is about 80% of the aerofoil chord length. When the spoiler is positioned in a direction normal to the aerofoil chord, the spoiler-aerofoil resembles a bluff body configuration with flow separation from the tip of the spoiler and the trailing edge of the aerofoil. The distance from the chord of the aerofoil to the tip of the spoiler is about $0.2c$, which gives a physical centre line at $y/c = 0.1$. However, the virtual centre of the wake, as is observed from the figure, is located at $y/c = 0.2$ where U^* is at its minimum. The deviation between the virtual centre and the physical centre could be due to the different flow separation angles at the two separation points: the separation angle at the spoiler tip is 90° with respect to the flow direction and it is 0° at the trailing edge of the aerofoil.

Shown in Fig 5.1b is the mean v velocity profile, again plotted as a non-dimensional parameter, V^* ($V^* = V/U_o$) against y/c . The velocity just outside the wake at $y/c = -0.2$ is 0.2, whereas V^* is nearly zero at the other side of the wake at $y/c = 0.7$. This indicates that, as a result of the unsymmetrical bluff body configuration, fluid is drawn across the wake towards the side where upstream the spoiler is located.

Root mean square values of u and v were also calculated and are presented in Fig 5.2 as non-dimensional groups of u'/U_o and v'/U_o vs y/c . In Fig 5.2a for u'/U_o , a familiar wake pattern is observed, with maximum peaks within the wake, and near zero values outside the wake. Values as high as 25% of the free stream velocity can be observed at the two peaks located at $y/c = 0.05$ and 0.33 , which are near the locations of the vortex centres. In Fig 5.2b, high values of v' occur near the centre of the wake, where the interaction between the alternate vortices is the strongest, and a maximum value of $v'/U_o = 0.4$ is reached at $y/c = 0.15$.

5.3. Measurements of Velocity Vector Fields

Through the hot-wire measurement performed across the wake behind the aerofoil, velocity vector fields were obtained for two activation times, which are $T = 1.1$ and 2.3 . For each measurement, it was repeated eight times and an ensemble averaging was performed over these eight runs to produce the final

result. The results are presented in terms of velocity vectors in a coordinate frame of y/c and time. On the time axis the time of $t = 0$ is defined at a time 10 ms prior to the activation of the spoiler. The velocity fields plotted in this way for the activation time of 11 ms or $T = 1.1$ are shown in Fig 5.3, where the scale of the velocity vectors is plotted in such a way that a length of 10 mm gives a magnitude of 90 m/s. In Fig 5.3a is the velocity vector field for $t = 20$ to 50 ms and in Fig 5.3b for $t = 50$ to 80 ms. The activation of the spoiler commences at $t = 10$ ms and the deployment is completed at $t = 21$ ms. However, since the measurement was conducted at a distance $0.5c$ aft of the trailing edge and the disturbances caused by the spoiler take time to travel downstream, no change occurs until after $t = 20$ ms. It is for this reason that the velocity vectors before $t = 20$ ms are not included.

In Fig 5.3a, there is no visible change in the velocity vectors until around $t = 29$ ms, when flow with high curvature is observed. At around 32 ms, the perturbation of the velocity vectors due to the movement of the spoiler occurs across almost the entire range of y/c from -0.12 to 0.68. Thereafter, there are continuous changes as t increases, and finally a regular pattern of perturbation can be observed from $t = 56$ ms onwards. Although visible changes can be readily identified, it is difficult to tell from these figures if there are any vortical structures present, since any vortical structures will be carried by the flow and the velocities associated with them will be small compared to the main flow component. In order to identify any vortical structures, a new frame of observation is needed. The new frame of observation is thus defined by subtracting a reference velocity away from the original u -velocity vectors. In the present case, a reference velocity equivalent to the convection velocity, which has a magnitude of $0.8U_o$, of the steady state vortices is chosen, and the resulting velocity vector fields are presented in Fig 5.4a and Fig 5.4b, where a length of 10 mm gives a magnitude of 45 m/s. Incidentally, the first few vortices shed following the activation of the spoiler have different convection velocities from the steady state vortices. These convection velocities will be illustrated in later sections.

By examining Fig 5.4a, where the spoiler is apparently to the right, it is clear that strong vortical structures are present in the flow field. A very strong vortical structure circulating in an anti-clockwise direction can be observed between $t = 29$ to 38 ms, with its centre of near zero velocity located at $y/c =$

0.32, $t = 33$ ms. This strong vortical structure is undoubtedly due to the passage of the initial vortex, which has been induced by the rapid movement of the spoiler. Following immediately behind the first vortex is another vortex, rotating in a clockwise direction. Although this second vortex has originated from the BSL, which is separating at the trailing edge of the aerofoil at $y/c = 0$, it appears that the second vortex has been drawn across the wake to the same level of y/c as that of the first vortex. This is apparently an induced effect and the vortex pair is convected together. Because the two vortices are located at the same level of y/c , the fluid between them is being squeezed into a strong jet, pumping fluid from the lower range to the higher range of y/c .

The flow field at some later time following the activation of the spoiler is shown in Fig 5.4b, for $t = 50$ ms onwards. There are regular pockets flow circulating in either a clockwise or an anti-clockwise directions, suggesting the presence of an alternate vortex street. The perturbation caused by the steady state vortices is less severe than by the initial vortex pair, and hence it only affects a small range of y/c , from 0 to 0.3, compared to $y/c = -0.12$ to 0.68 affected by the initial vortex.

In Fig 5.5 is shown the velocity vector field following the activation of the spoiler with a deployment time of 23 ms or $T = 2.3$. As in the previous case, the time axis is defined with $t = 0$ corresponding to the instant 10 ms before the activation of the spoiler. That is also to say, the spoiler starts moving at $t = 10$ ms and stops at $t = 33$ ms. As there are no significant changes in the velocity vector field before the perturbation caused by the deployment of the spoiler propagates downstream, only velocity vectors after $t = 30$ ms are shown. In this figure, visible changes begin at $t = 29$ ms, and there are pockets of flow with high curvature at $t = 32, 41$ and 44 ms. Again, in order to identify any vortical structures, it is necessary to subtract a reference velocity, which is equal to $0.8U_0$, away from the velocity vectors, and the resulting flow field is shown in Fig 5.6. It is clear from Fig 5.6 that a vortical structure with very strong circulation in an anti-clockwise direction exists at $t = 32$ ms and $y/c = 0.08$, although it spans across only a small range of y/c . A second vortex with a clockwise rotation, is located at $y/c = 0.08$ and $t = 40$ ms. There is also a third vortex, rotating in an anti-clockwise direction, located immediately behind the second vortex at $t = 44$ ms, $y/c = 0.4$. There are two features to be noted here. First, the interval between the instant when the flow field first

responds to the approach of the first vortex and the instant at which the third vortex occurs is less than the deployment time. Second, the first vortex is located at $y/c = 0.08$, near to the y/c -position of the trailing edge, whereas the third vortex is located at $y/c = 0.4$, well above the y/c -position of the tip of the spoiler when the spoiler is fully extended. The positions of these two vortices suggest that the first vortex is formed when the spoiler is inclined in a range of low angular positions whereas the third vortex is formed when the spoiler is at high angular positions. These two features therefore suggest that the interaction between the TSL and BSL takes place before the spoiler stops moving. This phenomenon will be discussed in depth in a later section when the distributions of vorticity are studied in conjunction with the variation of the base pressure.

5.4. Time Histories of u^* and v^* across the Wake

The discussion turns now to the study of the individual velocity time traces. Velocity measurement was carried out across a range of y/c in steps of 5 mm at 51 positions. In view of the large quantities of data collected, only half of them will be presented here. Shown in Fig 5.7 are the traces of u -velocity from $y/c = -0.2$ to 0.76 for $t_a = 11$ ms. In each of these traces, the velocity u is non-dimensionalised by U_o , which is the free stream velocity, and is denoted as u^* . On the time axis it is again defined with $t = 0$ corresponding to the instant 10 ms before the activation of the spoiler. In all traces, since the spoiler does not begin its movement until $t = 10$ ms, there should be little variation in the velocity traces before $t = 10$ ms. However, in the range of y/c from -0.08 to 0.12 , there are some fluctuations in u^* . This, as has been explained before, is due to the presence of the spoiler. Although the spoiler lies flat on the aerofoil surface at its closed position, its thickness causes a mild flow separation, which in turn causes a small wake behind the aerofoil.

Clearly the disturbances caused by the initial vortex do not reach the measurement stations immediately after the activation of the spoiler. Even when the disturbances arrive at the measurement stations, they do not do so at exactly the same time. Those measurement stations near $y/c = 0$ are the first to respond, and those further away at some later time. From the velocity vector plot shown earlier in Fig 5.4, it is observed that the vortex centre of the strong initial vortex is located at $t = 33$ ms, $y/c = 0.35$. Since the initial vortex is rotating in an anti-clockwise direction, those velocities measured above $y/c =$

0.35 respond to the initial vortex by increases in their values and for those velocities measured below $y/c = 0.35$, values of u^* are reduced by the approach of the initial vortex. At those measurement stations above $y/c = 0.35$, it can be noticed that the influence of the initial vortex is very strong indeed, and the increases in the values of u^* are thus very high.

After the passage of the initial vortex and the settling down to steady state vortex shedding, a familiar oscillation pattern, like those of the bluff bodies, can be observed. The frequency of the fluctuations is estimated to be about 90 Hz, which agrees well with the pressure measurement results discussed earlier. It can also be noticed that, within a range of y/c , there are ripples superimposed on top of the regular shedding pattern, indicating the measurement stations are located within the turbulent wake. Only at the virtual centre of the wake at $y/c = 0.2$ can a regular oscillation pattern due to the influence of the alternate vortices not be recognised, .

The traces of v^* for $t_a = 11$ ms are shown in Fig 5.8, in a similar manner as those for u^* presented in the previous section. Features similar to u^* can also be observed here. Before the activation at 10 ms, there are no significant variations in the velocity traces. Even after the activation of the spoiler, there are still no immediate variations in the traces of v^* until after a certain time when the perturbation of the initial vortex is felt. Also noted is that in the range of y/c from -0.08 to 0.08, there are signs of turbulence in the traces of v^* before the activation of the spoiler, due to the disturbances caused by the presence of the spoiler.

As the initial vortex approaches the measurement stations, the first effect of it is to induce a downward flow throughout the range of y/c , as the initial vortex is rotating in an anti-clockwise direction. This can be seen in the decreases in all the values of v^* at about 30 ms. Thereafter, v^* increases at each measurement station and reaches its first peak value. At each of those stations in the range of negative y/c , the first peak value is at a comparable level to its steady state fluctuations. Whereas at any measurement station in the range of positive y/c , the first peak value is comparatively much stronger than its steady state value. Throughout the range of y/c , the first peak values vary in a range from 0.3 to 0.8. It is noted in the velocity vector plot that the strong initial vortex induces a strong second vortex which, together with the first

vortex, squeeze the fluid between them into a jet-like structure. The first peak values are therefore associated with this jet-like structure.

5.5. Distributions of Vorticity

As the velocity vector plots indicate the presence of vortical structures, and the resulting flow field can best be explained with respect to these structures, it is therefore essential to study the flow field in terms of the distributions of vorticity. The vorticity is calculated assuming Taylor's hypothesis, as outlined in Chapter 2, and is presented here in terms of contours of constant vorticity. The vorticity values are non-dimensionalised as ω^* , with

$$\omega^* = \frac{\omega c}{U_o}$$

where ω is the vorticity, c is the aerofoil chord length and U_o is the free stream velocity. The y/c and time axes are plotted in such a way that if one unit of t is multiplied by U_c , where U_c is the convection velocity of the steady state vortices, the resulting $U_c \cdot t$ is equal to one unit of y/c . This is done so that the actual physical sizes of the vortices can be represented realistically.

For a start, the distribution of vorticity following the deployment of the spoiler with $t_a = 11$ ms or $T = 1.1$ is presented in Fig 5.9 for the duration from $t = 20$ to 45 ms. Clearly from the figure, there is a vortical structure of strong vorticity levels spanning across a large range of y/c , with its strongest vorticity level located at $t = 33$ ms, $y/c = 0.35$, in line with the strong circulating flow observed earlier in the velocity vector plot. This initial vortex is located above $y/c = 0$ and spans from $y/c = 0$ to 0.6. Its centre, within a contour level of $\omega^* = -10$, is located at $t = 33$ ms and $y/c = 0.35$. The vortical structure is elongated and inclined at an angle away from the y/c axis, which suggests that its shape may have been influenced by the trajectory of the spoiler movement. Immediately behind this vortex, there is a smaller second vortex, although its highest ω^* level is comparable to that of the initial vortex. The centre of the second vortex, within a contour level of 10, is located at $t = 39$ ms, or 6 ms away from the centre of the initial vortex. The y/c -position, though, is situated at the same level as that of the initial vortex. Therefore, the second vortex, which has originated from the BSL separating at the trailing edge of the aerofoil

at $y/c = 0$, is strongly influenced by the initial vortex that it is drawn across the wake.

Also shown in Fig 5.10 is the vorticity distribution at steady state, from $t = 60$ to 85 ms. In this figure, the magnitude of the highest vorticity contour level is at about 8, compared to 10 for the first two vortices in Fig 5.9. The vortex street is confined within a range of y/c from 0 to 0.4, with the cores of the clockwise rotating vortices at $y/c = 0.15$ and those of the anti-clockwise rotating vortices at $y/c = 0.25$. Clearly the cores of the first two vortices following the activation of the spoiler are located at much higher y/c -positions than those at the steady state. This is probably due to the different vortex formation mechanisms involved. For the initial vortex, it is formed when the spoiler is moving rapidly away from the surface of the aerofoil. While the spoiler is moving away at high acceleration, a region of very low suction is created aft of the spoiler, which in turn causes the TSL separating from the tip of the spoiler to roll up into a strong initial vortex. As the initial vortex traverses past the trailing edge of the aerofoil, the BSL is influenced by the strong induced velocity of the initial vortex to roll up into the second vortex. On the other hand at steady state, a near wake region of recirculating flow is established aft of the spoiler, and the vortices are formed as a result of the interaction between the TSL and BSL away from the trailing edge according to the formation mechanism described by Gerrard (1966), with destruction of vorticity in the near wake region.

To compare the various vortex strengths, values of circulation for the relevant vortices were obtained by performing vorticity integration, and they are presented here in terms of a non-dimensional parameter:

$$\Gamma^* = \frac{\Gamma}{U_0 c}$$

where Γ is the circulation, U_0 is the free stream velocity and h is the length of the aerofoil chord. The value of Γ^* calculated in this way for the initial vortex is thus equal to 0.86. The corresponding Γ^* for the second vortex and each of the steady state vortices are 0.57 and 0.33 respectively. The velocities of their vortex centres were also estimated to be $0.73U_0$ and $0.63U_0$ respectively, as opposed to $0.8U_0$ for the steady state vortices. Comparing these figures, the strength of the initial vortex is clearly very strong, 30% more than

the second vortex and more than double the strengths of the vortices at steady state. It is the suction exerted by this initial strong vortex that causes the adverse lift effect.

At steady state, it is known that the rate at which circulation is fed into the separated boundary layer is equal to $\frac{1}{2} U_b^2$, where U_b is the flow velocity at the separation point, and is related to C_{pb} , the base pressure, by $U_b^2 = U_o^2 \cdot (1 - C_{pb})$. Since C_{pb} and the period of vortex formation are known from the pressure measurements, it can be shown that the total amount of circulation from each separation point produced in a cycle of vortex formation is $\Gamma_T^* = 1.01$, by using the following formula:

$$\Gamma_T^* = \frac{U_o \cdot (1 - C_{pb})}{2 \cdot c \cdot f}$$

where f is the shedding frequency. According to the mechanism of vortex formation proposed by Gerrard (1966), part of the circulation from the separated shear layer is fed into the near wake region and is destroyed half a period later, part is fed into the opposite shear layer and is annihilated, and the remainder is fed into the growing vortex. Therefore it is interesting to note that, in the present case at the steady state, only 33% of the circulation shed in each cycle of vortex formation is retained in the vortex.

Although it is noted that the strength of the initial vortex is very strong and so has a profound effect on the adverse lift effect, it cannot be related directly to the adverse lift. As will be shown later, although the initial vortex in the case when $t_a = 23$ ms has a smaller strength than that in the present case, the resulting adverse lift is much higher. It has been explained earlier in the discussion of pressure measurements that the cause of the adverse lift is due to the changes in the overall pressure distribution, which involves the spatial mean suction aft of the spoiler and the compression ahead of it. So a higher value of Γ^* does not necessarily mean a higher C_{La} . What can be related here, though, is the absolute magnitude of each of the individual pressures aft of the spoiler. Therefore, the strength of the initial vortex will instead be studied closely in conjunction with the variation in the time history of C_{p16} , where C_{p16} is the pressure coefficient obtained at $x/c = 0.97$ on the upper surface of the aerofoil. It is for this reason that the trace of C_{p16} is shown here again in Fig 5.11.

In Fig 5.11, it can be seen that following the activation of the spoiler at $t = 10$ ms, C_{p16} first increases its suction level, until a value of -8.5 is reached at about $t = 26$ ms. This high suction value is clearly due to the passage of the first vortex, which has a circulation strength of $\Gamma^* = 0.86$. The maximum suction effect occurs at a time 14 ms after the spoiler starts moving or 4 ms after the spoiler stops moving. This shows that the initial vortex, which is fed by the TSL originating from the tip of the spoiler, convects past location 16 on the aerofoil at $x/c = 0.97$ only after the spoiler stops moving. This also means that the BSL can only interact with the initial vortex and the TSL after the spoiler stops moving. After the passage of the initial vortex, the pressure increases to a local maximum value of about -1 at $t = 33$ ms, due to the shedding of the second vortex. Thereafter, as t increases, C_{p16} settles down into the steady state with regular oscillation pattern.

The trend of C_{p16} described above is in good agreement with the variations in the traces of velocity. A plot of u^* at $y/c = 0.72$ is therefore included in Fig 5.12 for comparison. The peak value of u^* observed in this figure is due to the passage of the initial vortex and the first local minimum value is due to the second vortex. The interval between them is 7 ms, which is equal to the interval between the occurrence of the minimum pressure and the first local maximum pressure in the plot of C_{p16} .

The discussion of C_{p16} and u^* above might seem platitudinous, but it serves as a prelude to the following discussion for the case of $t_a = 23$ ms, when the study of the relationship between the variation of pressure and the passage of the vortices is critical. Shown in Fig 5.13 is the vorticity contour plot for $t_a = 23$ ms or $T = 2.3$, from $t = 30$ to 55 ms. From the velocity vector plot shown earlier in Fig 5.6, it is realised that strong vortical structures occur throughout the flow field, and this can be confirmed in the present vorticity contour plot. An initial vortex, with its centre located at $t = 32$ ms, $y/c = 0.05$, extends from $y/c = 0$ to 0.2, and from $t = 30$ to 35 ms. Following the first vortex, there is a smaller vortex, with maximum $\omega^* = 8$, located at $t = 40$ ms, and almost at the same level of y/c as that of the initial vortex. A much larger but more diffused vortex can also be identified between $t = 40$ to 45 ms and $y/c = 0.1$ to 0.5, with a strongest vorticity level of -6. The fourth vortex, at around $t = 53$ ms, is a transitional vortex towards the establishment of a steady state vortex street. It

is therefore proposed to place emphasis on the discussion of the first three vortices only.

Compared with the first vortex in the case where the deployment rate is $T = 1.1$, the first vortex here is much smaller in size physically, and is located slightly above $y/c = 0$, although the strongest vorticity contour level here is -16, much higher than the level of -10 in the case of $T = 1.1$. It is known that as the spoiler starts moving, the TSL separating from the tip of the spoiler begins to roll up into an anti-clockwise rotating vortex which, while continuing to grow in size, convects downstream across the aerofoil surface aft of the spoiler. Whether or not the spoiler has already stopped moving when the initial vortex convects past the trailing edge of the aerofoil depends on the rate of spoiler deployment. It is observed in Fig 5.13 that the first vortex, however, extends across a time interval of only 5 ms, far less than the activation time of 23 ms. Both the length and the location of this vortex suggest that it is formed and has its link severed from the TSL when the spoiler is still moving at low angular positions. This can be clearly illustrated with the aid of the velocity trace of u^* at $y/c = 0.72$, as shown in Fig 5.14. In this figure, u^* does not respond until about $t = 22$ ms. The first peak in u^* , due to the passage of the first vortex, occurs at about 30 ms, and the second peak, due to the passage of the third vortex, occurs at 43 ms. A careful study of the timings suggests that, if the first response of u^* at 22 ms is due to the onset of the spoiler deployment, then the third vortex is formed 21 ms after the spoiler starts moving. Therefore, it seems that the first three vortices, which are observed in Fig 5.14, are formed within the spoiler deployment time of 23 ms, as will be confirmed in the following study of C_{p16} .

In the absence of the BSL, the initial vortex will continue to grow both in size and in strength with the feeding of vorticity from the TSL, while at the same time, its centre will move downstream. However, with the presence of the trailing edge where the BSL separates, there is a possibility of interaction between the TSL and the BSL before the spoiler stops moving for slow activation times, as witnessed in the present case.

To understand the present phenomenon more clearly, the trace of C_{p16} is presented in Fig 5.15. It can be seen that, although the spoiler starts moving at $t = 10$ ms, its influence is first felt in C_{p16} at $t = 18$ ms, and the first suction

peak occurs at about $t = 26$ ms, indicating that the first vortex is formed 8 ms after the spoiler starts moving, which agrees well with the timings in the trace of u^* in Fig 5.14. The second suction peak, which is due to the passage of the third vortex, occurs at $t = 39$ ms or 21 ms after the spoiler starts moving, which also agrees well with the plot of u^* trace in Fig 5.14. As the suction effect of the first vortex is felt in C_{p16} at $t = 26$ ms or 16 ms after the spoiler starts moving, it therefore suggests that while the spoiler is still moving, the first vortex has already convected past the trailing edge of the aerofoil. As a result, the TSL, which has been feeding the first vortex, now has the opportunity to interact with the BSL by inducing it to roll up into a second vortex. At the same time when the interaction between the TSL and BSL is taking place, the source of vorticity feeding into the first vortex from the TSL, is cut off by the vorticity of opposing sign from the BSL. While all this is happening, the TSL, which separates from the tip of the moving spoiler, although with its link being cut off from the first vortex by the BSL, continues to be influenced by the suction created by the rapid moving away of the spoiler, and as a result rolls up to form a third vortex in the region aft of the spoiler but ahead of the trailing edge of the aerofoil. This third vortex then convects downstream and passes location 16 at $t = 39$ ms, which results in a second suction peak in C_{p16} with a value of -2. The interval between the first two suction peaks, due to the passage of the first and third vortices respectively, is 13 ms, which is also equal to the time between the centres of the first and third vortices in the vorticity contour plot shown in Fig 5.13.

It is also noted from the above argument that the TSL continues to roll up into a third vortex when its link with the first vortex is cut off by the BSL at the instant when the first vortex has just convected past the trailing edge. Therefore, it is reasonable to suggest that the second vortex and the third vortex grow simultaneously at different x/c locations, with the second vortex, which is being fed by the BSL, growing at a distance further downstream. As the third vortex is formed when the spoiler is at its later stage of deployment and hence at higher angular positions, it is therefore located at a higher level of y/c than those of the other two vortices. The simultaneous growth of the second and the third vortices also explains the proximity of the two vortex centres, which are only separated in time by 2.5 ms in the plot of vorticity contour. Finally, while the BSL is rolling up into the second vortex, part of its vorticity may be induced by the vorticity of the TSL and find its way into the region aft of the spoiler, which causes the distribution of vorticity of the third vortex to be more diffused.

The strength of each of the vortices is also calculated here for the purpose of comparison. The values of Γ^* for the first three vortices are 0.42, 0.19 and 0.43 respectively, compared to the steady state value of 0.33. Their vortex centre velocities were also estimated to be $0.7U_0$, $0.9U_0$ and $0.85U_0$ respectively, compared to the value of $0.8U_0$ for the steady state vortices. Although the value of Γ^* for the first vortex is far less than the value of 0.86 for the first vortex in the case of $T = 1.1$, it induces a suction level of about -8 in C_{p16} , only slightly less than the value of -8.5 in the case of $T = 1.1$. To explain this, it is essential to consider the parameters affecting the peak suction value. Clearly, the peak suction value is directly related to Γ^* and $\omega_{\max}^*$, and is inversely related to r , where r is the distance between the pressure measurement station on the surface of the aerofoil and the vortex centre. Hence,

$$C_{p16, \text{peak}} = F(\Gamma^*, \omega_{\max}^*, \frac{c}{r})$$

Clearly in the previous case where $T = 1.1$, although Γ^* is twice the value here, its value of $\omega_{\max}^*$ is only -10, and r is much larger. Whereas in the present case, although Γ^* is much less than the case of $T = 1.1$, $\omega_{\max}^*$ is higher with a value of about -16, and r is also smaller, thus giving a value of $C_{p16, \text{peak}} = -8$, marginally less than the value of -8.5 in the case of $T = 1.1$. Similarly, although the value of Γ^* for the third vortex is about the same as the first vortex, it has a lower level of $\omega_{\max}^*$ and is located farther away from the surface of the aerofoil, thus resulting in a C_p level of only -2.

5.6. Variations of Velocity Traces with t_a

To understand the effect of different activation times on the initial vortex, a number of measurements of velocity was obtained at $y/c = 0.72$. This location is chosen because it is located outside the turbulent wake and hence any variations in the velocity traces are due to the passage of vortices only. The activation times are $t_a = 11, 20, 23, 28$ and 40 , giving $T = 1.1, 2, 2.3, 2.8$ and 4 . These are also the activation times used in the measurement of pressures at selected locations as discussed in the previous chapter.

The traces of u^* velocity are presented in Fig 5.16a to Fig 5.16e, for $T = 1.1$ to 4 respectively. Common features can be observed in this series of

graphs: traces of u^* do not respond immediately after the spoiler starts moving and there is an attainment of steady oscillation state at large t . At high deployment rates, i.e small T , single dominant peaks can be observed, whereas at moderate T , double peaks can be observed, and finally at large T , irregularities in the traces of u^* can be observed before steady state is reached. Clearly the variations in the traces of u^* are closely related to the formation of the initial vortex/vortices, and it has also been demonstrated that the variations in the traces of u^* can best be understood if they are studied in conjunction with the corresponding C_{p16} . It is for this reason that traces of C_{p16} for $T = 1.1$ to 4 are also shown here in Fig 5.17a to Fig 5.17e. Marked in each of these figures is t_s , the instant when the spoiler stops moving. Although the relationship between the variations in u^* and C_{p16} for $T = 1.1$ has been discussed in the study of vorticity distributions, it is recapped here to aid the discussion for other values of T .

It can be seen from Fig 5.17a that, at $T = 1.1$, C_{p16} does not respond until $t = 18$ ms, and it reaches a peak suction value of -8.5 at $t = 26$ ms, just 15 ms after the spoiler stops moving. It can be considered that the duration between the instant when C_{p16} first responds and the instant when the suction peak occurs is equivalent to the time required for the initial vortex to form fully. Therefore the initial vortex takes about 8 ms to form, and it convects past measurement station 16 only after the spoiler stops moving. This also means that the BSL does not interact with the TSL, nor the first vortex, before the spoiler stops moving. Therefore only a strong first vortex, as observed in Fig 5.9, is identified. It is due to the passage of this initial vortex past the hot wire measurement location at $y/c = 0.72$ that a prominent peak in the trace of u^* is observed at $t = 33$ ms, or 23 ms after the spoiler starts moving. A trough is also observed in trace of u^* at $t = 40$ ms, 7 ms apart from the peak value. It is noted from the trace of C_{p16} that the first crest following the peak suction value also occurs after 7 ms from the peak suction. Therefore the trough in the trace of u^* is due to the passage of the second vortex. So in the present case, there is only one initial vortex, which is formed while the spoiler is moving.

At $T = 2$, it can be seen from Fig 5.16b that the trace of u^* exhibits the same features as in the case of $T = 1.1$, with a prominent peak value at $t = 36$ ms or 26 ms after the spoiler starts moving, and a trough in the trace of u^* at $t = 43$ ms or 7 ms after the passage of the first vortex. The corresponding C_{p16}

is shown in Fig 5.17b, and it can be seen from this figure that a peak suction value of -9.2 is attained at $t = 28$ ms, just before the spoiler stops moving at $t = 30$ ms. It must be noted that C_{p16} is measured at $x/c = 0.97$, and it is safe to assume that by the time the first vortex convects past the trailing edge at $x/c = 1$, the spoiler is near to its fully extended position. So, it is also safe to say that the BSL does not roll up into the second vortex while the spoiler is still moving. This is verified by the instant at which the first crest in C_{p16} following the peak suction occurs. The crest in C_{p16} , which is due to the shedding of the second vortex, occurs at $t = 35$ ms, 5 ms after the spoiler stops moving. Therefore, at the present rate of deployment, as in the case of $T = 1.1$, only one initial vortex is formed. Also, it is noted that the value of the peak u^* is 1.26, less than the peak value of 1.33 in the case of $T = 1.1$. As the peak value of u^* is directly related to the strength of the initial vortex, the lower value of peak u^* therefore suggests that the circulation strength of the first vortex here is less than that in case of $T = 1.1$.

The traces of u^* and pressure for $T = 2.3$ are shown in Fig 5.16c and Fig 5.17c respectively. As has been explained earlier in the discussion of vorticity distributions, the first vortex convects past the trailing edge of the aerofoil, causing the first hump in u^* at $t = 30$ ms, and a peak suction in C_{p16} at $t = 26$ ms, while the spoiler is still in the process of deployment. As a result, the BSL responds to the influence of the first vortex by rolling up into a second vortex and cuts off the link between the TSL and the first vortex. At the same time, while the TSL has severed its link with the first vortex, it continues to grow into a third vortex until the spoiler has completed its deployment. It is due to this third vortex that a second, much stronger crest is observed in the trace of u^* at $t = 43$ ms. There is also a second suction peak, albeit with much smaller level, occurs in the trace of C_{p16} at $t = 39$ ms. To relate the events to the deployment of the spoiler, it is noted that in Fig 5.17c, C_{p16} does not respond until $t = 18$ ms, and the second suction peak due to the passage of the third vortex occurs at $t = 39$ ms. If the initial response of C_{p16} at $t = 18$ ms is due to the onset of the spoiler deployment, then the third vortex is formed at $t = 39$ ms or 21 ms after the spoiler starts moving. Therefore, as the deployment time of the spoiler is 23 ms, all three vortices in the present case are formed while the spoiler is still moving, and hence they can all be considered as initial vortices. It is also noted that the two peak values in the trace of u^* , which are due to the passage of the first and third vortices, are at much lower levels compared with the case of $T = 1.1$. This is because the circulation strength of

the two vortices is much weaker than the first vortex at $T = 1.1$, as has been shown in the discussion of vorticity distributions.

At $T = 2.8$, the trace of u^* as shown in Fig 5.16d, exhibits a similar pattern to that of $T = 2.3$. Again, it can be deduced from the trace of C_{p16} that, as shown in Fig 5.17d, the first two suction peaks occur before the spoiler stops moving, thus three initial vortices are formed during the deployment of the spoiler. Also noted is that the magnitude of the first suction peak is less than that of the second suction peak, which indicates that the third vortex has a stronger circulation strength than the first vortex. It can also be seen from Fig 5.17b that C_{p16} does not respond until $t = 18$ ms, and the first suction peak is reached at about $t = 25$ ms or 15 ms after the spoiler starts moving. The time interval after the spoiler starts moving can be expressed in terms of the state of spoiler deployment, by taking the ratio of any duration from the instant when the spoiler starts moving to the spoiler activation time. By expressing the time at which the first vortex arrives at the trailing edge of the aerofoil in this way, it can be shown that the spoiler has completed only 15/28 or about 50% of its deployment before the BSL cuts off the supply of vorticity from the TSL to the first initial vortex.

In the previous case at $T = 2.3$, the first initial vortex reaches the trailing edge of the aerofoil at 16 ms after the spoiler starts moving, or when the spoiler has completed 16/23 or about 70% of its deployment. The strength of the vortex is related to Γ , which is in turn related to U_b :

$$\text{i.e. } \frac{d\Gamma}{dt} = F(U_b^2)$$

and U_b is related to the spoiler angular position, the higher the δ , the bluffer the section and the higher the U_b . So although at both deployment rates, i.e. $T = 2.3$ and 2.8, the first initial vortices take about the same time, i.e. 15 and 16 ms, to reach the trailing edge of the aerofoil, the spoiler in the case of $T = 2.3$ has a higher mean angular position and hence a higher mean U_b than the present case. Therefore the resulting Γ for the first vortex in the case of $T = 2.3$ is also higher than the present case, with the consequence that the first suction peak for the present case is lower than that at $T = 2.3$.

Also noted is that in the present case for $T = 2.8$, the spoiler has completed only 50% of its deployment when the first vortex reaches the trailing edge. For the remaining 50% of its deployment, a third vortex is formed. Although this third vortex is also formed during 50% of the spoiler deployment time, the spoiler is generally at much higher angular positions than its first 50% deployment, resulting in higher values of U_b and hence higher Γ . Therefore the third vortex is much stronger than the first vortex, giving a higher second suction peak at $t = 35$ ms.

Shown in Fig 5.16e and Fig 5.17e are the traces of u^* and C_{p16} for $T = 4$ respectively. In the present case, before the TSL has sufficient time to roll up into a strong initial vortex, the effect of the BSL is already felt, so alternate vortex shedding commences before the spoiler actually stops moving. This has also been explained in the previous Chapter in terms of the rate of change of the spoiler height. In the trace of C_{p16} , though, there is a suction peak at $t = 40$ ms. This is because during the later stage of the spoiler deployment where U_b is generally higher than at the initial stage of the spoiler deployment, part of the TSL is able to roll up into a vortex of relatively higher strength. This relatively strong vortex also registers its presence in terms of a peak u^* value at $t = 54$ ms in Fig 5.16e.

6. CONCLUSIONS AND RECOMMENDATIONS FOR FURTHER WORK

The experimental investigation into the aerodynamics of 2-D stationary and rapidly acting spoilers was successfully carried out. The experiments were performed in two phases. In phase one for an aerofoil with a stationary spoiler, measurement of surface pressures and forces was carried out, and both time mean and unsteady characteristics were studied. In phase two, measurement was carried out with the spoiler activated at different deployment times. Transient characteristics were studied in terms of the development of transient pressures and lift. In addition, hot-wire measurement was also carried out in the transient wake that is generated by the spoiler deployment to obtain information about the transient flow field and initial vortices shed by the moving spoiler.

In phase one for the stationary spoiler, experiments were conducted to determine the time mean and unsteady characteristics for the aerofoil and the spoiler separately. It was found that with the addition of the spoiler, the flow field around the aerofoil changes dramatically, with a compression ahead of the spoiler and a near wake with constant mean base pressure aft of the spoiler. The effectiveness of the spoiler in terms of lift reduction is evidently good across a range of aerofoil incidence. Only at high aerofoil incidence where the stalling phenomenon of the aerofoil occurs does the spoiler lose its effectiveness.

For a given aerofoil incidence, the magnitude of lift reduction depends both on the location and size of the spoiler. This is because the addition of the spoiler causes the aerofoil configuration to change from a streamlined body to a bluff body, with flow separation occurring at fixed separation points. If the configuration of the spoiler changes, there is a corresponding change in the bluntness of the body, which in turn causes the magnitude of lift reduction to change.

As the lift reduction is achieved by forcing flow separation to occur, unsteady characteristics arise inevitably. The unsteady features were measured in terms of the fluctuating pressures, the fluctuating forces and the vortex shedding frequencies. Basically, the unsteady flow is dominated by the vortex shedding phenomenon, although the hinge bubble and the near wake also play a significant part in local pressure fluctuations. The rate at which the TSL and BSL interact with each other depends on two parameters, the afterbody length,

l_a , and the spoiler projection height, H . However, l_a and H change in an opposing trend as the aerofoil incidence changes, and so no universal Strouhal number can be found, which makes the prediction of shedding frequency difficult.

Measurements of pressure loading on the aerofoil section near the trailing edge at $x/c = 0.97$ were also obtained. It was found that the change of pressure across the trailing edge is quite drastic, which suggests that there is a large pressure gradient between the two pressure measurement locations at $x/c = 0.97$ on both the upper and lower surface of the aerofoil. As the aerofoil incidence changes, the pressure gradient also changes tremendously, which poses a problem for numerical modelling.

On the spoiler itself, force coefficients in the range from 1 to 1.6 were observed for the range of aerofoil incidence tested. As for the unsteady characteristics, the fluctuations are dominated by the influence of vortex shedding. However, on the front surface, there is also a substantial level of random fluctuations, due to the presence of the hinge bubble.

In phase two, the measurement was carried out with the spoiler activated in a range of deployment times, from $T = 1$ to 10. When the spoiler is activated rapidly, i.e. at $T < 3$, the adverse lift effect can be observed. As the activation time tends to large values, the transient lift follows a quasi-steady variation, with no clear adverse lift effect. When there is an adverse lift effect, the magnitude of the adverse lift is not directly related to the deployment time, and the maximum adverse lift does not occur at the quickest activation time either. It was realised from the surface pressure measurements that the transient lift has two major contributions: the overshoot of compression pressure ahead of the spoiler and the high suction aft of the spoiler due to the passage of the initial vortex. C_{La} occurs at a time when the combination of these two parts give the most optimal result, and not necessary at the time when the absolute maximum suction peak occurs near the trailing edge of the aerofoil. It was also found that the maximum values of C_{La} for the two final spoiler angles tested are at a comparable level.

From the measurement of selected surface pressures at different deployment rates, it was noted that the initial vortex shed from the moving

spoiler causes a very high level of suction, as witnessed in the measurements of C_{p16} , at which a value as high as -9 was obtained. This high suction level of C_{p16} was observed over a range of T from 1 to about 2.5. Also, the suction peak of C_{p16} always occurs at a time before the spoiler stops moving, except at $T = 1.1$ when it occurs just after the spoiler stops moving. This suggests that at most deployment times, the first vortex sweeps past the trailing edge, and causes the BSL to interact with the TSL before the spoiler stops moving.

The transient flow fields in the wake developed following the activation of the spoiler to $\delta_{ss} = 90^\circ$ were obtained for two activation times through hot-wire measurements. Basically the transient flow field is dominated by the vortices that are formed while the spoiler is still moving.

It was found that at $T = 1.1$, a very large initial vortex with strong circulation was observed. This strong vortex causes the BSL to roll up into a second vortex, and the two vortices form a strong induced vortex pair. It is due to the strong first vortex that C_{p16} has a suction peak of -9. However at $T = 2.3$, three vortices are formed during the duration of spoiler deployment. The way of how these vortices come about has been explained in Chapter 5. Basically, it involves the interaction between the TSL and BSL. Although the first vortex for $T = 2.3$ has a smaller circulation than the initial vortex for $T = 1.1$, it is very near the aerofoil surface and causes a high suction peak of about -7.5 in C_{p16} .

To sum up, the stationary spoiler causes the aerofoil configuration to behave like a bluff body, dominated by the interaction between the TSL and BSL. When the spoiler is activated rapidly, the transient flow field, like the flow field generated by the stationary spoiler, is also dependent on how the TSL and BSL are allowed to interact. The TSL separating from the spoiler tip following the activation of the spoiler always rolls up into an initial vortex. The strength and size of this initial vortex, and the subsequent interaction with the BSL depends on the rate at which the spoiler moves or rather the rate at which the spoiler projection height, H , changes.

As the study of spoiler aerodynamics is essentially the study of how the TSL and BSL interact with each other, it is therefore recommended that further study could be done with emphasis placed on manipulating the way at which the

TSL and BSL interact. This can be done by non-solid spoilers or by having different activation profile instead of the ramp motion used in the present investigation. For example, it can be carried out with multiple-step activation as employed by Kalligas (1986). Instead of measuring the adverse lift, as Kalligas (1986) did, the emphasis of the study should be placed on how the multiple-step activation will disturb the interaction between the TSL and BSL. So, by understanding the way TSL rolls up and its subsequent interaction with the BSL, it will enable one to design appropriate activation profiles to either enhance or reduce the adverse lift effect.

In addition, further work can be carried out with oscillating spoilers, to see how different oscillating patterns can be used to enhance dynamic lift. Essentially, the employment of different oscillating patterns is just another way of controlling the interaction between the TSL and BSL. By controlling the interaction of the TSL and BSL continuously, the size of the shed vortices and the spacing between them can be altered to obtain the desired results.

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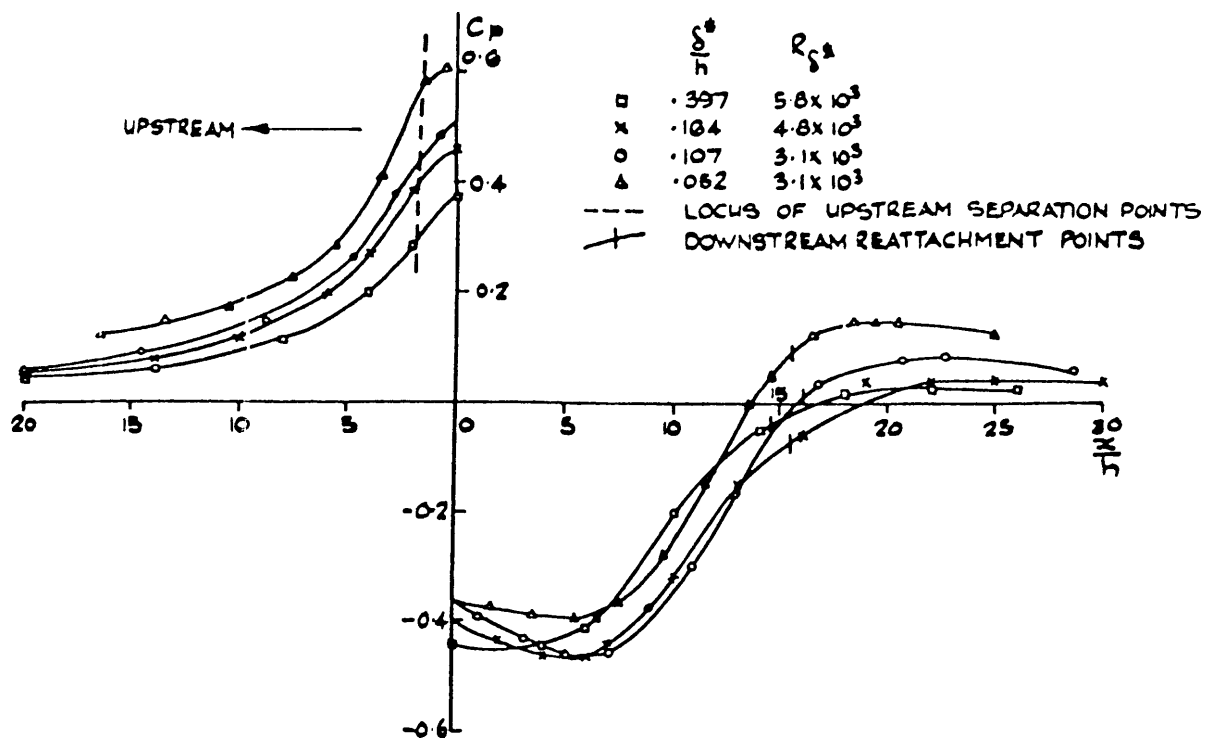


Fig 1.1 Pressure distributions on a flat plate due to the presence of spoilers, Barnes (1966)

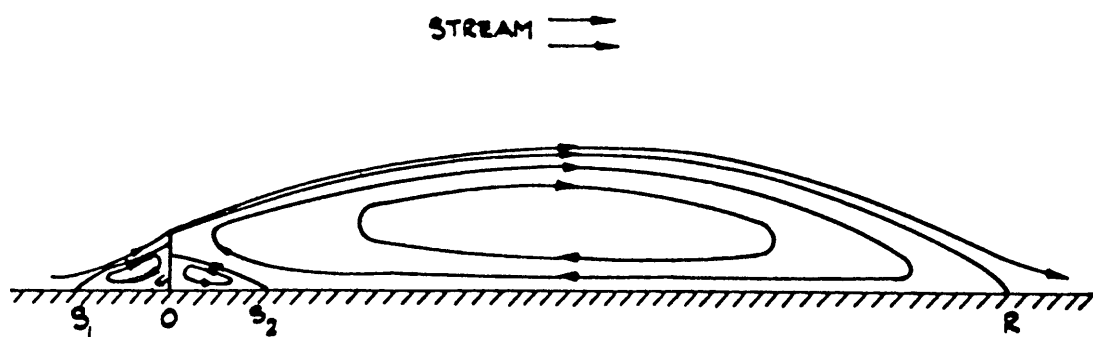


Fig 1.2 Flow pattern for a normal spoiler on a flat plate, Barnes (1966)

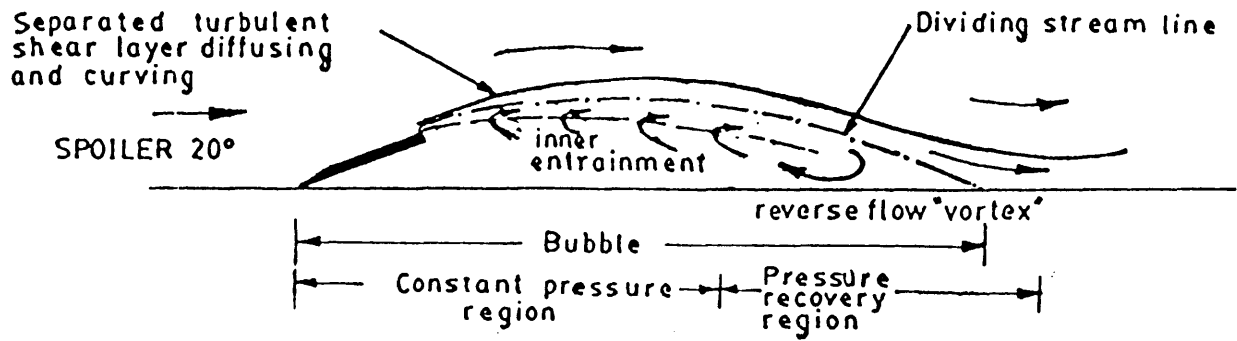


Fig 1.3 Flow pattern for an inclined spoiler on a flat plate, Siddalingappa & Hancock (1980b)

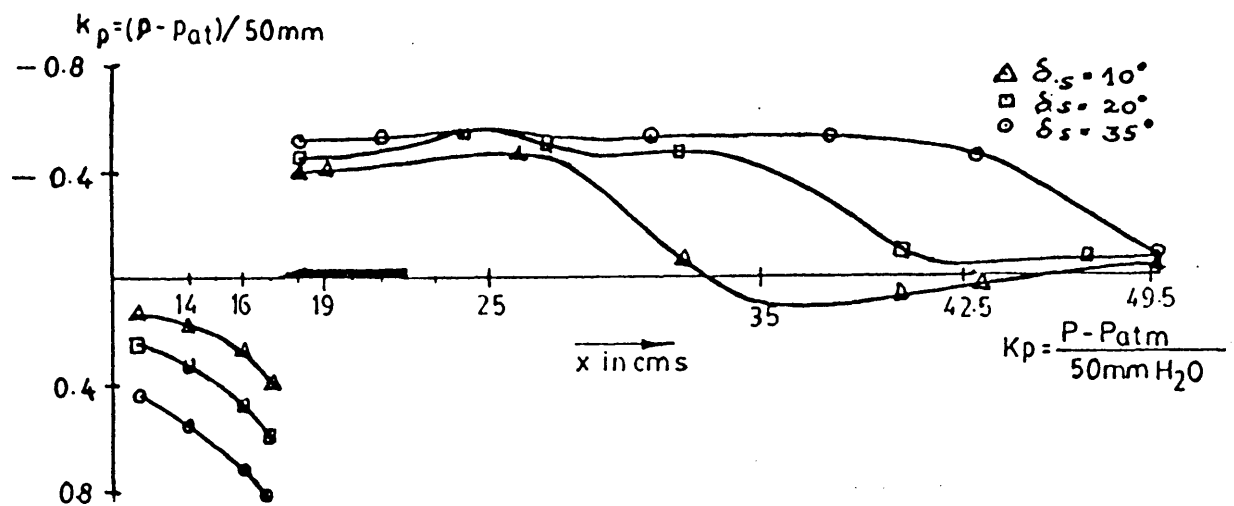


Fig 1.4 Pressure distributions for an inclined spoiler on a flat plate, Siddalingappa & Hancock (1980b)

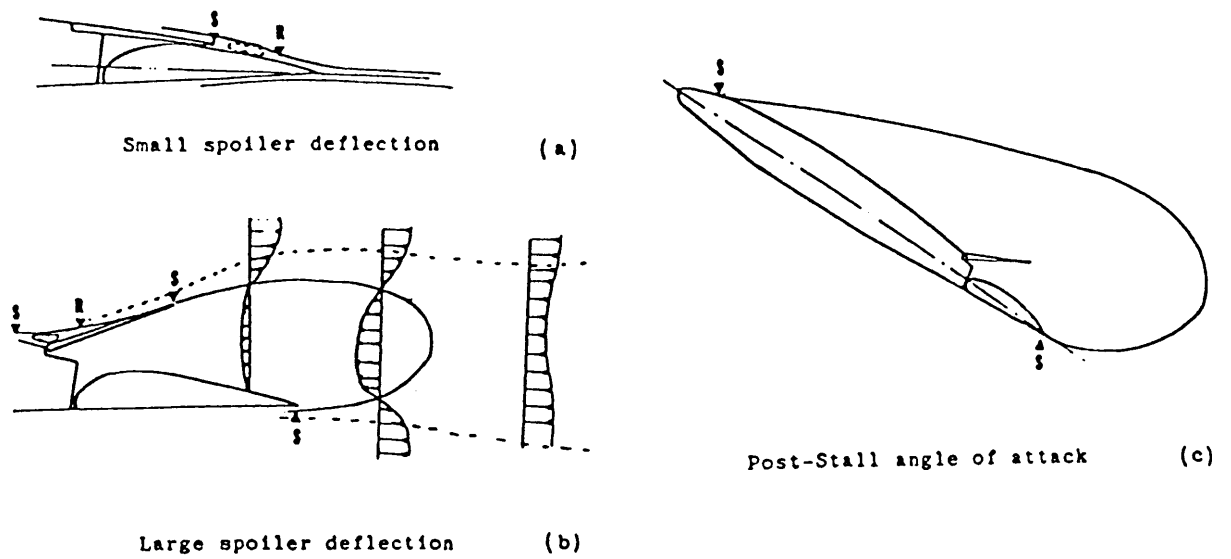


Fig 1.5 Flow patterns on aerofoil with inclined spoilers, Mack et al (1979)

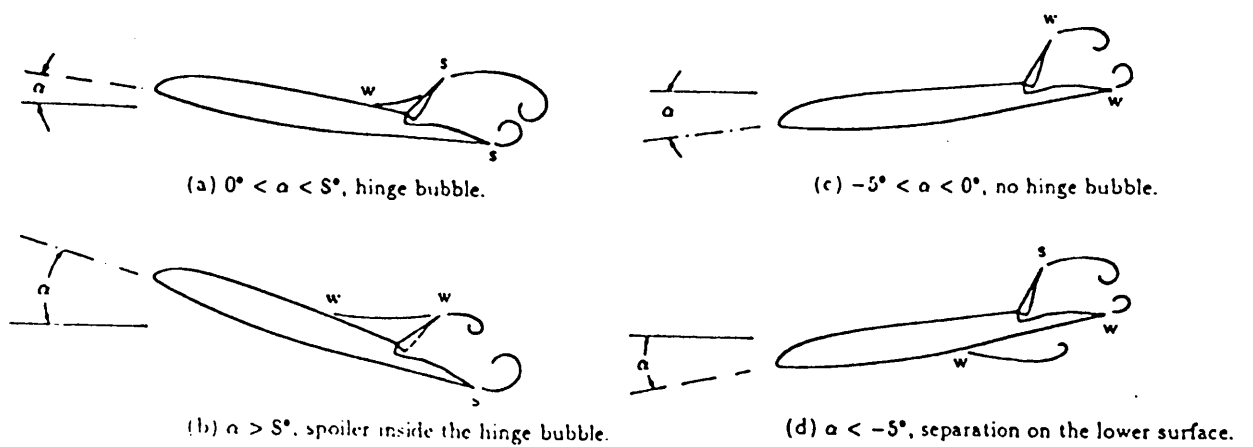


Fig 1.6 Flow patterns on aerofoil with inclined spoilers, Lee & Bodapati (1988)

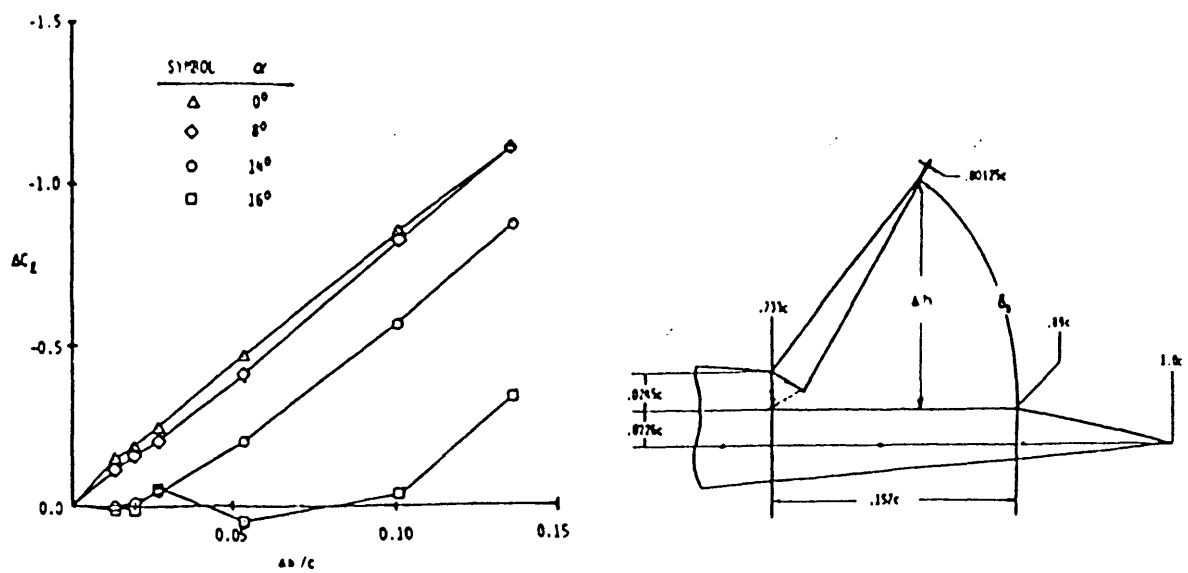


Fig 1.7 Spoiler effectiveness for 2-D spoiler on aerofoil, Wentz et al (1981)

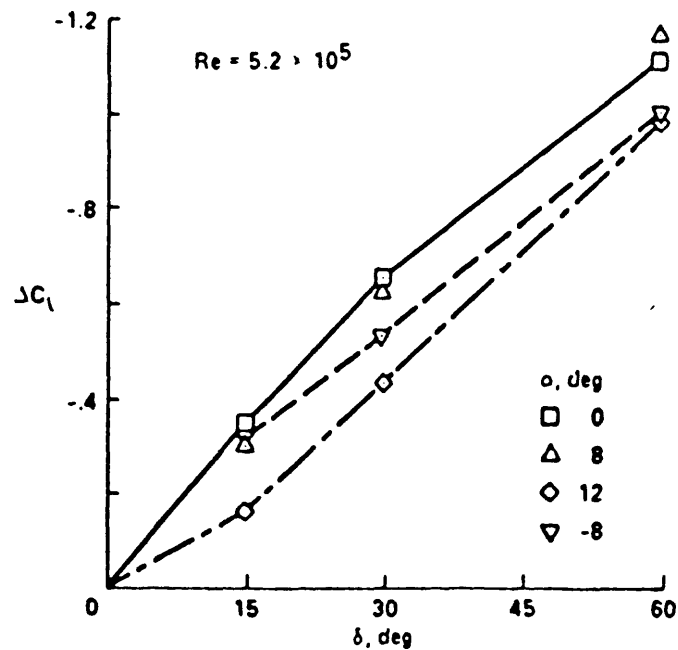


Fig 1.8 Spoiler effectiveness for 2-D spoiler on aerofoil, McLachlan et al (1982)

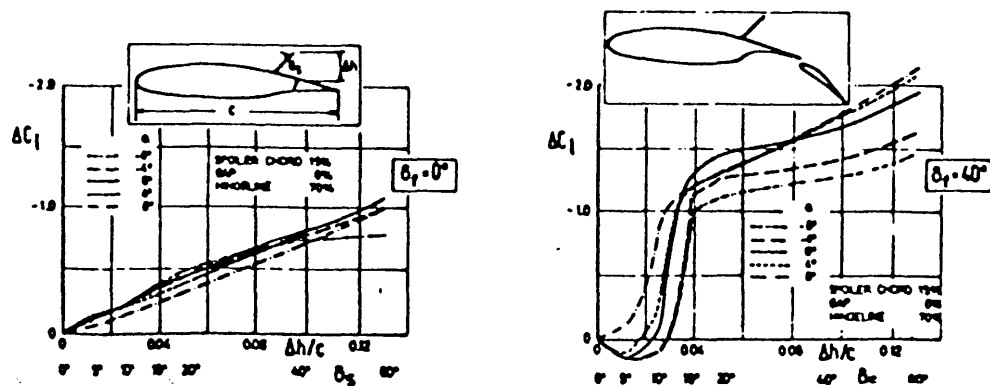


Fig 1.9 Spoiler effectiveness for 2-D spoiler on aerofoil with 2-D fowler-flap, Jones et al (1979)

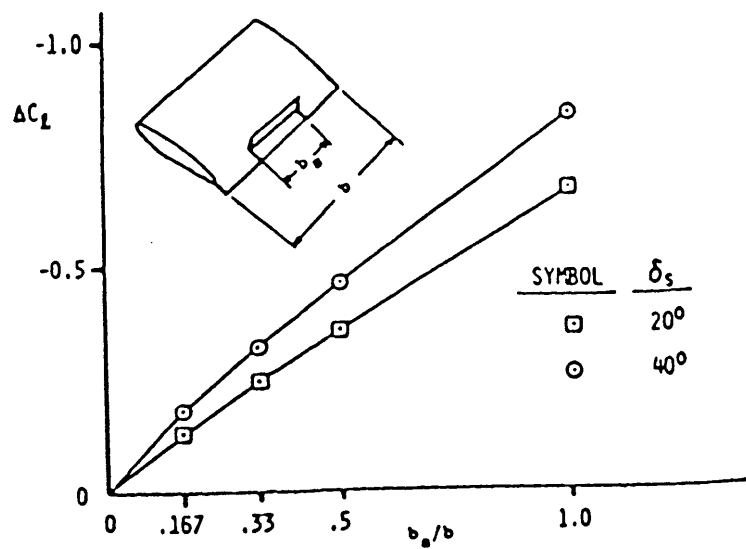


Fig 1.10 Spoiler effectiveness of finite spoiler on a 2-D aerofoil, Wentz et al (1981)

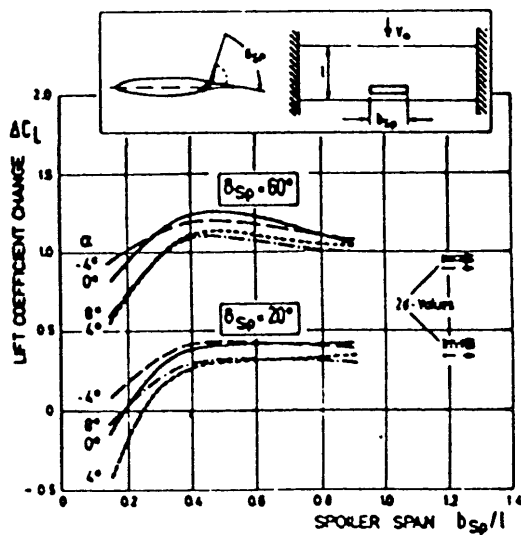


Fig 1.11 Spoiler effectiveness of finite spoiler on a 2-D aerofoil, Jones (1979)

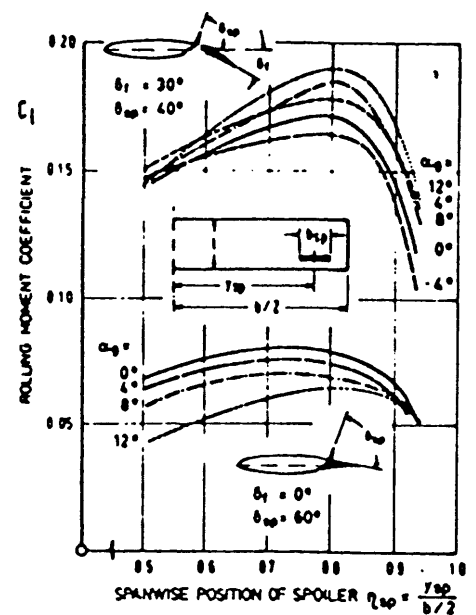


Fig 1.12 Variation of moment coefficient for finite spoiler on 2-D aerofoil, Jones (1979)

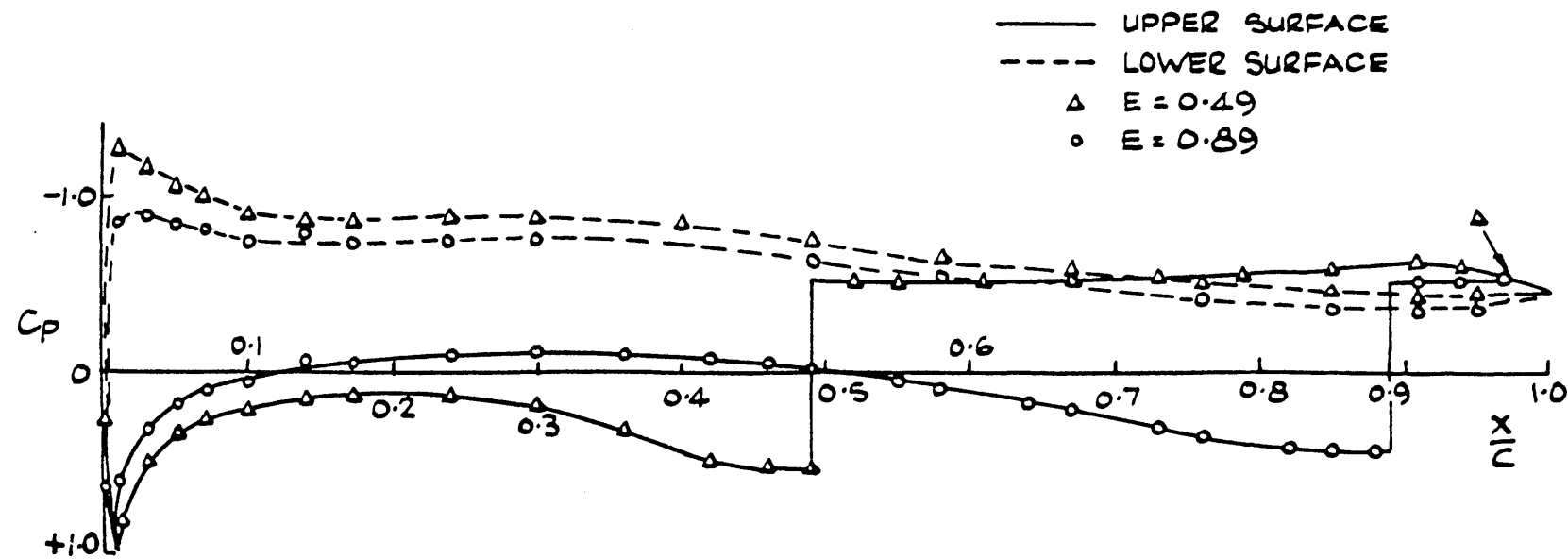


Fig 1.13 Pressure distributions on aerofoil with a normal spoiler, Barnes (1966b)

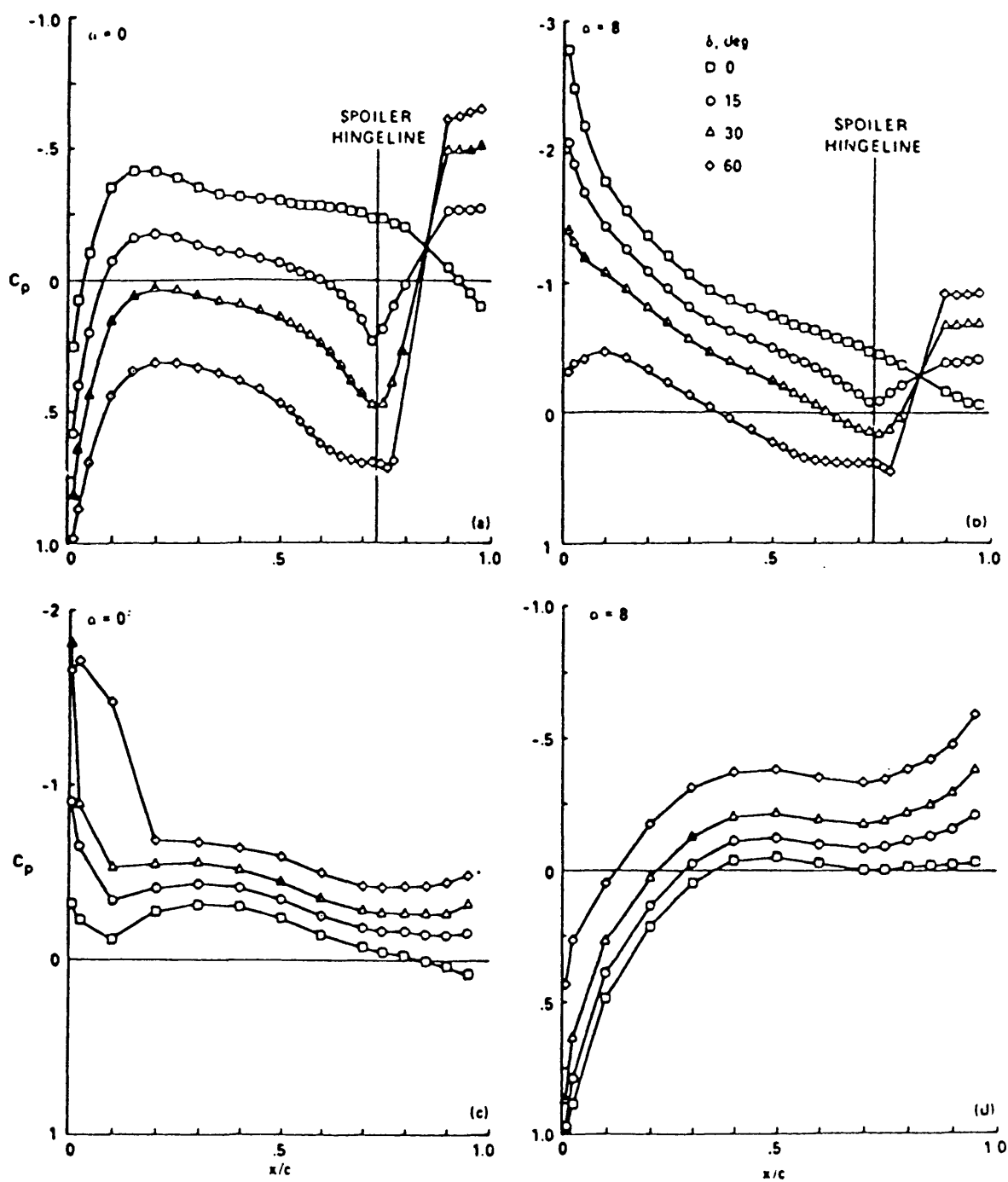


Fig 1.14 Pressure distributions on an aerofoil due to inclined spoilers at two angles of attack, McLachlan et al (1982)

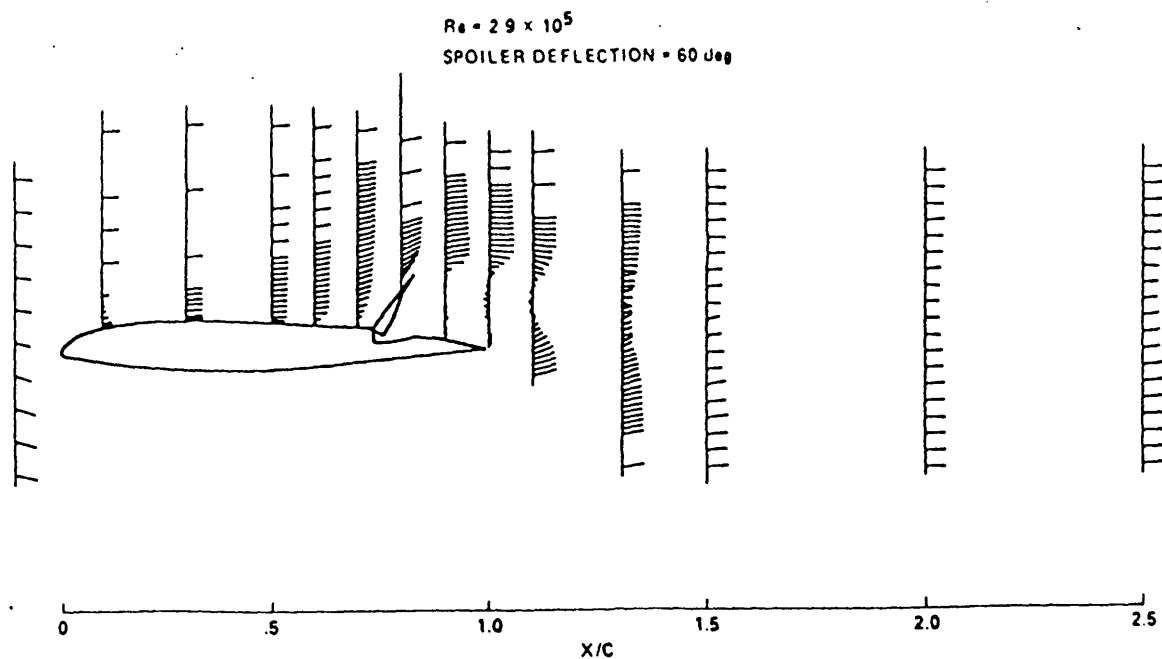


Fig 1.15 Flow field around an aerofoil with spoiler, Bodapati & Karamcheti (1982)

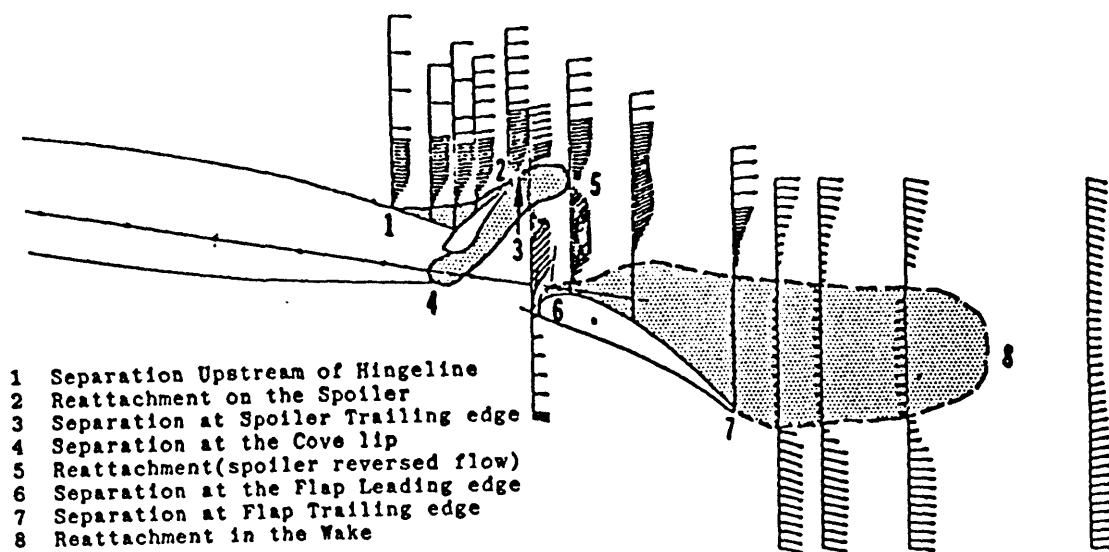


Fig 1.16 Flow field around an aerofoil with flap and spoiler, Mack et al (1979)

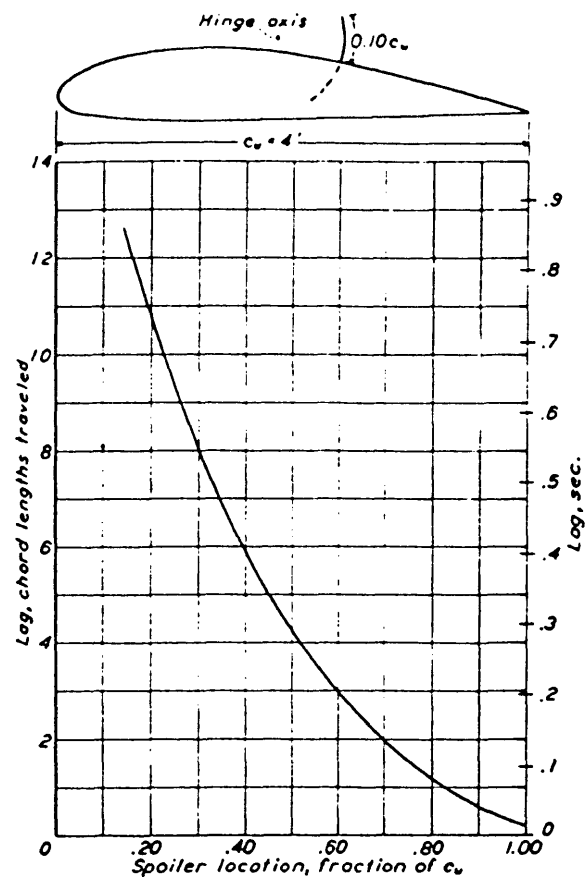


Fig 1.17 Time lag against the location of spoiler on aerofoil, Shortal (1961)

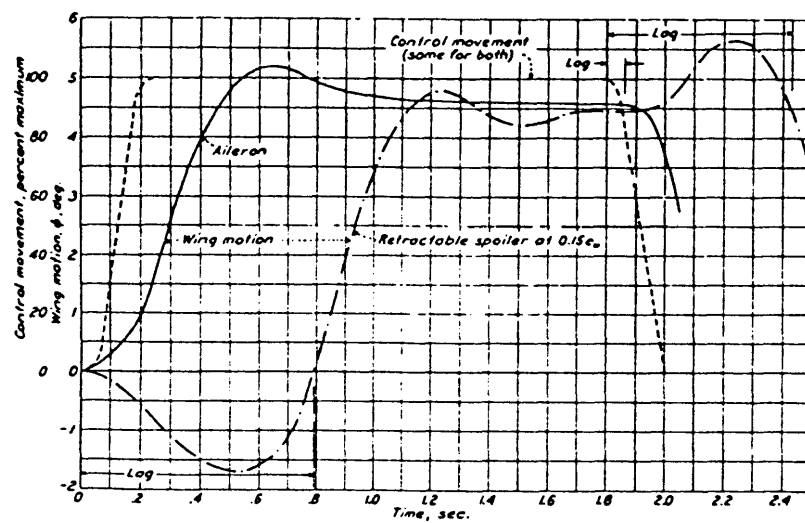


Fig 1.18 Time history of the control motion of spoiler, Shortal (1961)

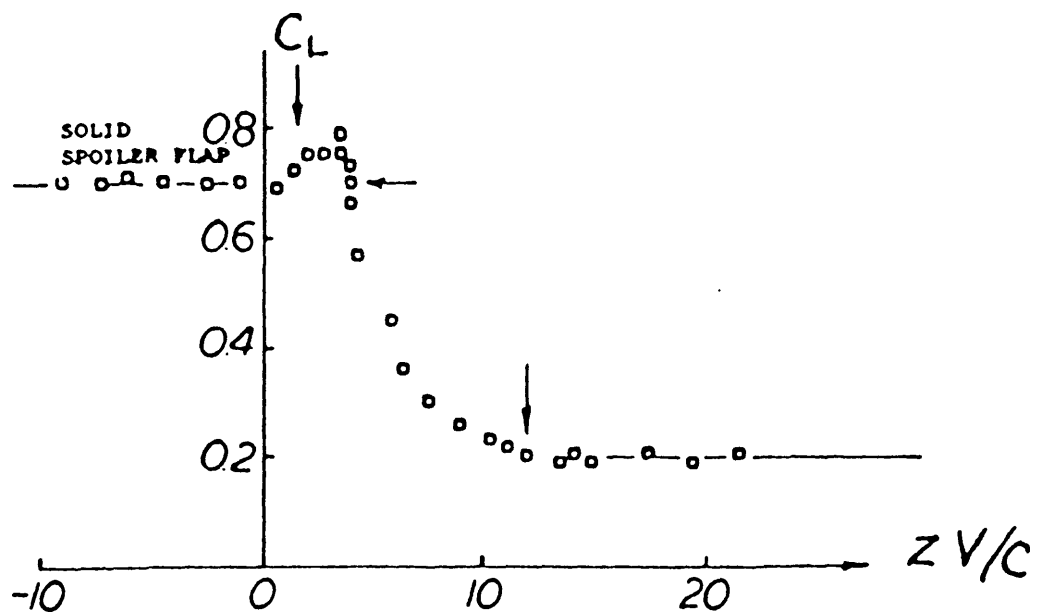


Fig 1.19 Adverse lift effect, Hoerner (1975)

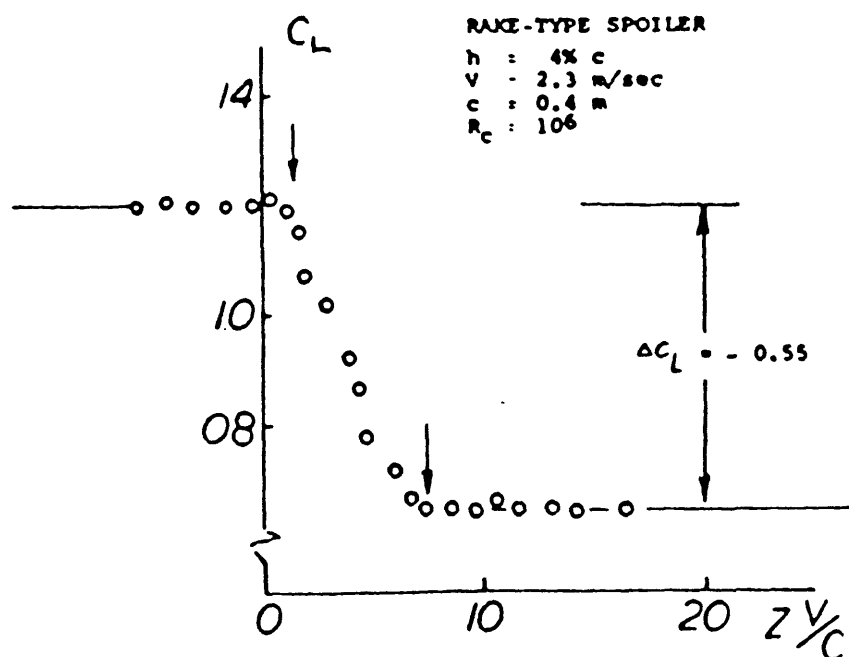


Fig 1.20 Effect of adverse lift and time lag with rake type spoiler, Hoerner (1975)

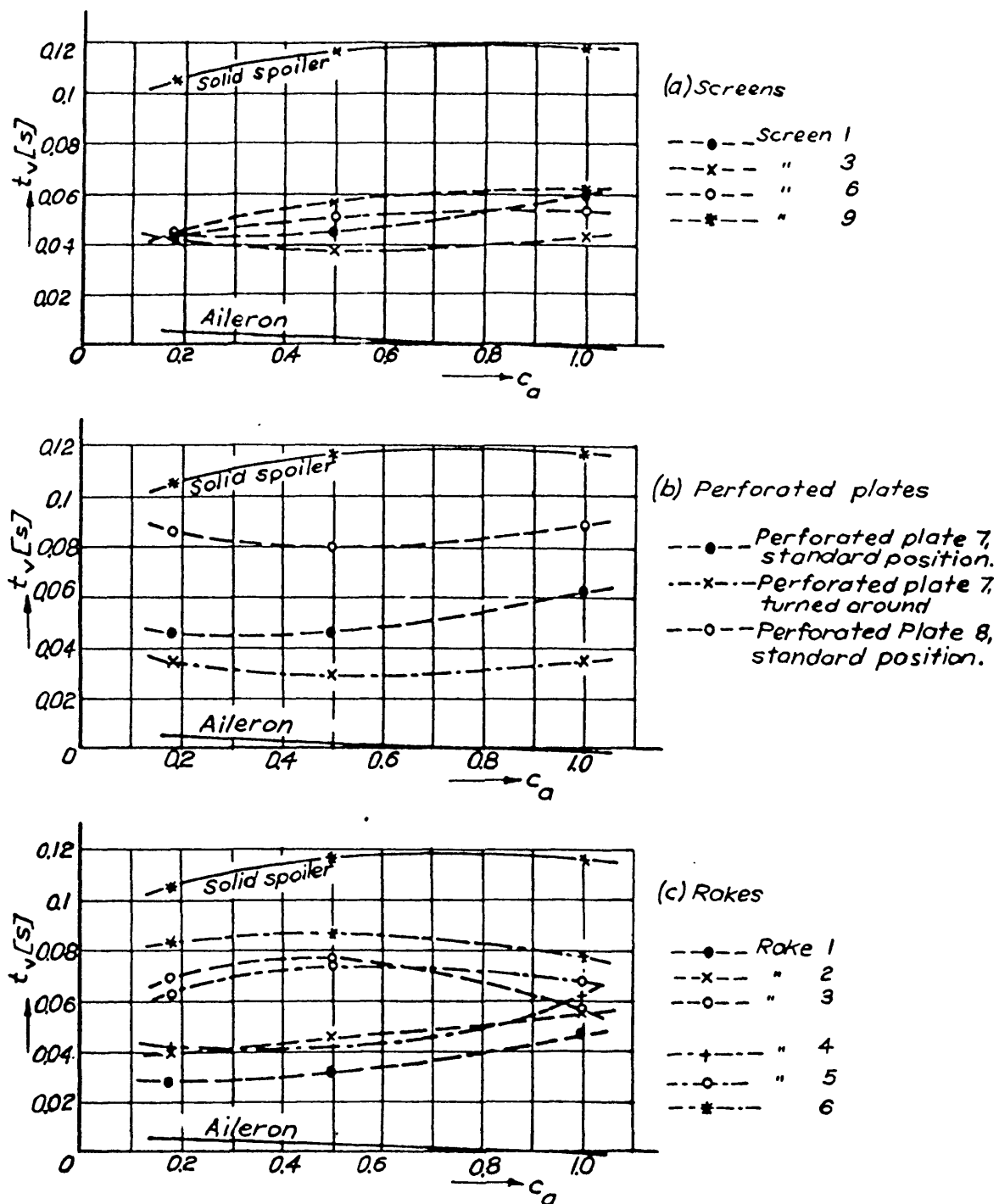


Fig 1.21 Effect of time lag with various spoilers, Kramer et al (1951)

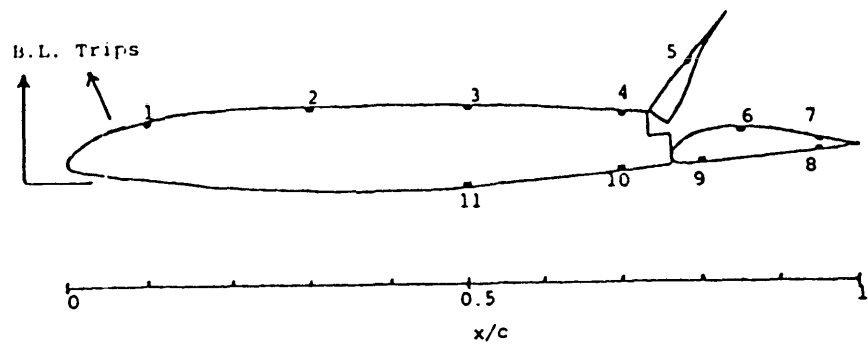


Fig 1.22a Aerofoil used, Ayoub et al (1982)

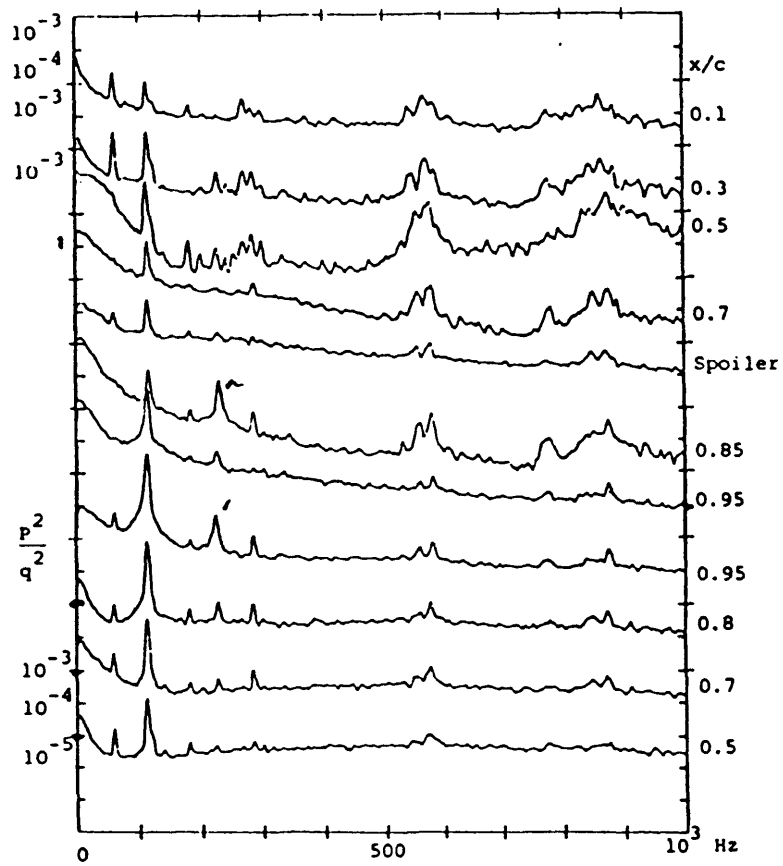


Fig 1.22b Power spectra of surface pressures, Ayoub et al (1982)

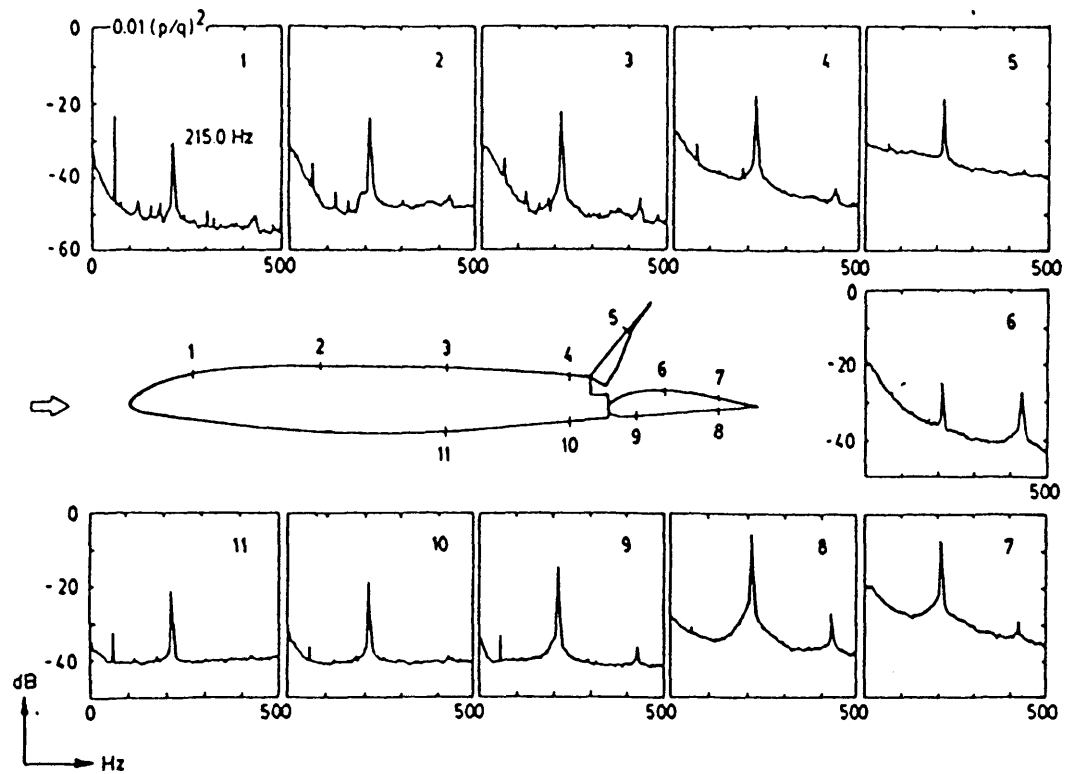


Fig 1.23 Power spectra of surface pressures, McLachlan et al (1983)

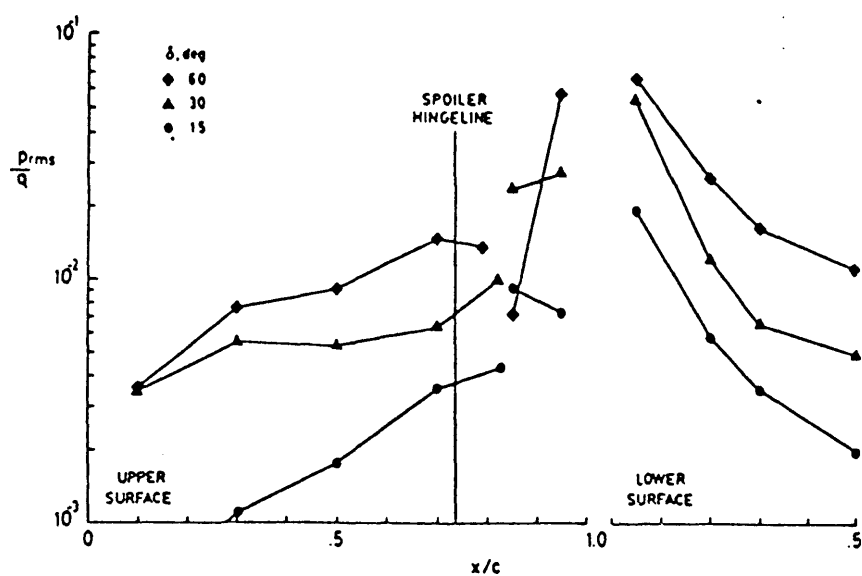


Fig 1.24 Fluctuating pressure distributions of aerofoil with spoiler, McLachlan et al (1983)

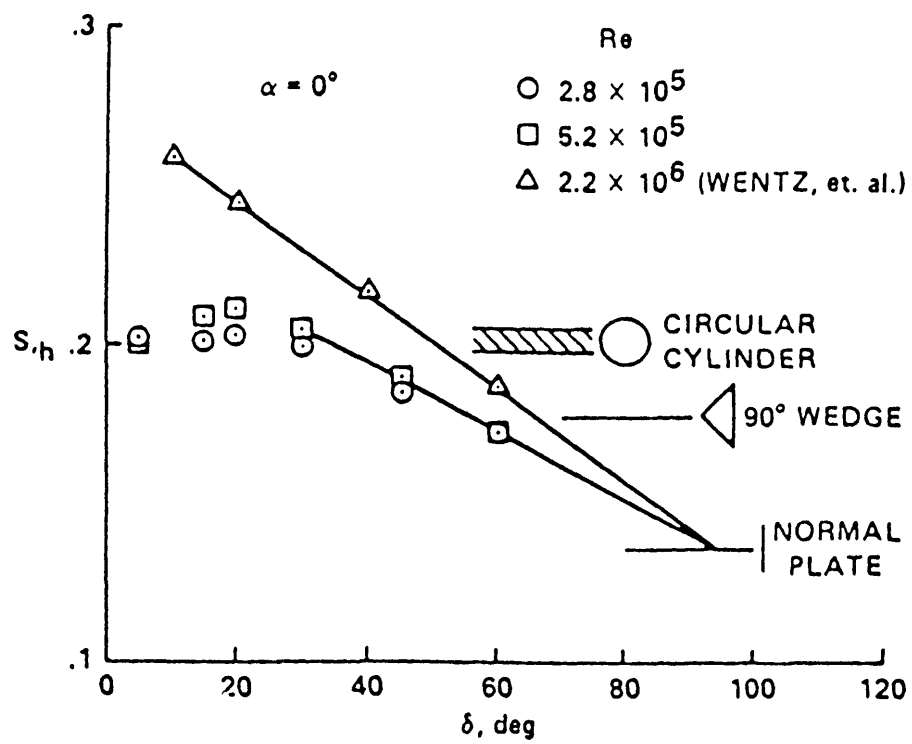
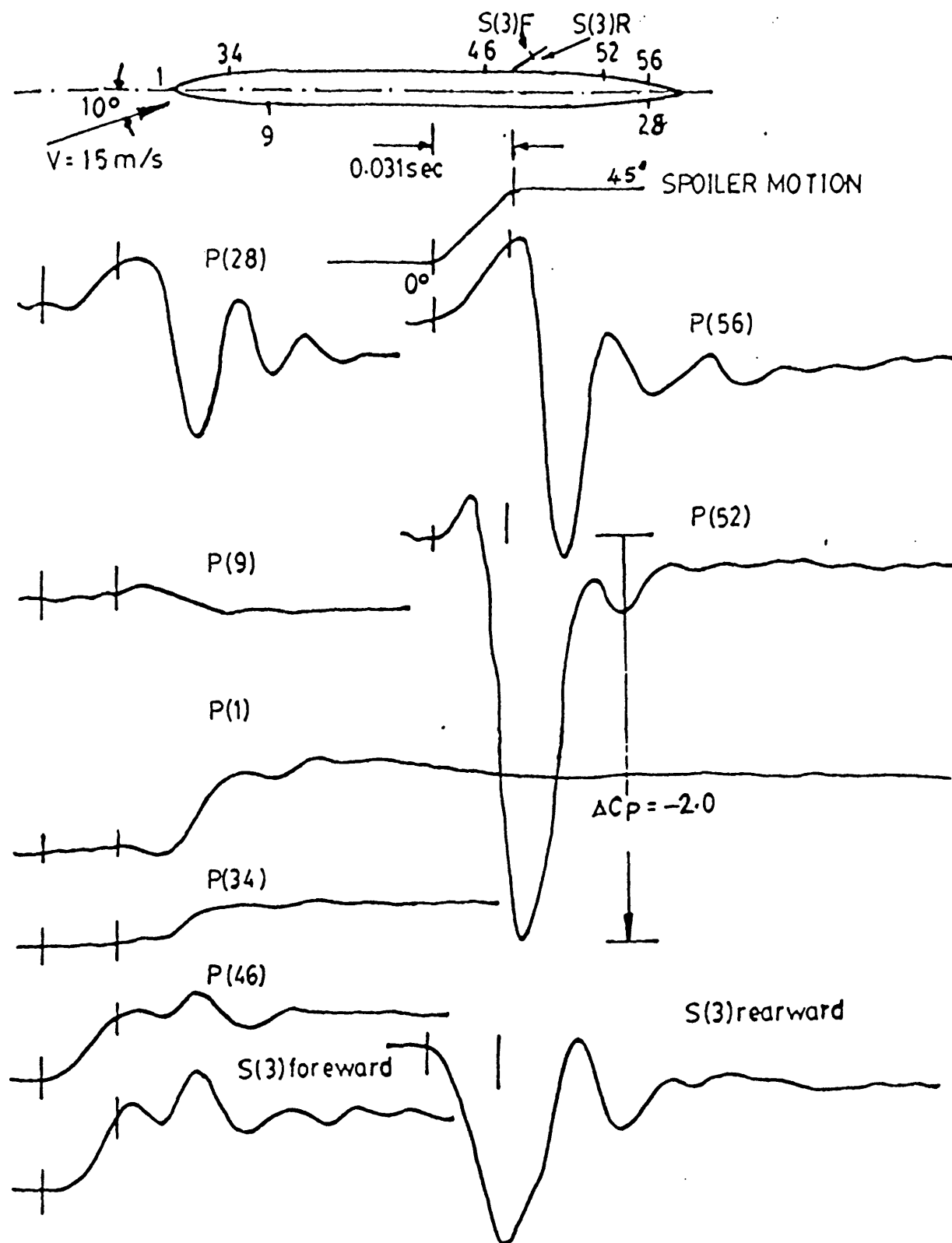
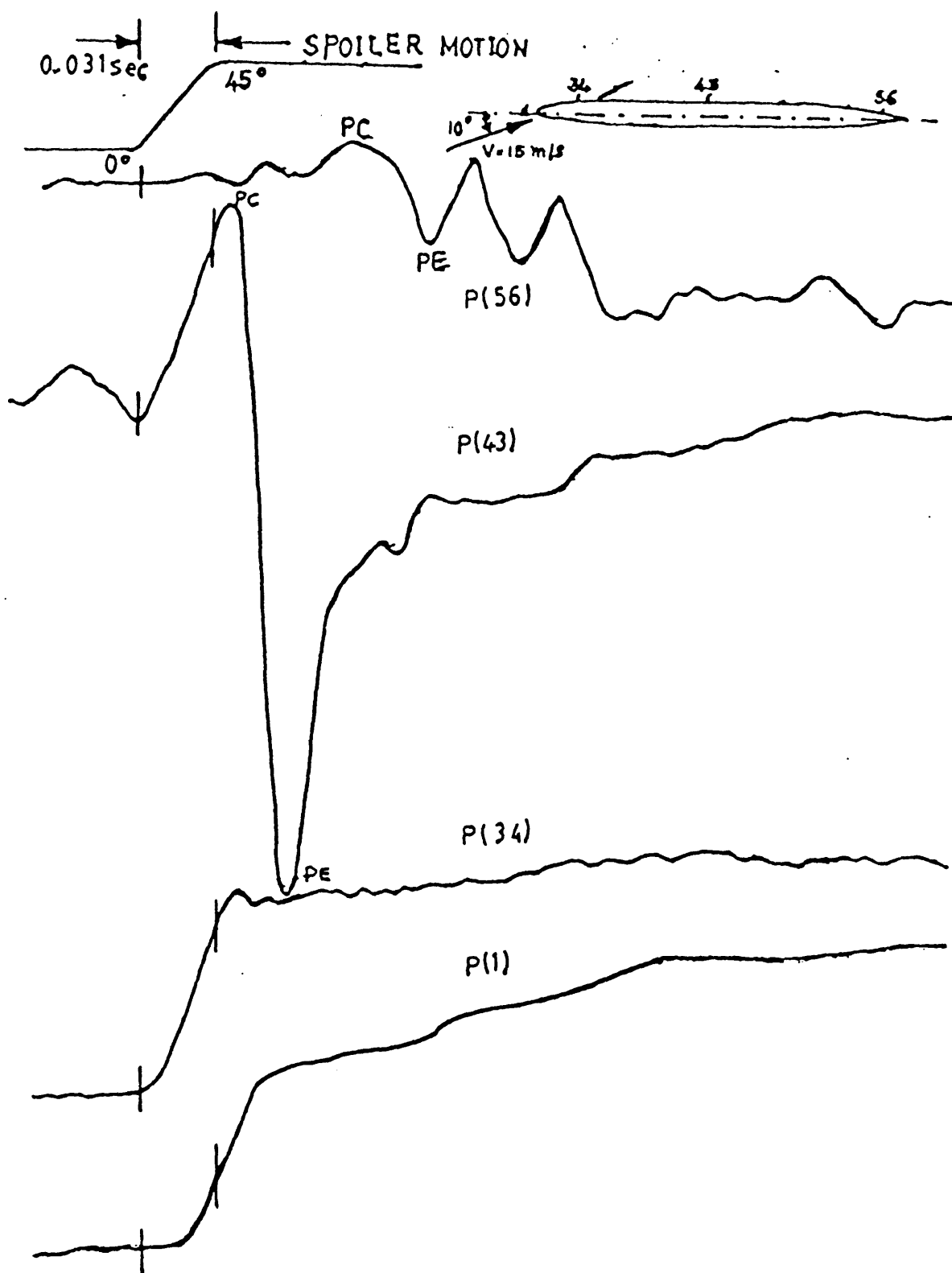


Fig 1.25 Strouhal numbers for aerofoil with spoiler, McLachlan et al (1983)



SPOILER AT 68% CHORD UPPER SURFACE, $\alpha = 10^\circ$,
GAP = 0%. RAMP SPOILER $0^\circ \rightarrow 45^\circ$ IN OPEN JET

Fig 1.26 Pressure traces on aerofoil for a spoiler at 68% chord, Siddalingappa & Hancock (1980c)



spoiler at 16 % chord : upper surface
 $\alpha = 10^\circ$. gap = 0%. ramp spoiler $0^\circ - 45^\circ$
 in open jet working section

Fig 1.27 Pressure traces on aerofoil for a spoiler at 16% chord,
 Siddalingappa & Hancock (1980c)

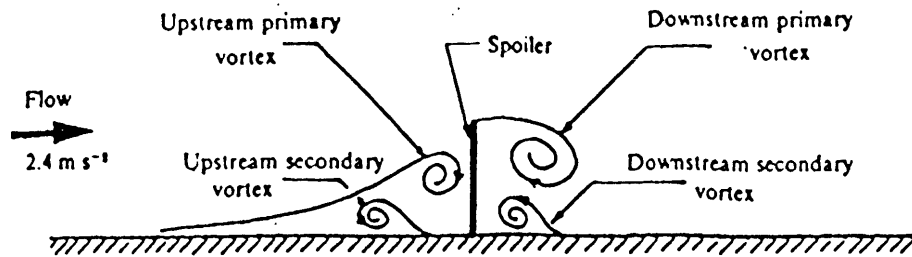


Fig 1.28 Sketch of 2-D vortices developed for starting flow past spoiler, Finaish et al (1985)

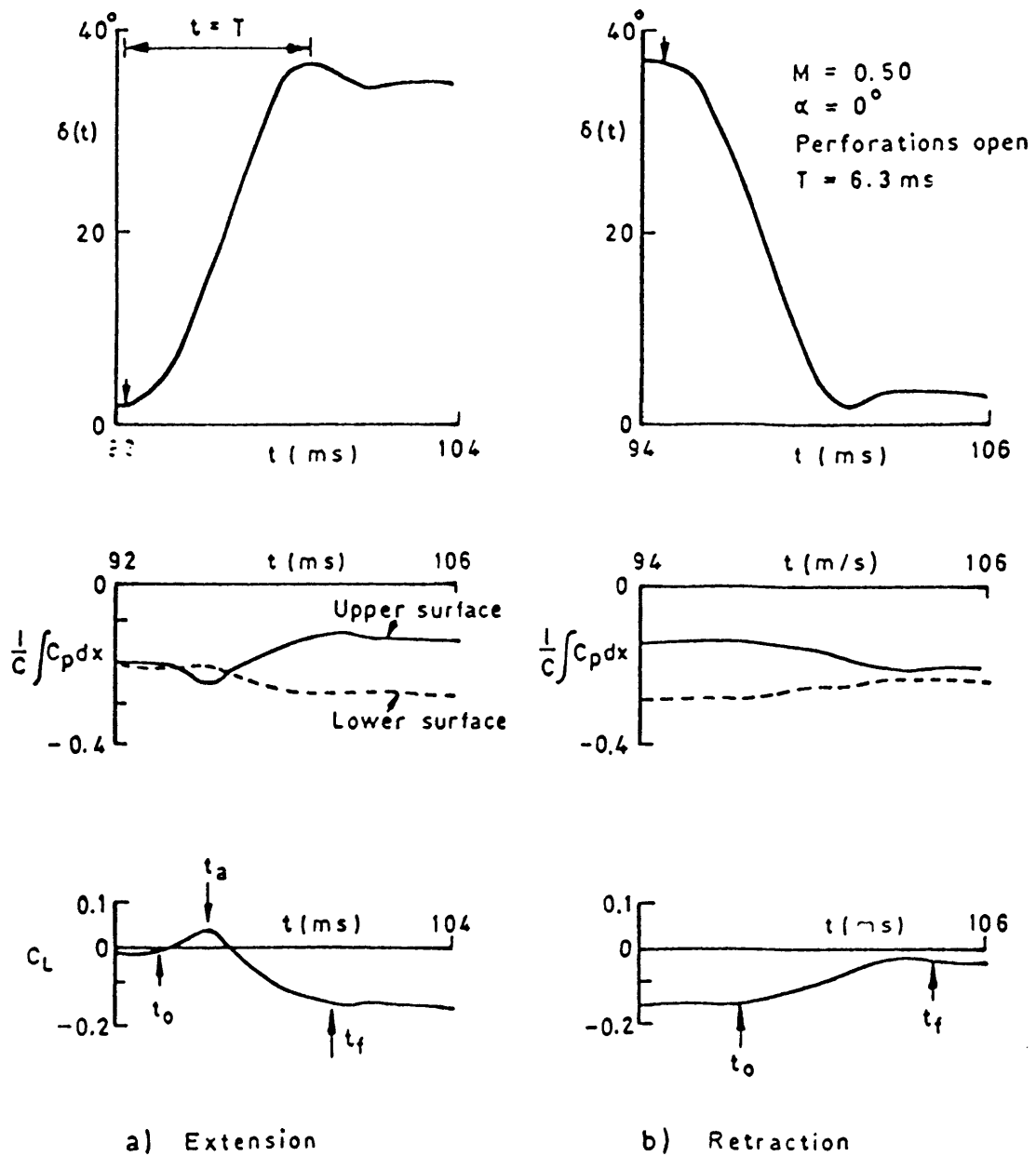


Fig 1.29 Transient lift on aerofoil due to a rapidly acting spoiler, Mabey et al (1982)

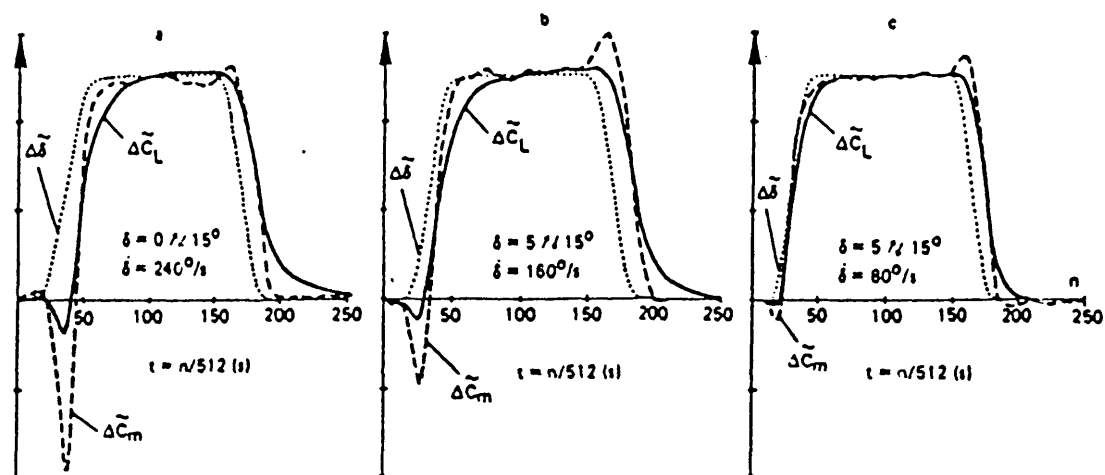


Fig 1.30 Transient lift on aerofoil due to a rapidly acting spoiler, Costes et al (1986)

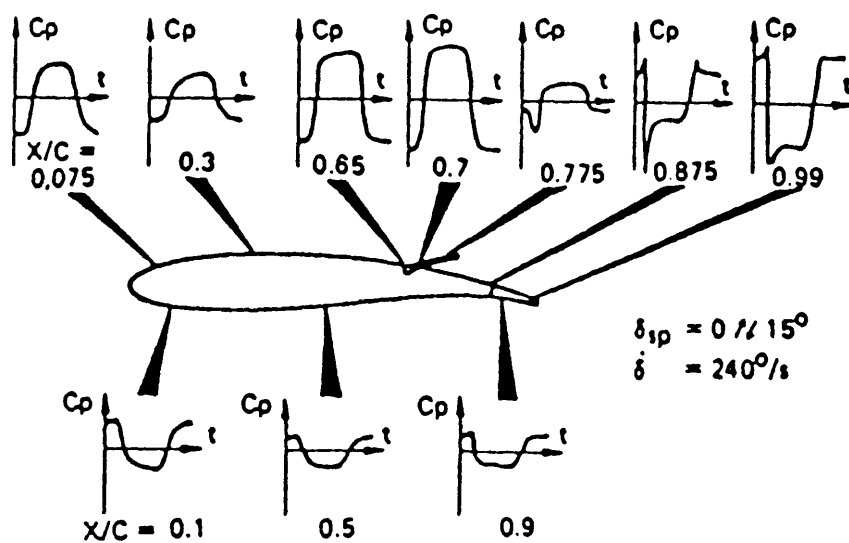
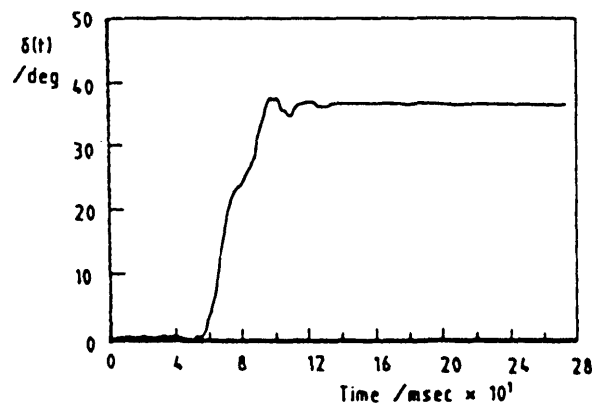
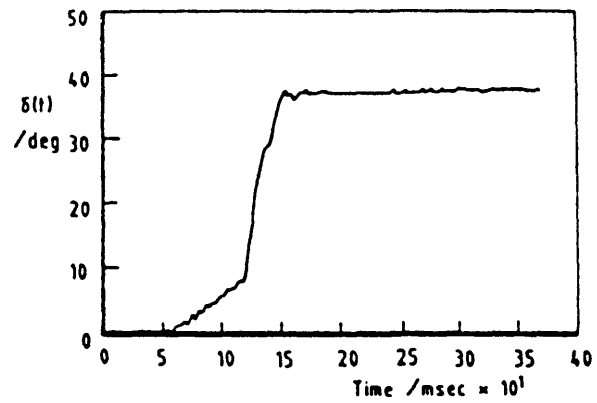


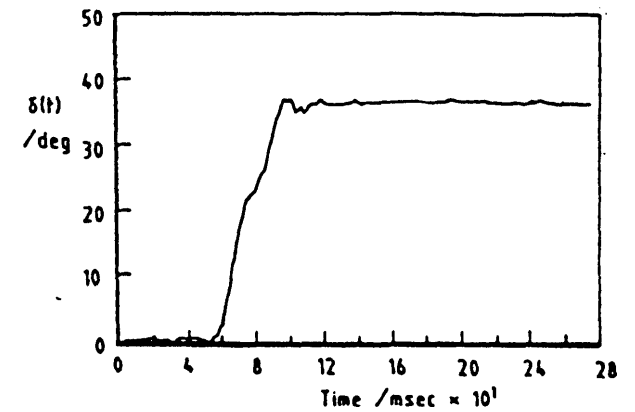
Fig 1.31 Pressure traces on aerofoil due to a rapidly acting spoiler, Costes et al (1986)



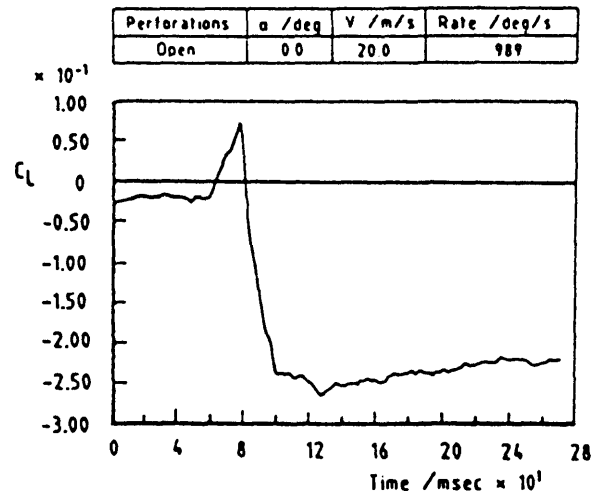
(a) Spoiler displacement trace



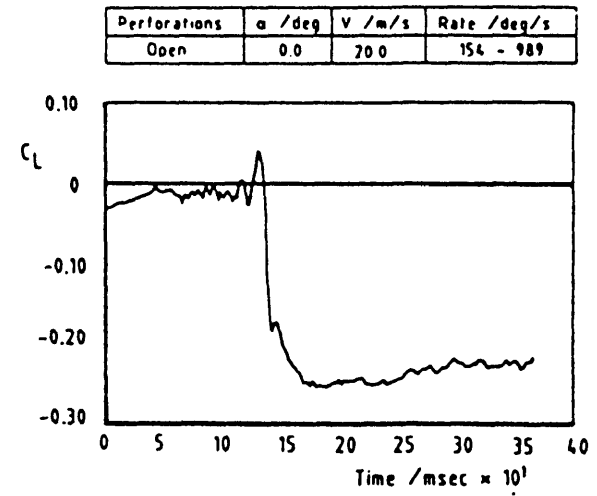
(c) Spoiler displacement trace



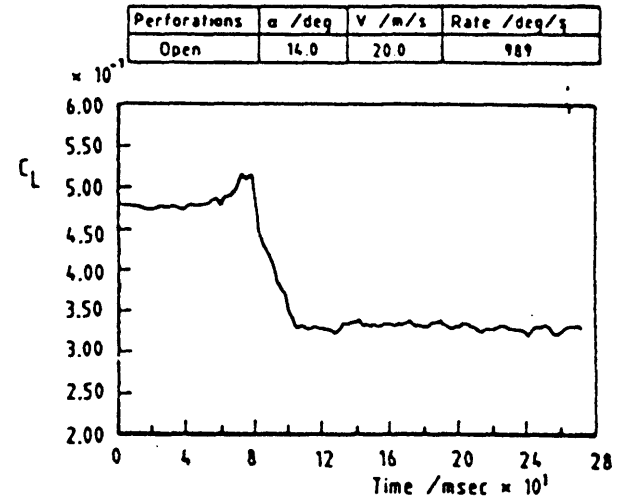
(e) Spoiler displacement trace



(b) Aerofoil lift time history

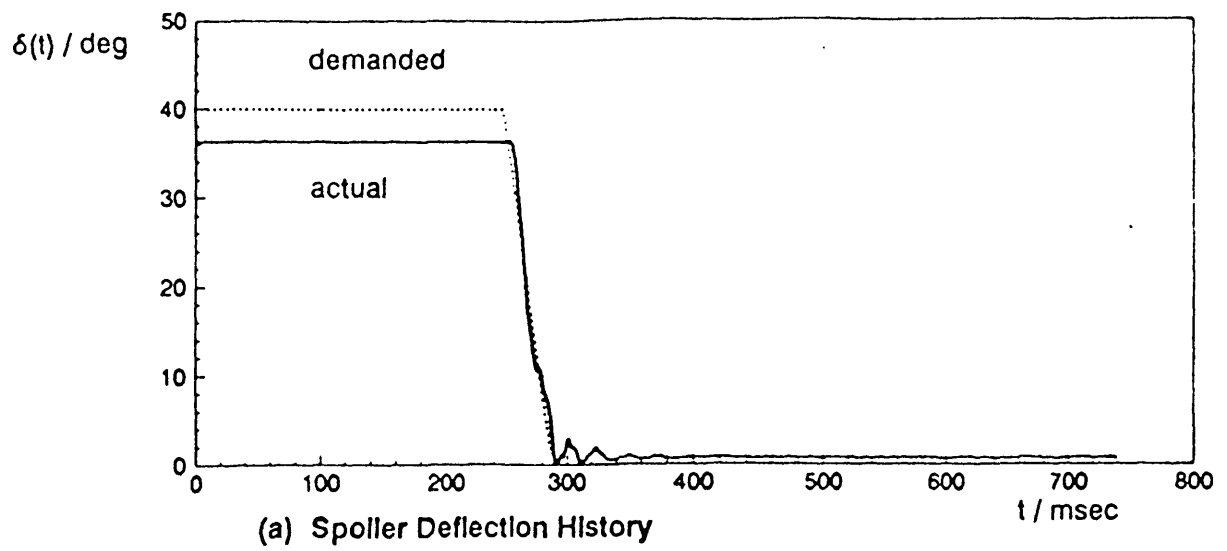


(d) Aerofoil lift time history



(f) Aerofoil lift time history

Fig 1.32 Transient lift on aerofoil due to different spoiler activation profiles, Kalligas (1986)



Forward Spoiler : Retraction

Spoiler at $x/c = 0.15$, $\alpha = 12^\circ$, Rate = $1000^\circ/\text{sec}$

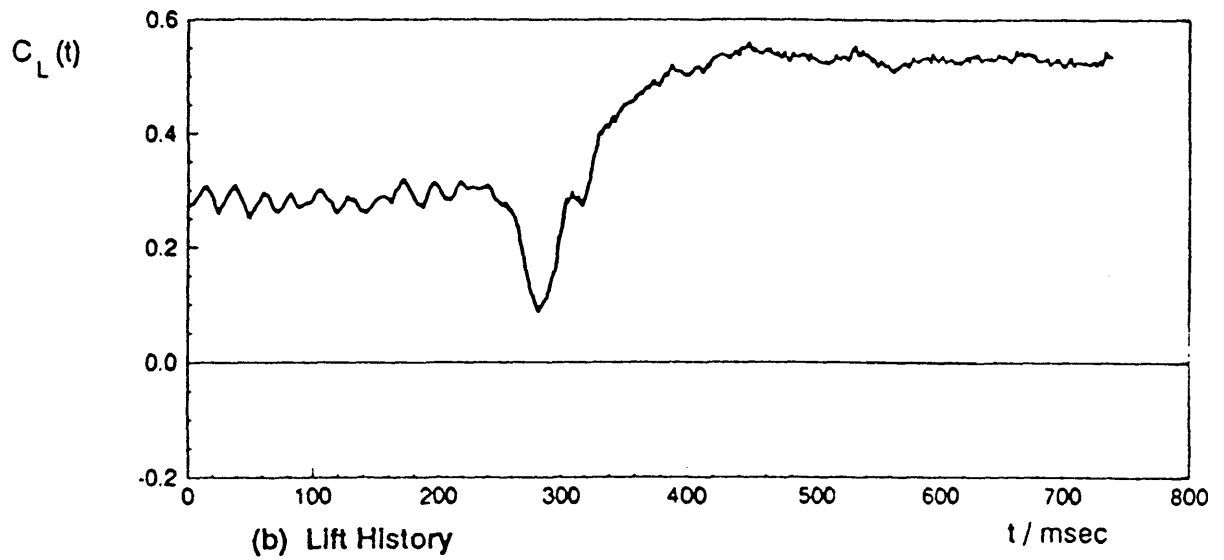


Fig 1.33 Transient lift on aerofoil due to retraction of forward spoiler, Myers (1990)

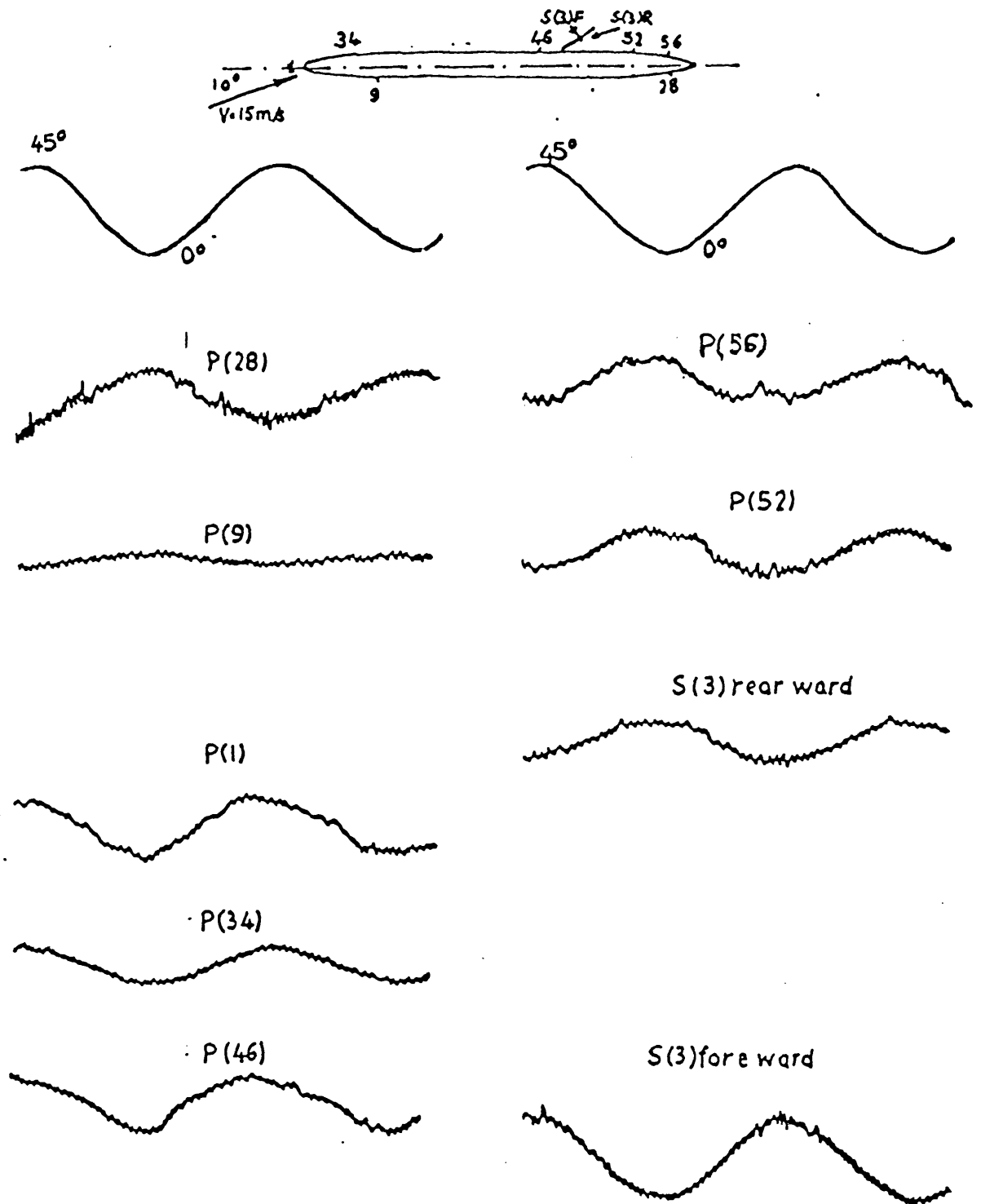


Fig 1.34 Pressure traces on aerofoil with a spoiler oscillating at 1 Hz, Siddalingappa & Hancock (1980c)

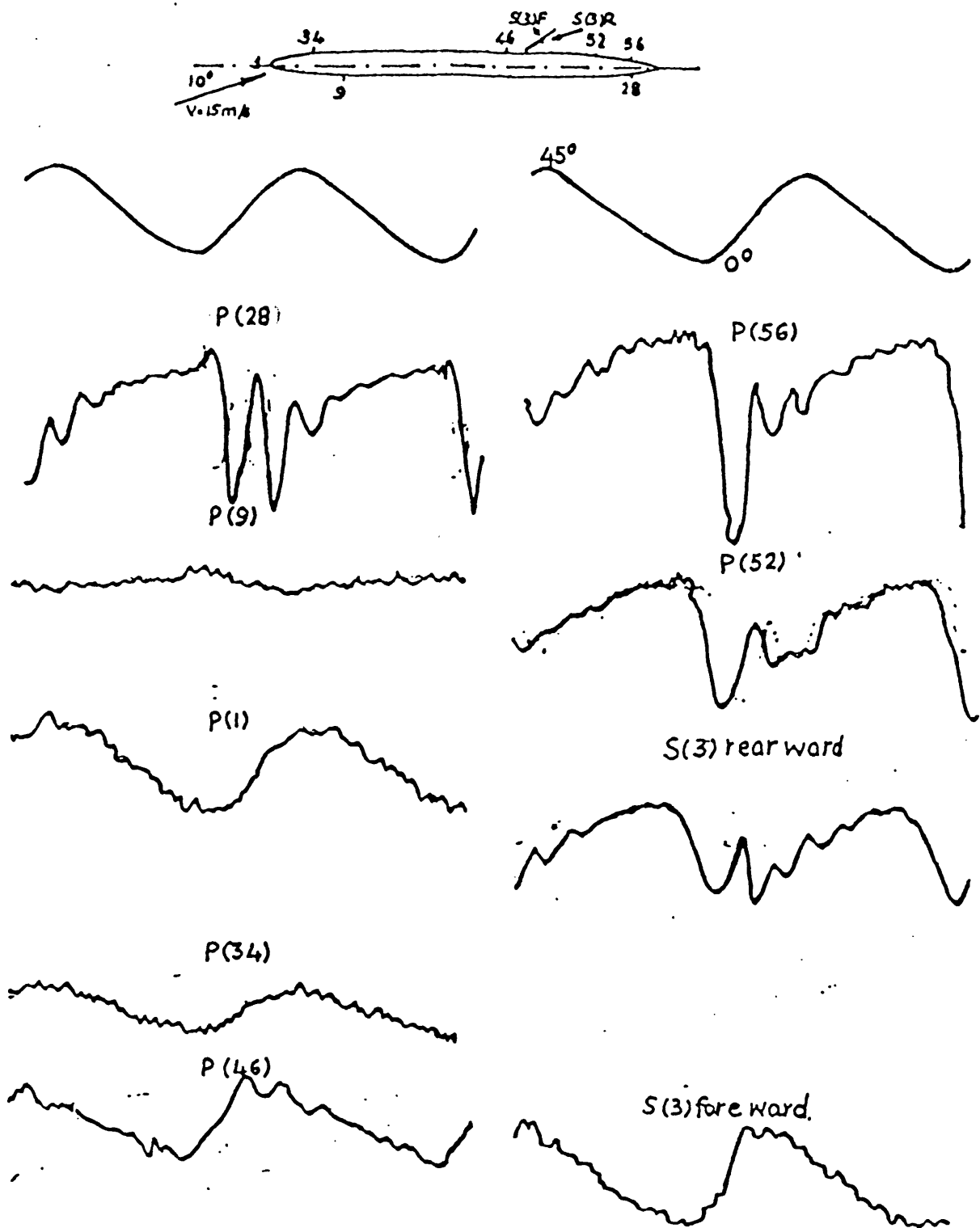


Fig 1.35 Pressure traces on aerofoil with a spoiler oscillating at 5 Hz, Siddalingappa & Hancock (1980c)

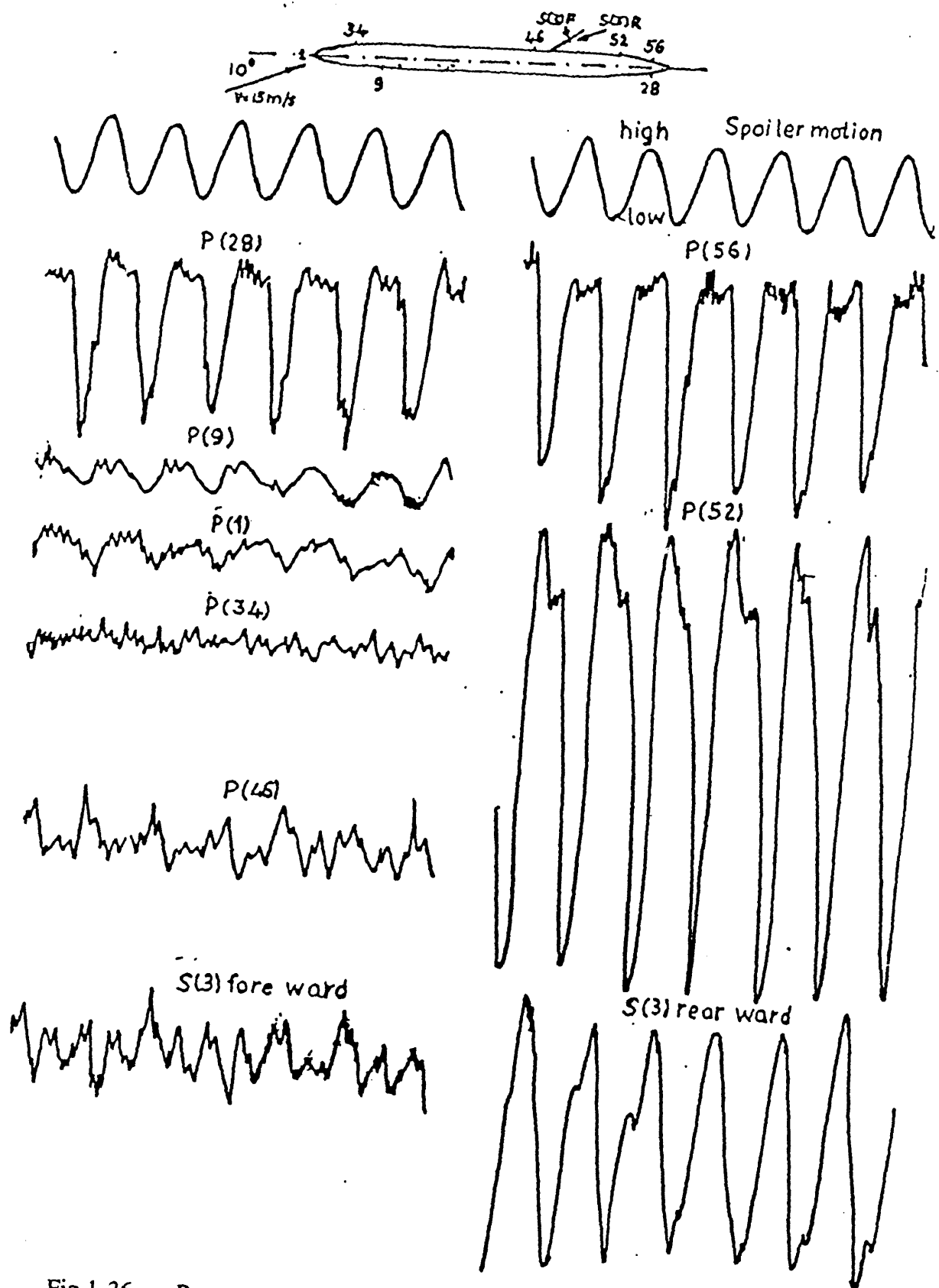


Fig 1.36 Pressure traces on aerofoil with a spoiler oscillating at 18 Hz, Siddalingappa & Hancock (1980c)

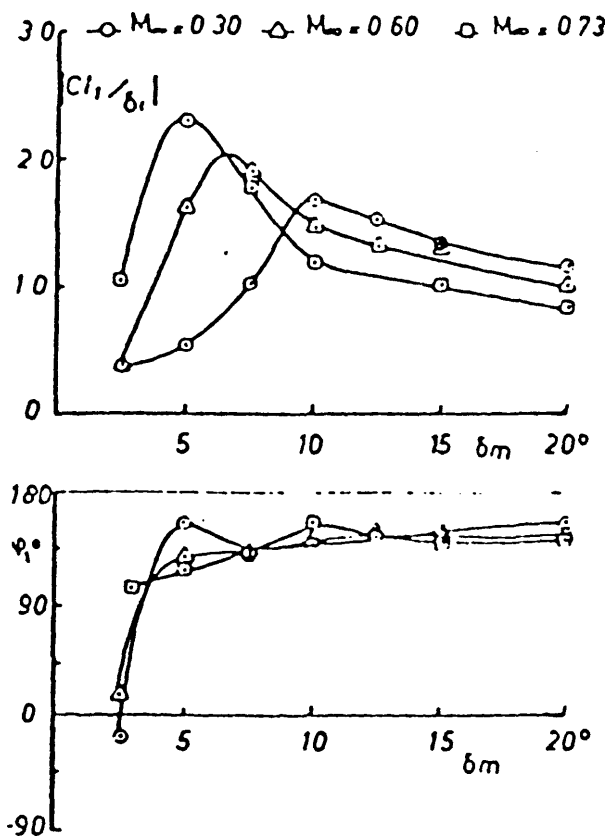


Fig 1.37 Effects of spoiler deflection & Mach number on oscillating spoiler, Consigny et al (1984)

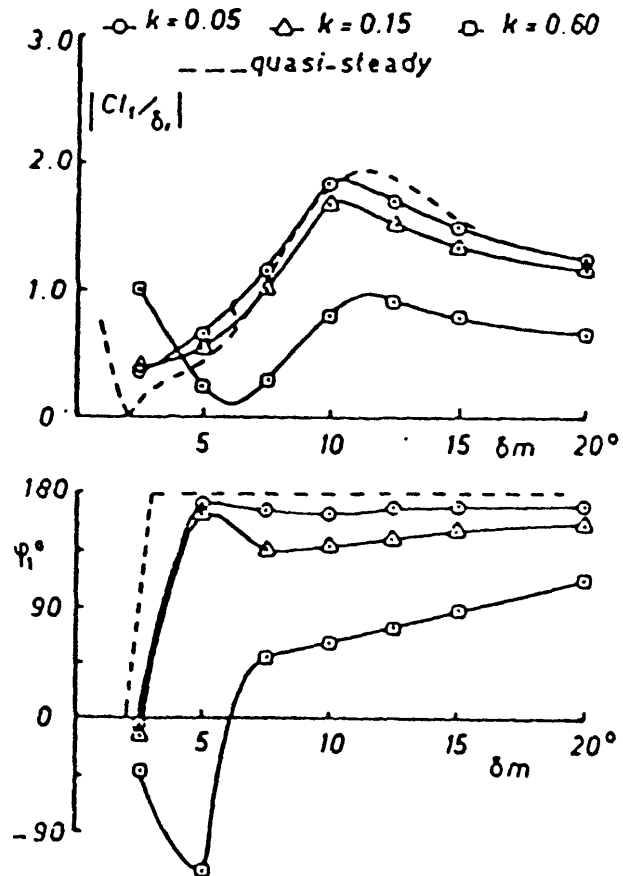


Fig 1.38 Effect of spoiler oscillation frequency on spoiler, Consigny et al (1984)

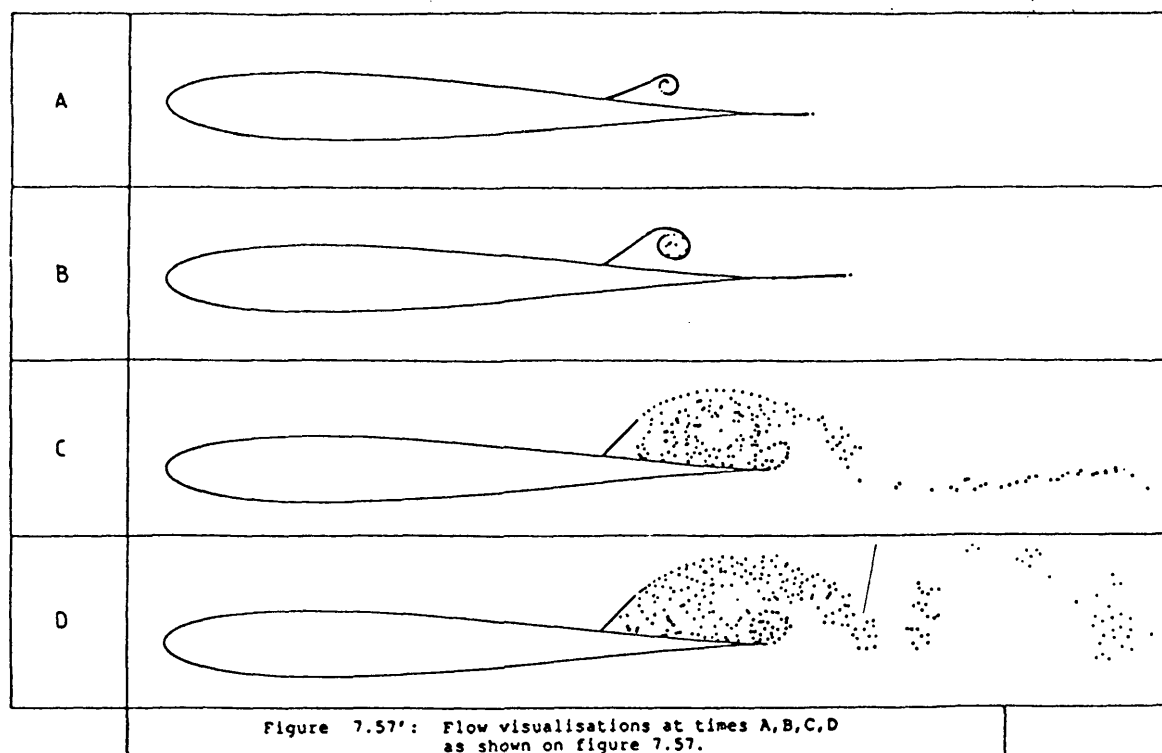


Fig 1.39 Numerical flow visualization of flow generated by rapidly acting spoiler, Kalkanis (1988)

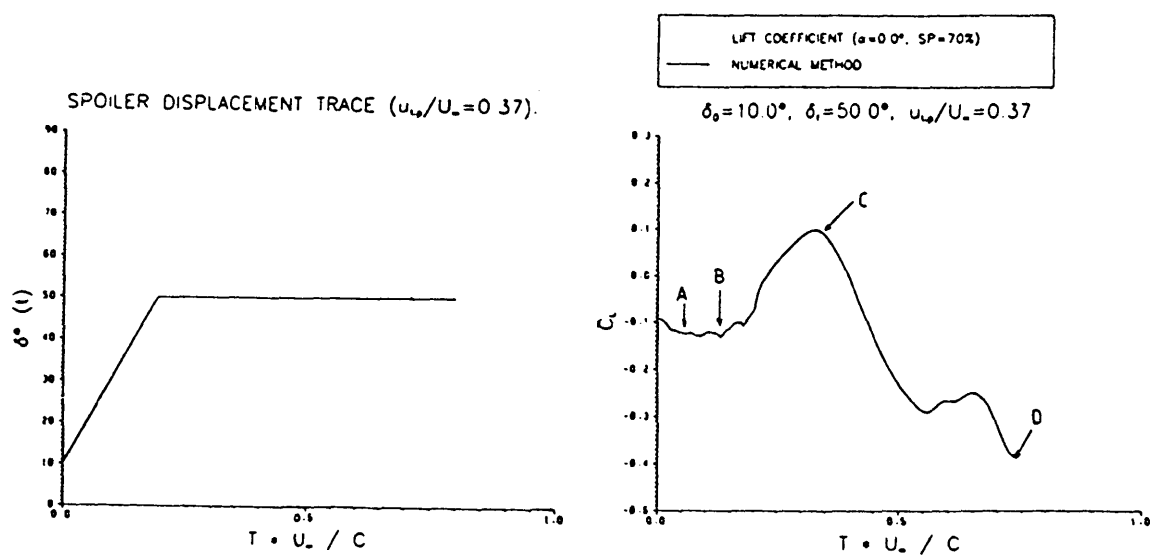


Fig 1.40 Numerical prediction of transient lift due to rapidly acting spoiler, Kalkanis (1988)

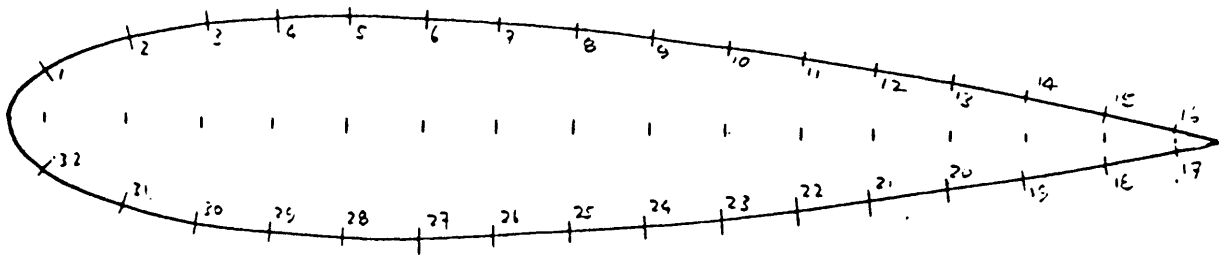


Fig 2.1a Profile of NACA 0018 and locations of pressure tappings

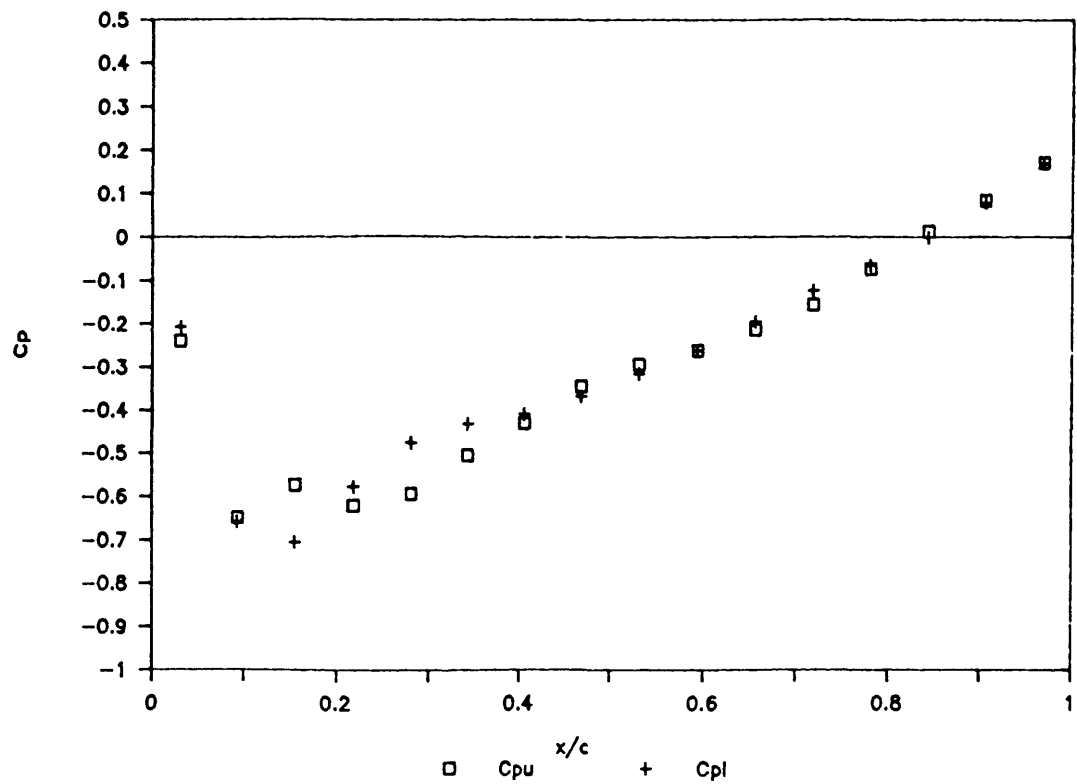


Fig 2.1b Pressure distribution of NACA 0018 at $\alpha = 0^\circ$, $Re = 4.17 \times 10^5$

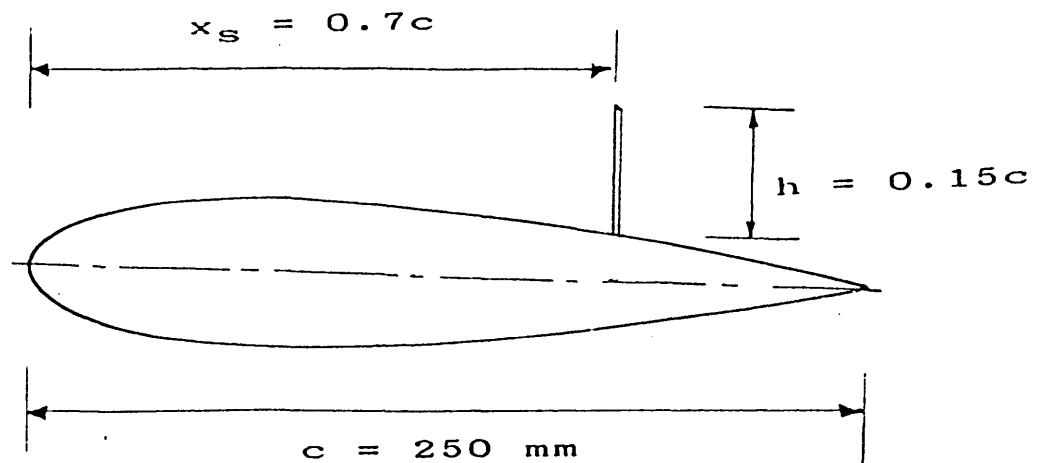


Fig 2.2 Aerofoil with a spoiler of height 15% chord and located at 70% chord

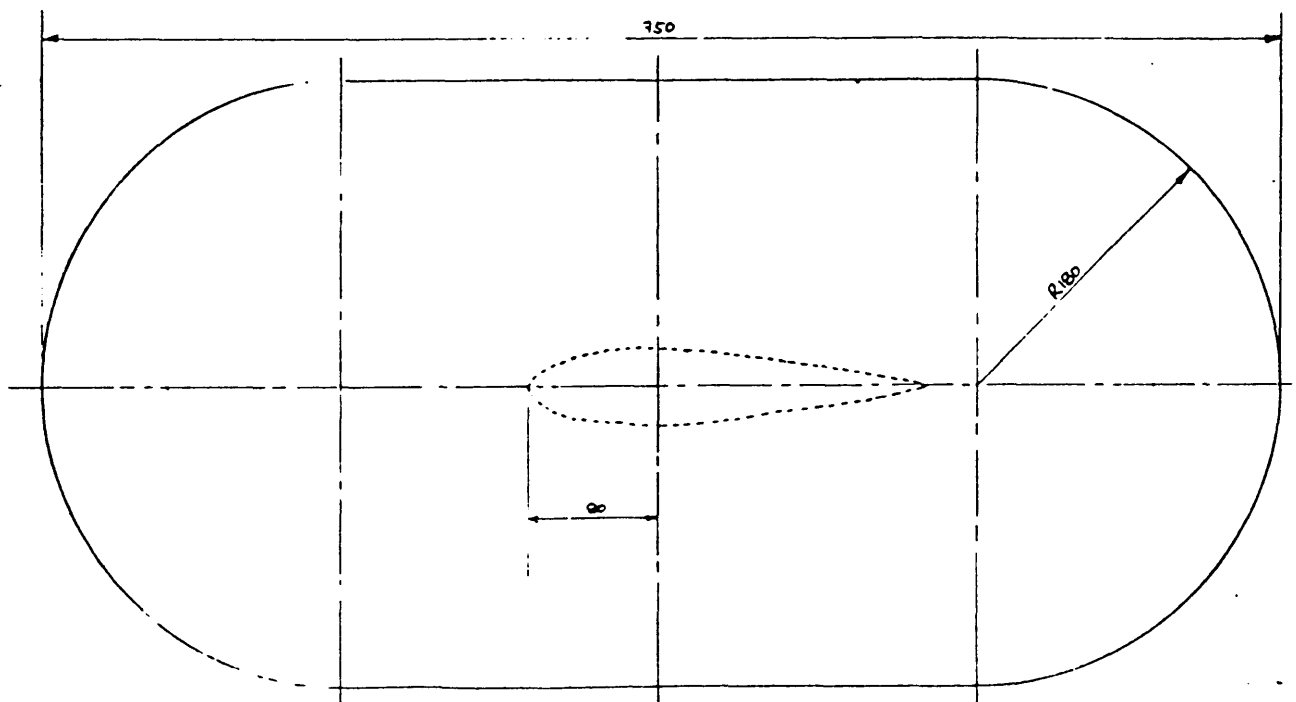


Fig 2.3 Details of the end plate used

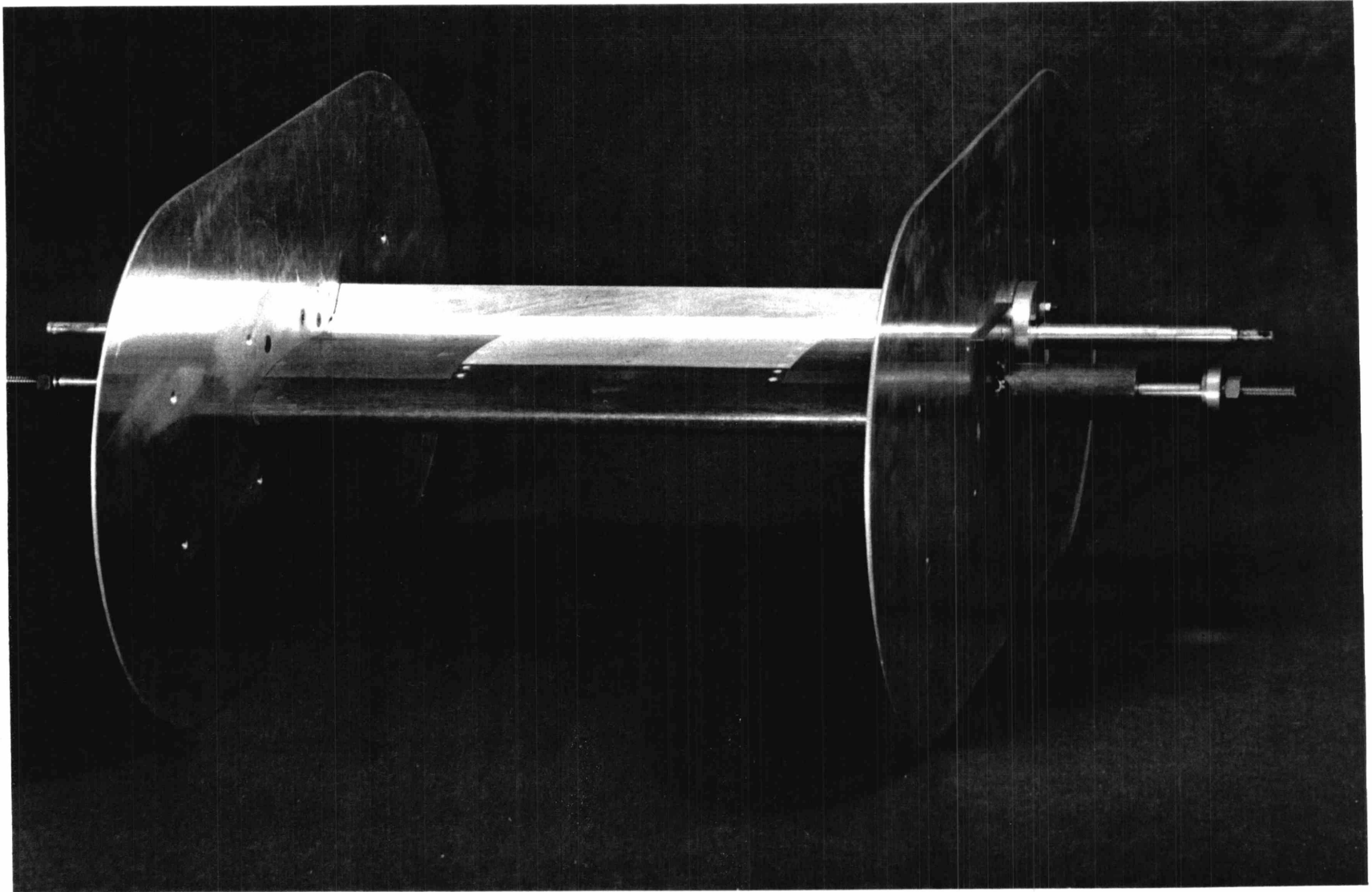


Fig 2.4 Photograph of the aerofoil mounted with spoiler and end plates

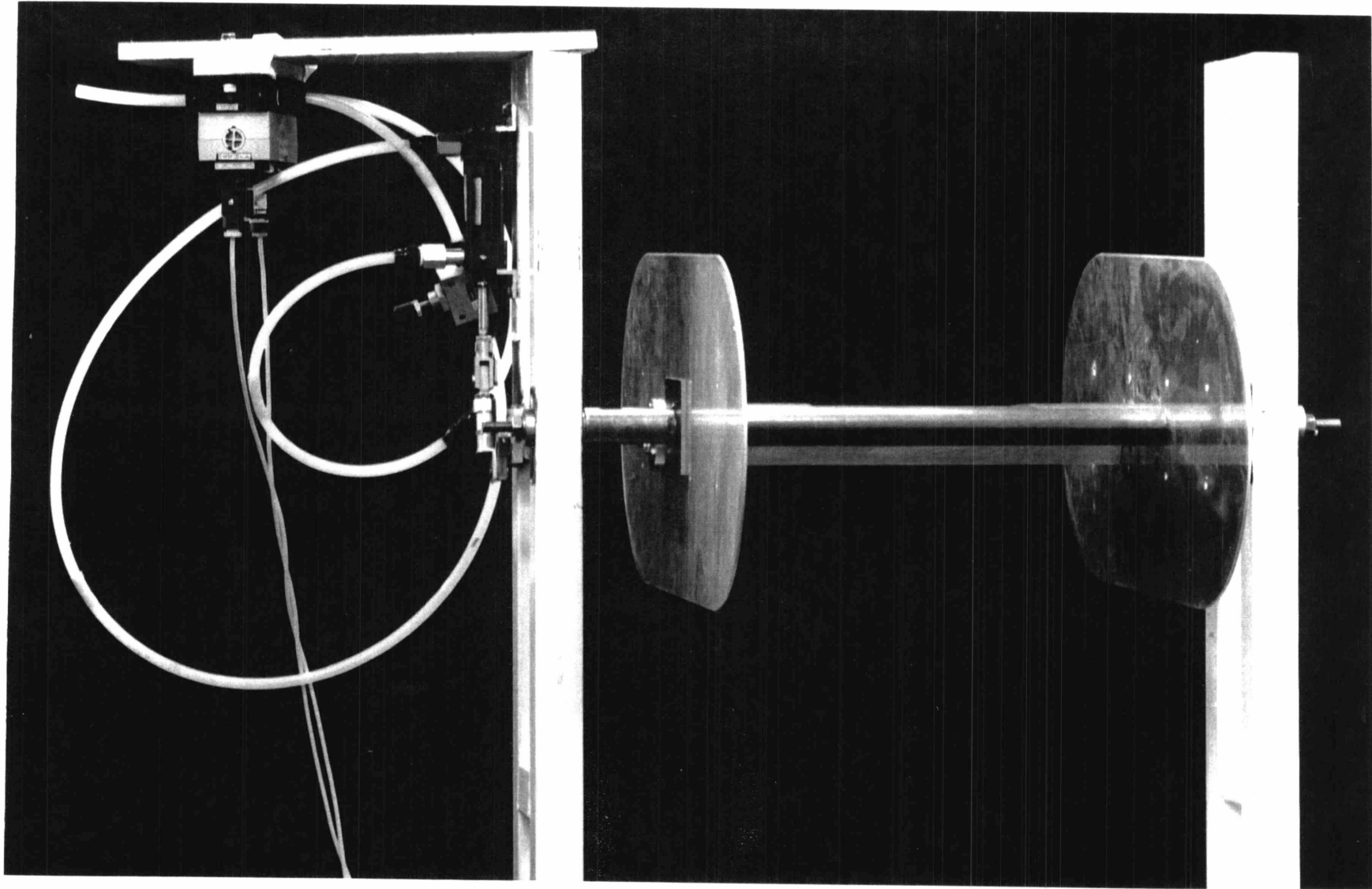


Fig 2.5 Photograph of the aerofoil and spoiler, outside the wind tunnel, showing the pneumatic actuation mechanism.

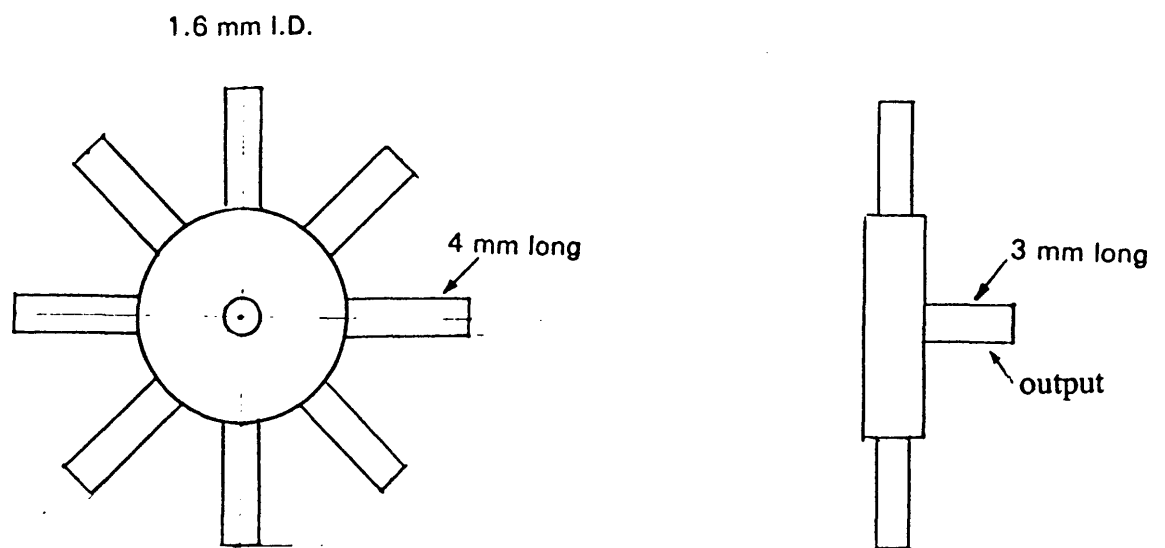


Fig 2.6 Details of pressure averager

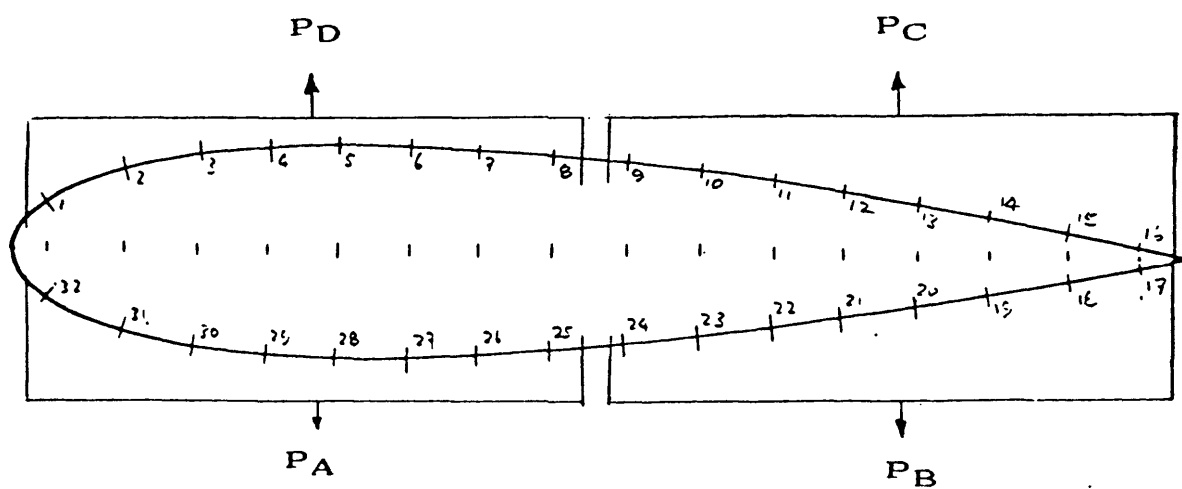


Fig 2.7 Plan of pressure averaging connection

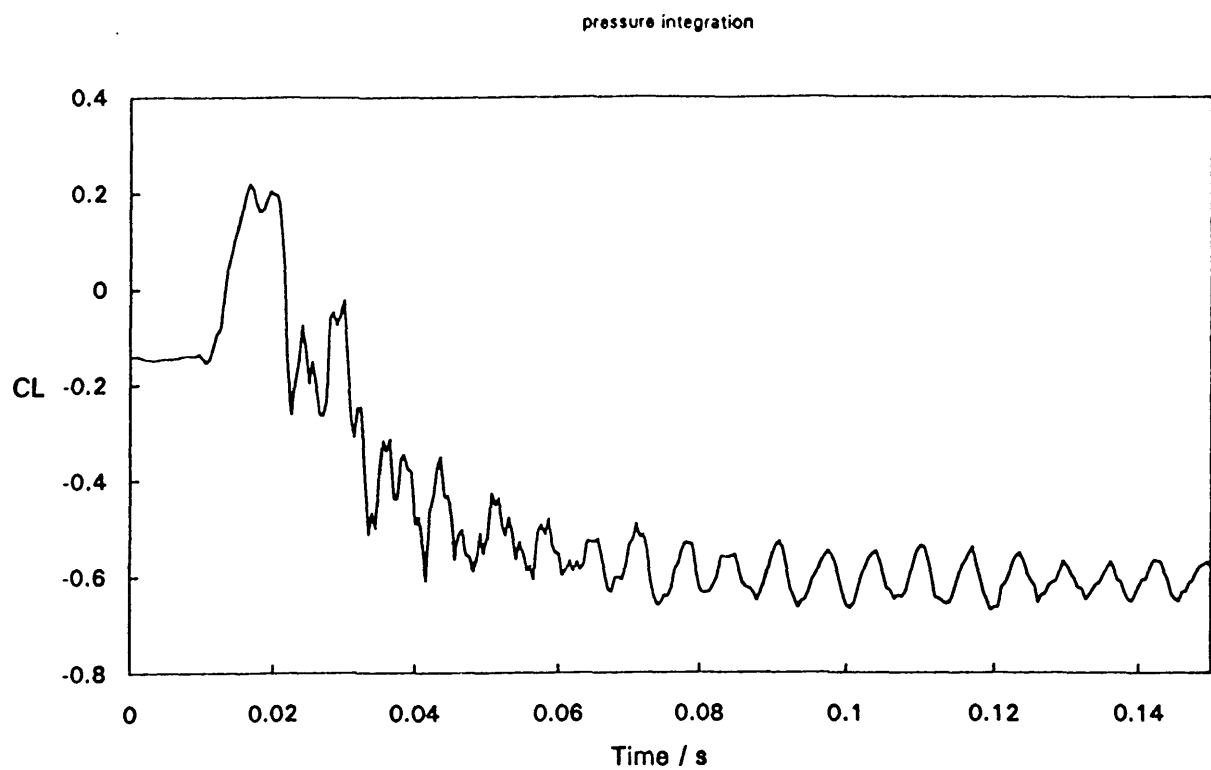


Fig 2.8a C_N obtained through pressure through individual pressure tapping measurements

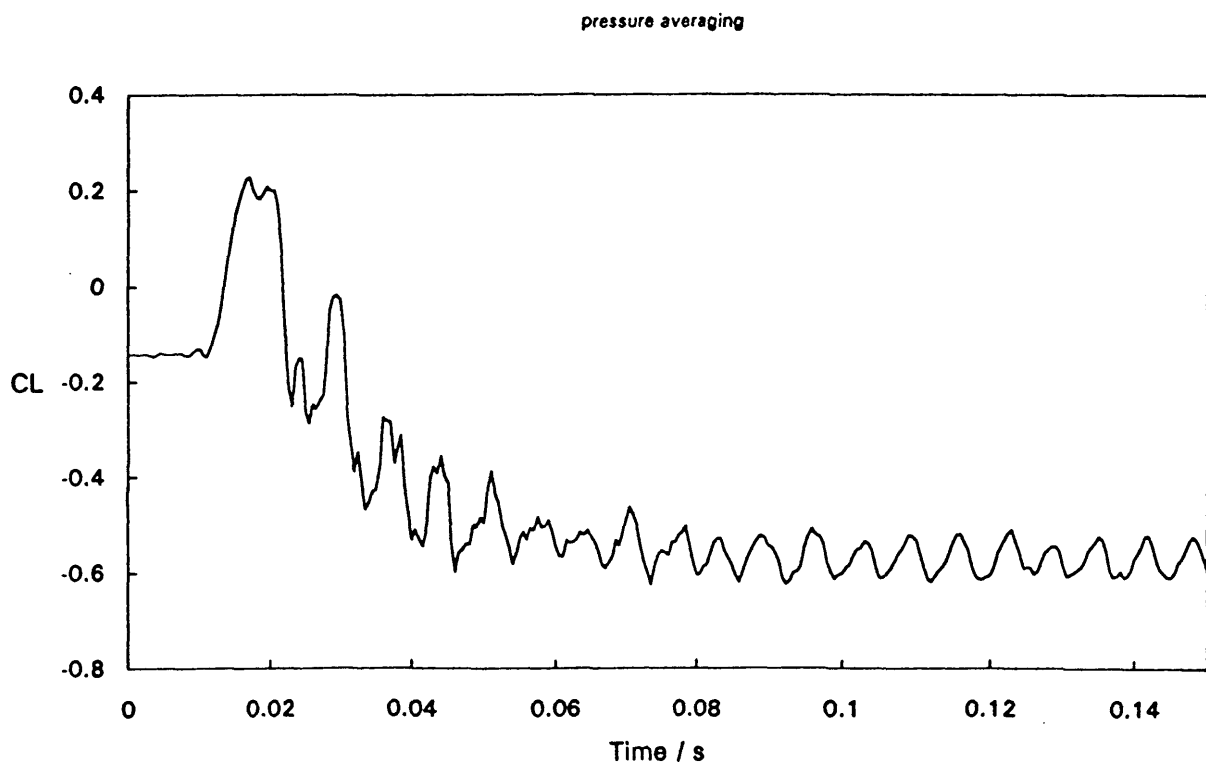


Fig 2.8b C_N obtained through pressure through individual pressure averaging technique

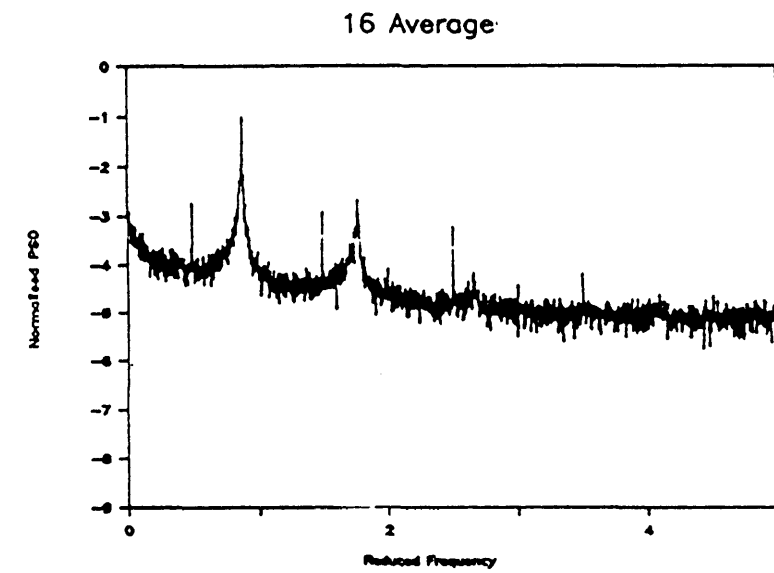
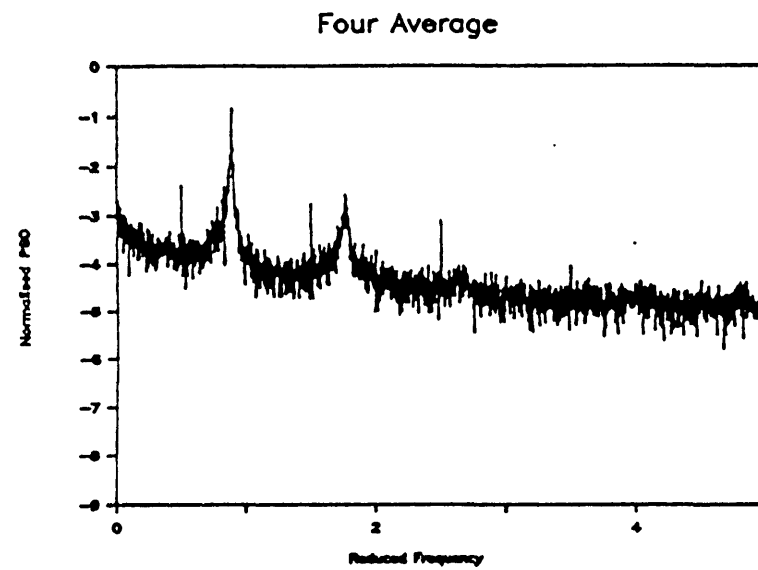
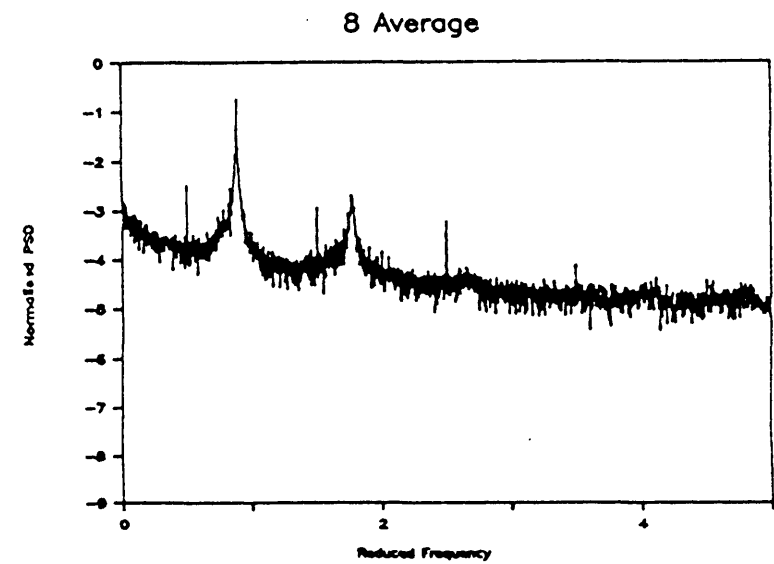
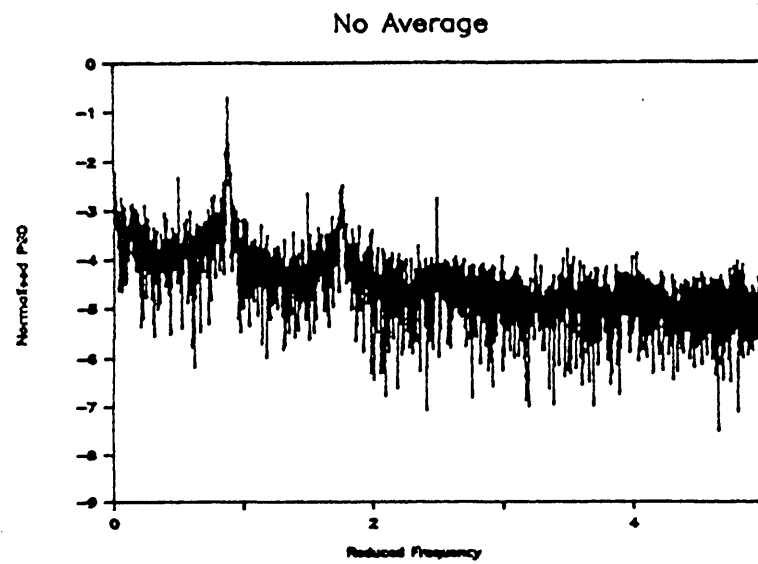


Fig 2.9 Power spectra obtained with different levels of ensemble smoothing

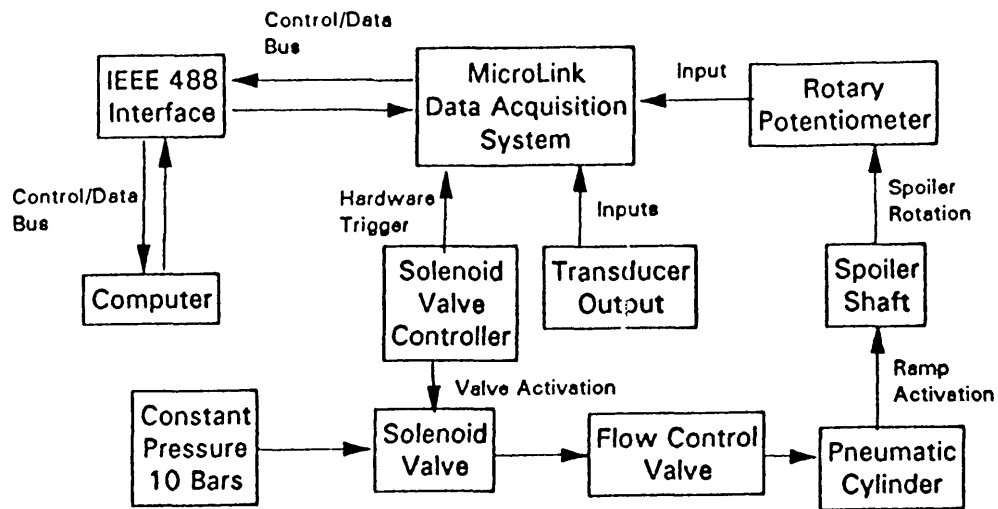


Fig 2.10 Block diagram of connection of various components

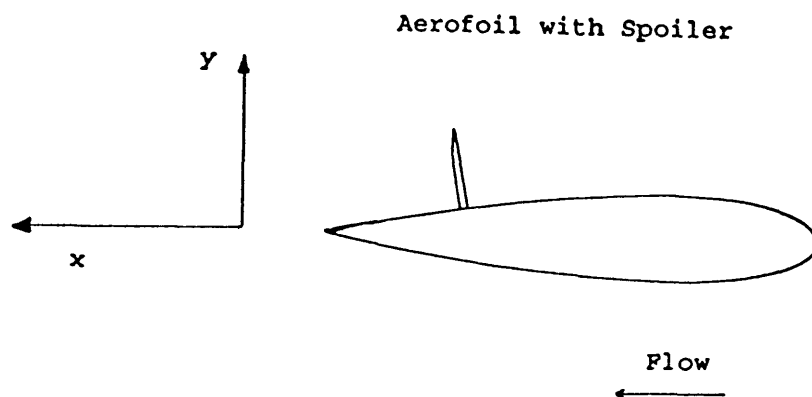


Fig 2.11 x-y plane of aerofoil for hot-wire measurement

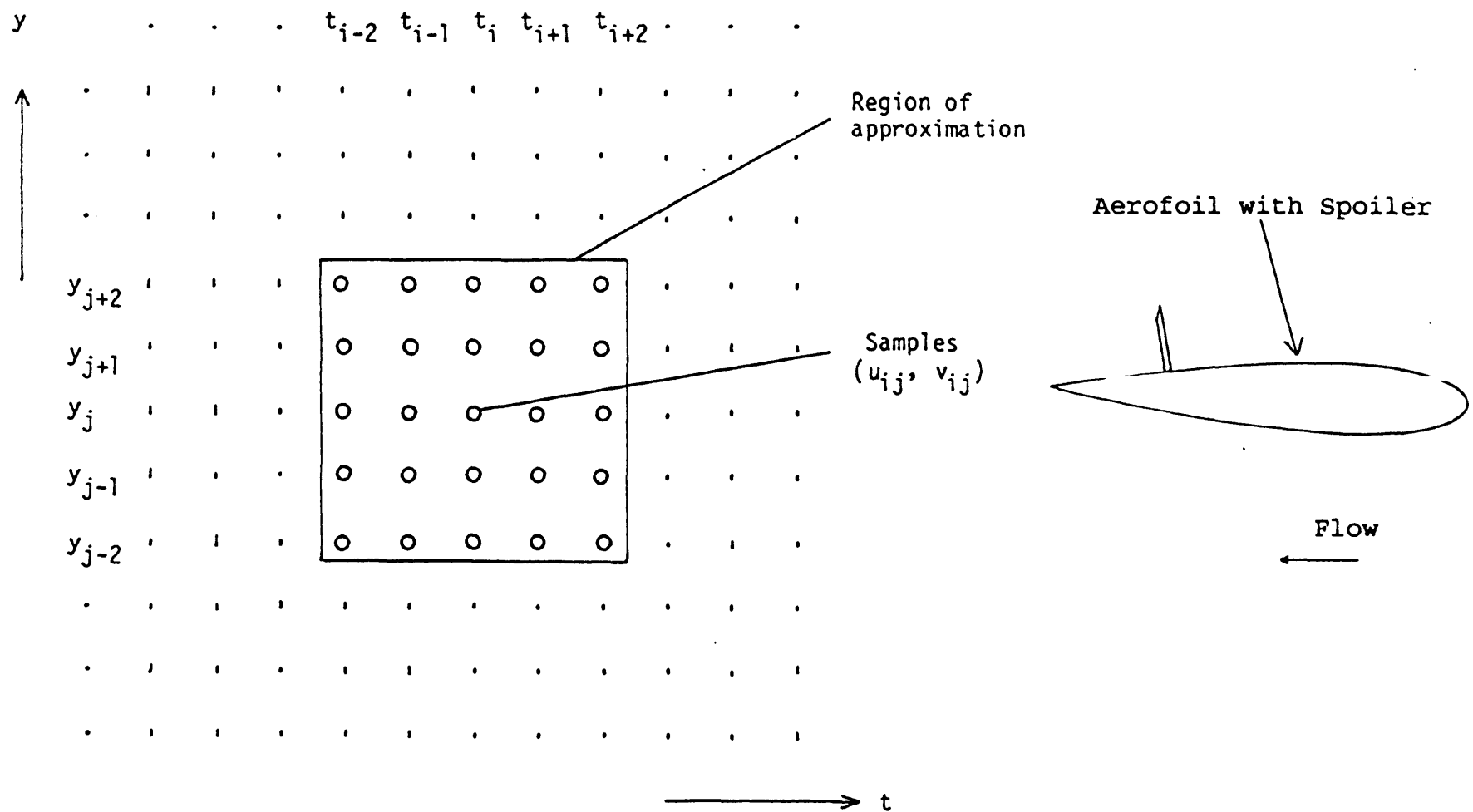


Fig 2.12 t - y plane for calculation of vorticity

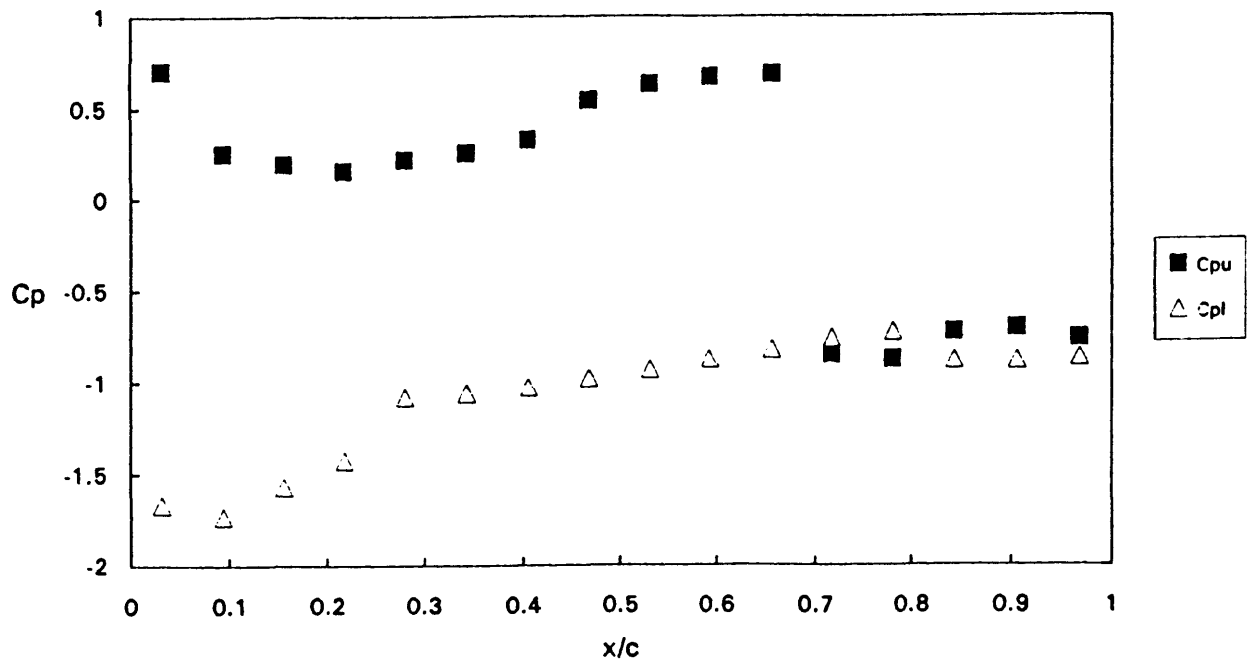


Fig 3.1 Pressure distribution on aerofoil with a spoiler of $h/c = 0.15$ at $x_s/c = 0.7$

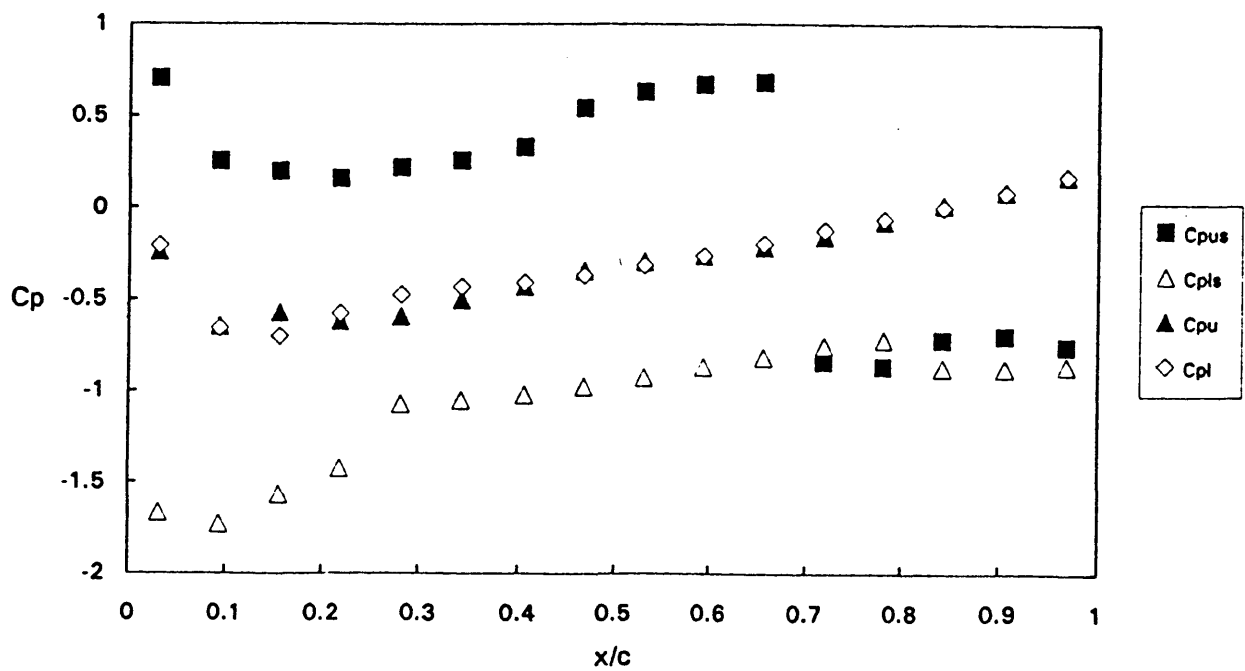


Fig 3.2 Pressure distributions on aerofoil with & without spoiler

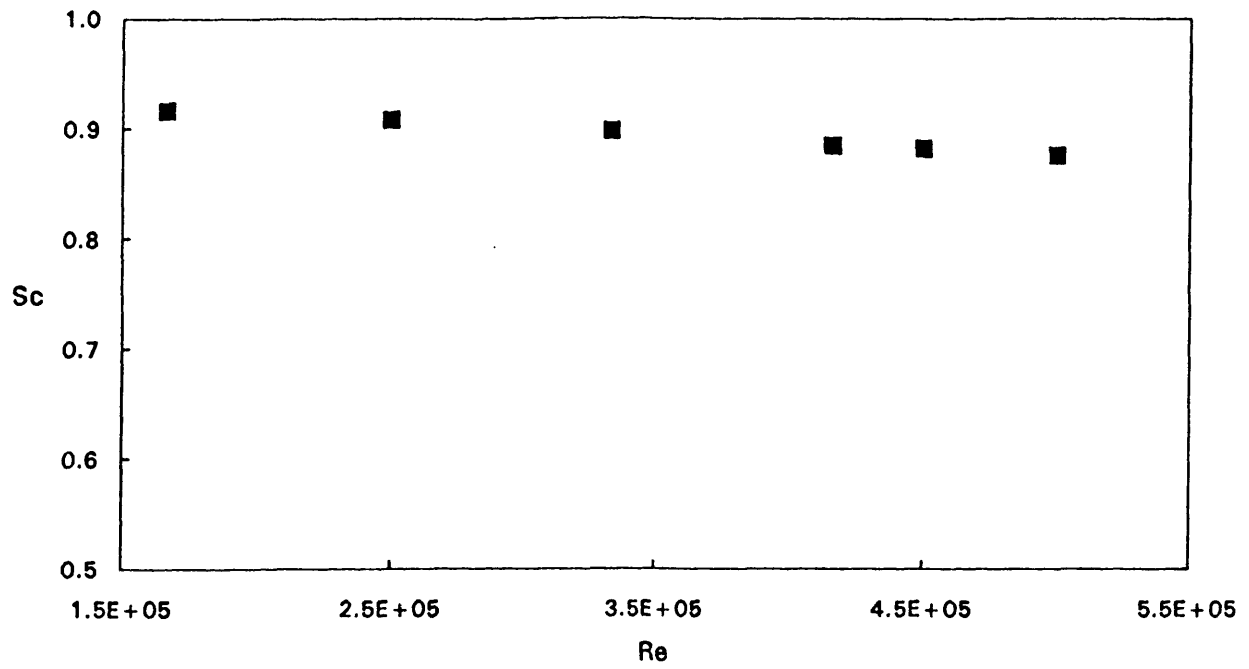


Fig 3.3 Variation of S_c with Re , for $h/c = 0.15$, $x_s/c = 0.7$

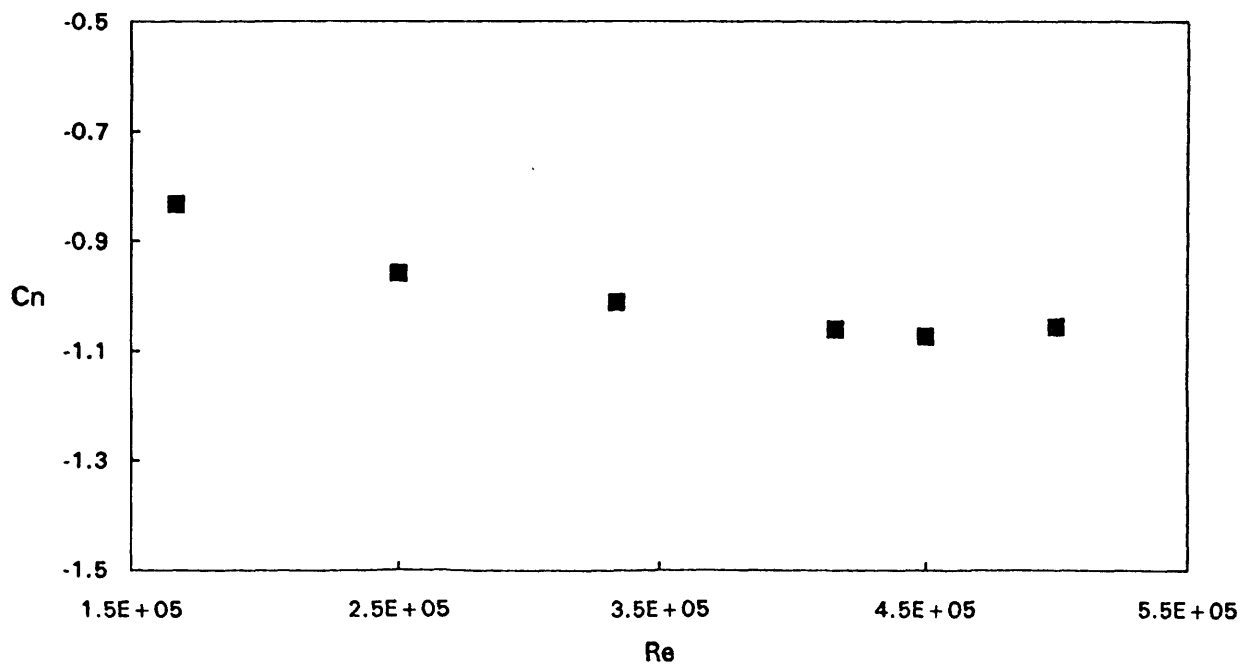


Fig 3.4 Variation of C_N with Re , for $h/c = 0.15$, $x_s/c = 0.7$

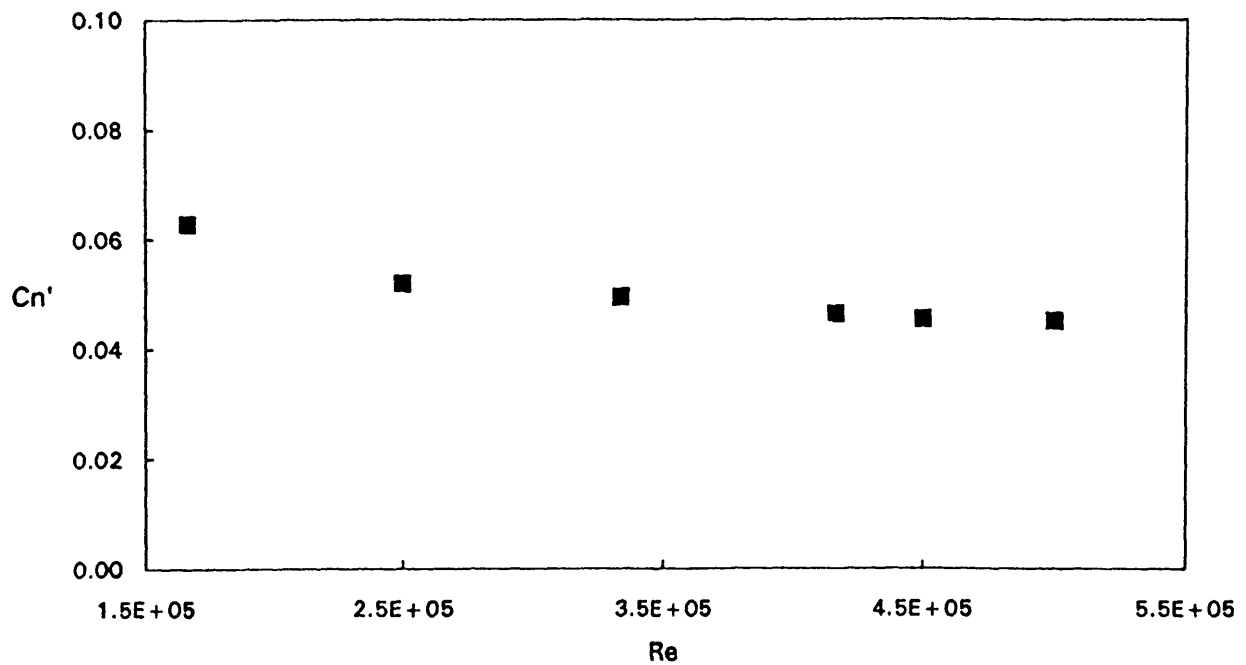


Fig 3.5 Variation of C_N' with Re , for $h/c = 0.15$, $x_g/c = 0.7$

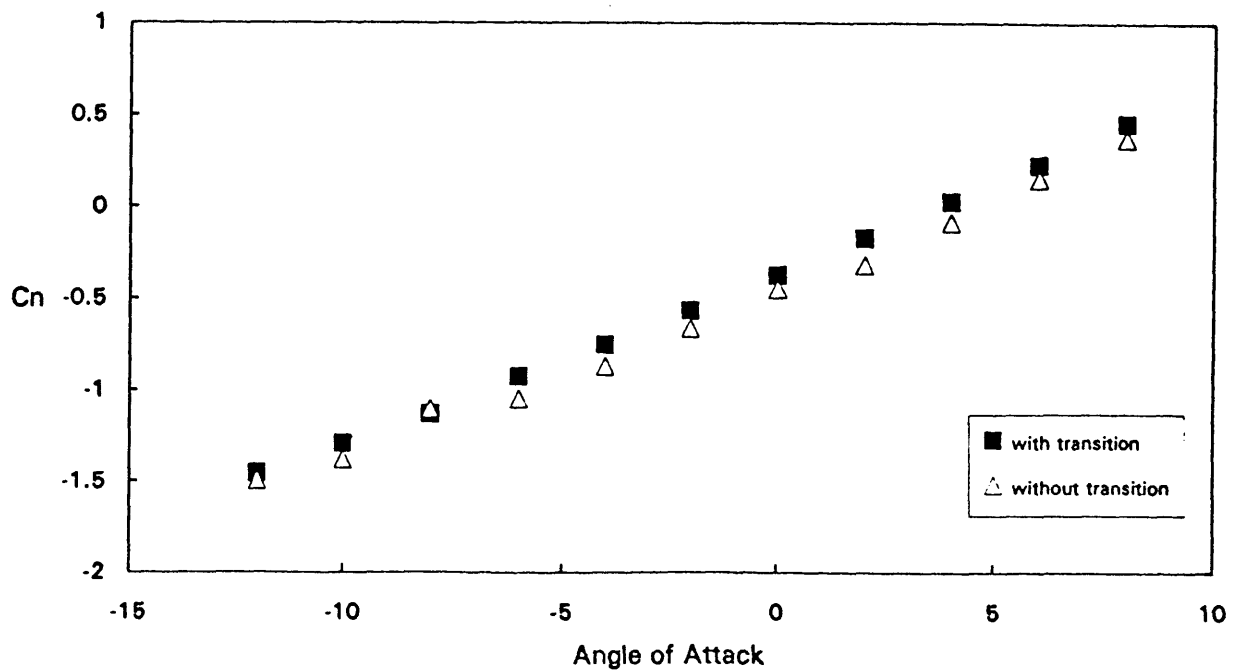


Fig 3.6 Variation of C_N with α , with and without transition strips fitted

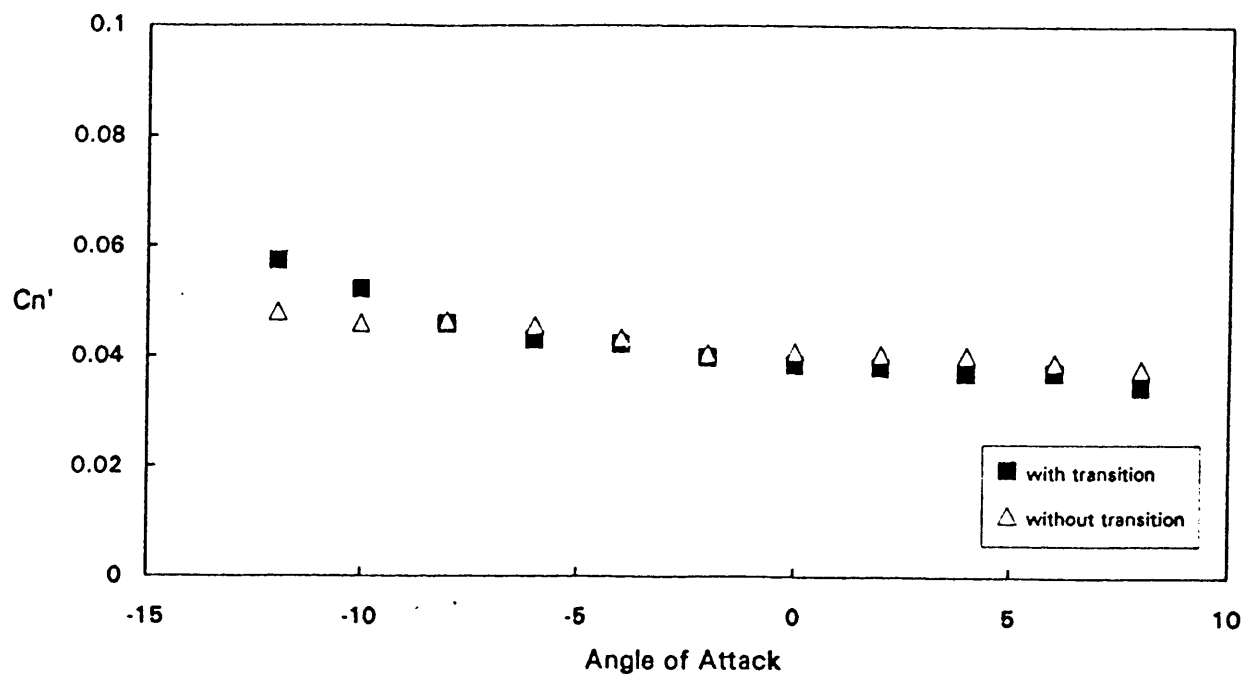


Fig 3.7 Variation of C_N' with α , with and without transition strips fitted

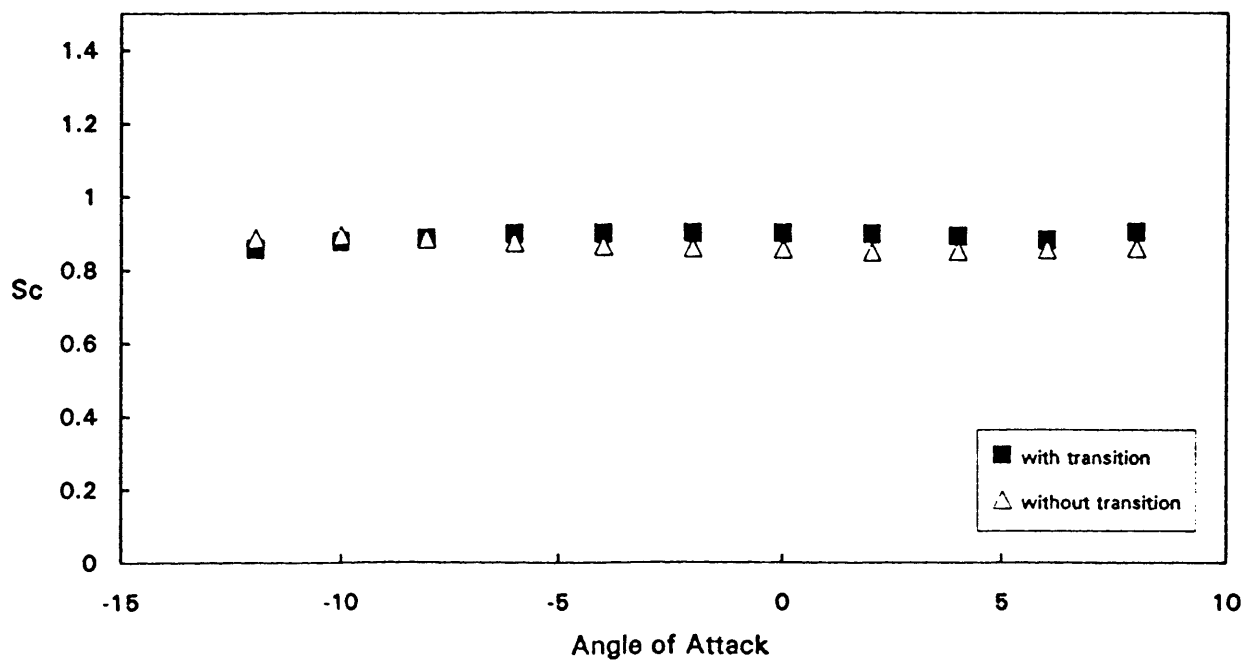


Fig 3.8 Variation of S_c with α , with and without transition strips fitted

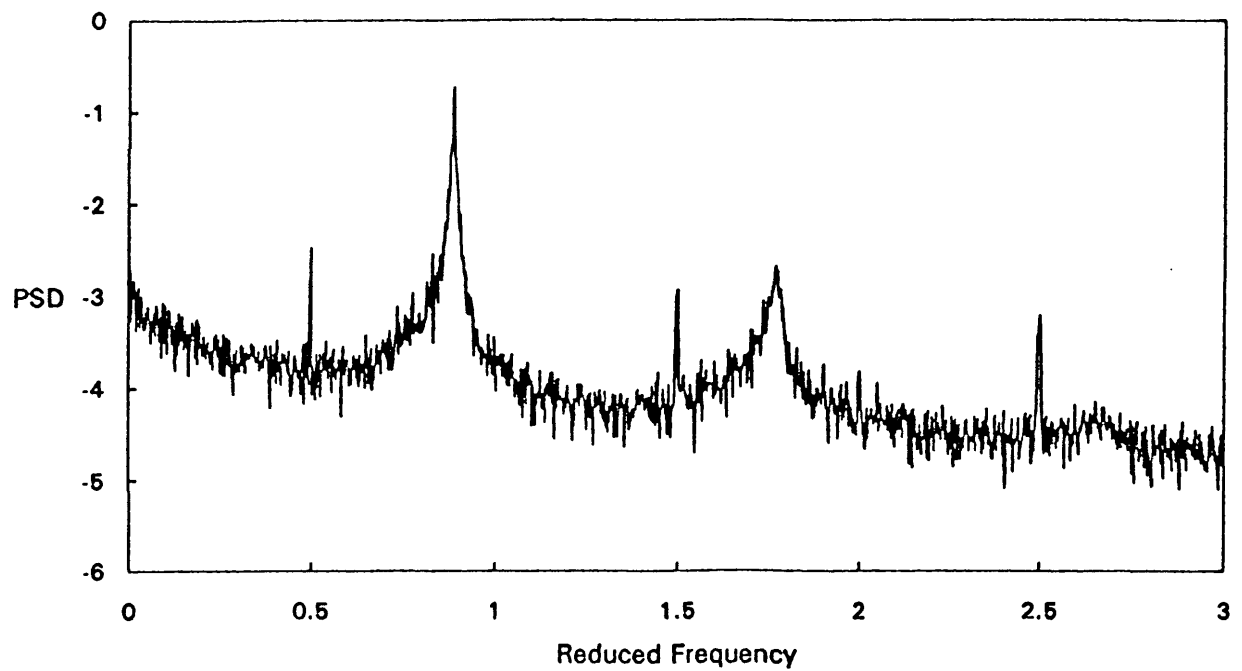


Fig 3.9a Power spectrum of C_N at $\alpha = -2^\circ$, with no transition strips fitted

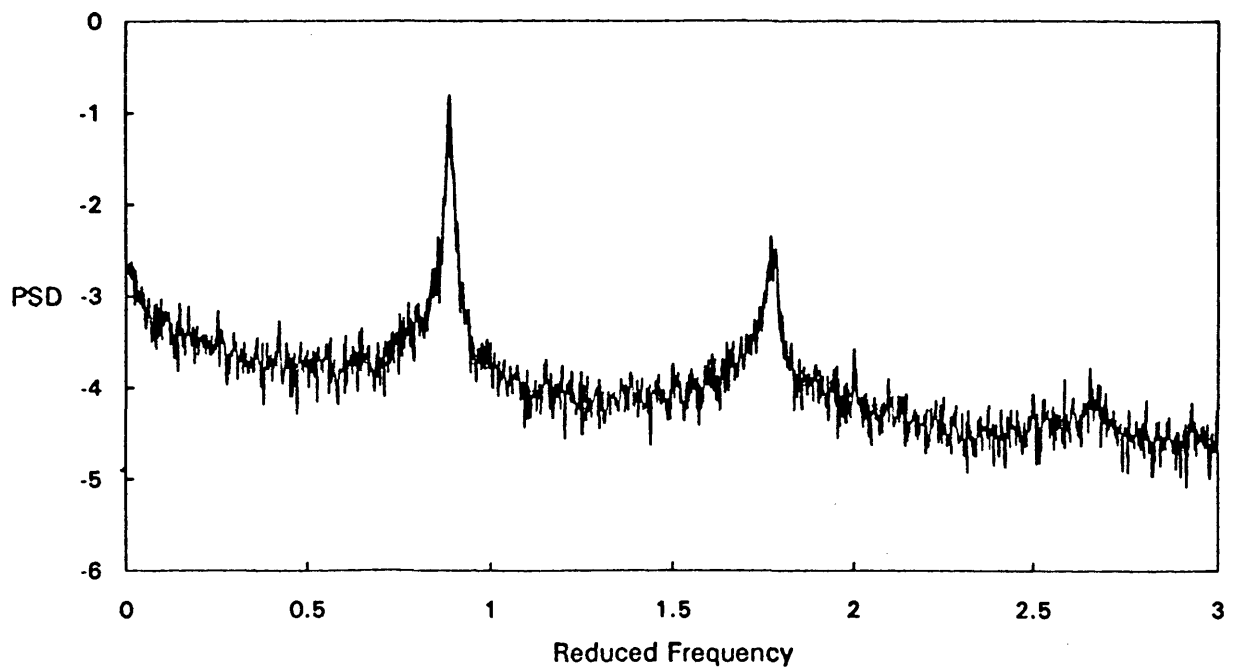


Fig 3.9b Power spectrum of C_N at $\alpha = -2^\circ$, with transition strips fitted

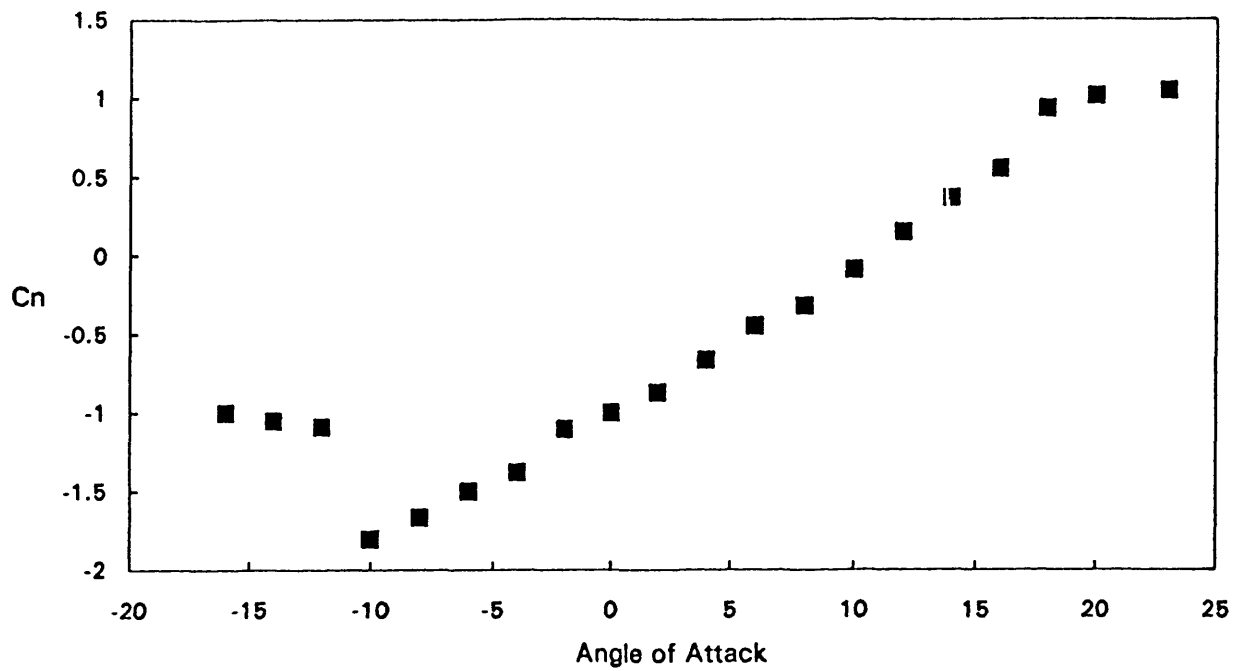


Fig 3.10a Variation of C_N with α for $h/c = 0.15$, $x_s/c = 0.7$

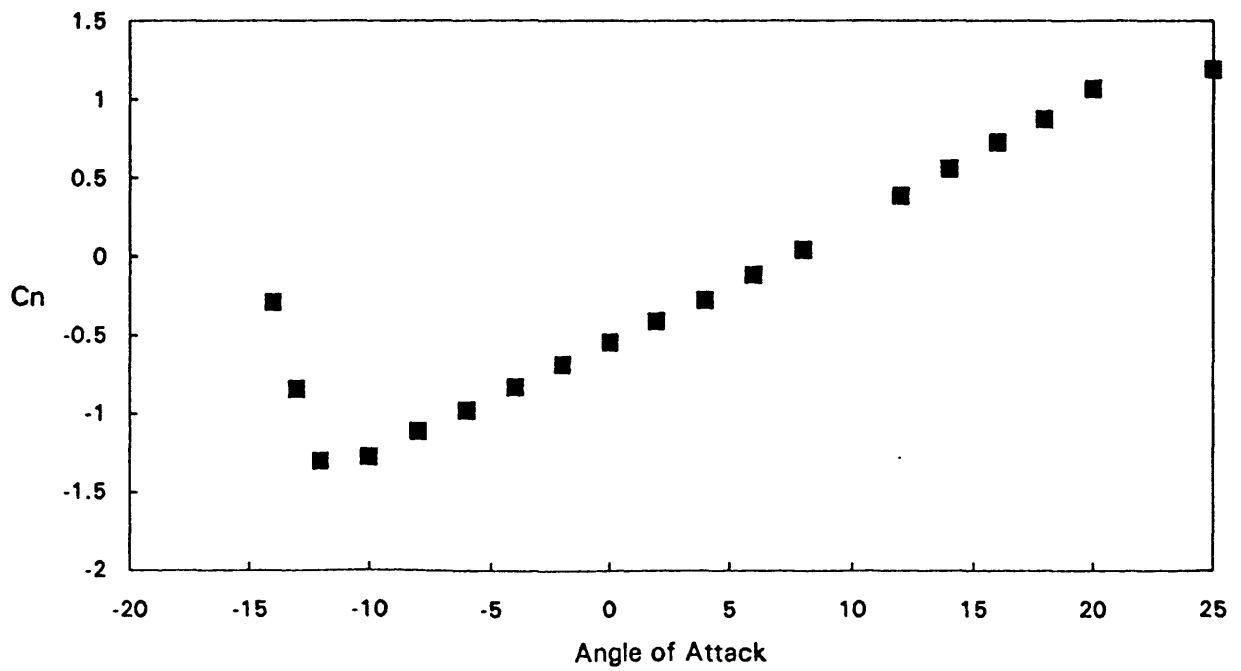


Fig 3.10b Variation of C_N with α for $h/c = 0.10$, $x_s/c = 0.3$

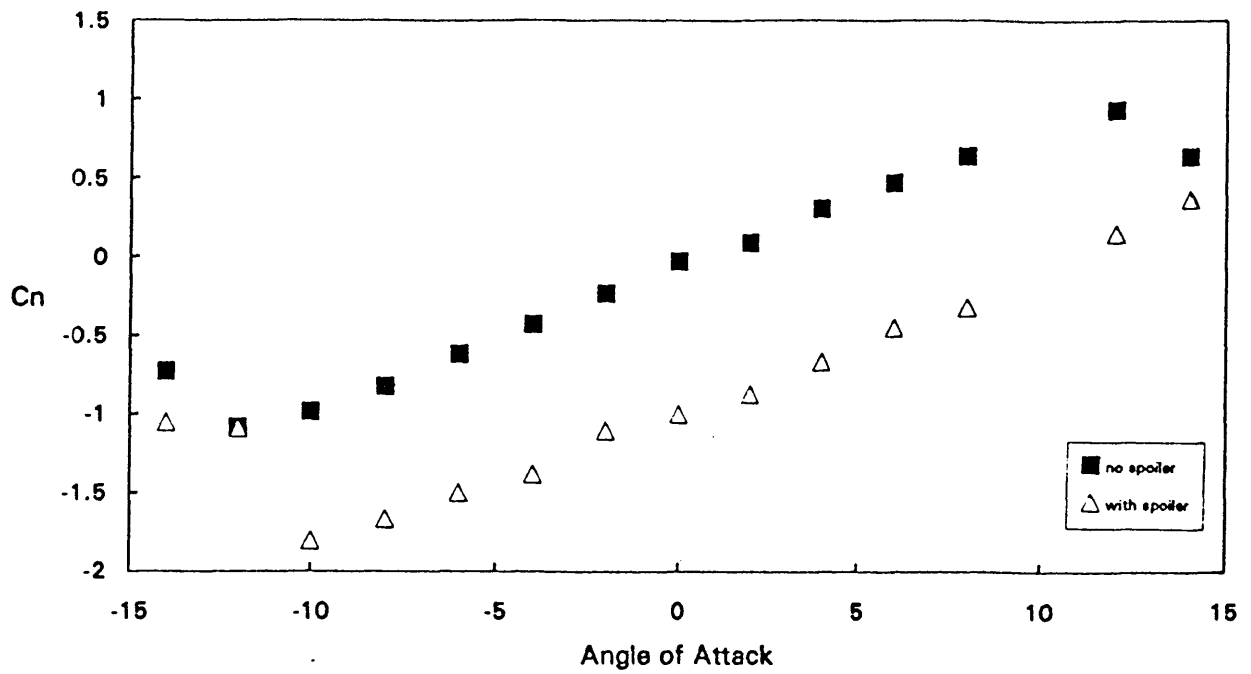


Fig 3.11 Variation of C_N with α , for aerofoil with and without spoiler of $h/c = 0.15$, $x_s/c = 0.7$

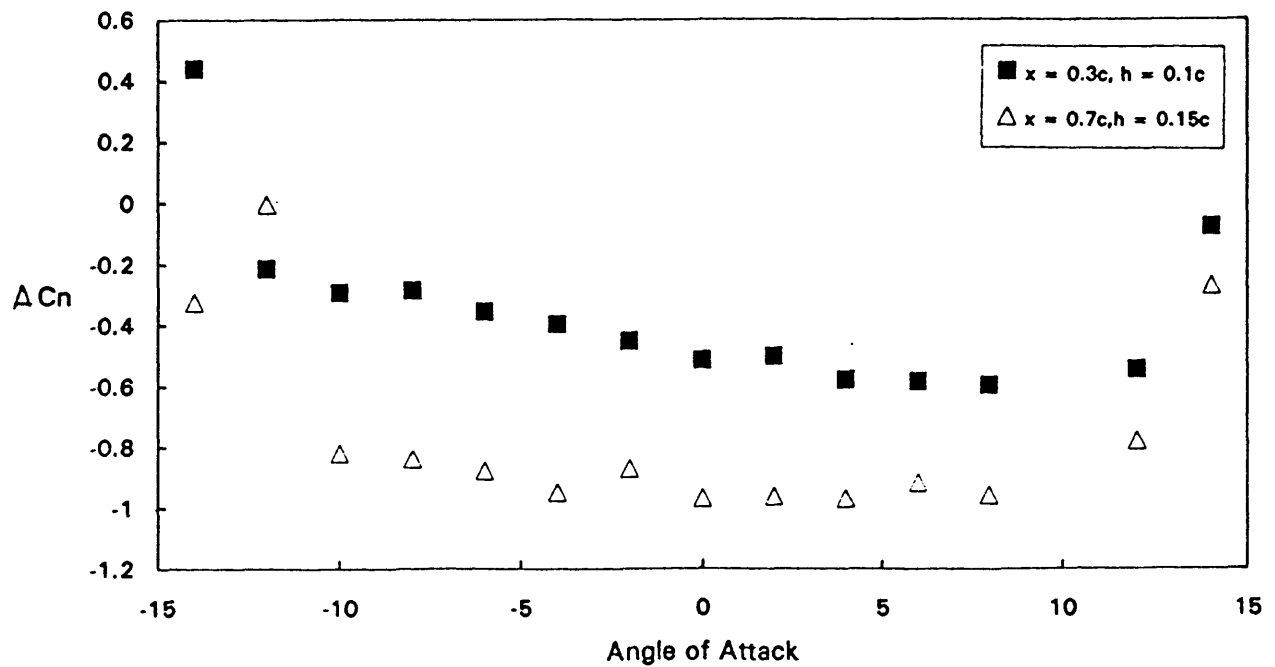


Fig 3.12 Variation of spoiler effectiveness in terms of differential C_N with α , for two spoiler configurations

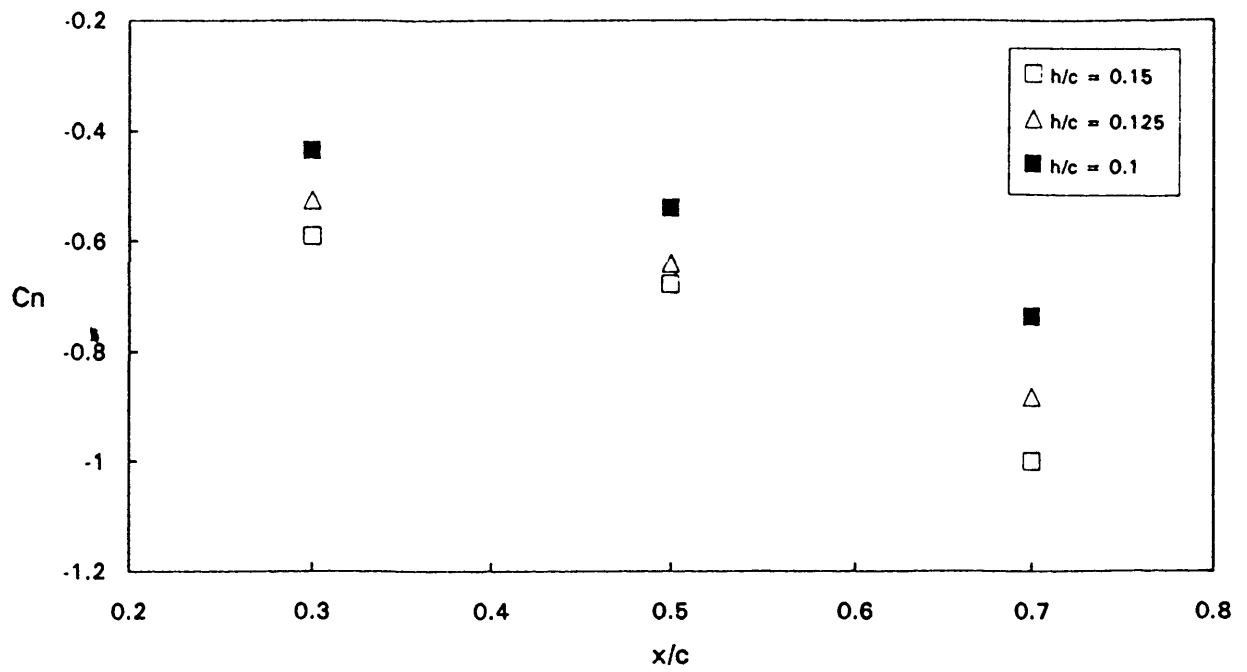


Fig 3.13a Variation of C_N with x_s/c for different h/c , $\alpha = 0^\circ$

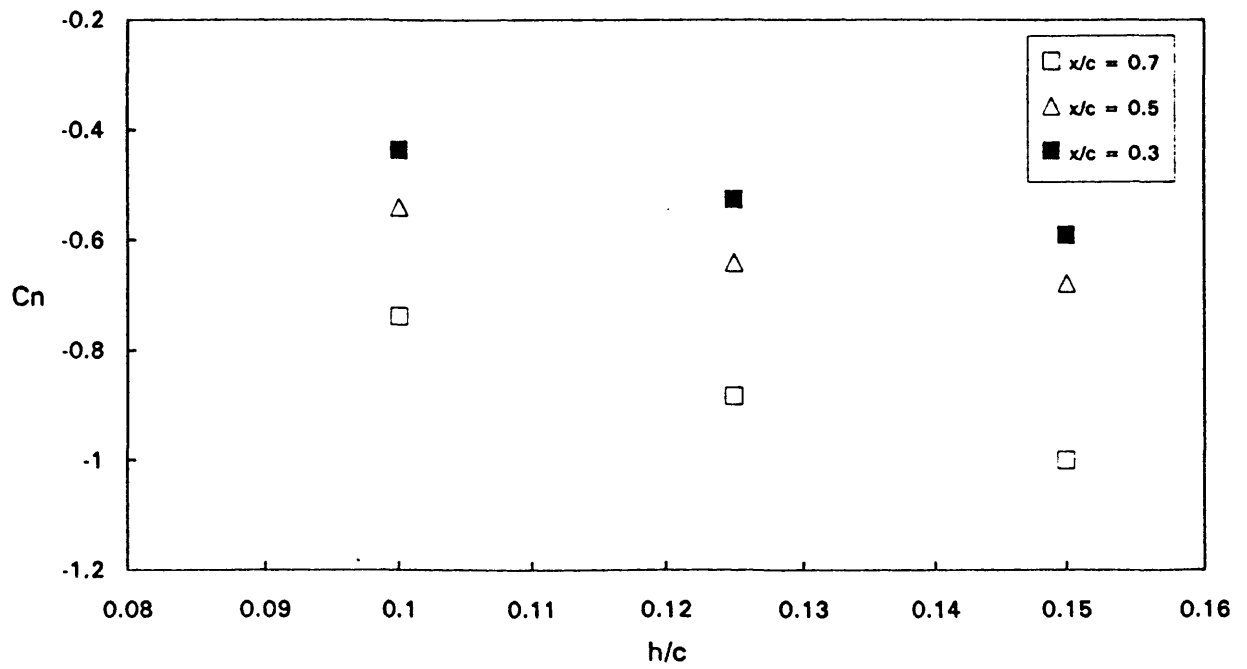


Fig 3.13b Variation of C_N with h/c for different x_s/c , $\alpha = 0^\circ$

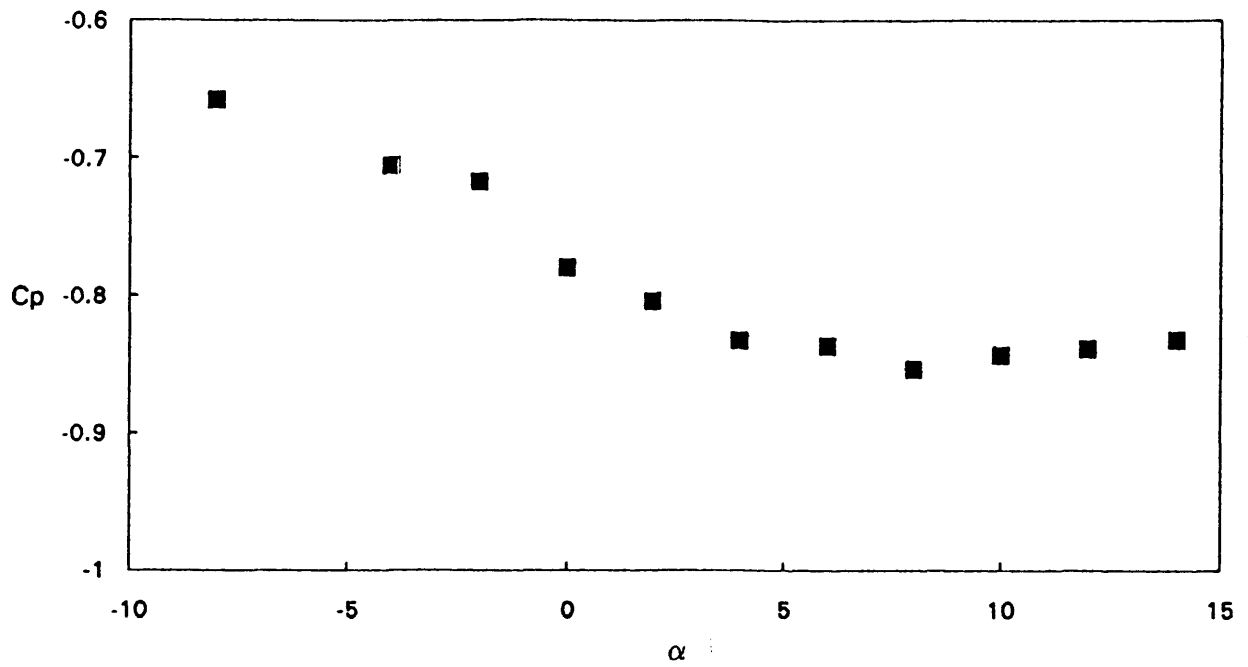


Fig 3.14 Variation of base pressure with α , for $h/c = 0.15$, $x_s/c = 0.7$

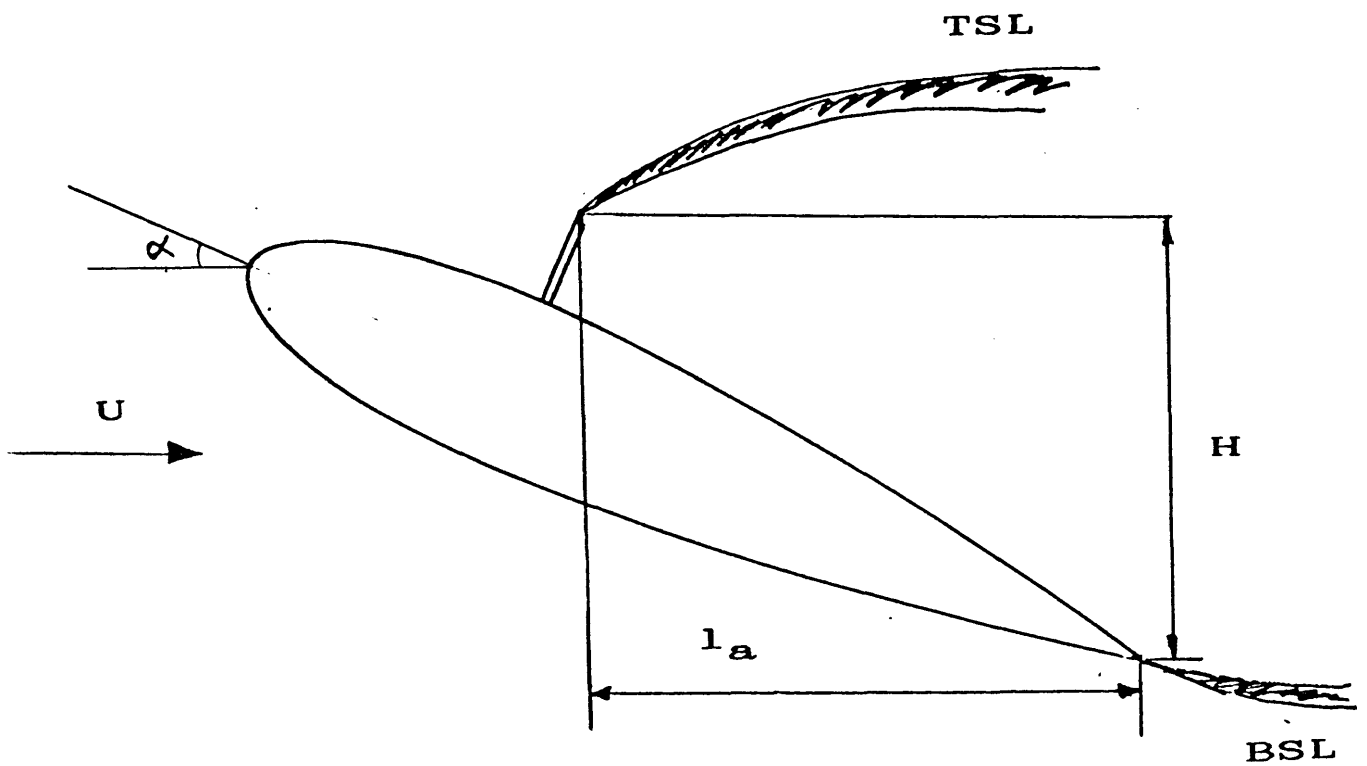


Fig 3.15 Definitions of parameters related to an aerofoil with spoiler

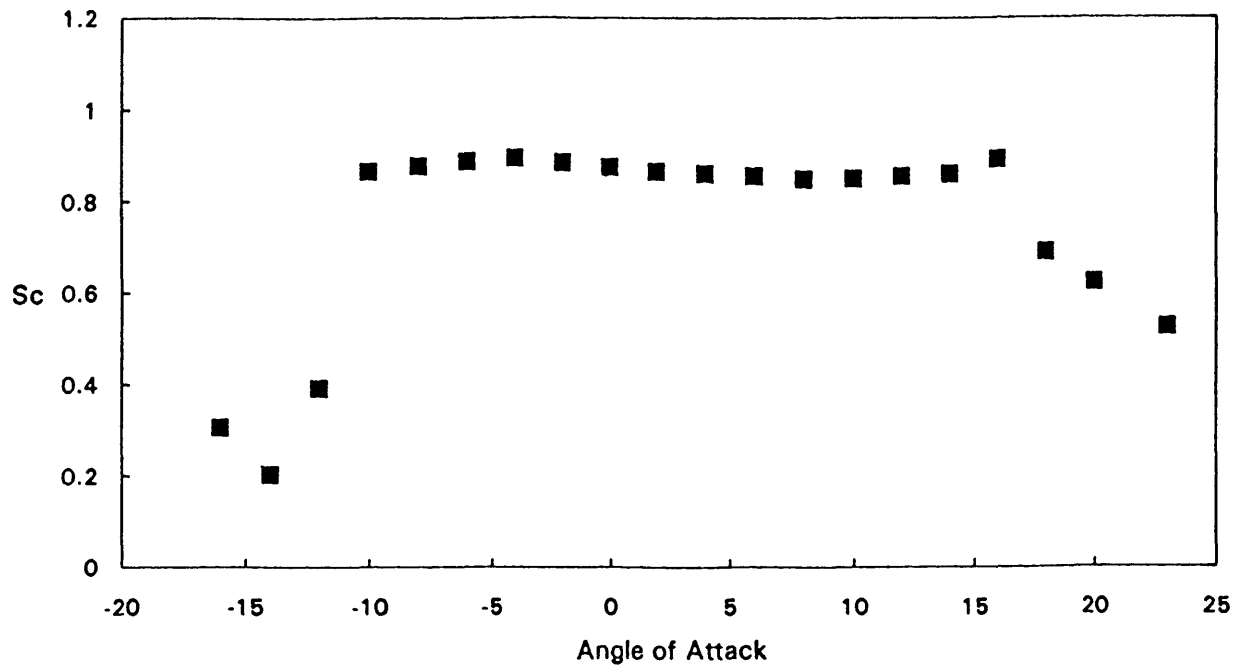


Fig 3.16a Variation of Sc with α , for $h/c = 0.15$, $x_s/c = 0.7$

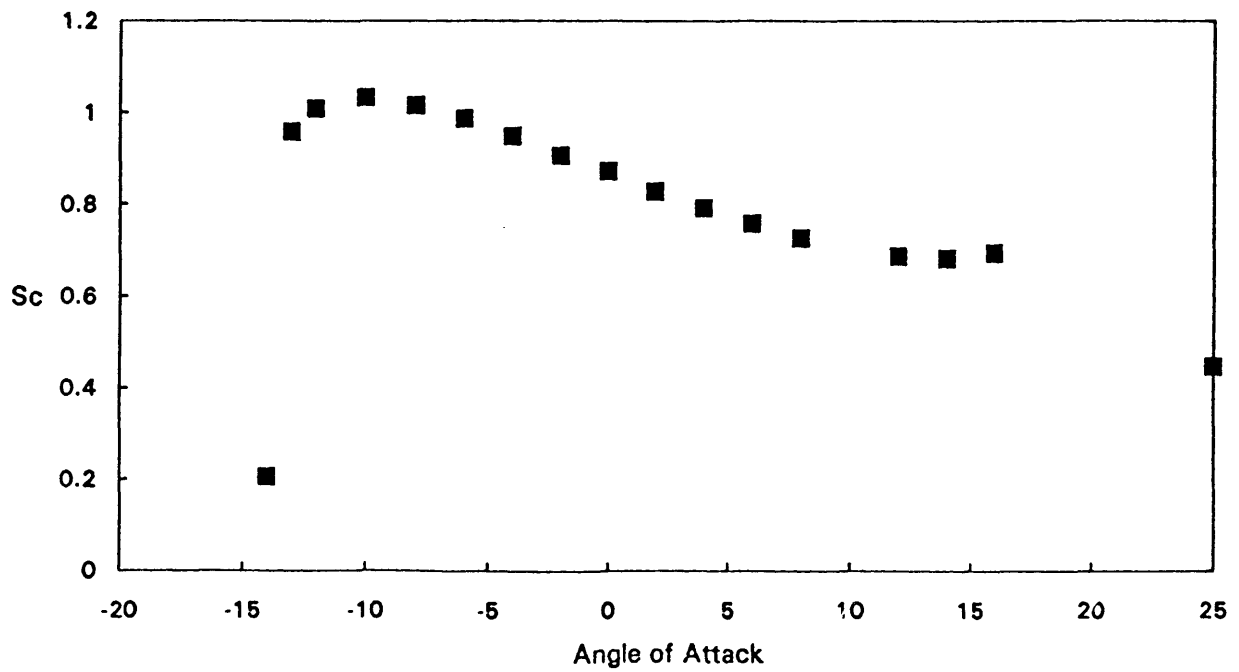


Fig 3.16b Variation of Sc with α , for $h/c = 0.10$, $x_s/c = 0.3$

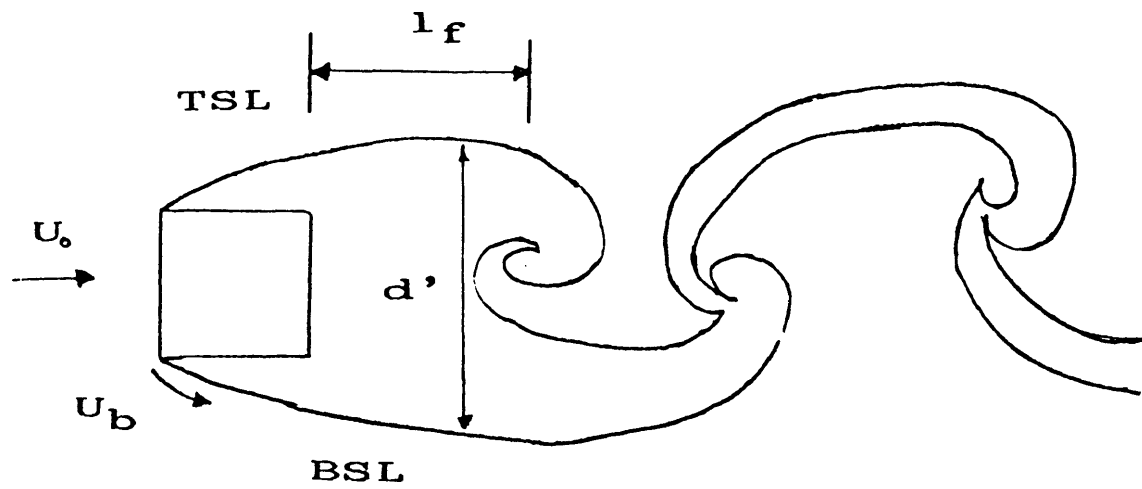


Fig 3.17 Definitions of parameters related to flow past symmetrical bluff body

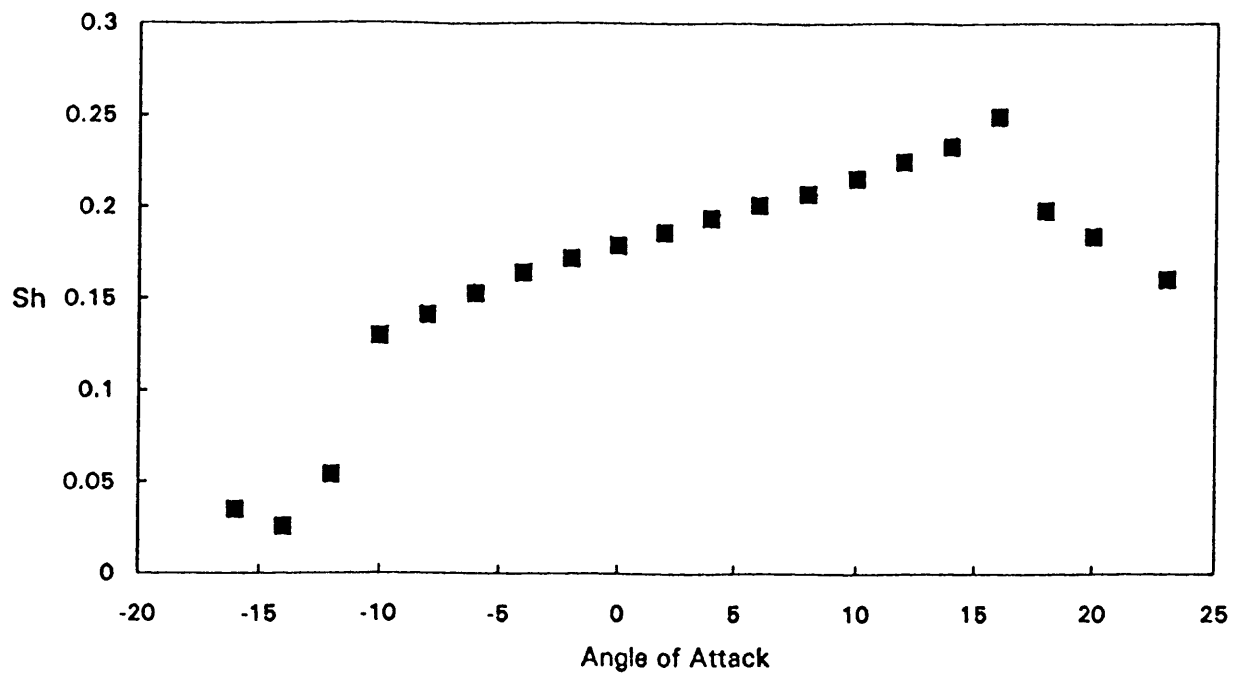


Fig 3.18a Variation of S_H with α , for $h/c = 0.15$, $x_s/c = 0.7$

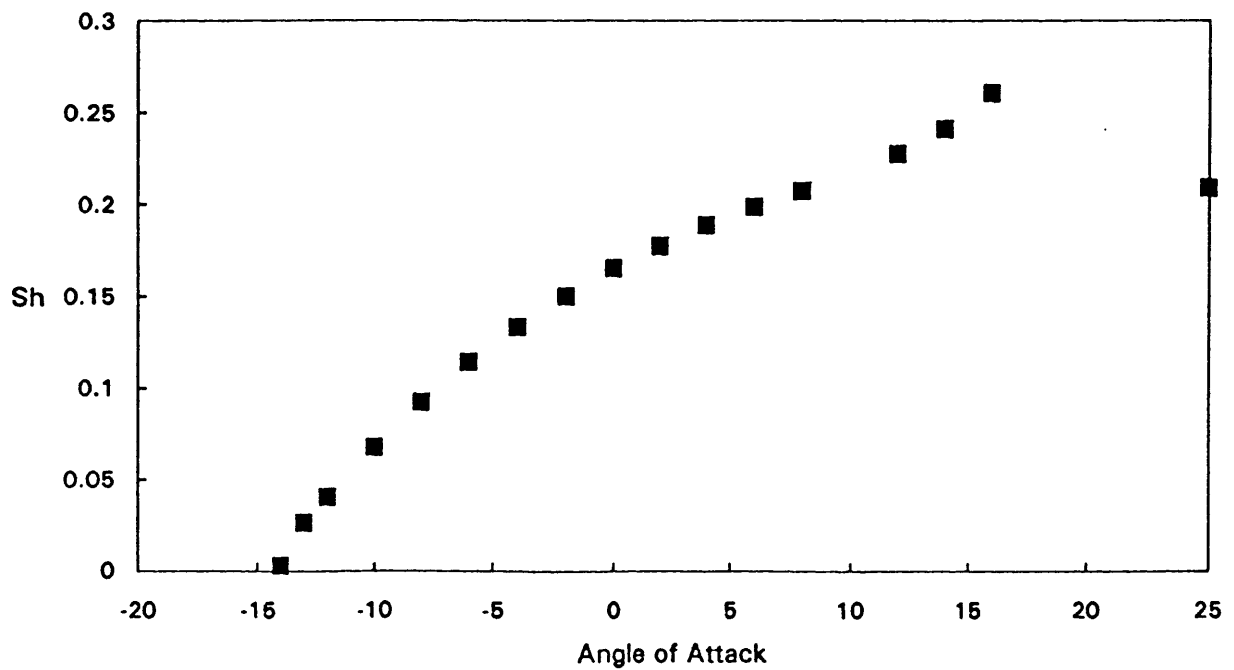


Fig 3.18b Variation of S_H with α , for $h/c = 0.10$, $x_s/c = 0.3$

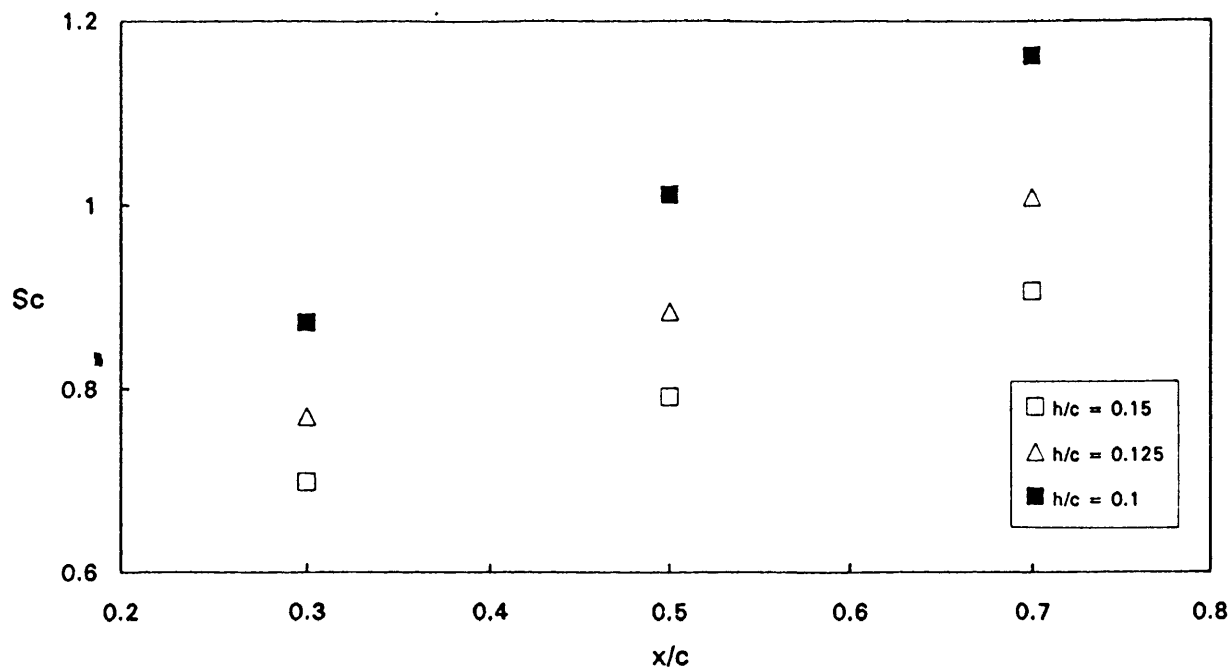


Fig 3.19a Variations of S_c with x_s/c , for different h/c at $\alpha = 0^\circ$

Fig 3.15b

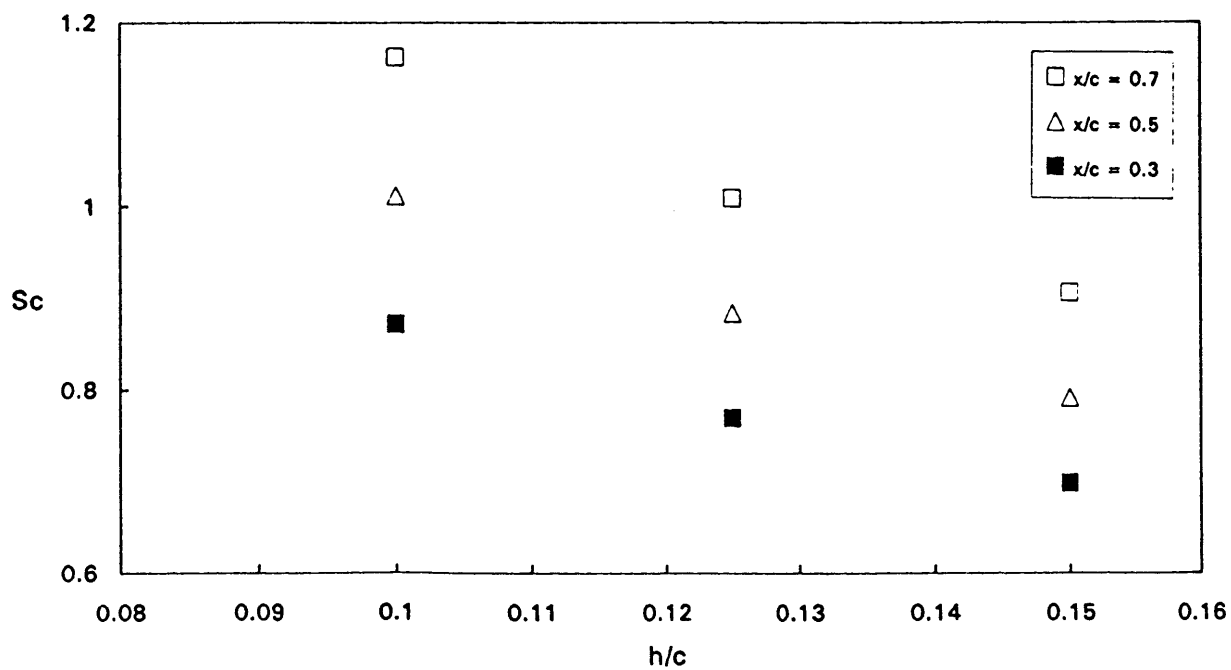


Fig 3.19b Variations of S_c with h/c , for different x_s/c at $\alpha = 0^\circ$

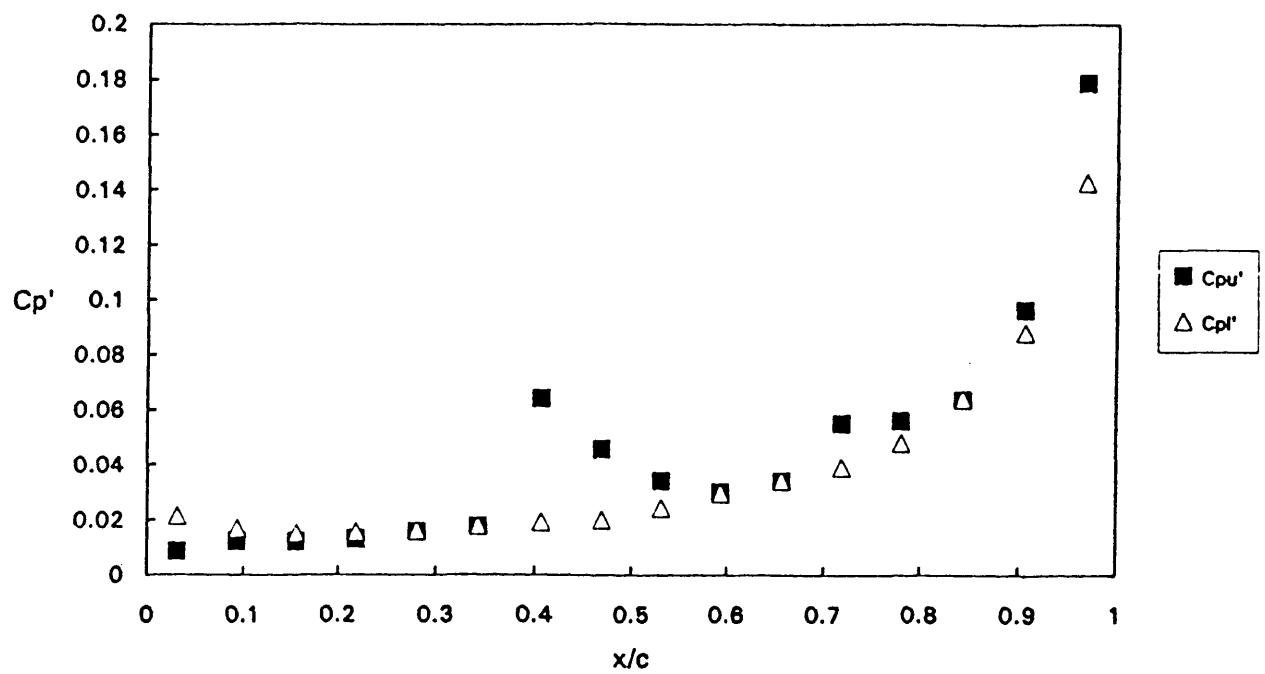


Fig 3.20 Rms pressure distributions on aerofoil with spoiler of $h/c = 0.15$, $x_s/c = 0.7$ at $\alpha = 0^\circ$

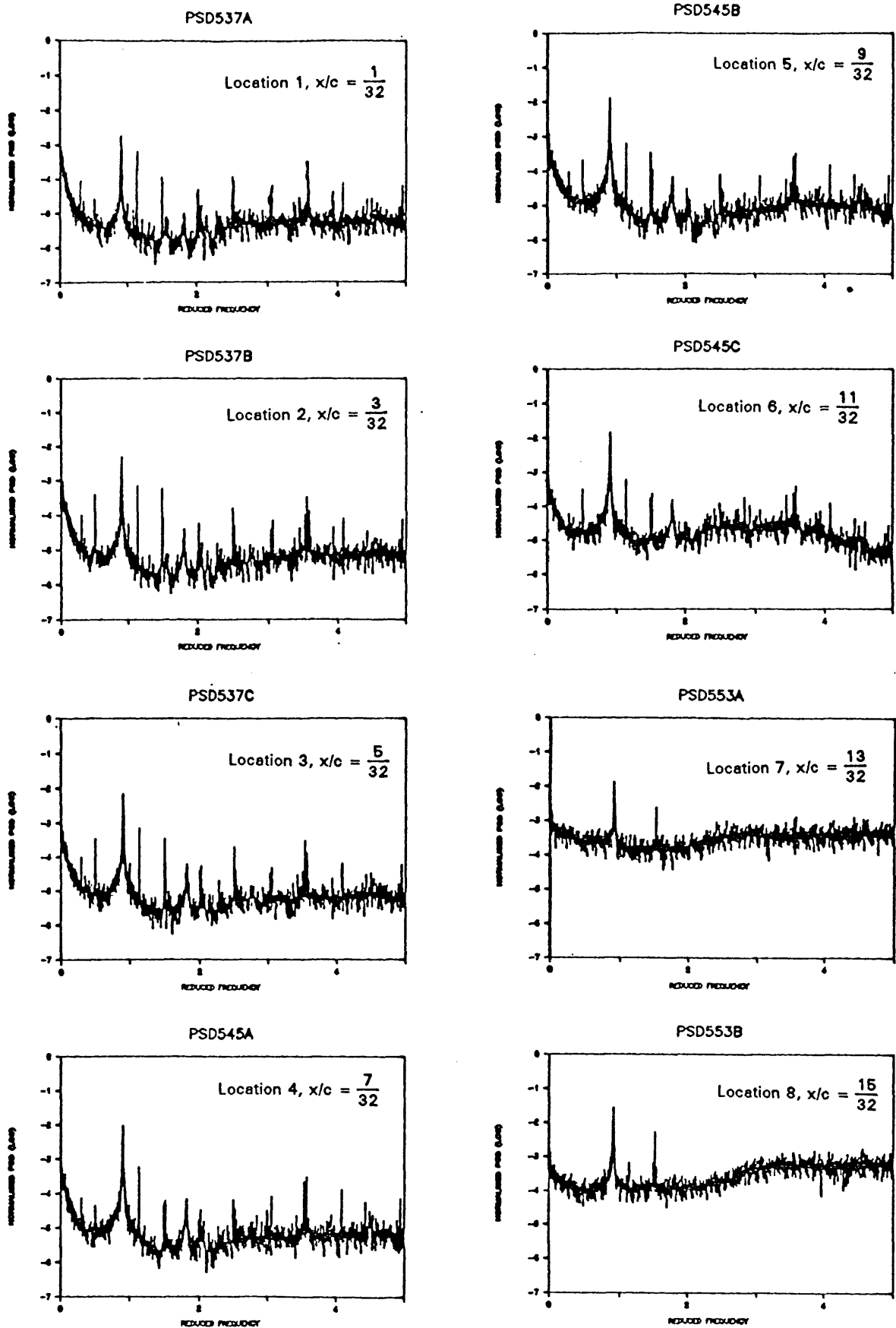


Fig 3.21a Power spectra of C_p at $\alpha = 0^\circ$, with spoiler of $h/c = 0.15$, $x_s/c = 0.7$ on aerofoil upper surface

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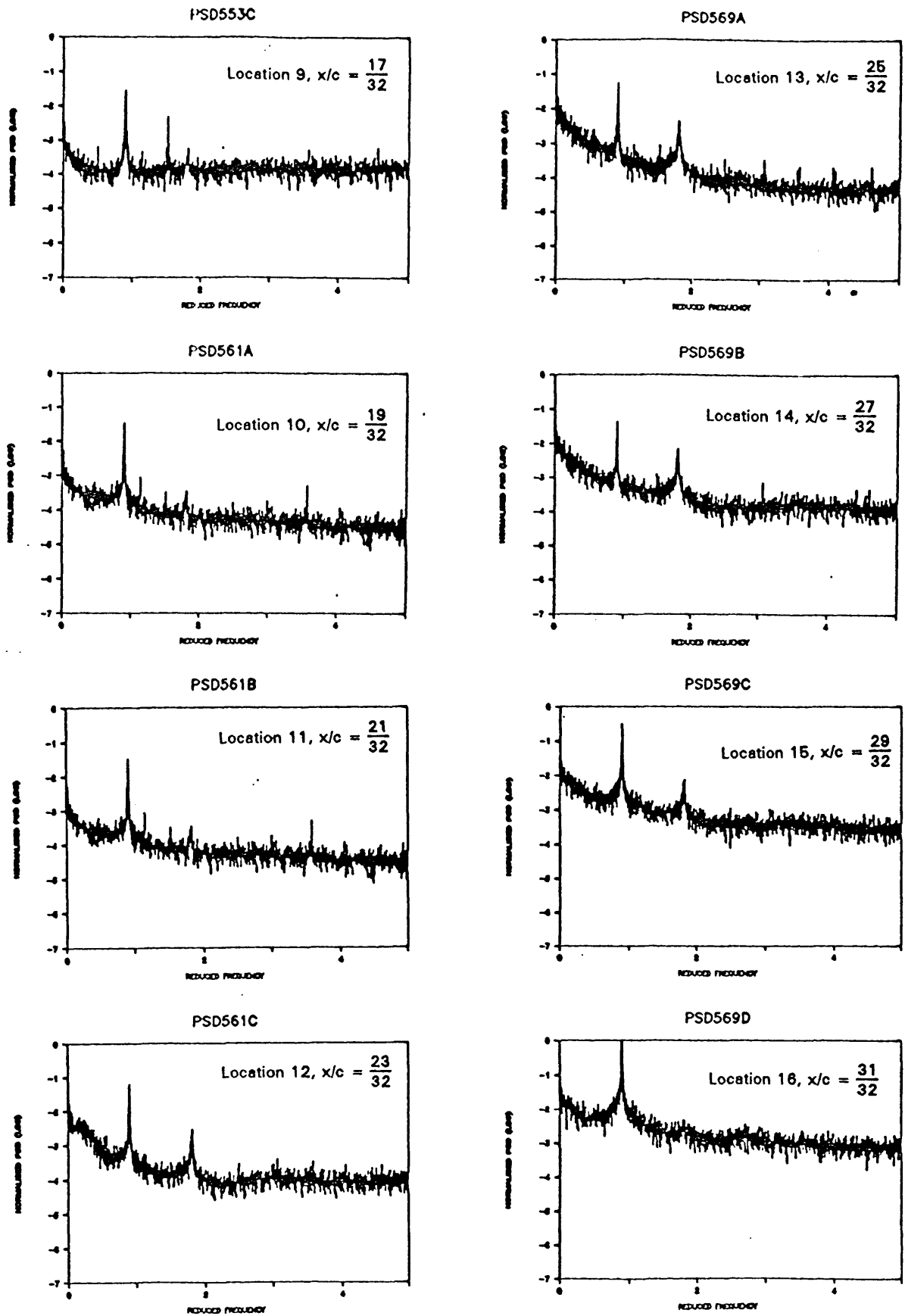


Fig 3.21a Power spectra of C_p at $\alpha = 0^\circ$, with spoiler of $h/c = 0.15$, $x_s/c = 0.7$ on aerofoil upper surface

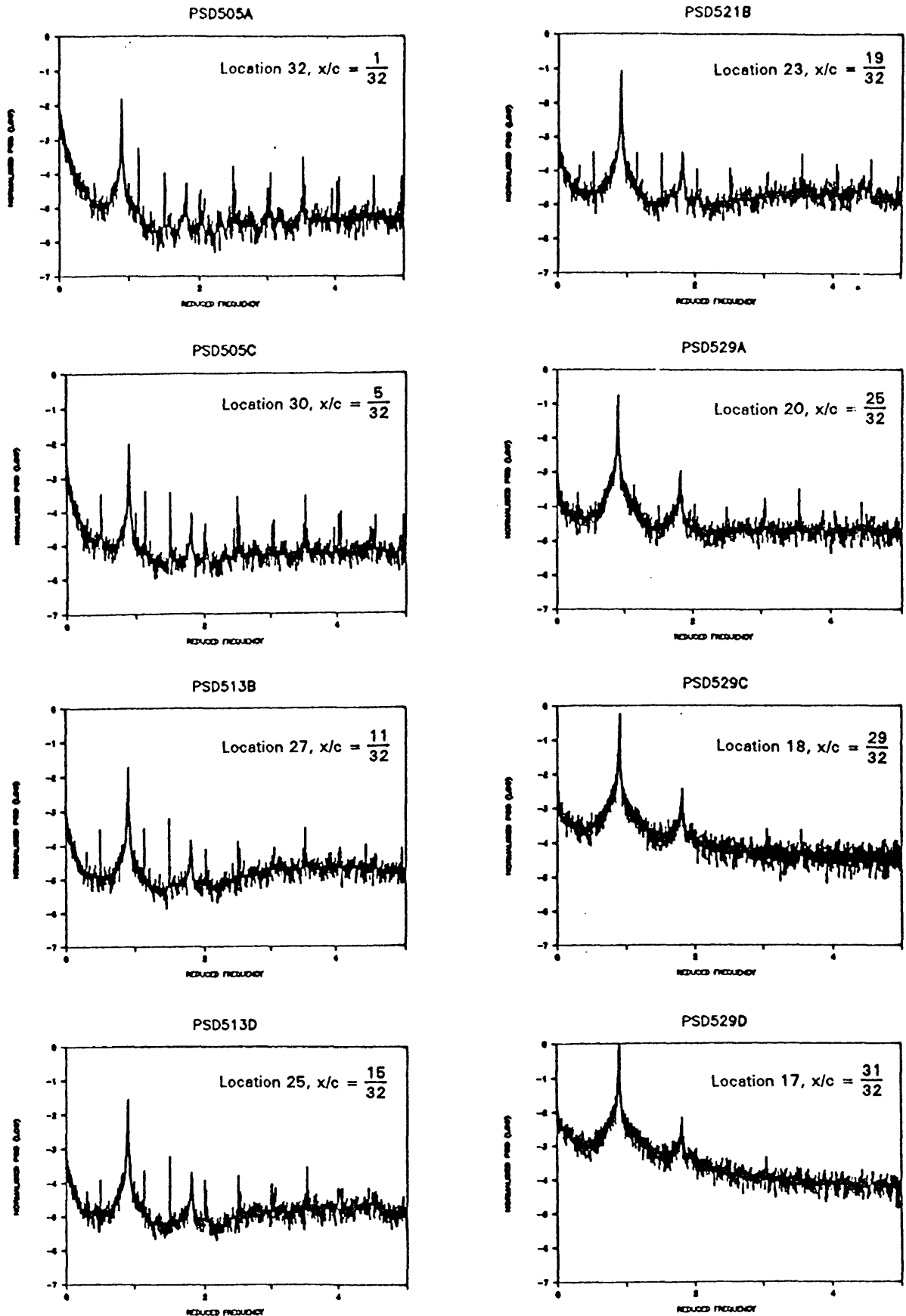


Fig 3.21b Power spectra of C_p at $\alpha = 0^\circ$, with spoiler of $h/c = 0.15$, $x_s/c = 0.7$ on aerofoil lower surface

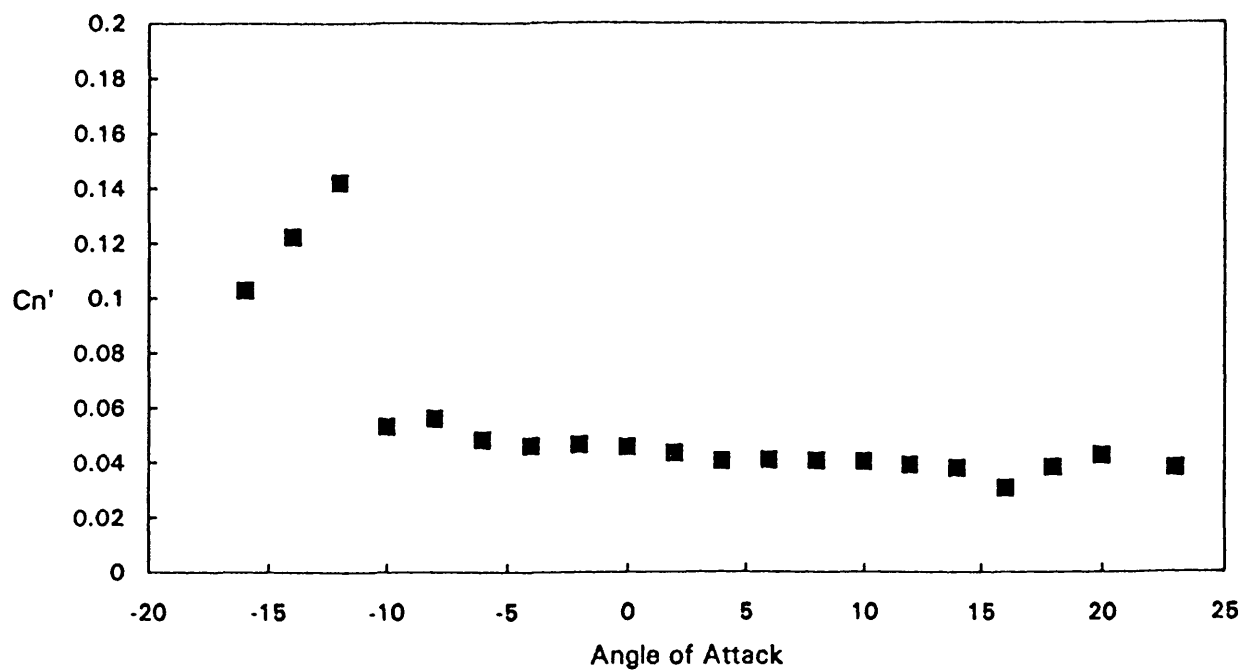


Fig 3.22a Variation of C_N' with α , for $h/c = 0.15$, $x_s/c = 0.7$

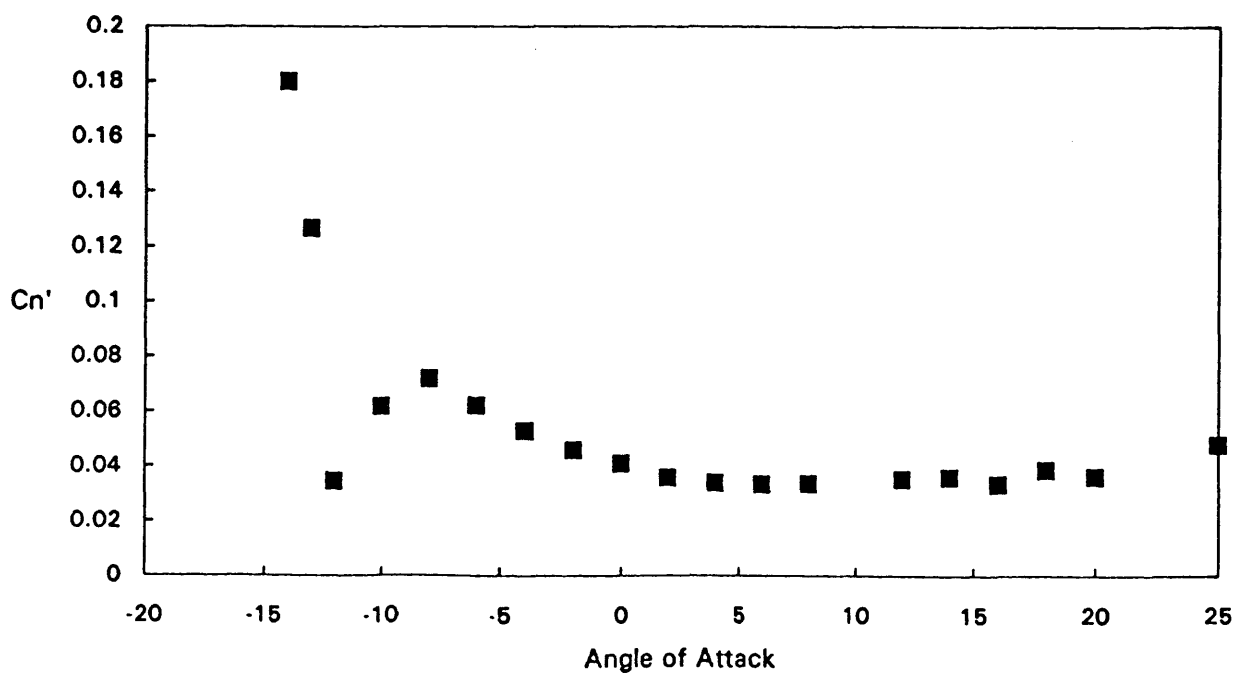


Fig 3.22b Variation of C_N' with α , for $h/c = 0.10$, $x_s/c = 0.3$

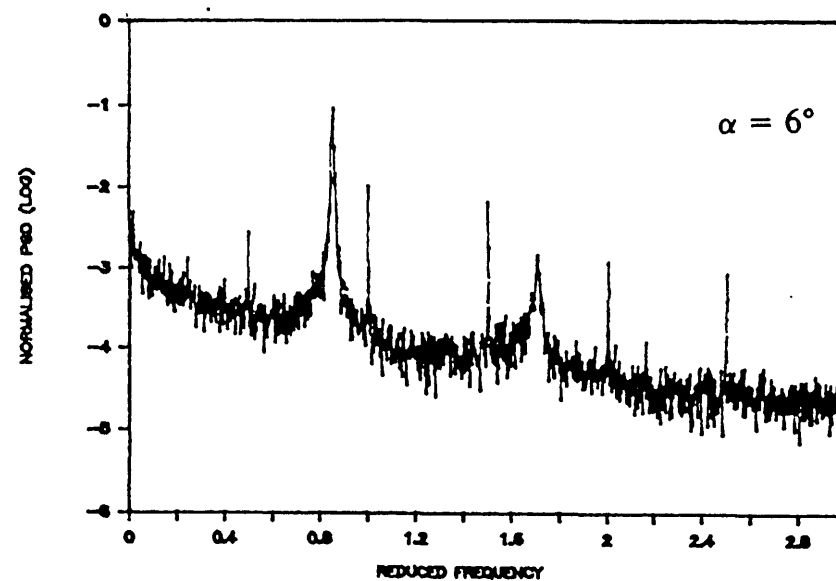
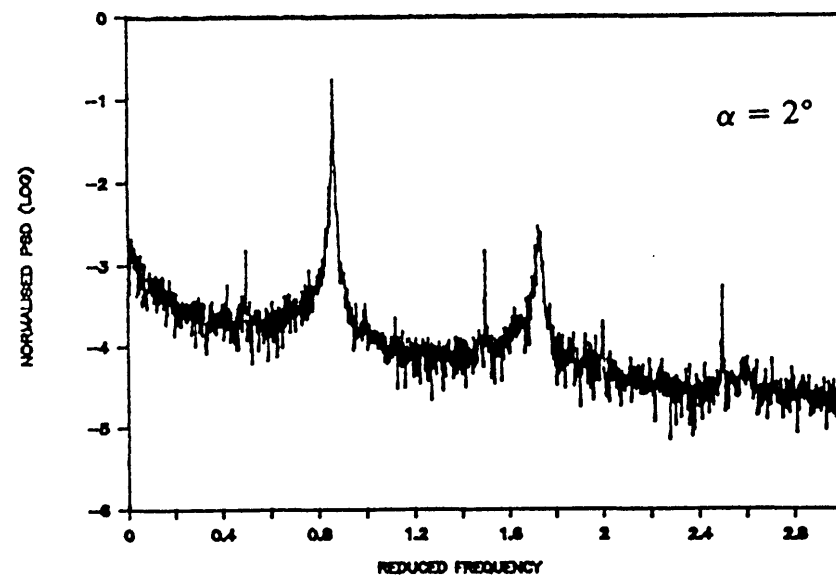
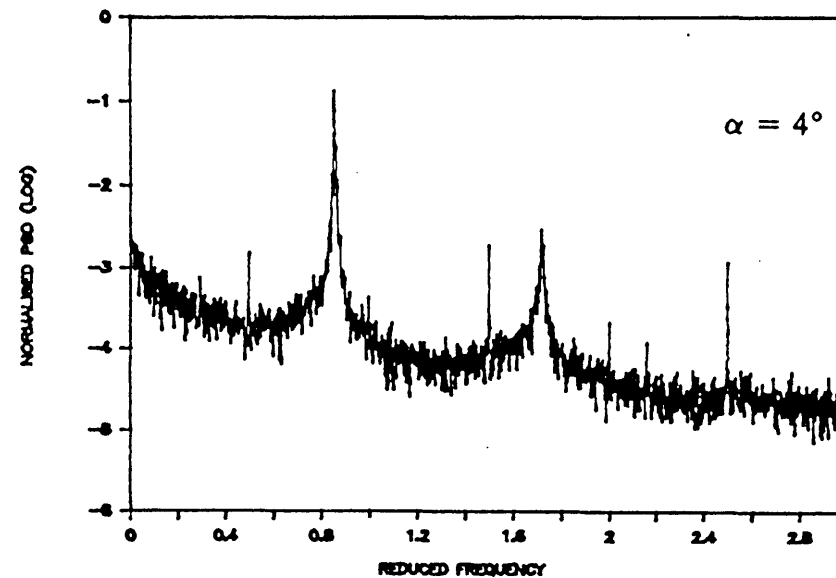
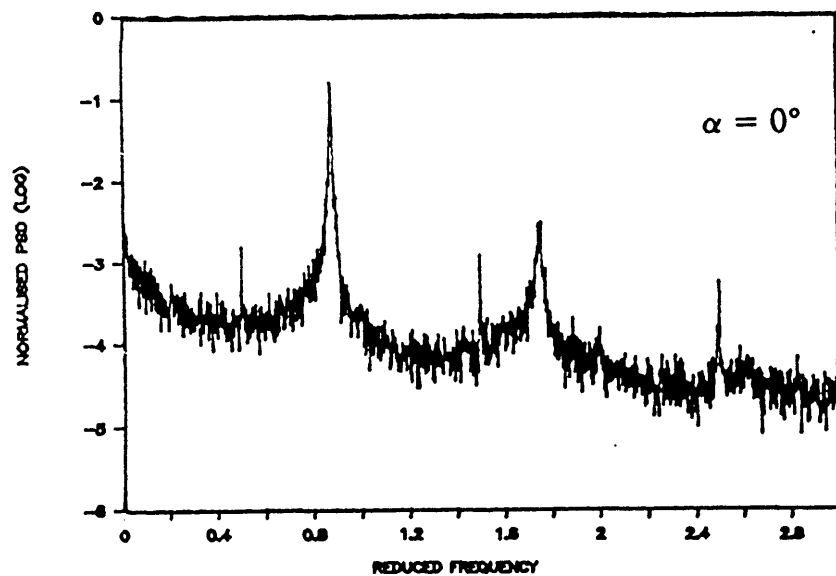


Fig 3.23 Power spectra of C_N for $h/c = 0.15$, $x_s/c = 0.7$ at different α

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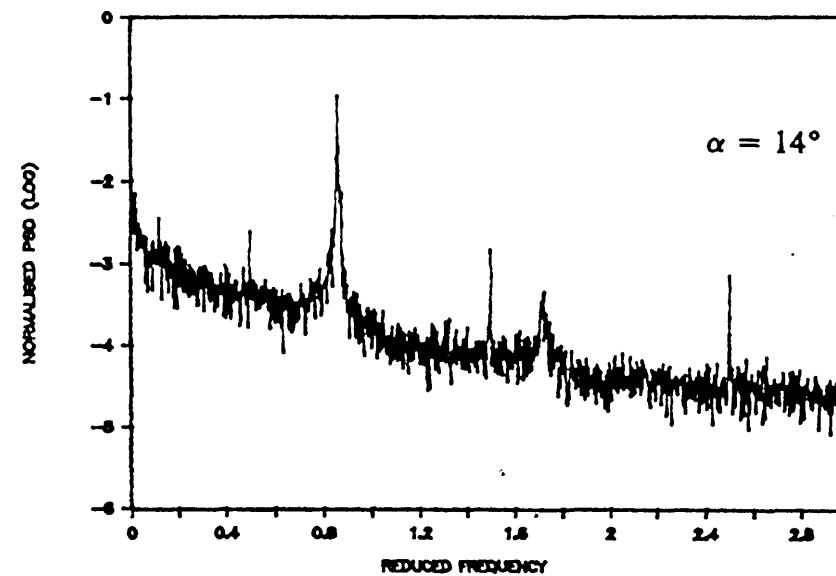
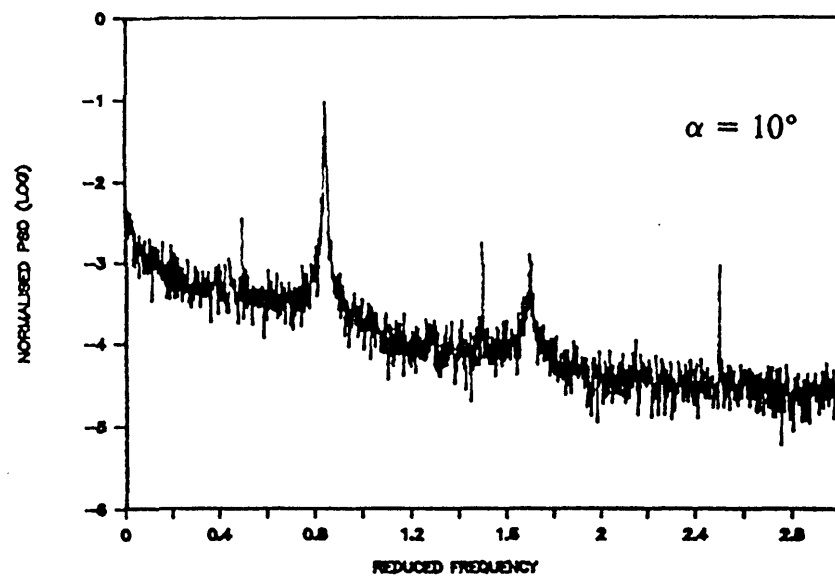
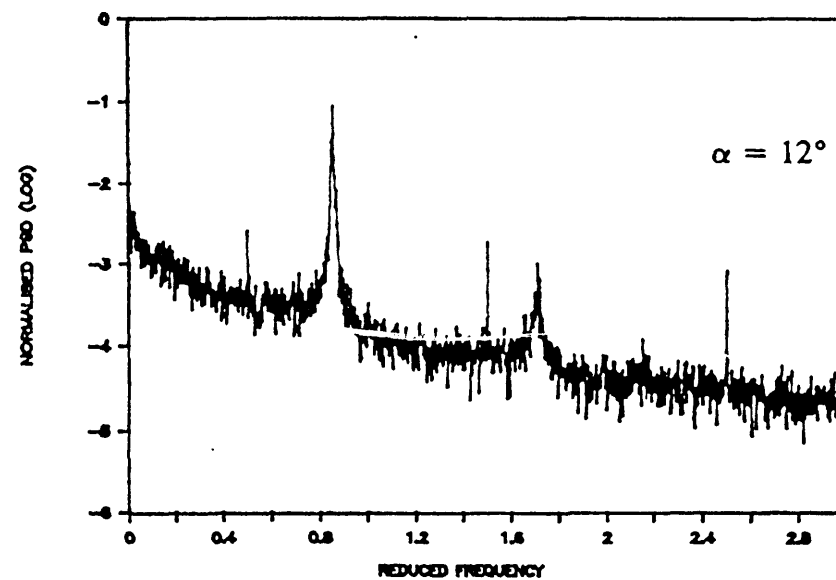
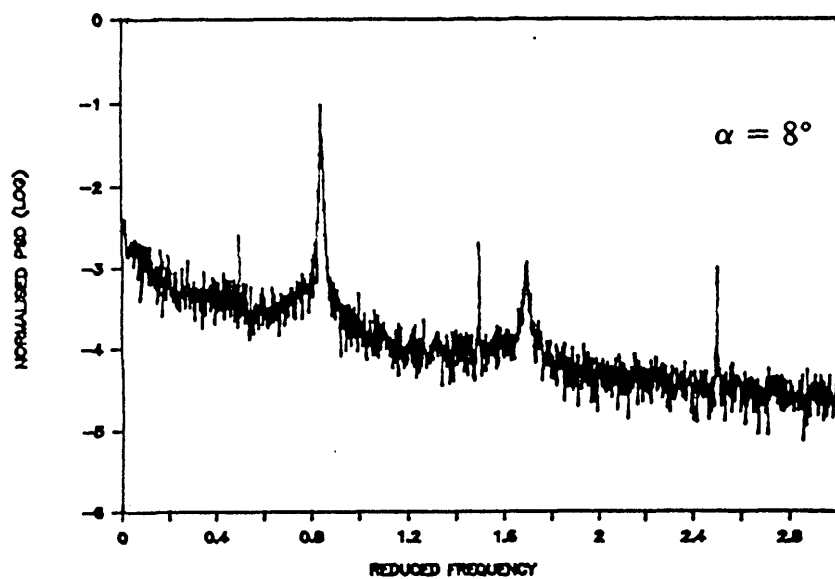


Fig 3.23 Power spectra of C_N for $h/c = 0.15$, $x_s/c = 0.7$ at different α

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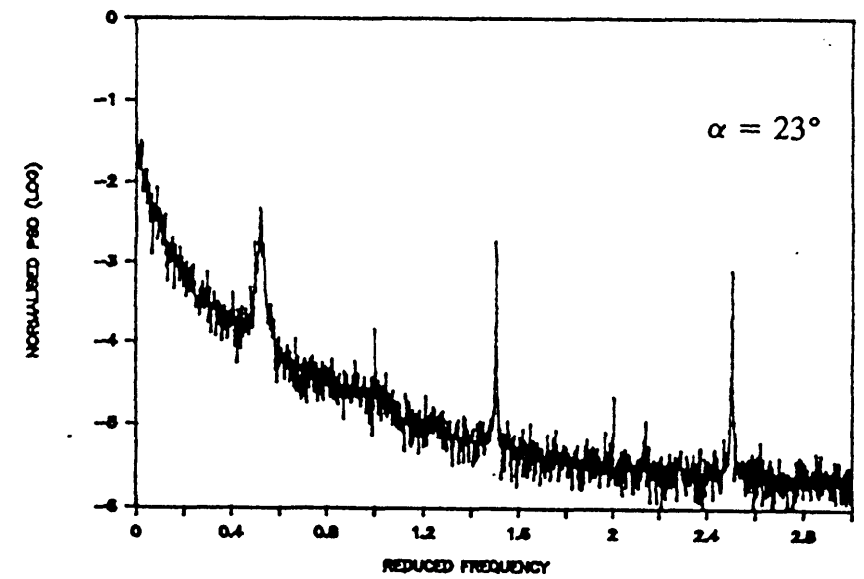
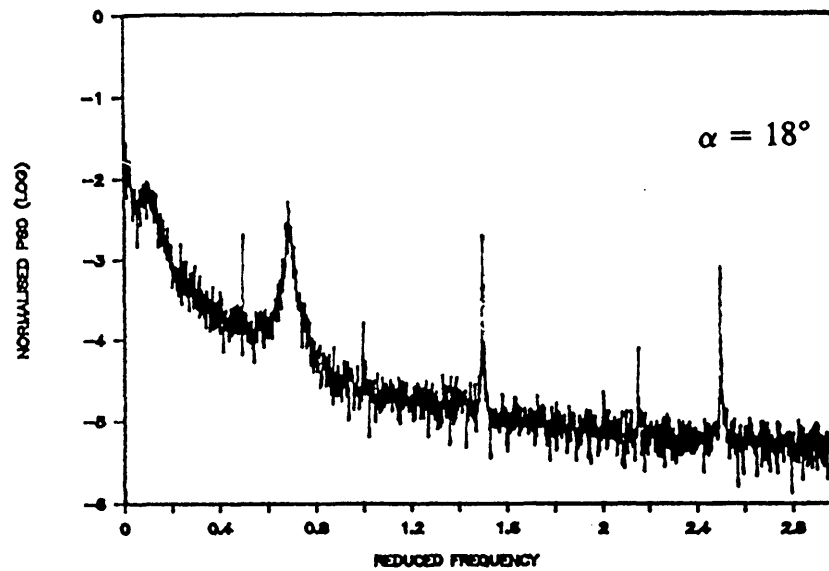
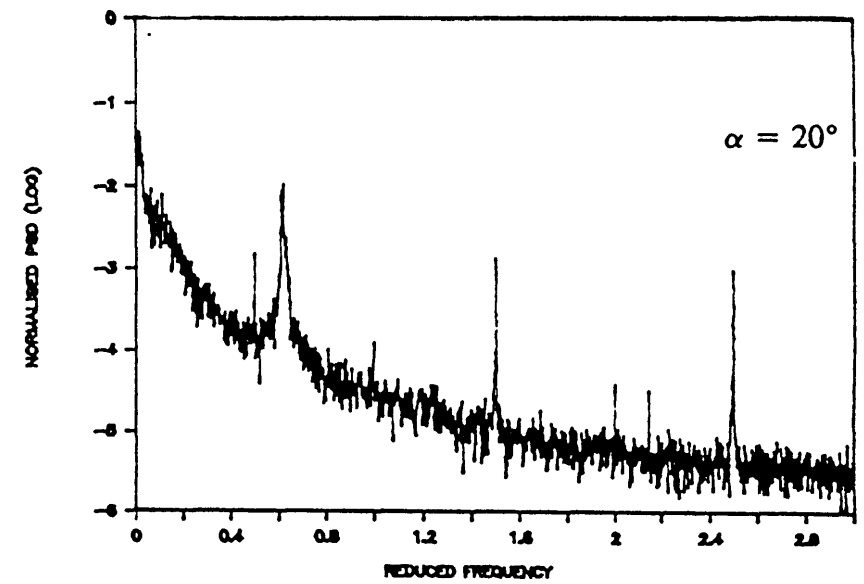
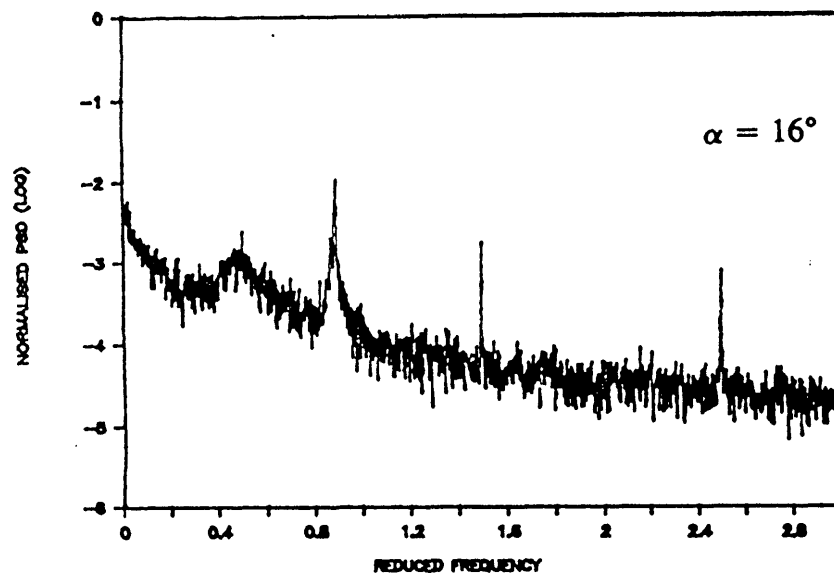


Fig 3.23 Power spectra of C_N for $h/c = 0.15$, $x_s/c = 0.7$ at different α

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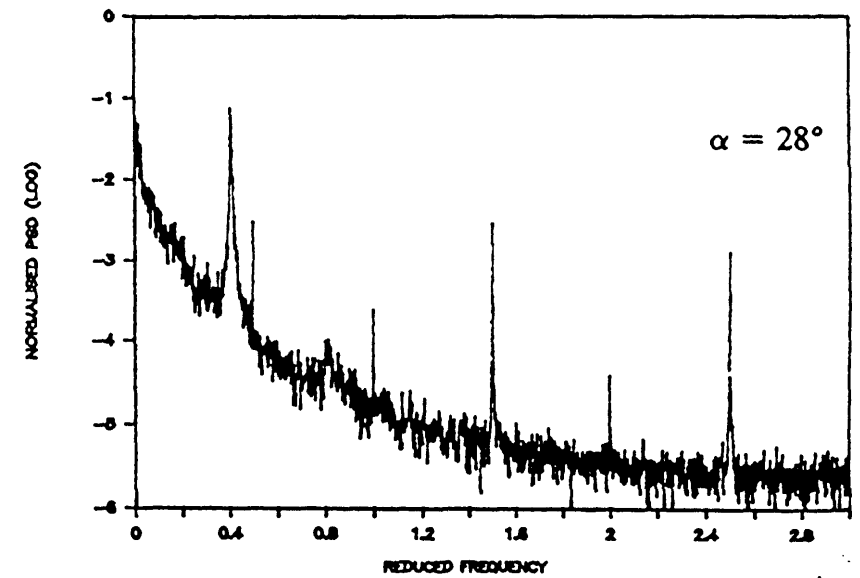
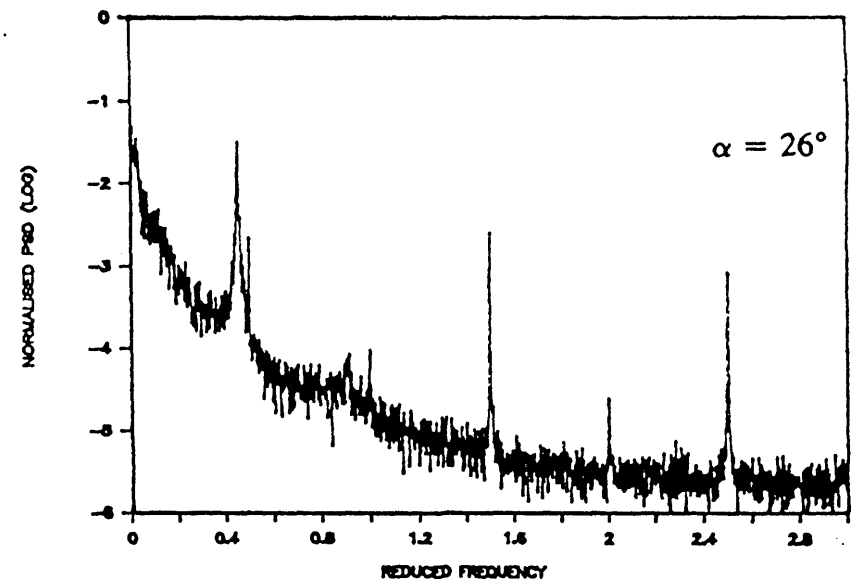


Fig 3.23 Power spectra of C_N for $h/c = 0.15$, $x_s/c = 0.7$ at different α

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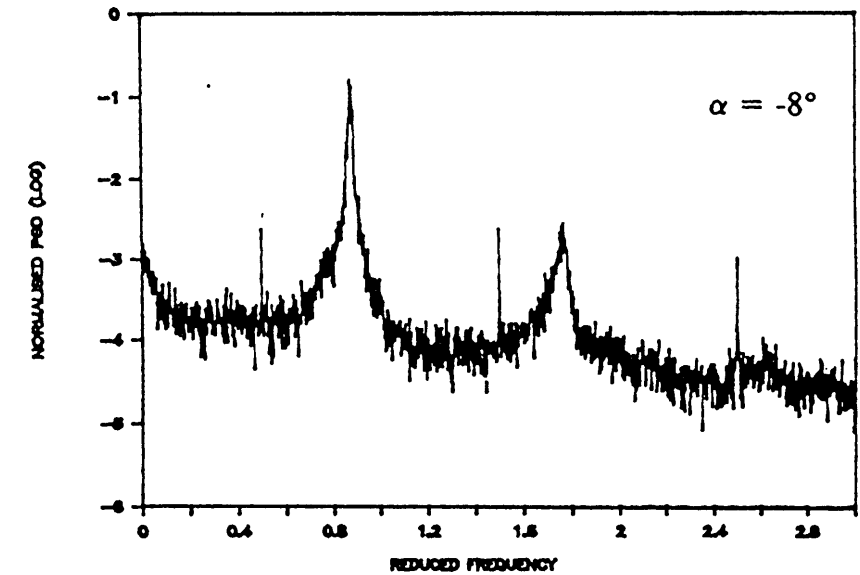
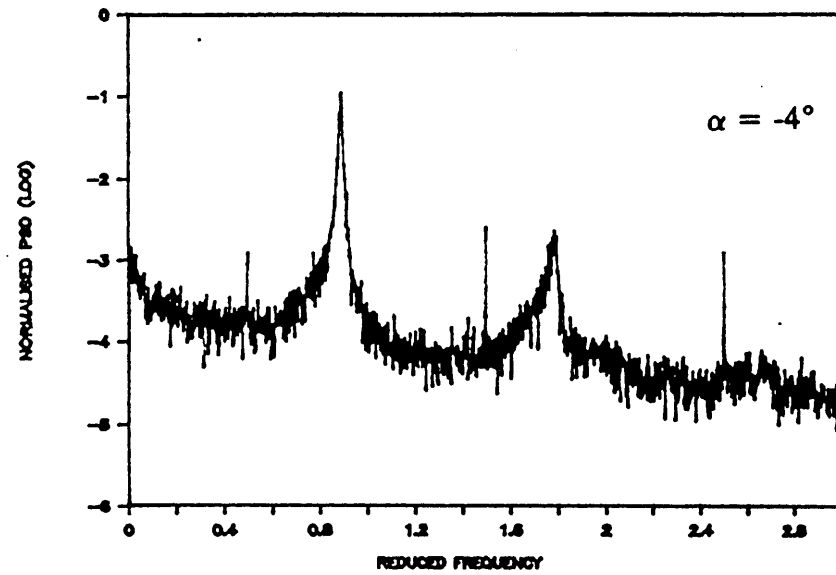
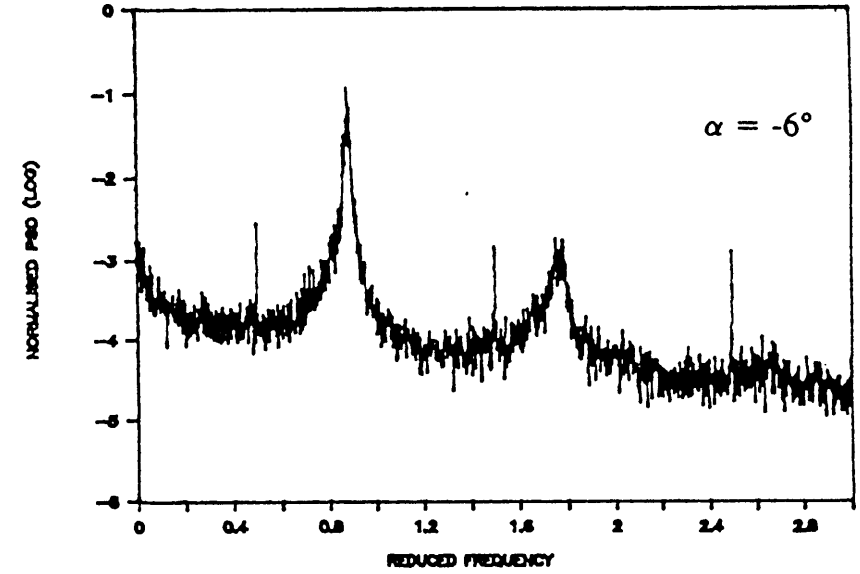
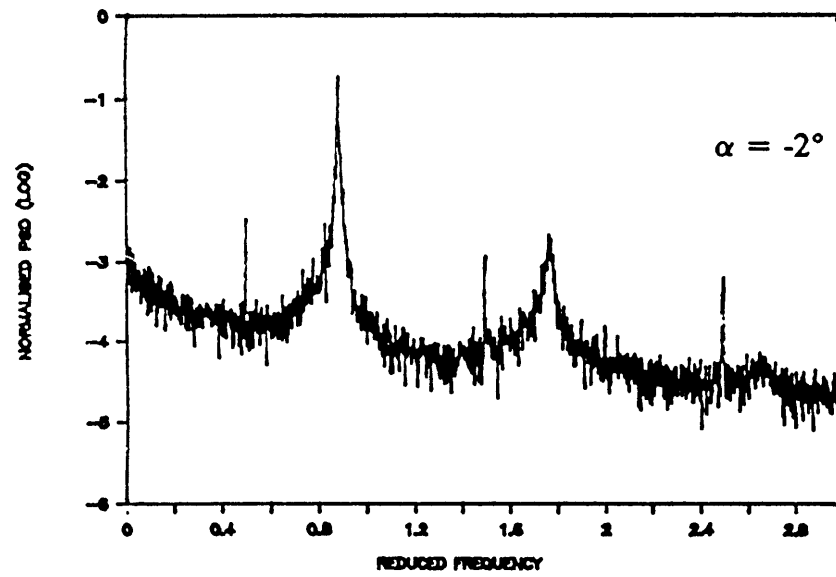


Fig 3.23 Power spectra of C_N for $h/c = 0.15$, $x_s/c = 0.7$ at different α

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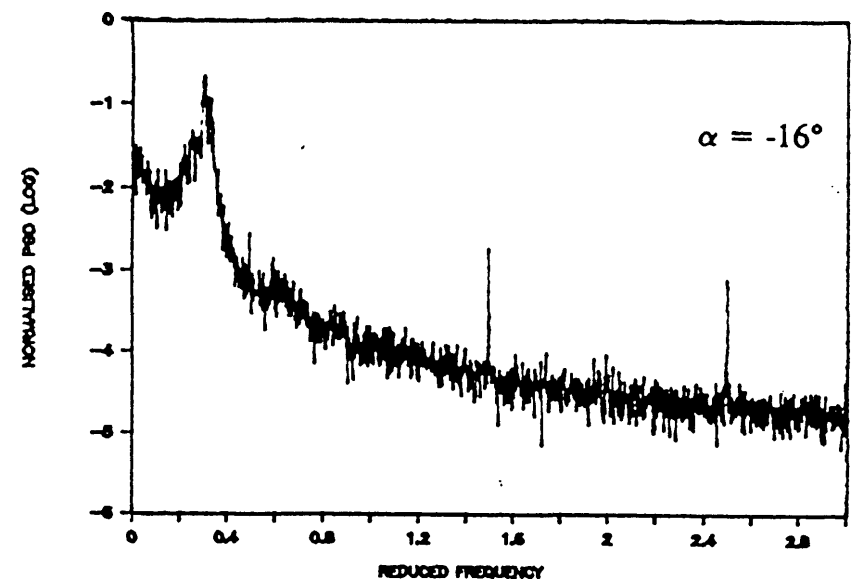
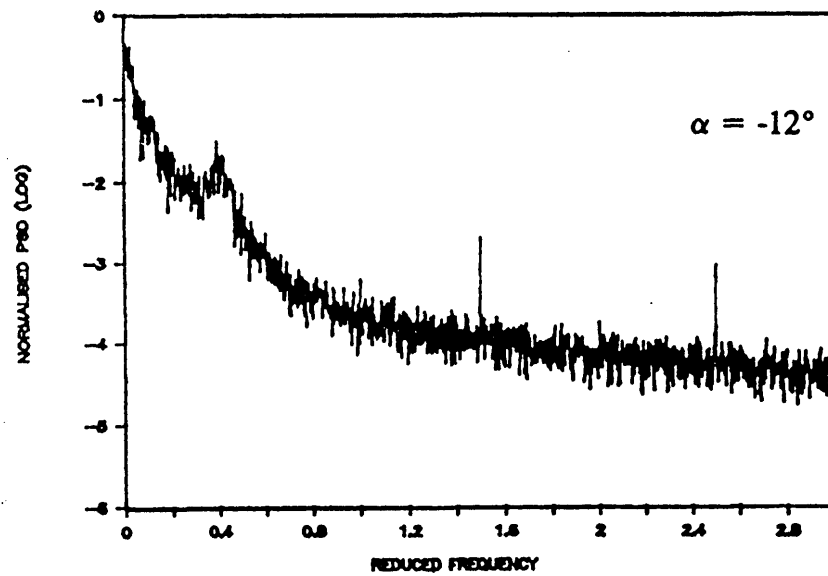
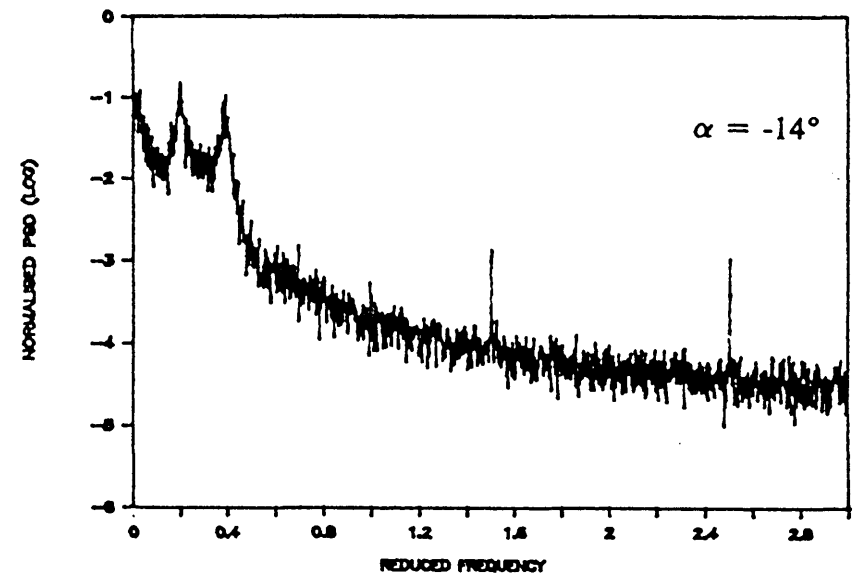
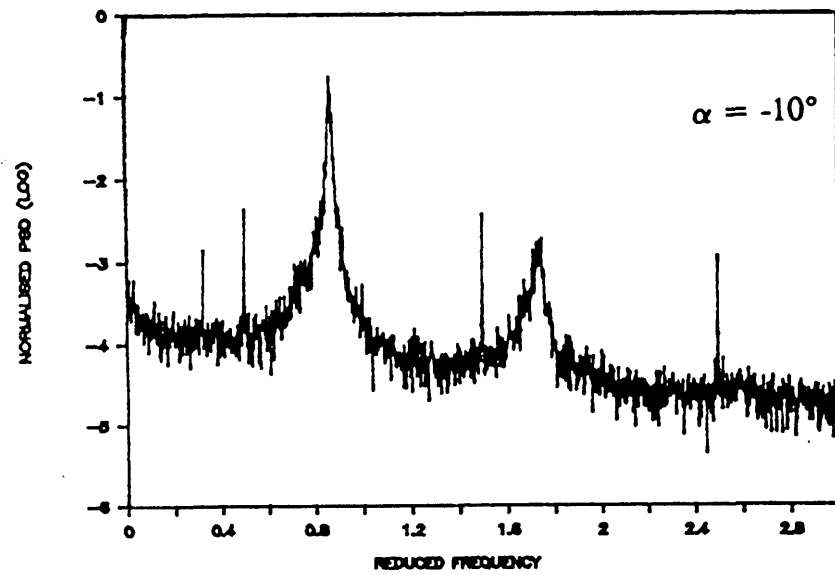


Fig 3.23 Power spectra of C_N for $h/c = 0.15$, $x_s/c = 0.7$ at different α

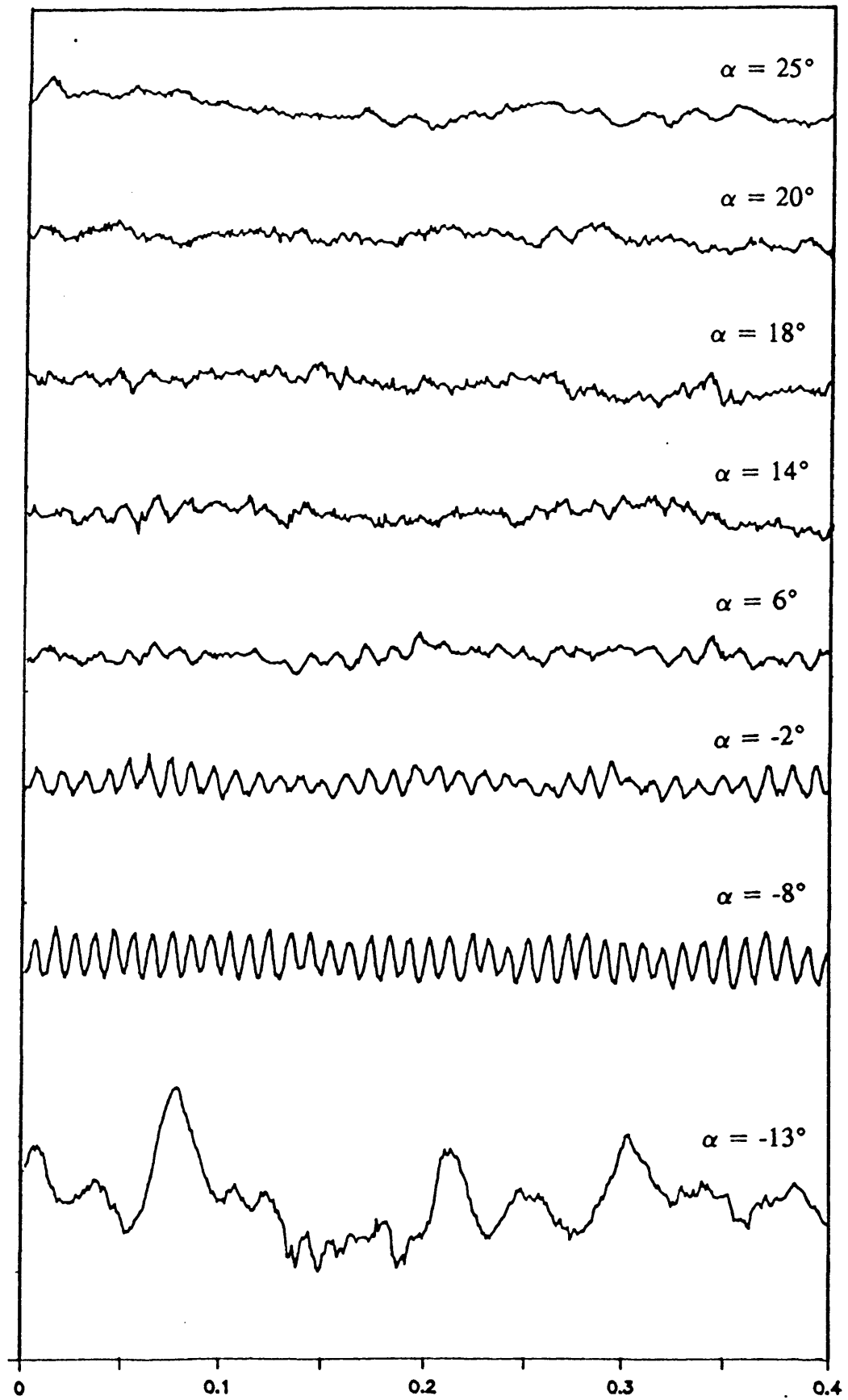


Fig 3.24 Time histories of C_N at selected α for $h/c = 0.15$, $x_s/c = 0.7$

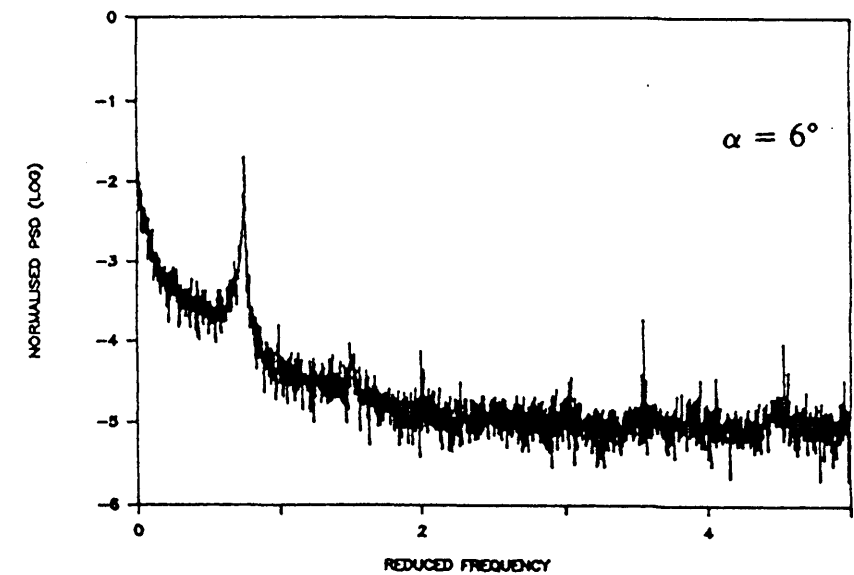
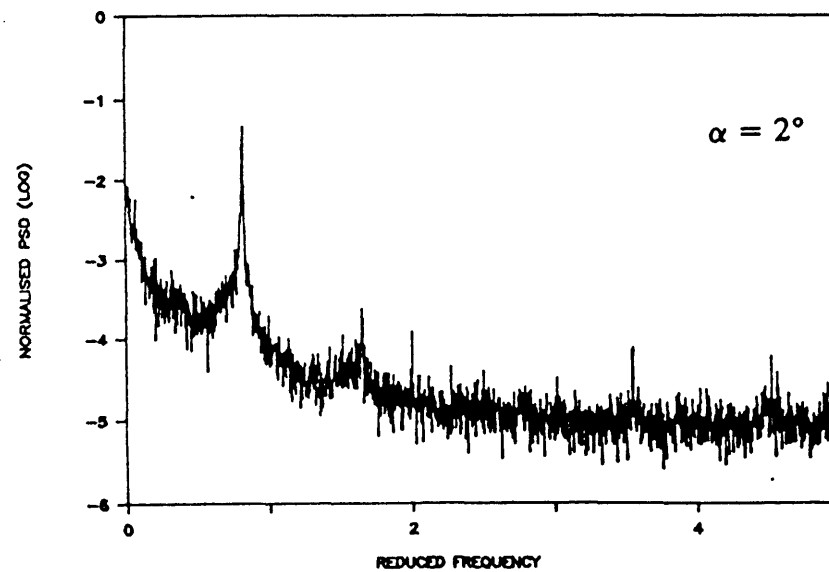
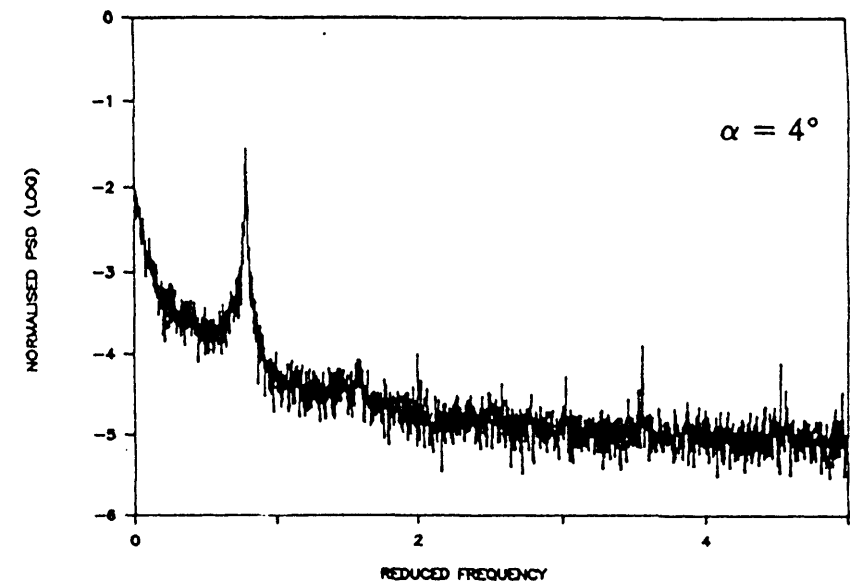
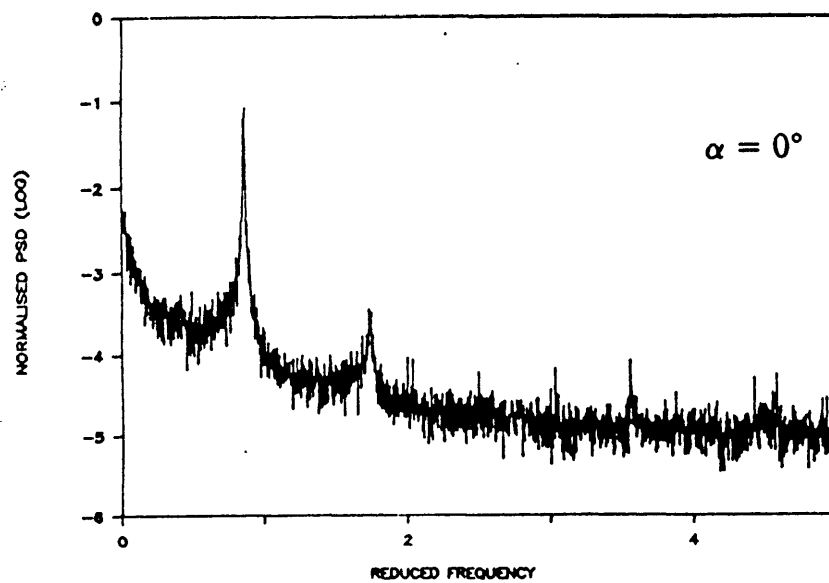


Fig 3.25 Power spectra of C_N for $h/c = 0.10$, $x_s/c = 0.3$ at different α

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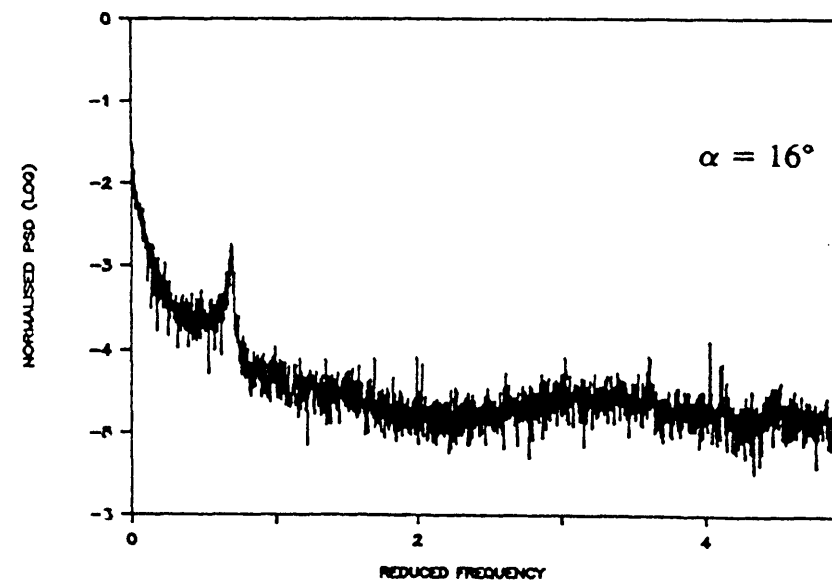
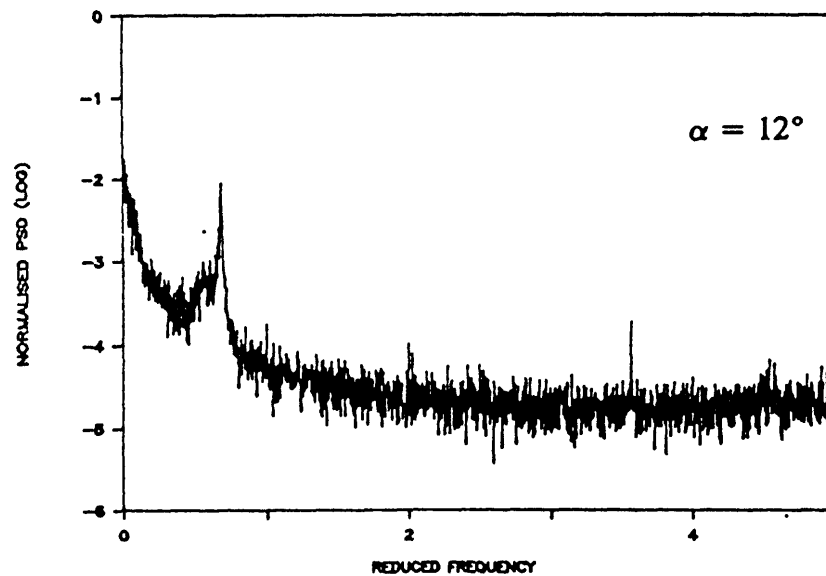
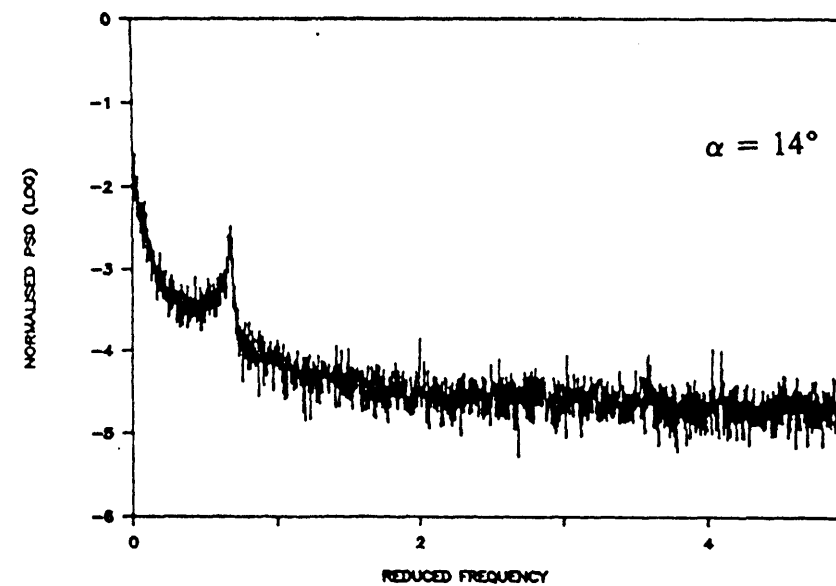
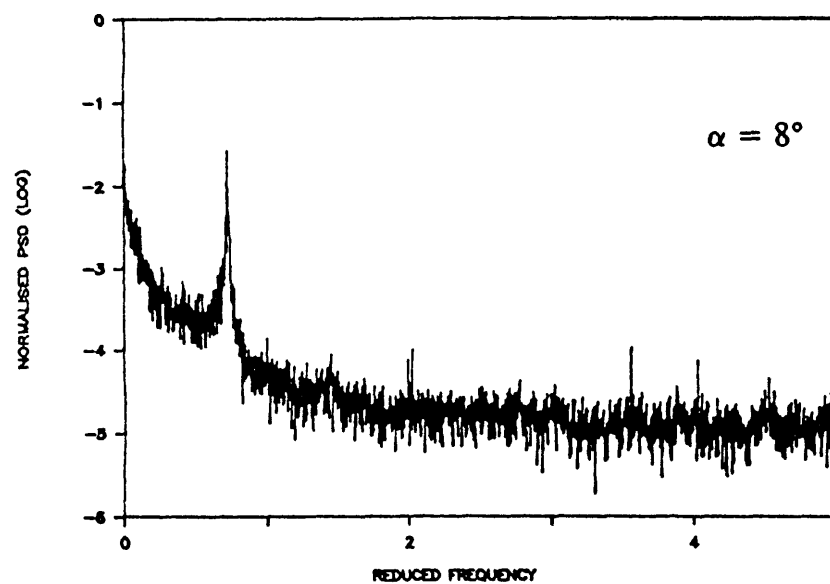


Fig 3.25 Power spectra of C_N for $h/c = 0.10$, $x_s/c = 0.3$ at different α

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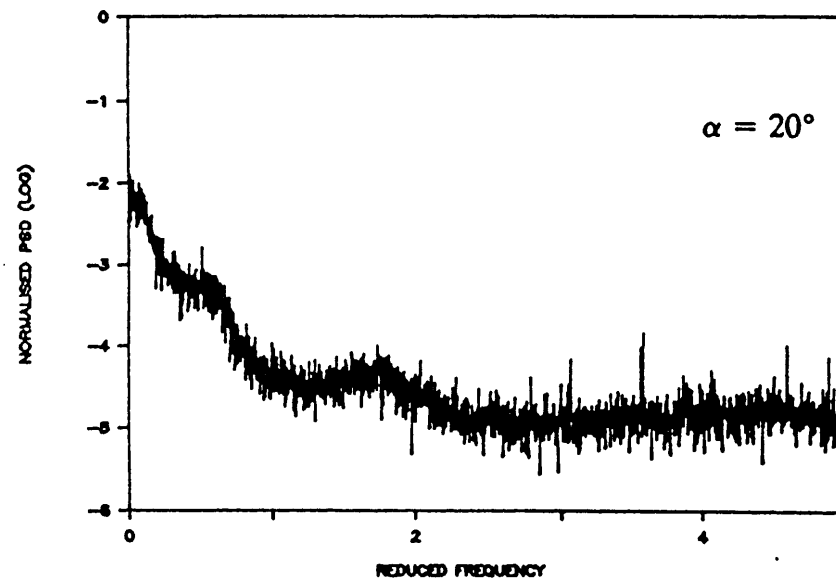
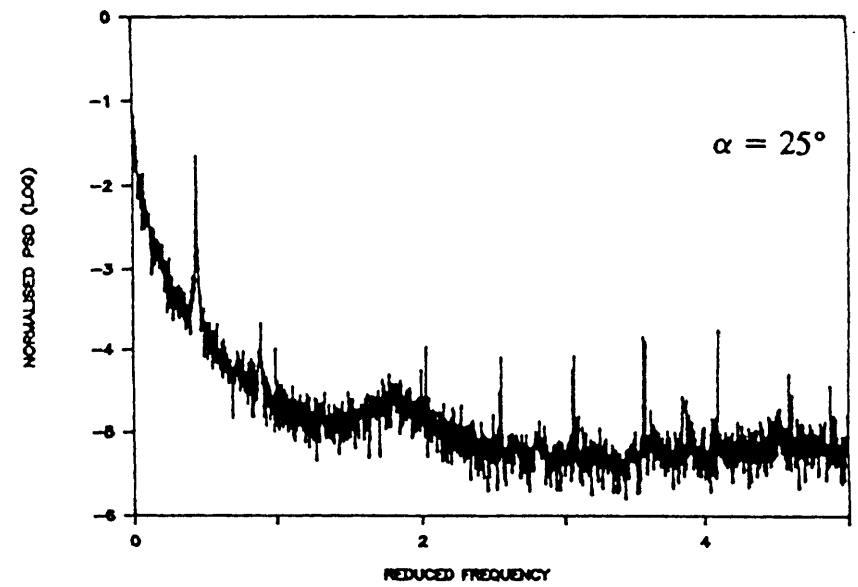
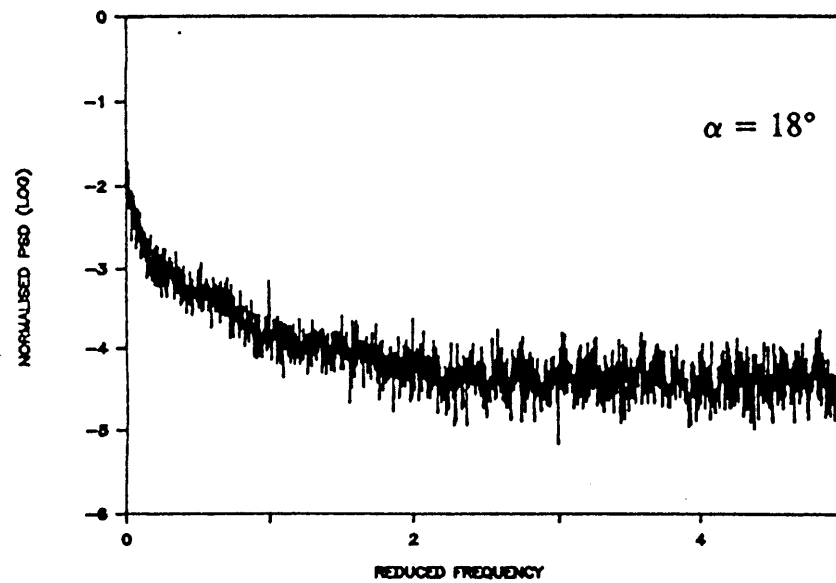


Fig 3.25 Power spectra of C_N for $h/c = 0.10$, $x_s/c = 0.3$ at different α

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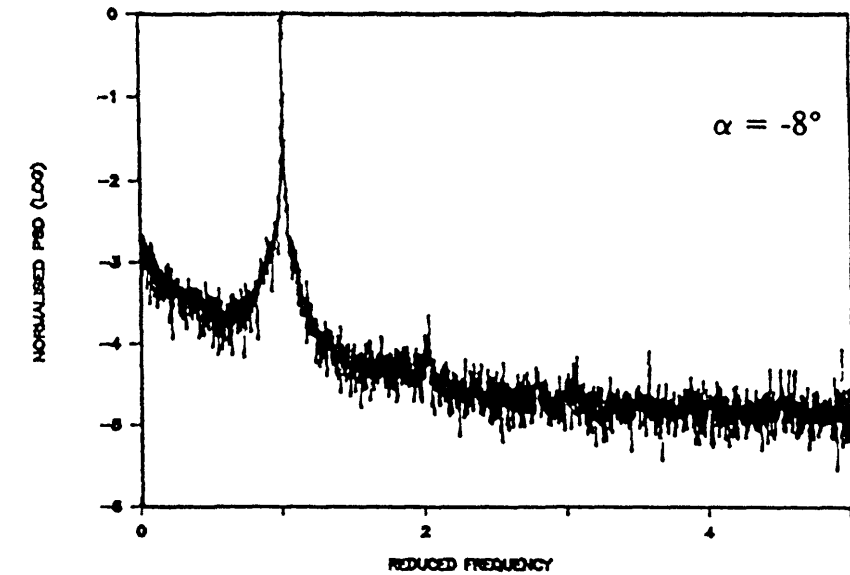
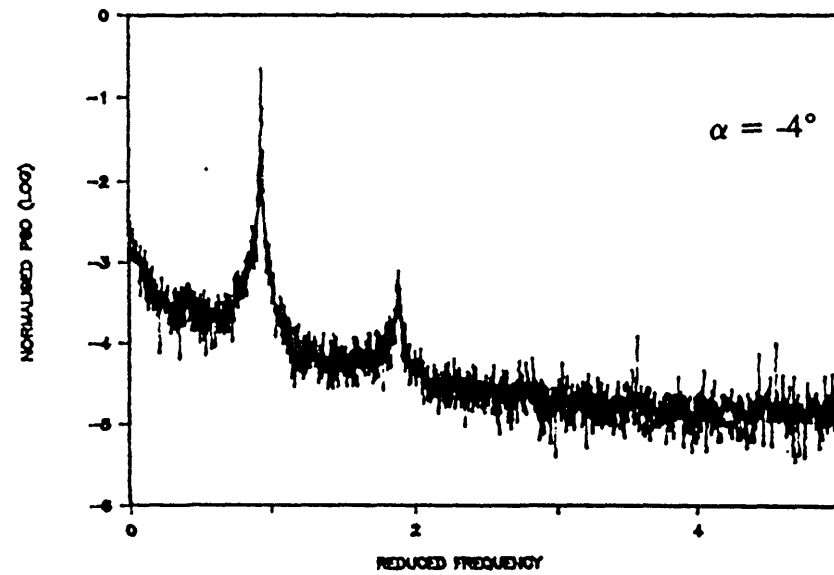
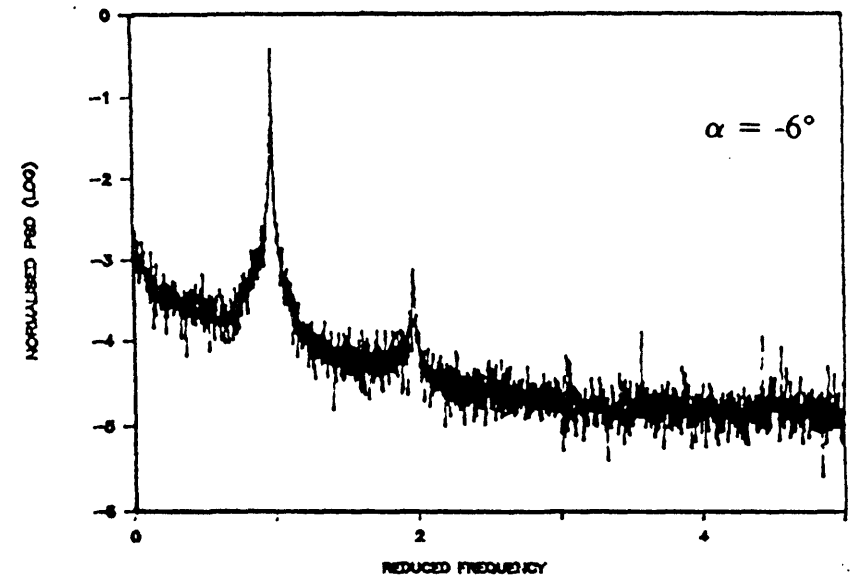
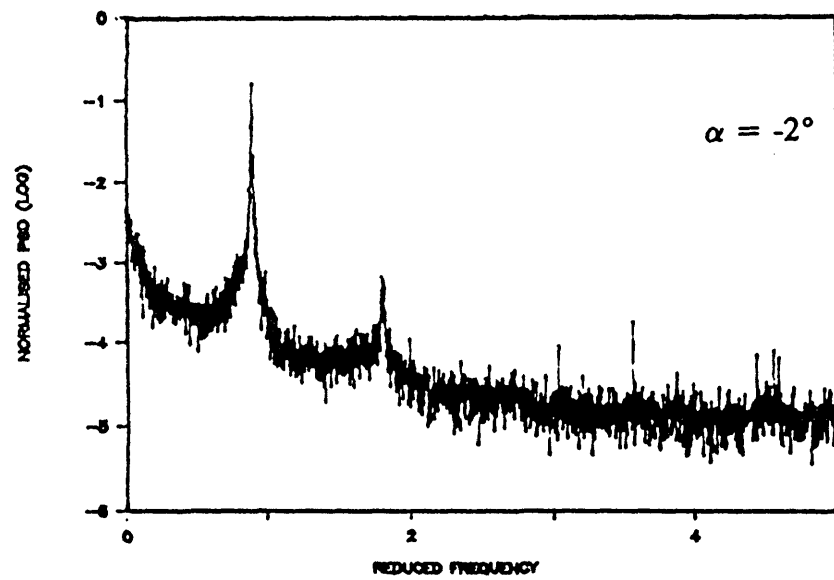


Fig 3.25 Power spectra of C_N for $h/c = 0.10$, $x_s/c = 0.3$ at different α

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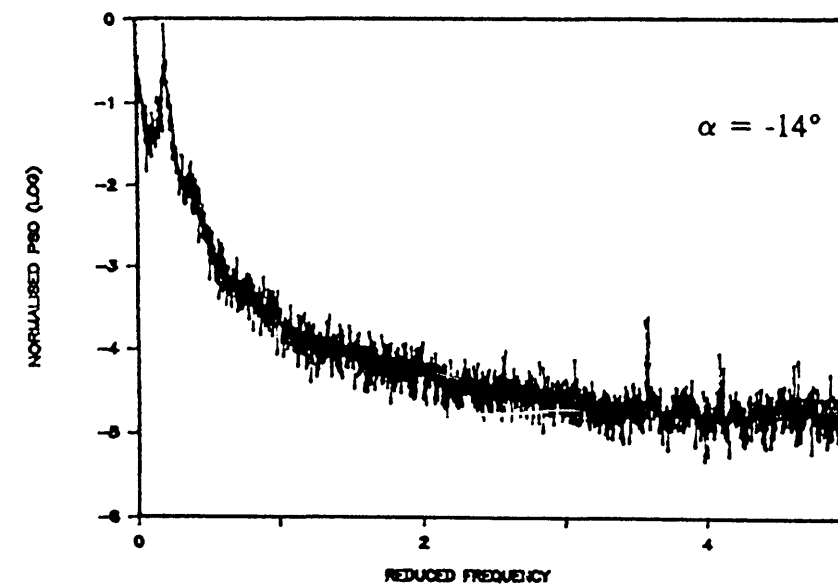
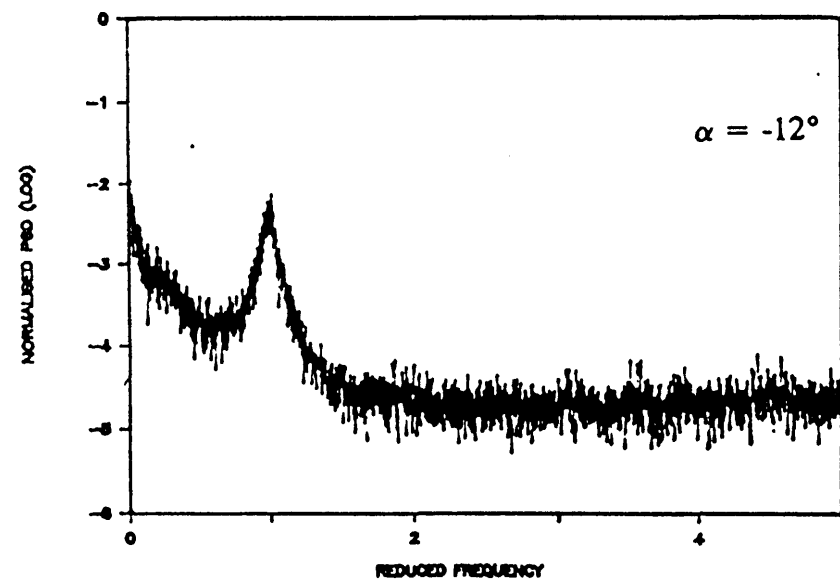
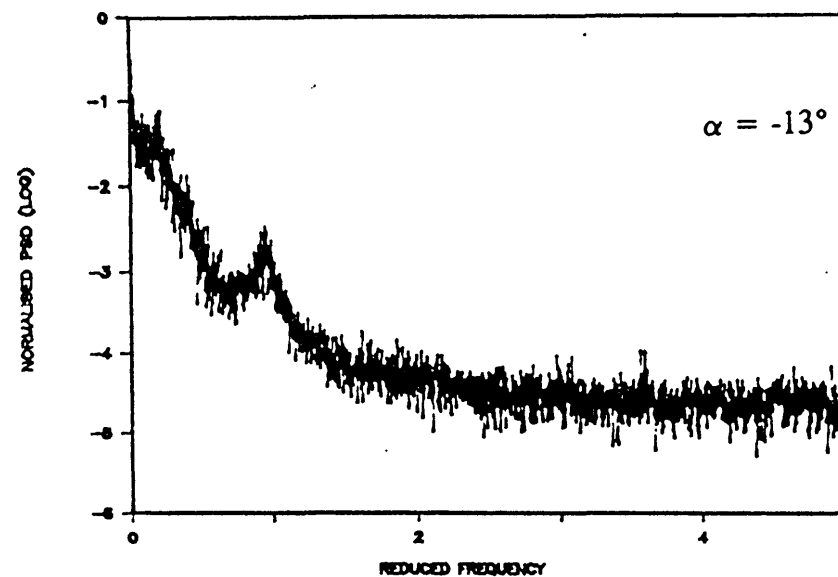
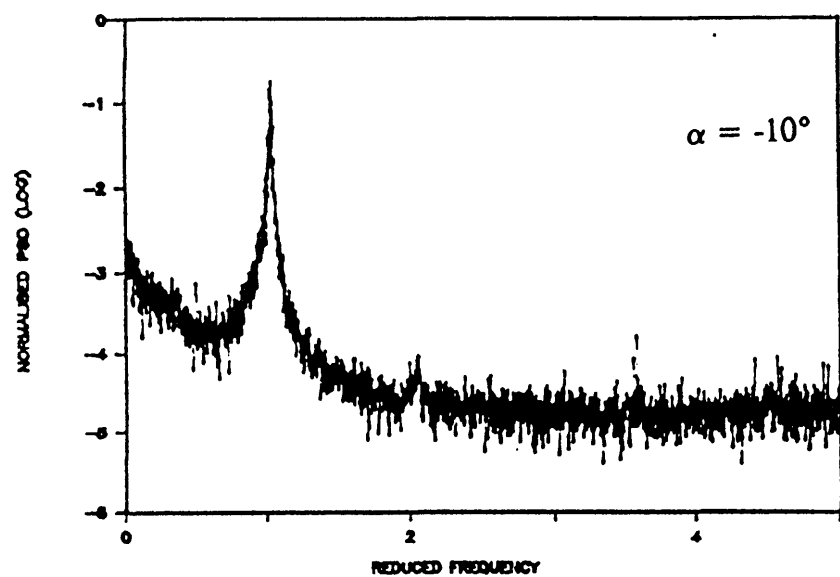


Fig 3.25 Power spectra of C_N for $h/c = 0.10$, $x_s/c = 0.3$ at different α

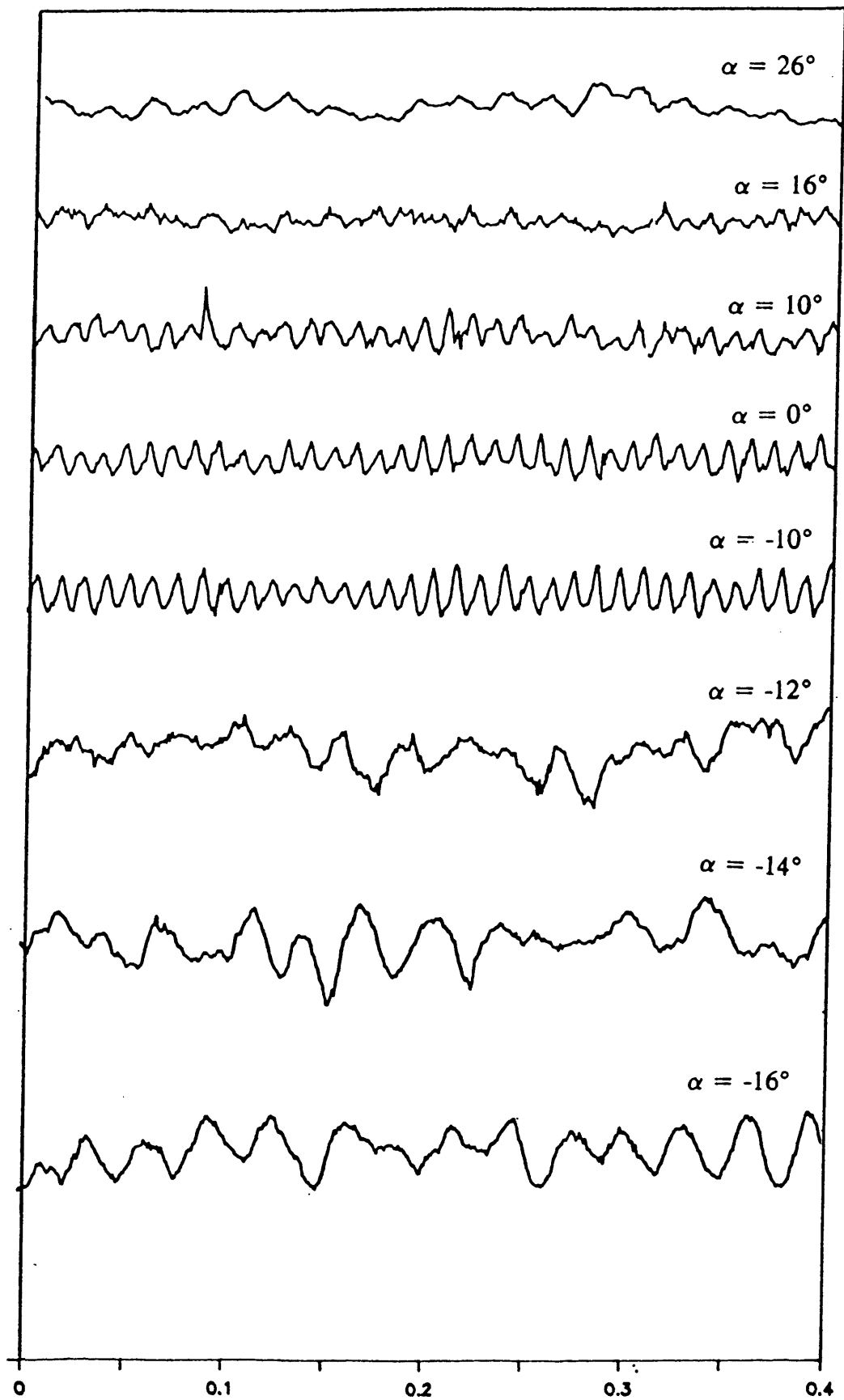


Fig 3.26 Time histories of C_N at selected α for $h/c = 0.10$, $x_s/c = 0.3$

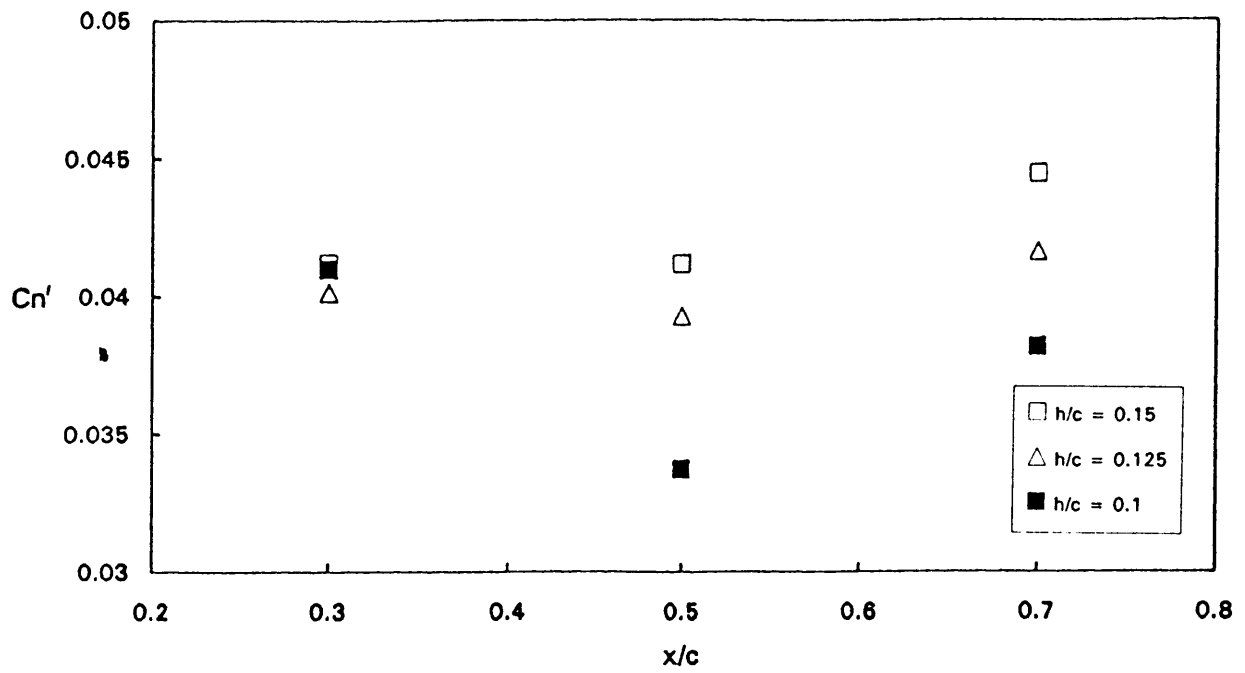


Fig 3.27a Variation of C_N' with x_s/c at different h/c for $\alpha = 0^\circ$

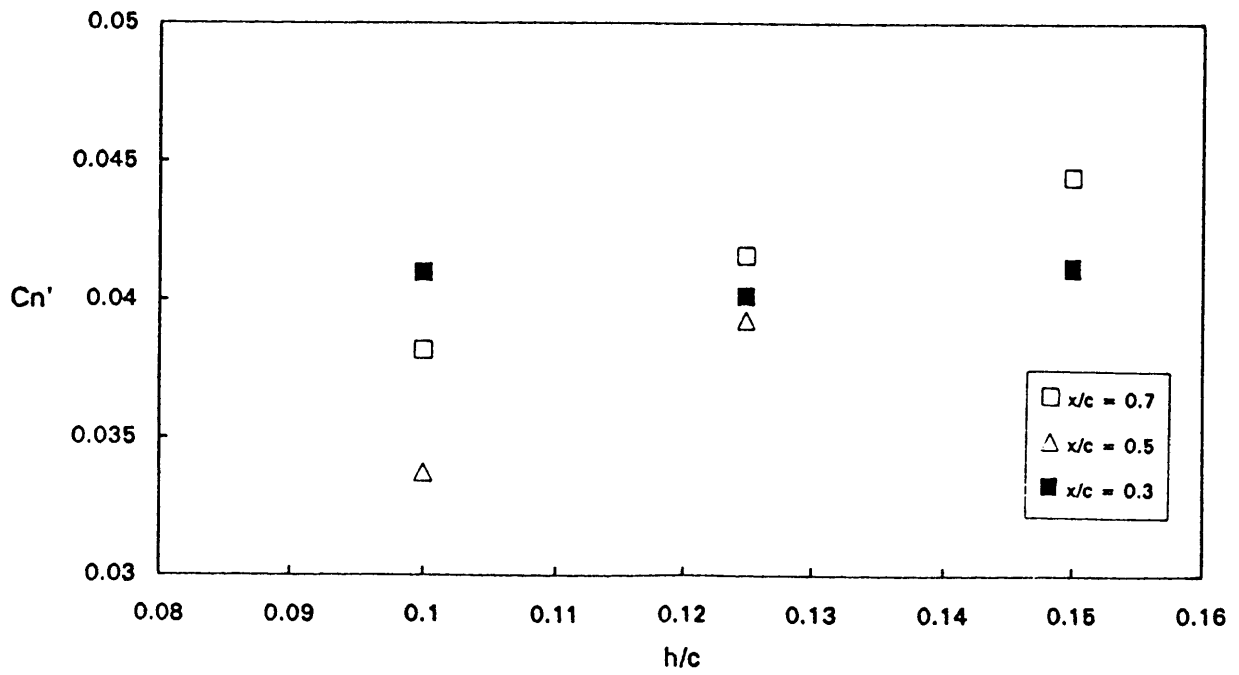


Fig3.27b Variation of C_N' with h/c at different x_s/c for $\alpha = 0^\circ$

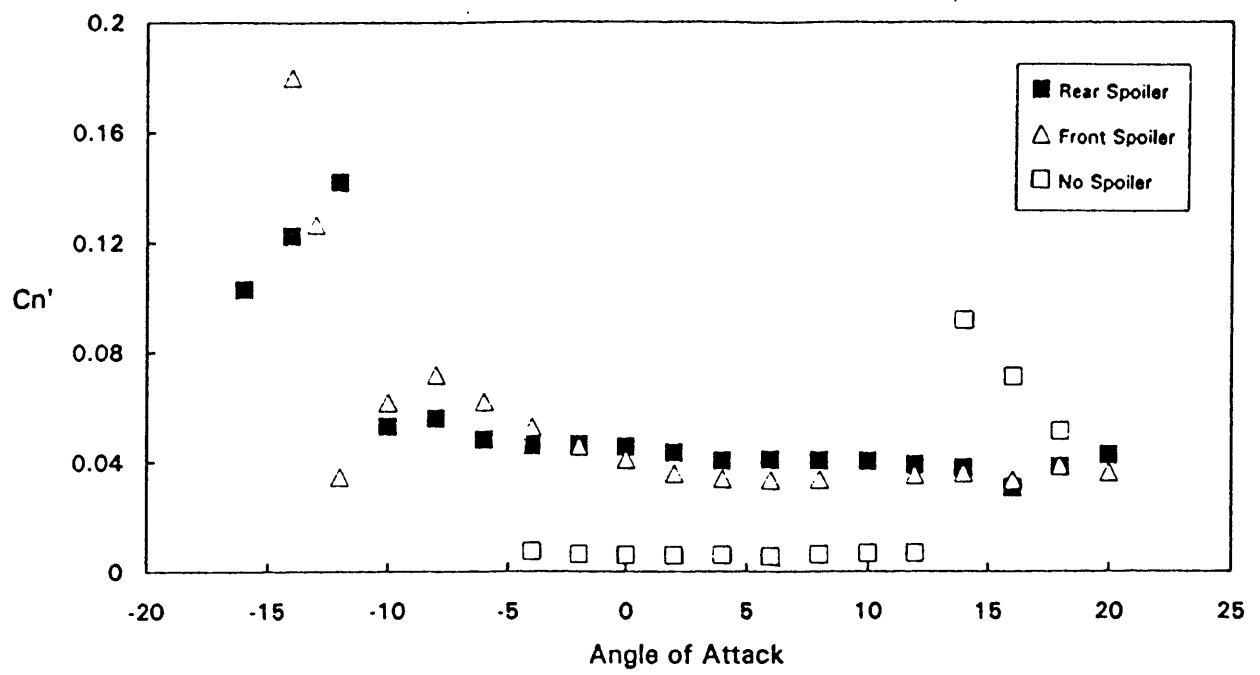


Fig 3.28 Variation of C_N' for aerofoil with α , for an aerofoil with and without spoilers

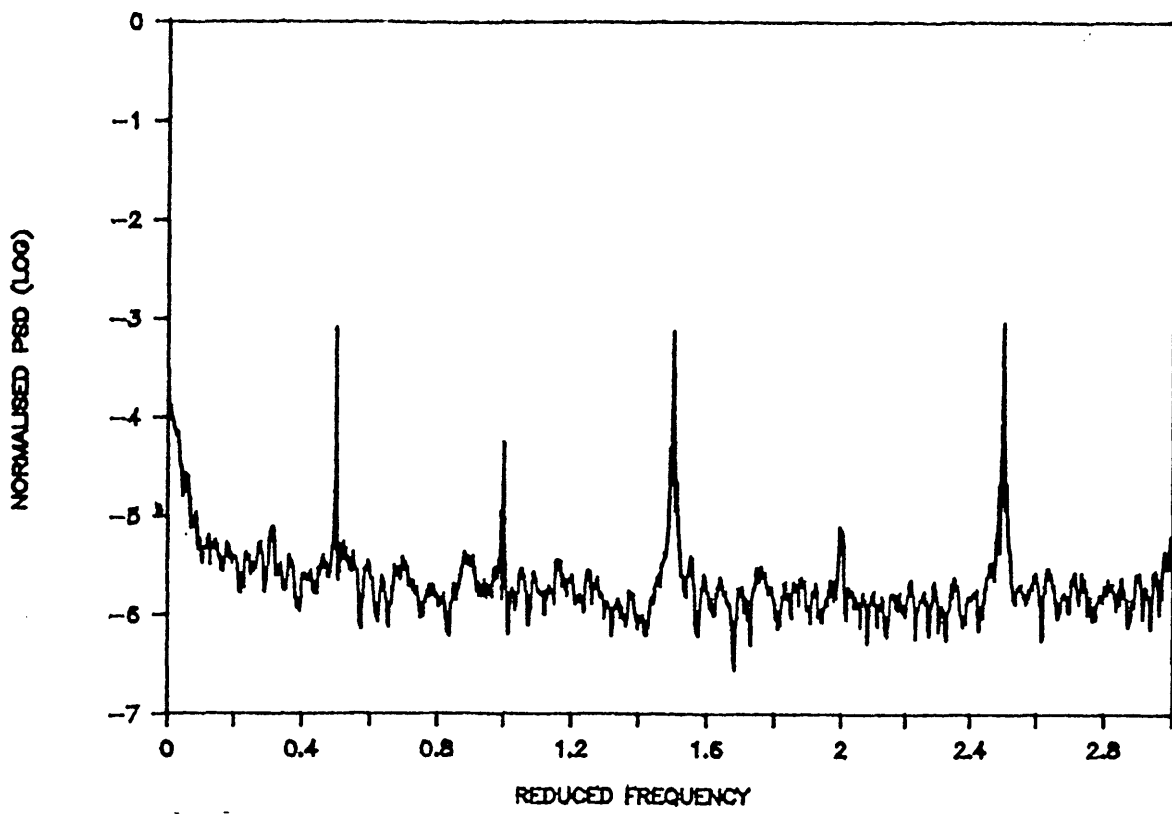


Fig 3.29a Power spectrum of C_N' for aerofoil with no spoilers at $\alpha = 0^\circ$

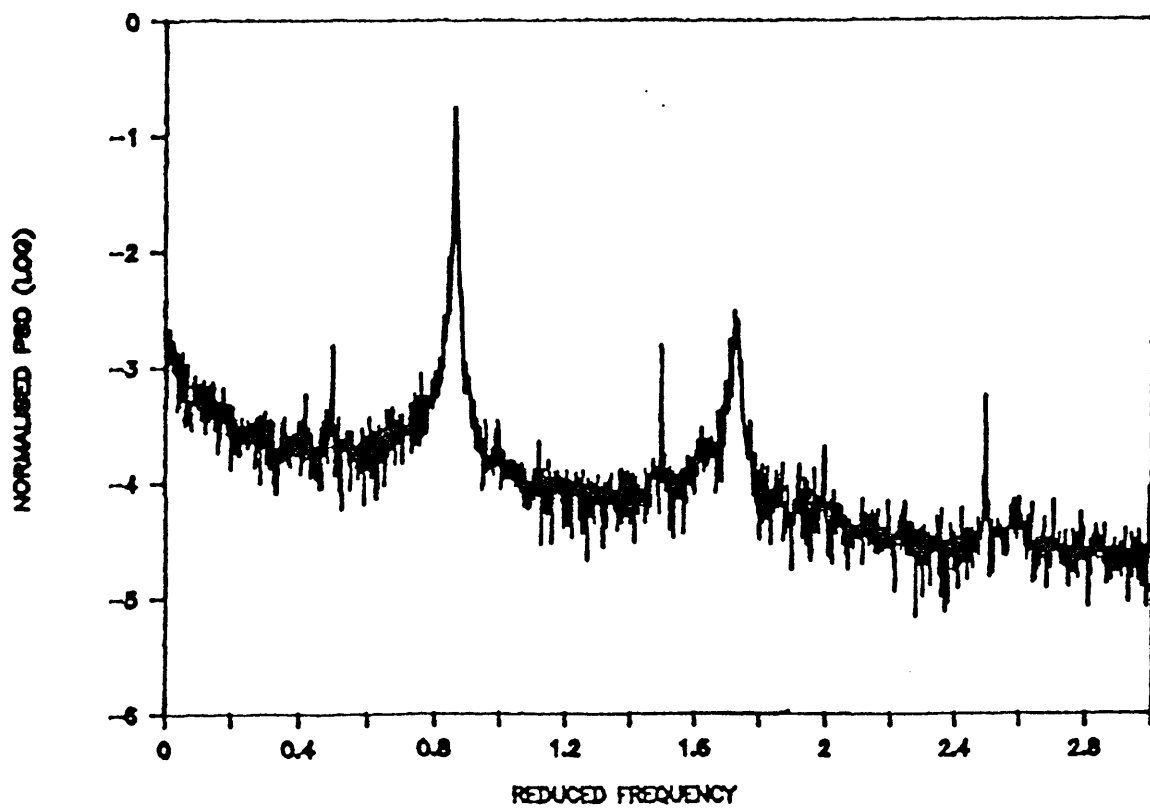


Fig 3.29b Power spectrum of C_N' for aerofoil with rear spoiler at $\alpha = 0^\circ$

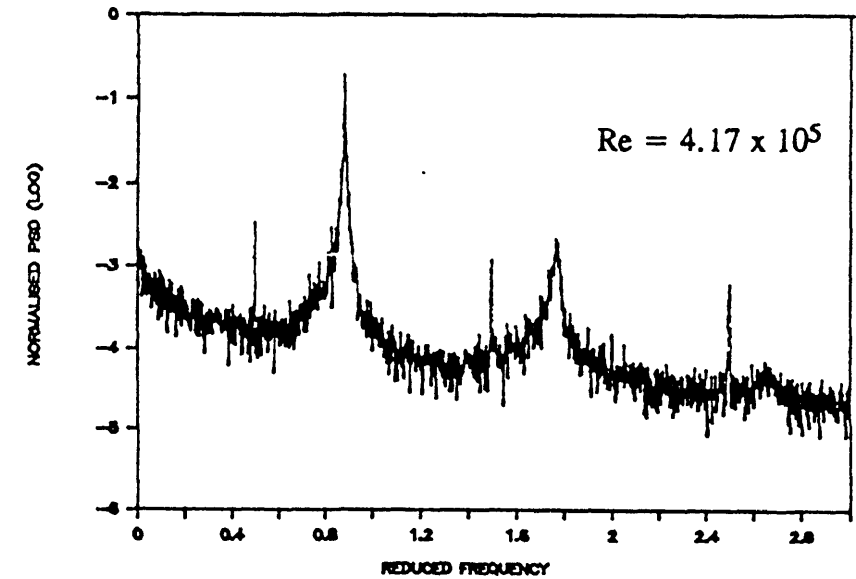
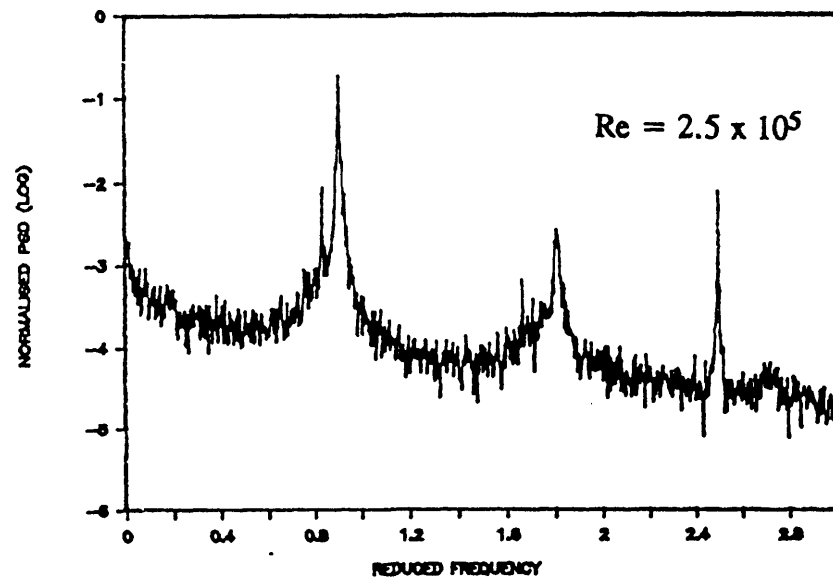
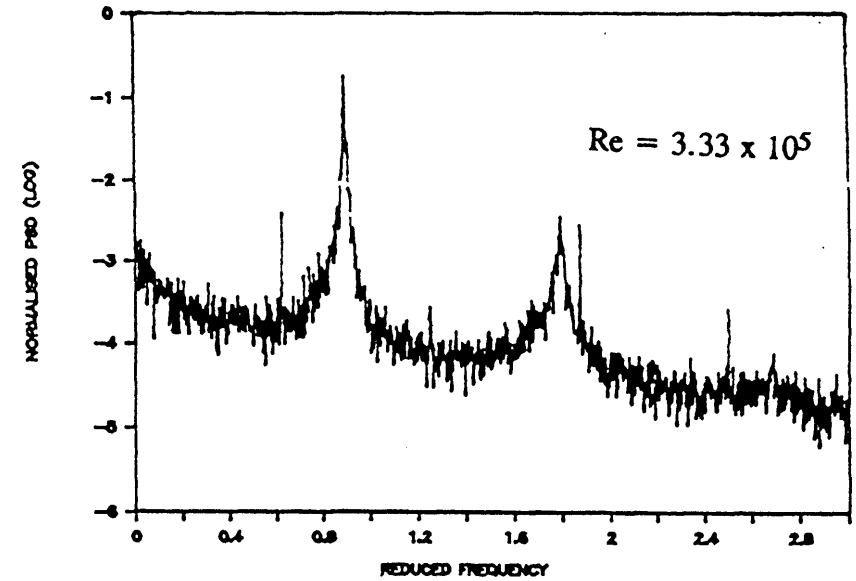
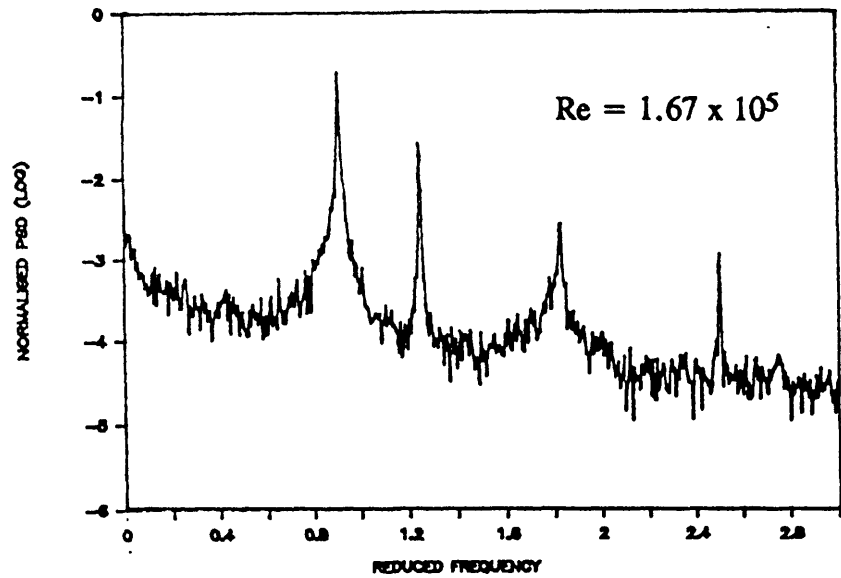


Fig 3.30 Power spectrum of C_N at different Re , at $\alpha = 0^\circ$

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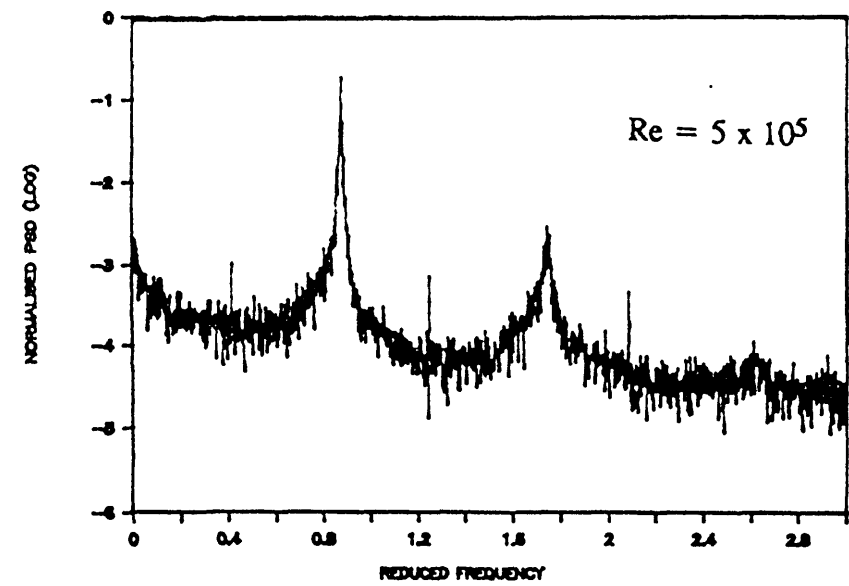
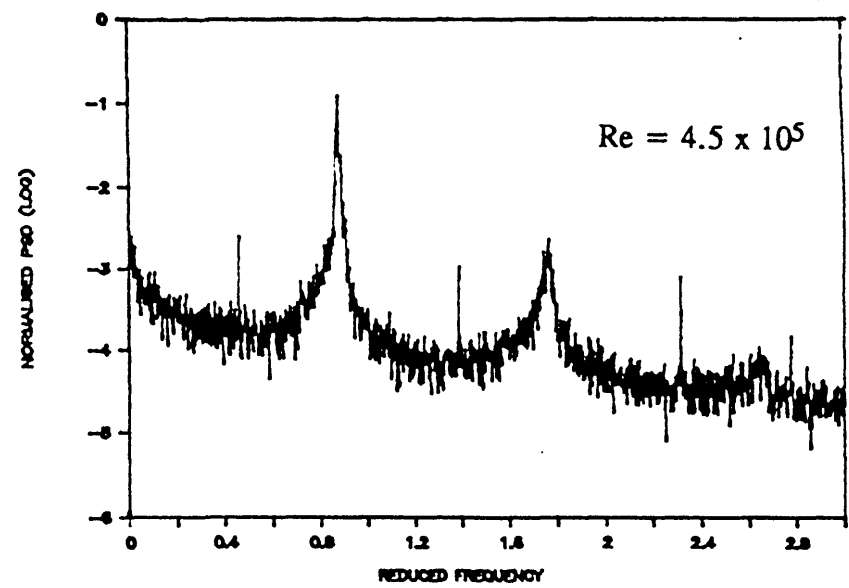


Fig 3.30 Power spectrum of C_N at different Re , at $\alpha = 0^\circ$

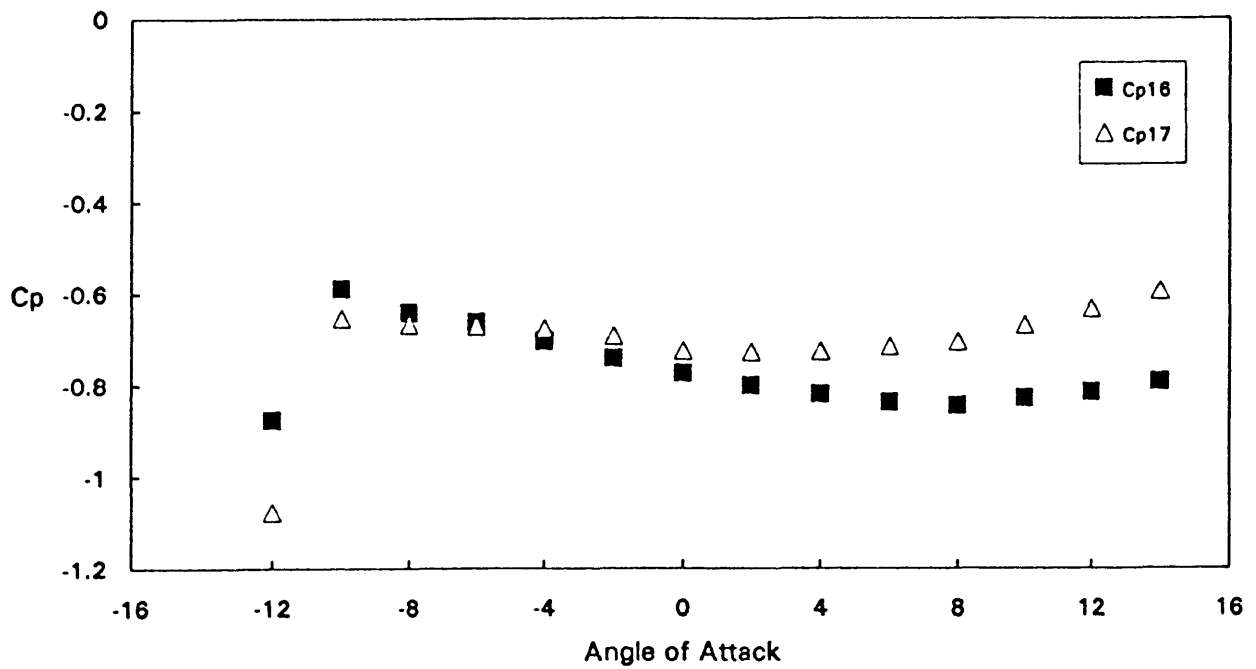


Fig 3.31 Variations of mean C_{p16} and C_{p17} with α

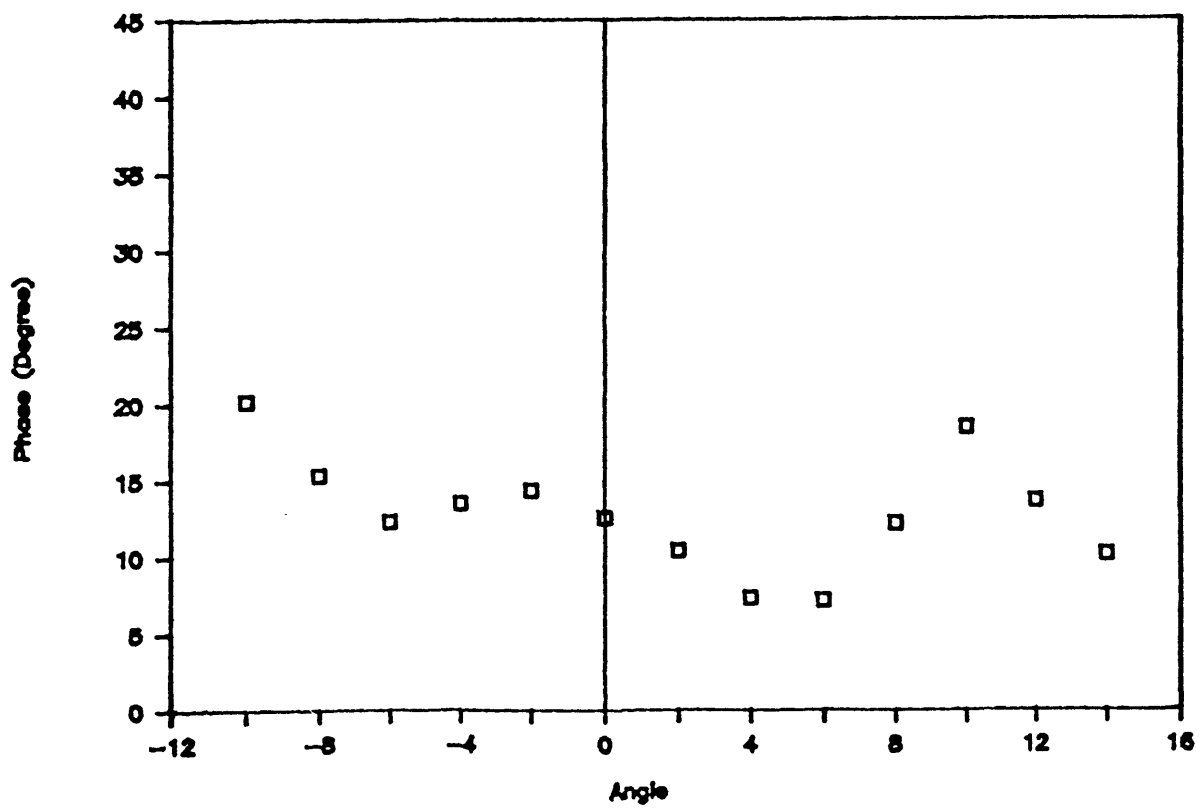


Fig 3.32 Variation of phase difference between C_{p16} and C_{p17} with α

Fig 4.10

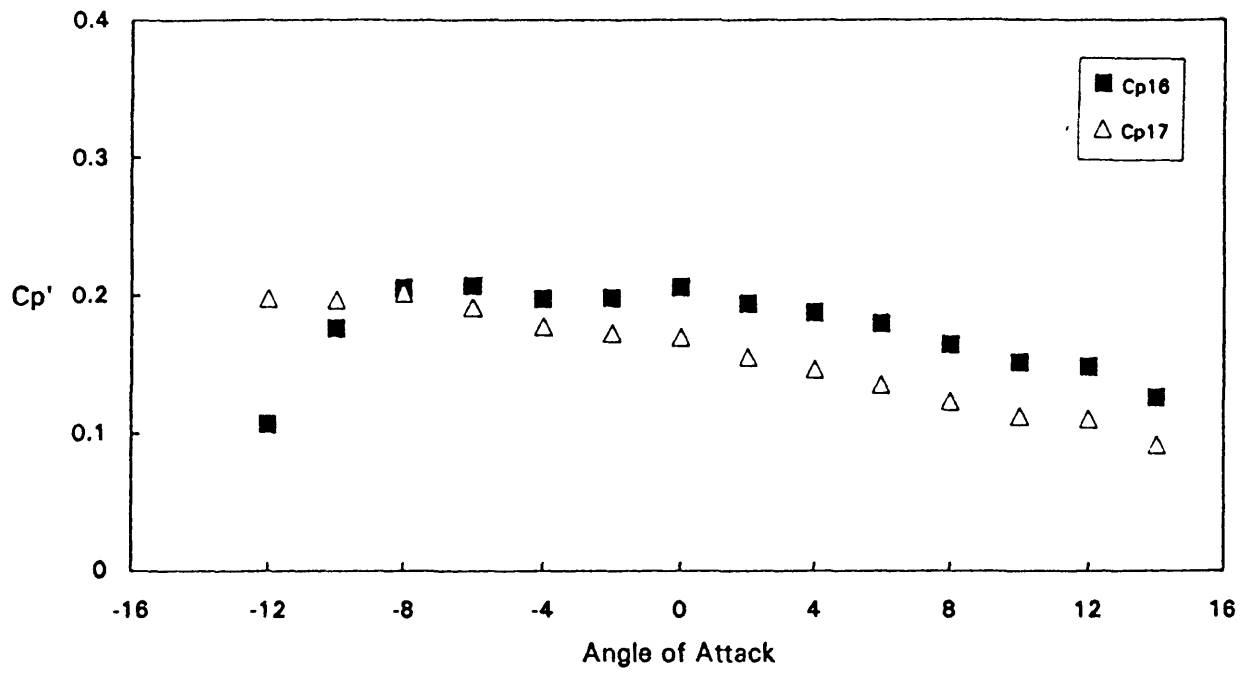


Fig 3.33 Variations of C_{p16}' and C_{p17}' with α

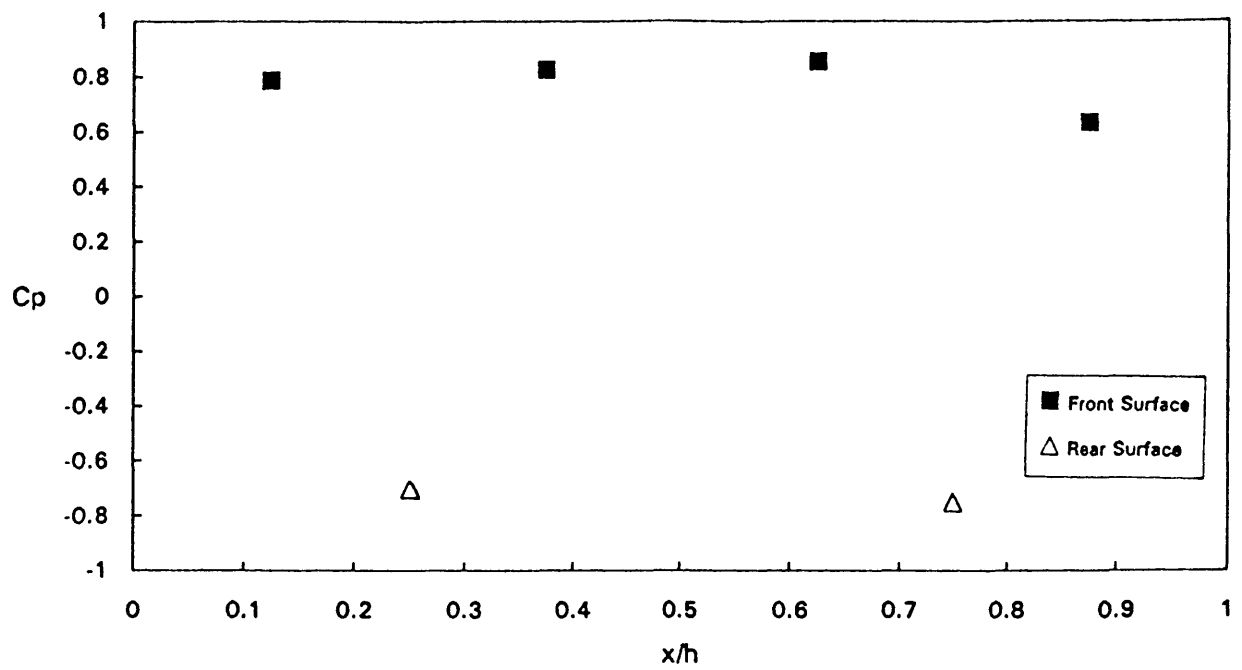


Fig 3.34 Distribution of C_p on spoiler at $\alpha = 0^\circ$

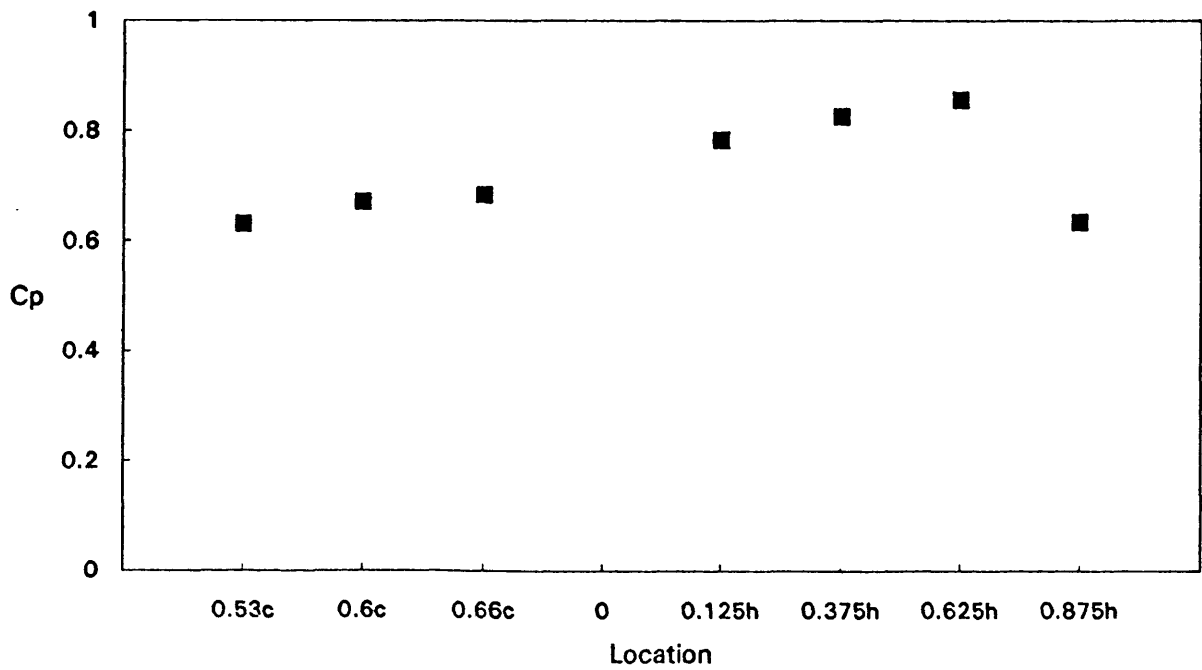


Fig 3.35 Distribution of C_p in the vicinity of spoiler hinge line

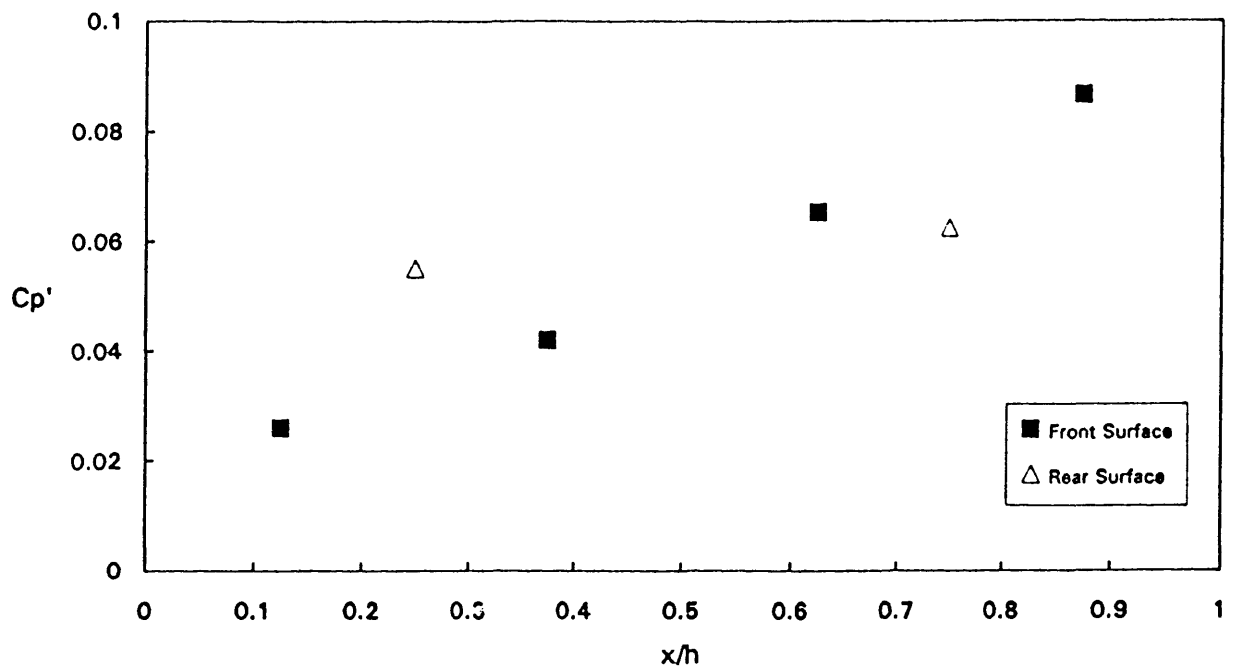


Fig 3.36 Distribution of C_p' on spoiler at $\alpha = 0^\circ$

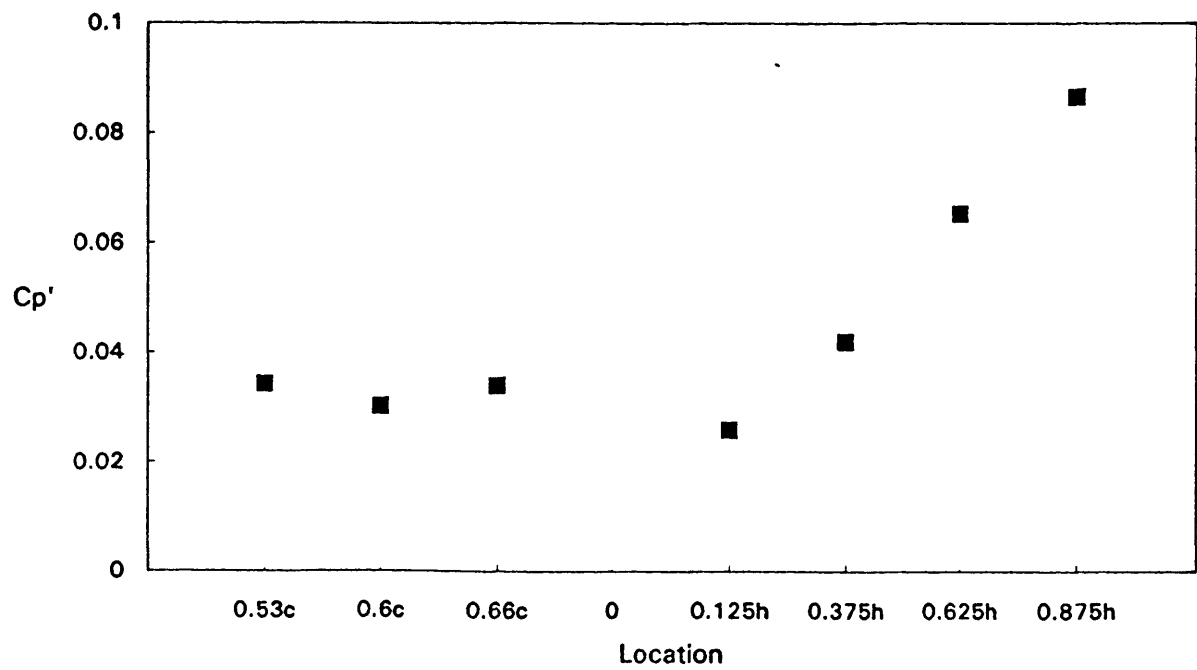


Fig 3.37 Distribution of C_p' in the vicinity of spoiler hinge line

PSD1027A

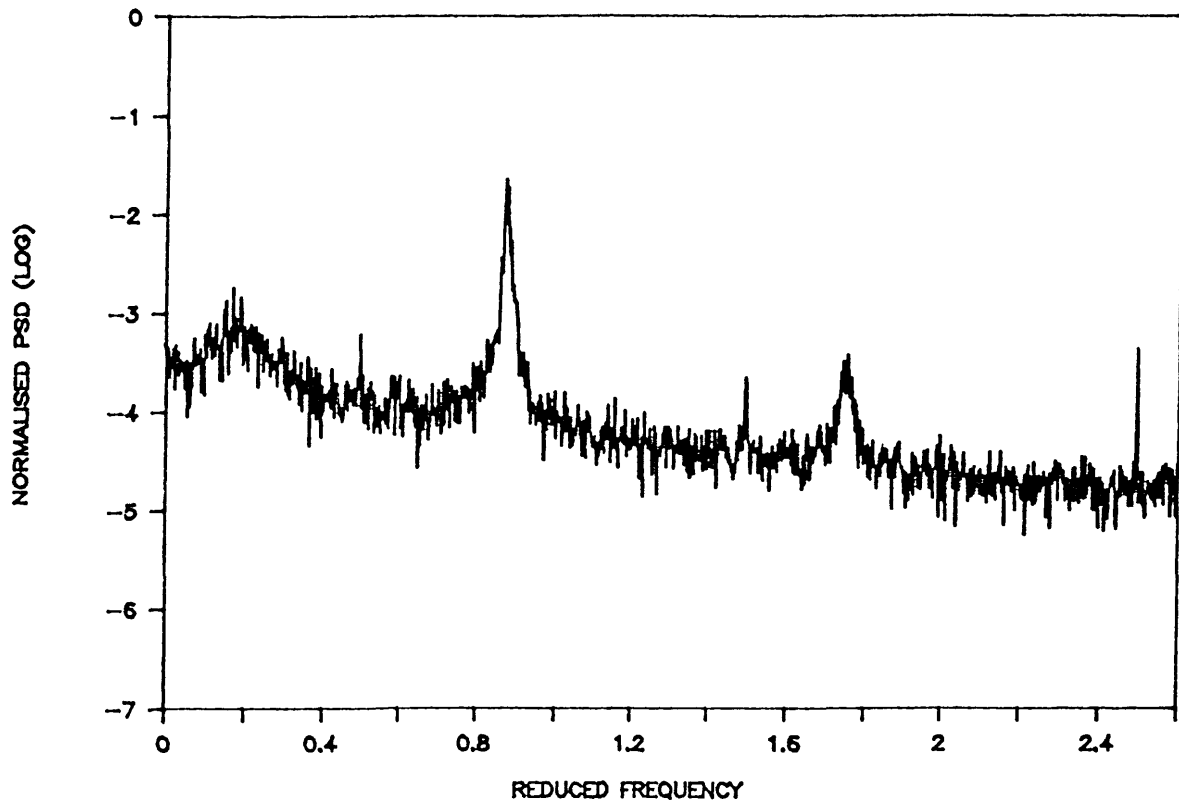


Fig3.38a Power spectrum of C_p at $x/h = 1/8$ on spoiler front surface

PSD1027B

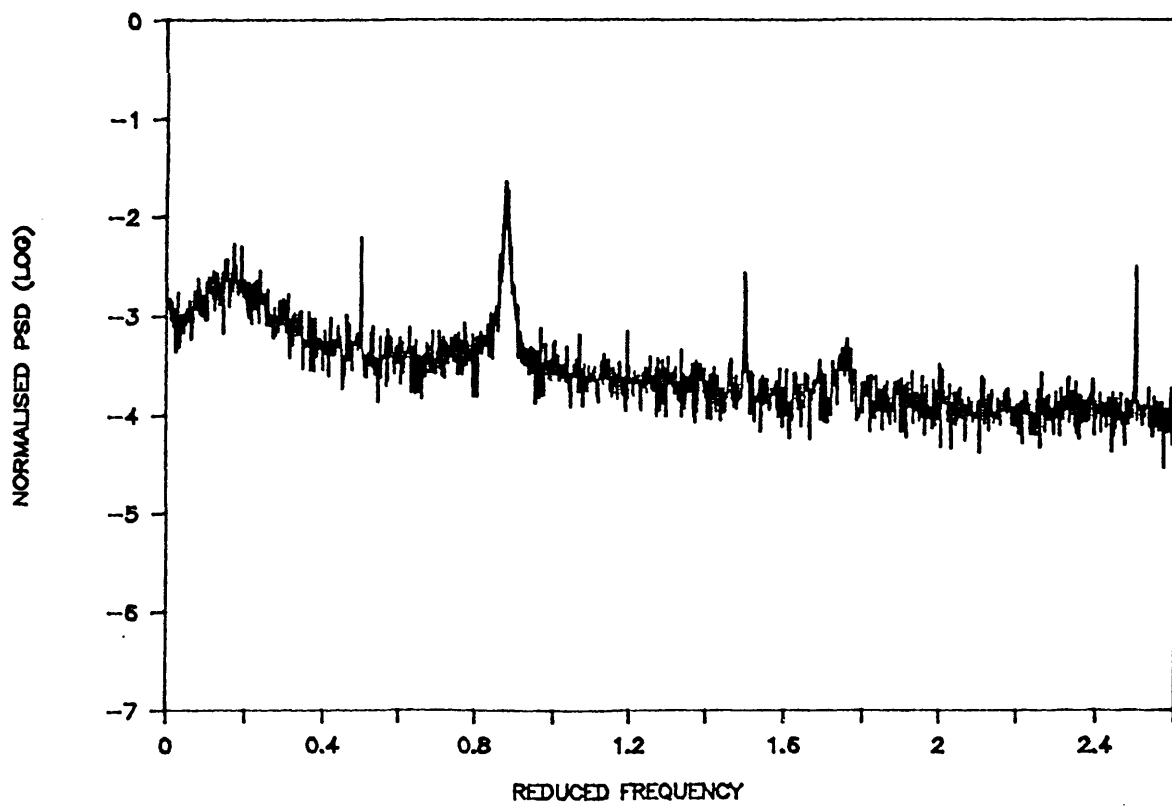


Fig3.38b Power spectrum of C_p at $x/h = 3/8$ on spoiler front surface

PSD1027C

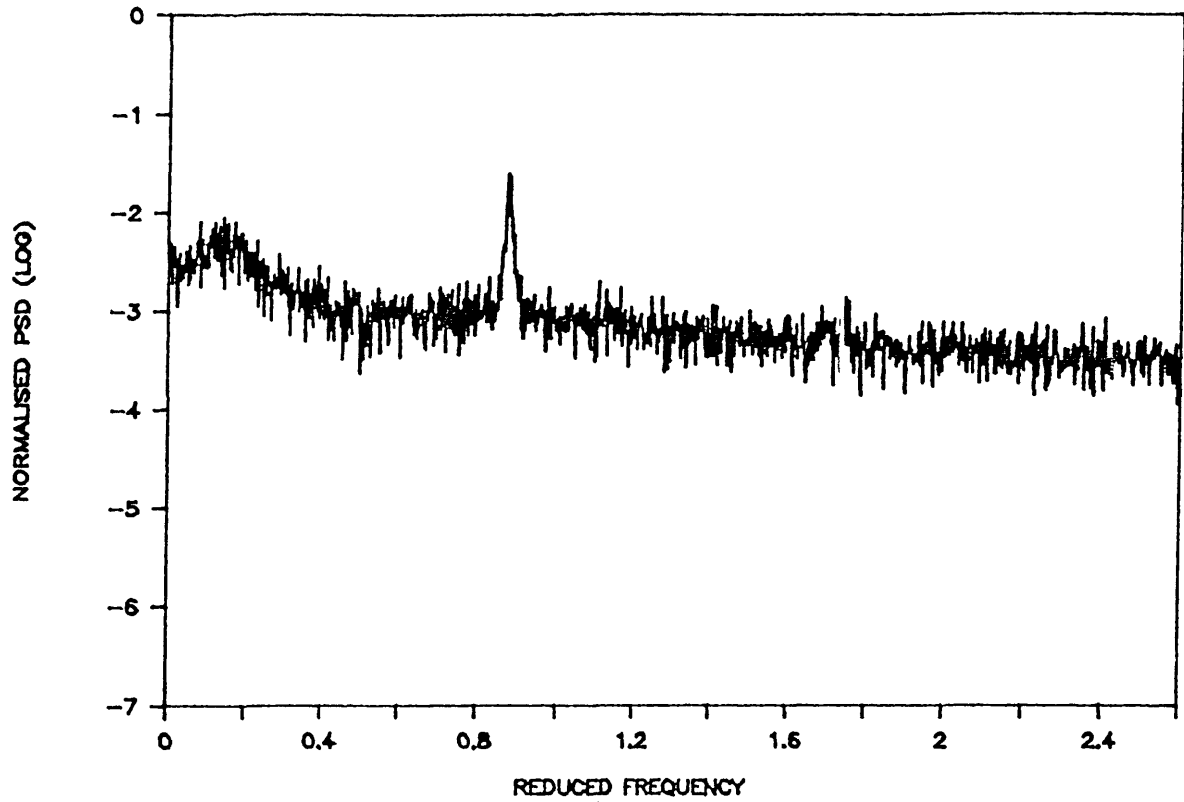


Fig3.38c Power spectrum of C_p at $x/h = 5/8$ on spoiler front surface

PSD1027D

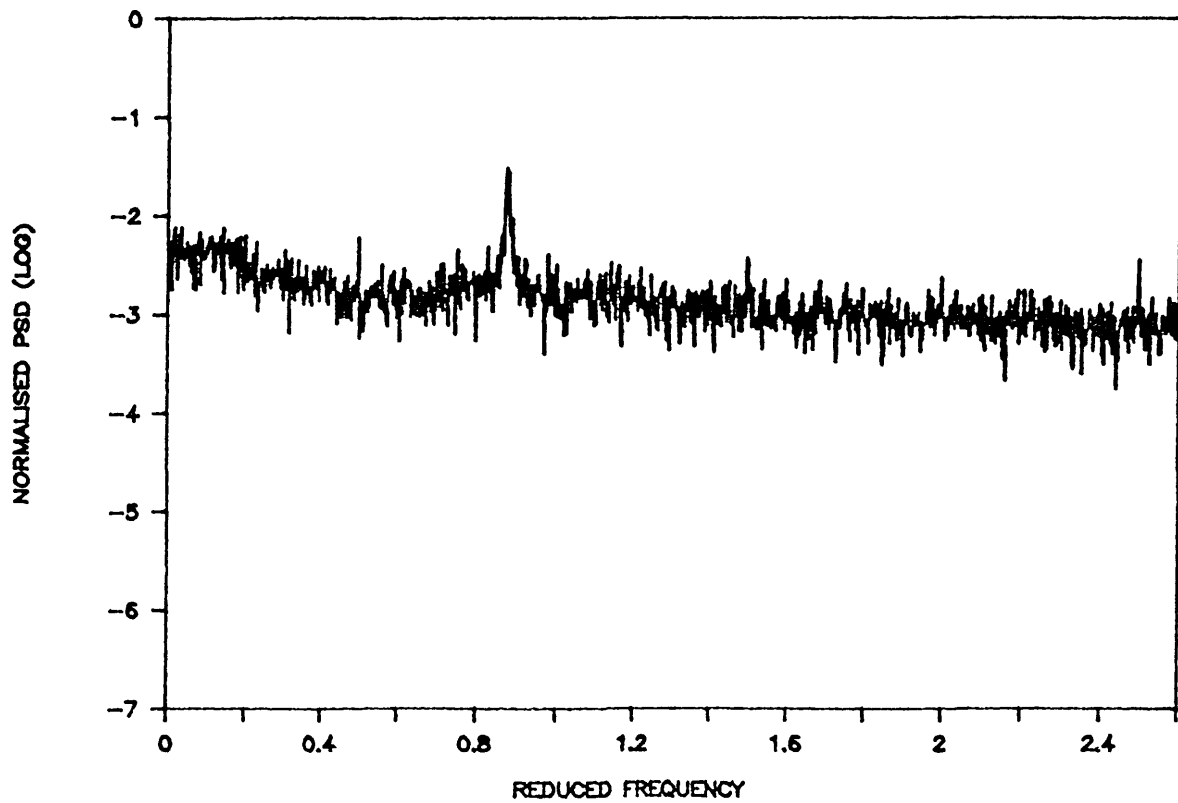


Fig3.38d Power spectrum of C_p at $x/h = 7/8$ on spoiler front surface

PSD1035A

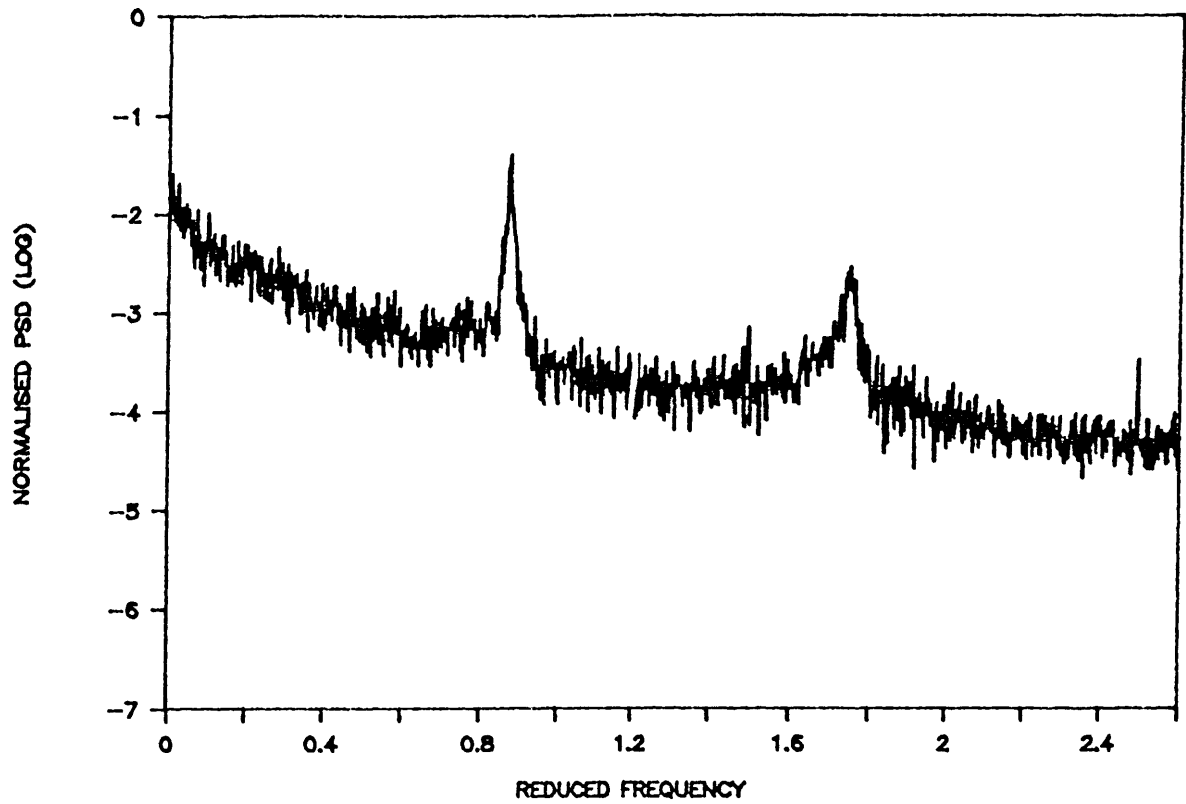


Fig3.39a Power spectrum of C_p at $x/h = 1/4$ on spoiler rear surface

PSD1035B

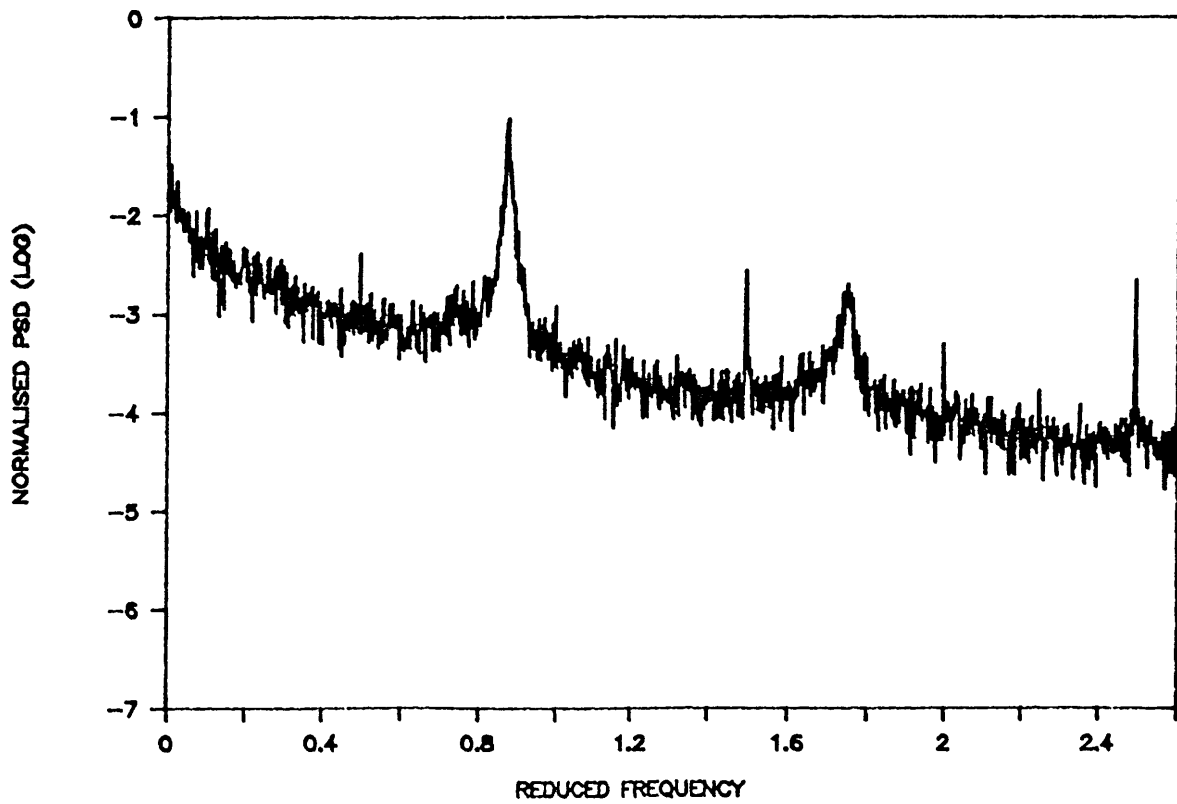


Fig3.39b Power spectrum of C_p at $x/h = 3/4$ on spoiler rear surface

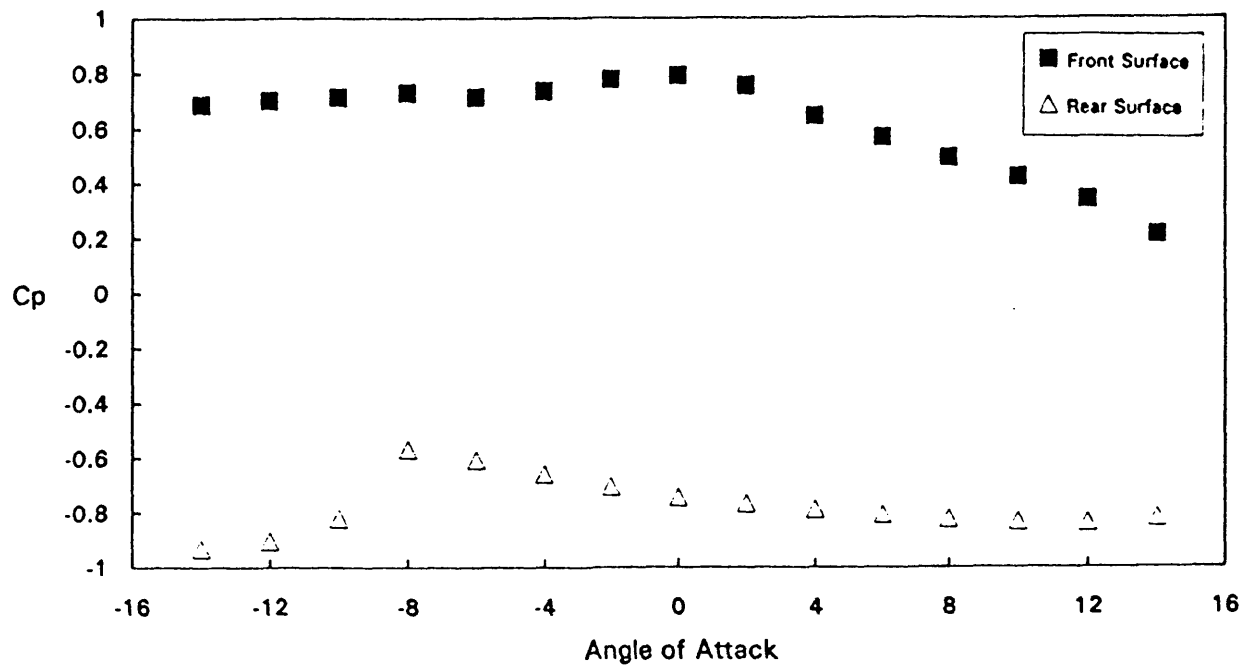


Fig 3.40 Variations of C_{px} and C_{py} with α

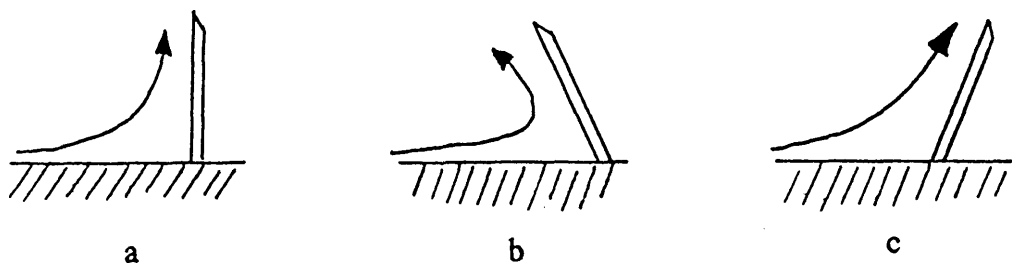


Fig 3.41 Spoiler-aerofoil configurations presented to the flow at different α

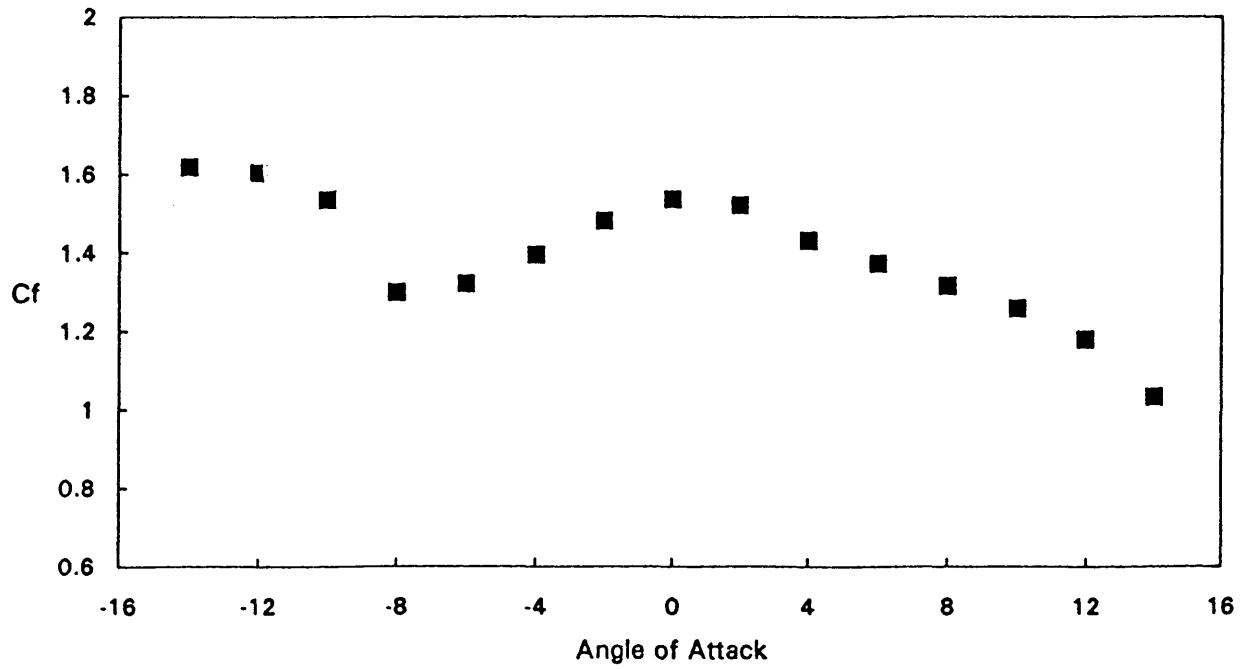


Fig 3.42 Variation of C_F with α

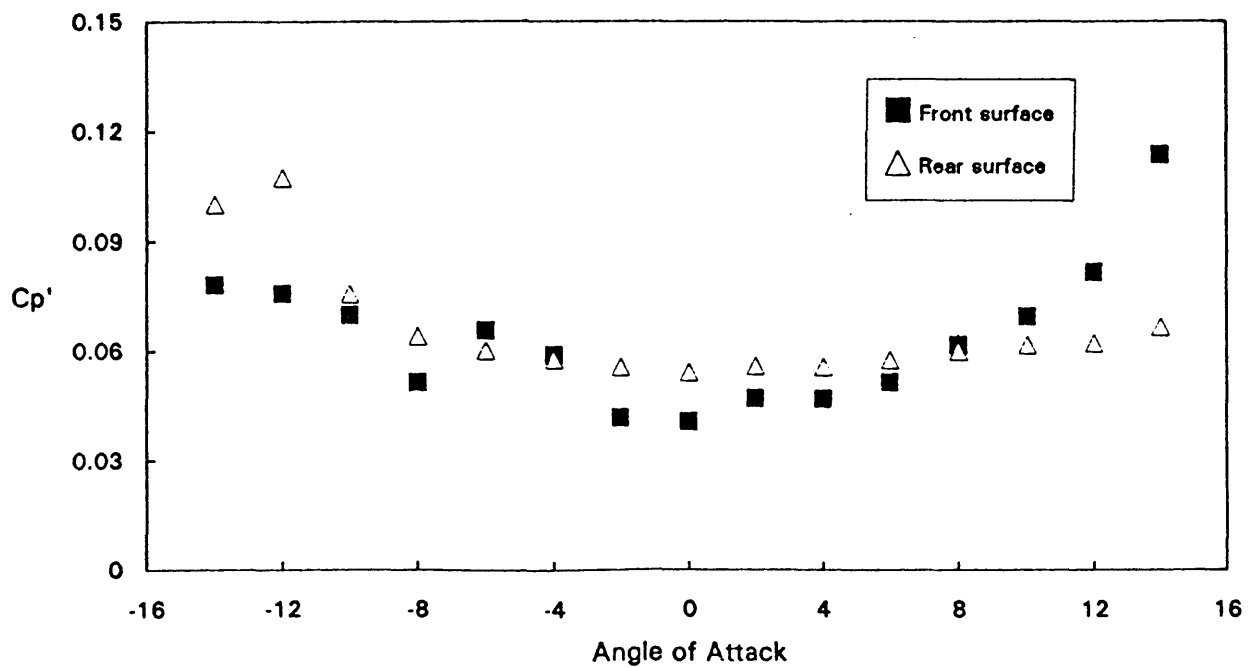


Fig 3.43 Variations of $C_{px'}$ and $C_{py'}$ with α

PSD907A

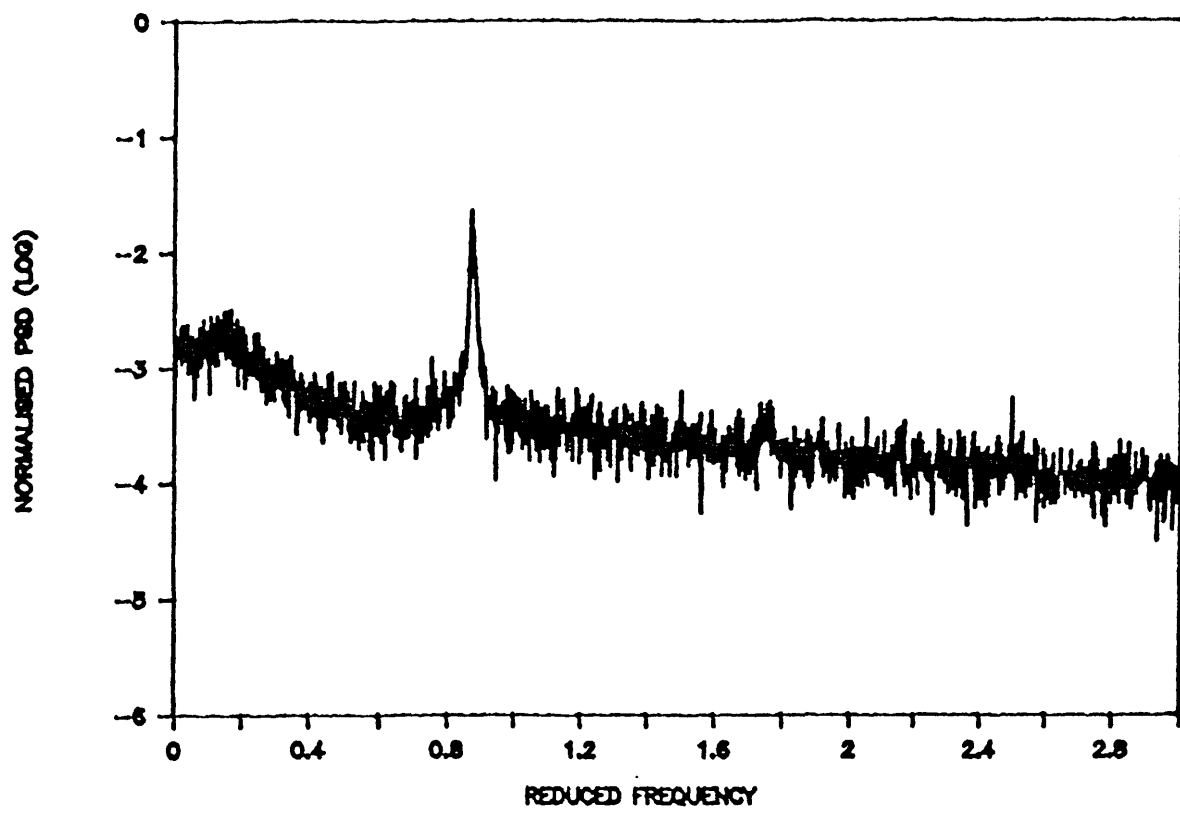


Fig 3.44a Power spectrum of C_{px} at $\alpha = 0^\circ$

PSD907B

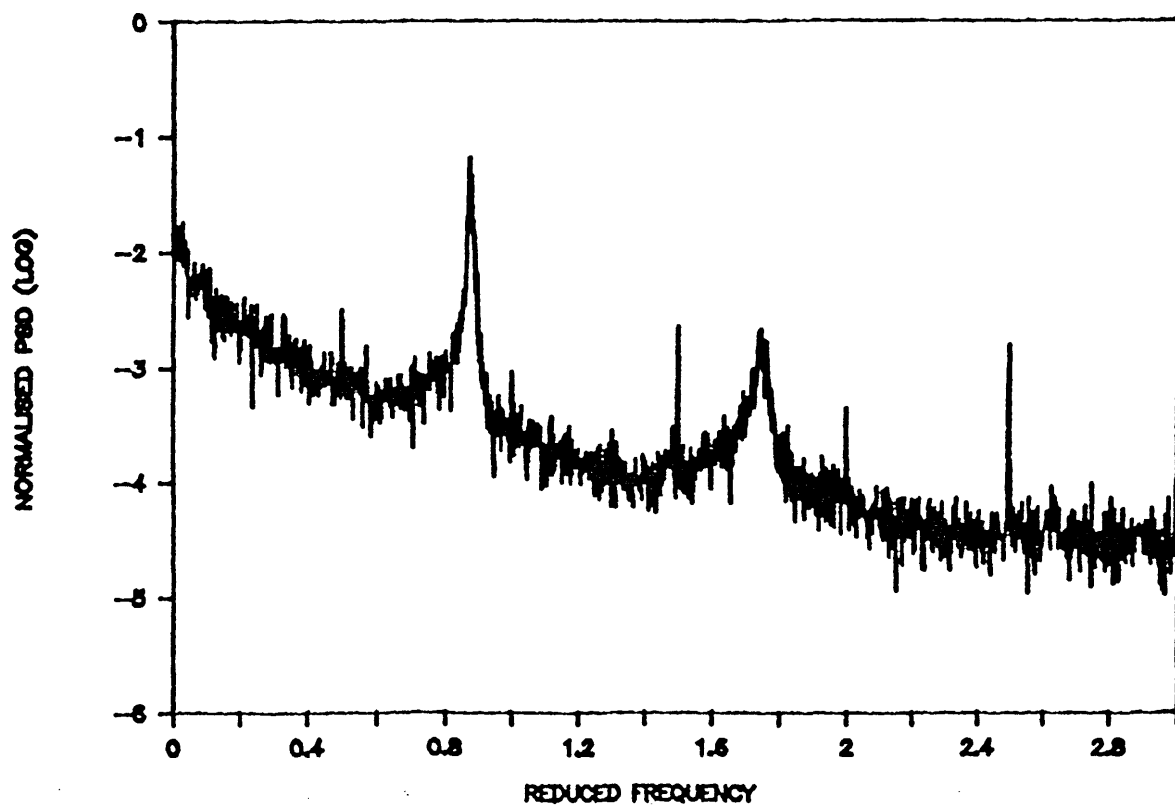


Fig 3.44b Power spectrum of C_{py} at $\alpha = 0^\circ$

PSD931A

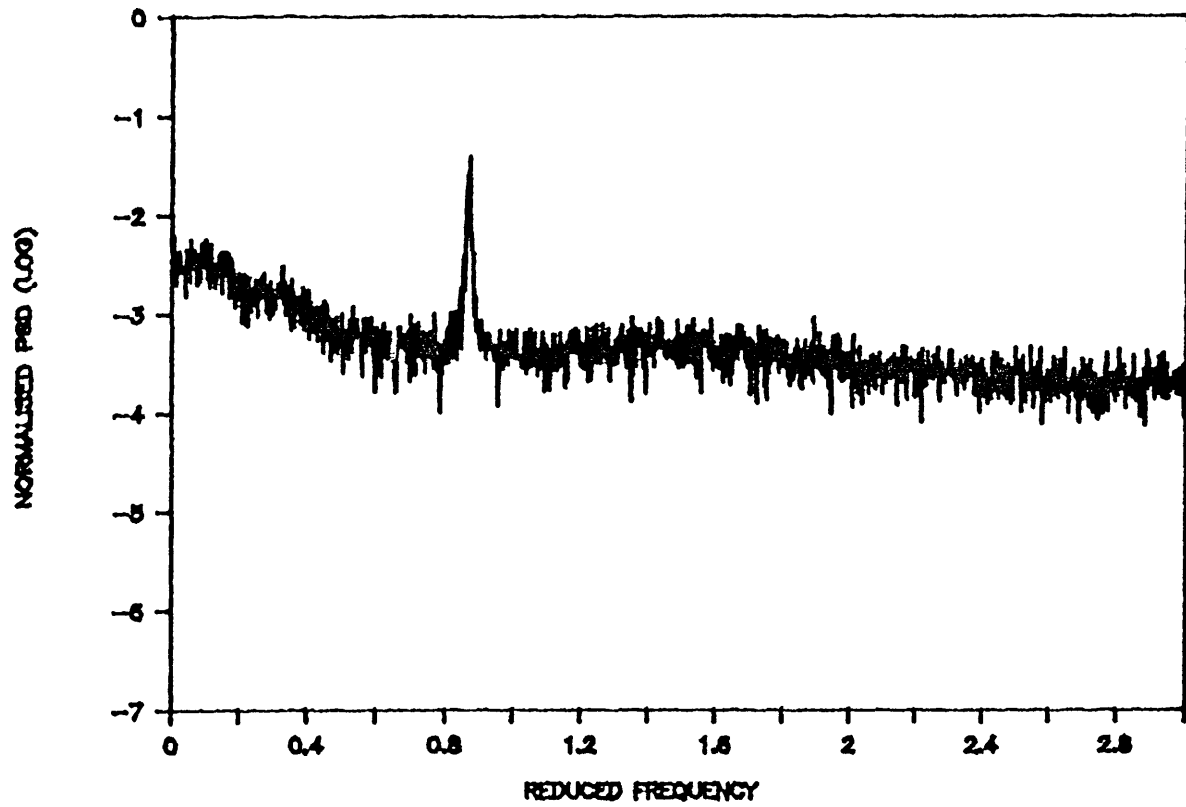


Fig 3.45a Power spectrum of C_{px} at $\alpha = 6^\circ$

PSD931B

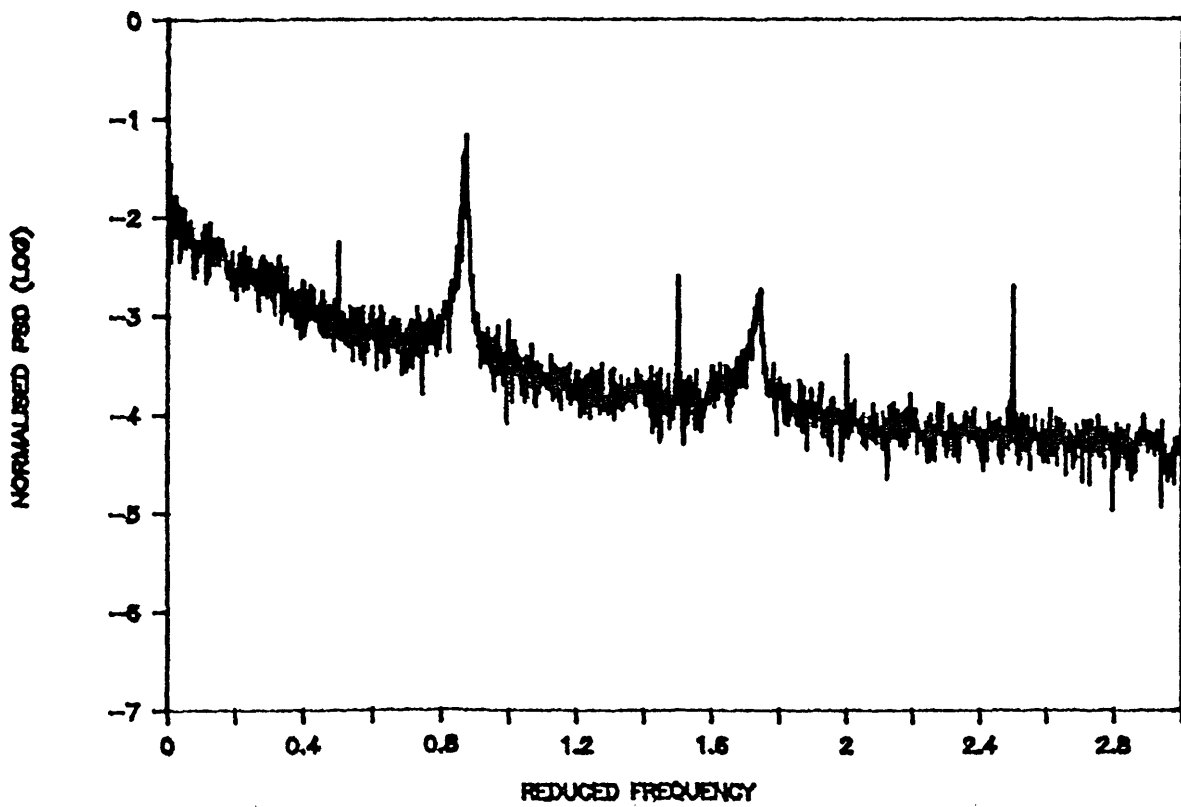


Fig 3.45b Power spectrum of C_{py} at $\alpha = 6^\circ$

PSD939A

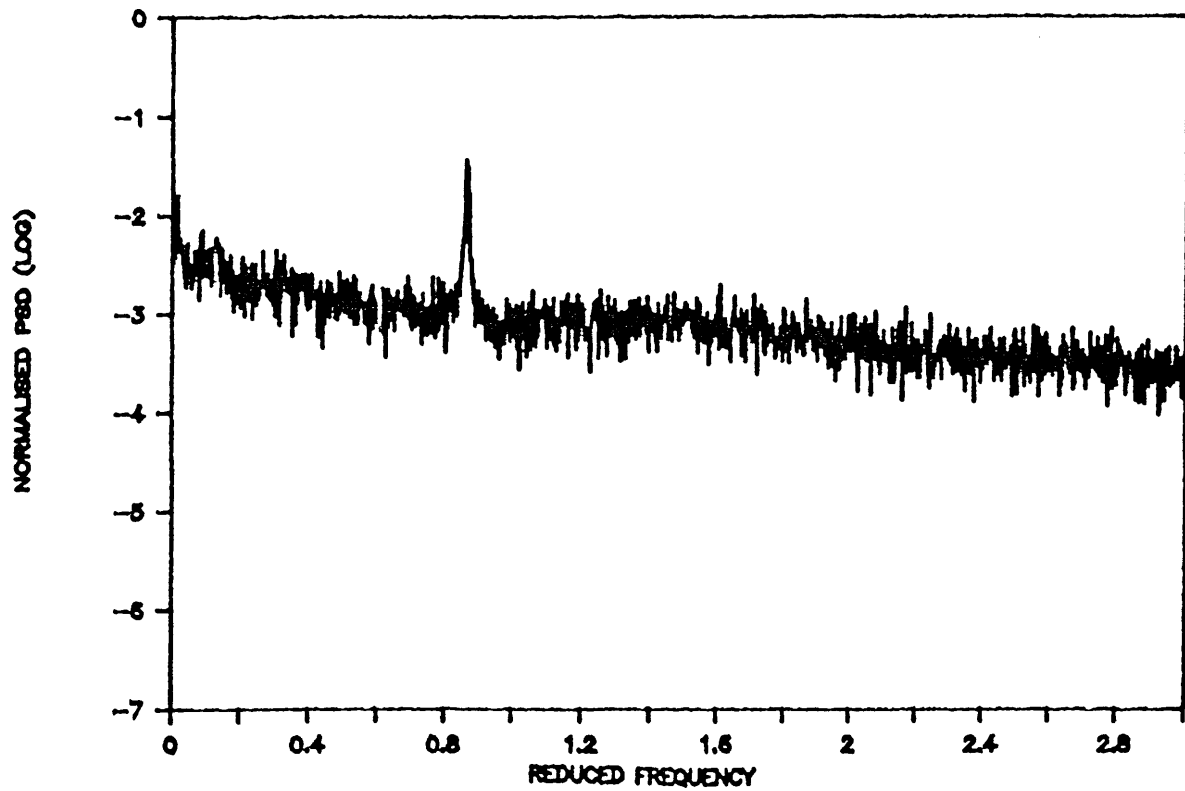


Fig 3.46a Power spectrum of C_{px} at $\alpha = 8^\circ$

PSD939B

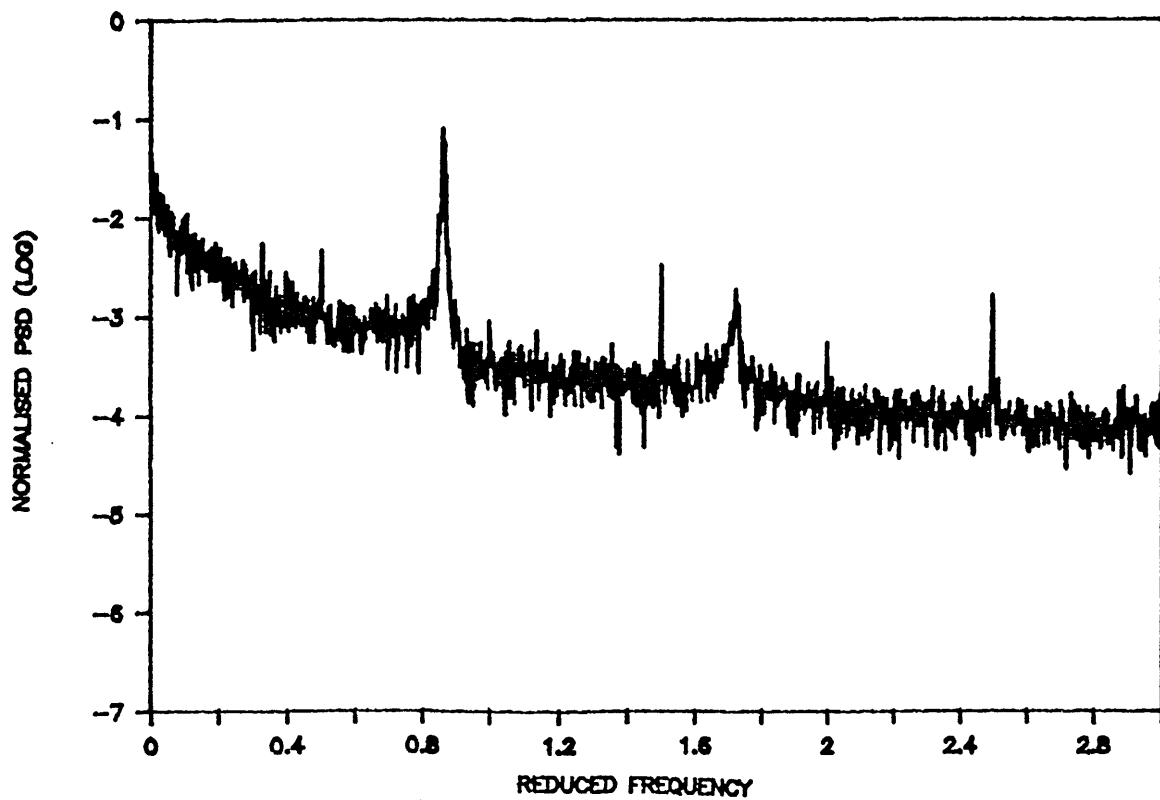


Fig 3.46b Power spectrum of C_{py} at $\alpha = 8^\circ$

PSD1019A

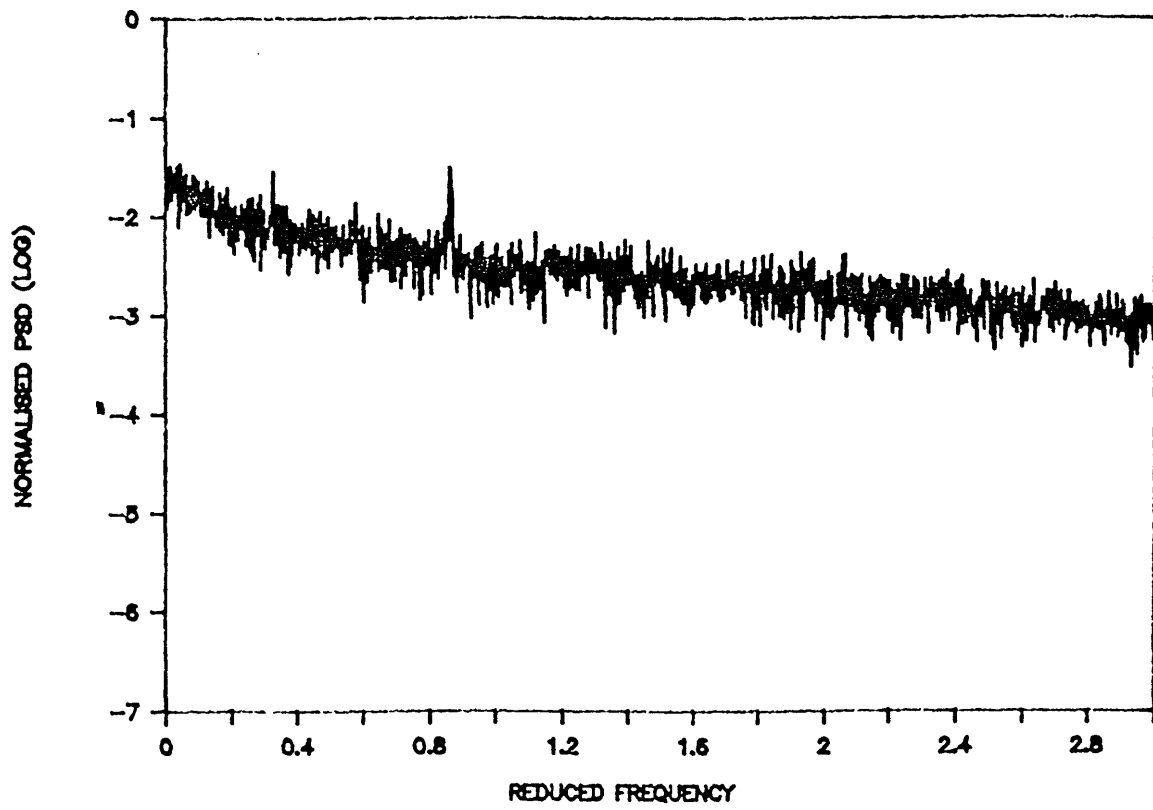


Fig 3.47a Power spectrum of C_{px} at $\alpha = 12^\circ$

PSD1019B

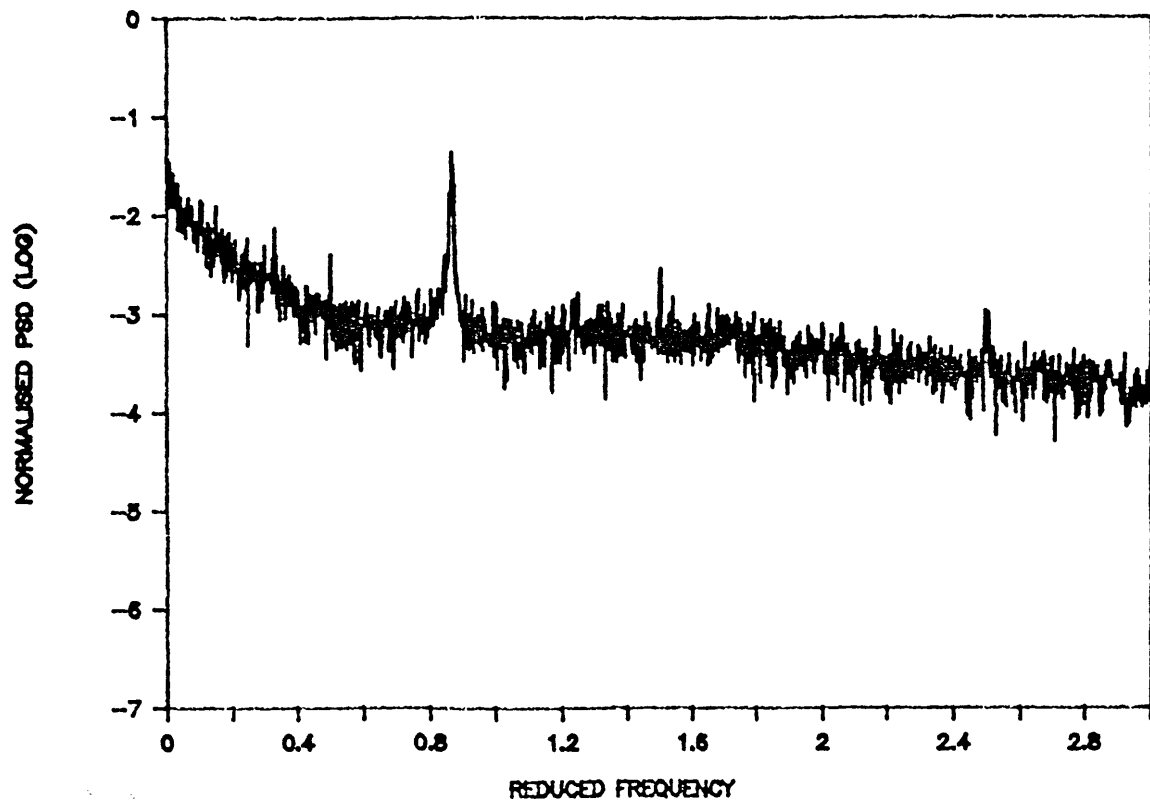


Fig 3.47b Power spectrum of C_{py} at $\alpha = -6^\circ$

PSD979A

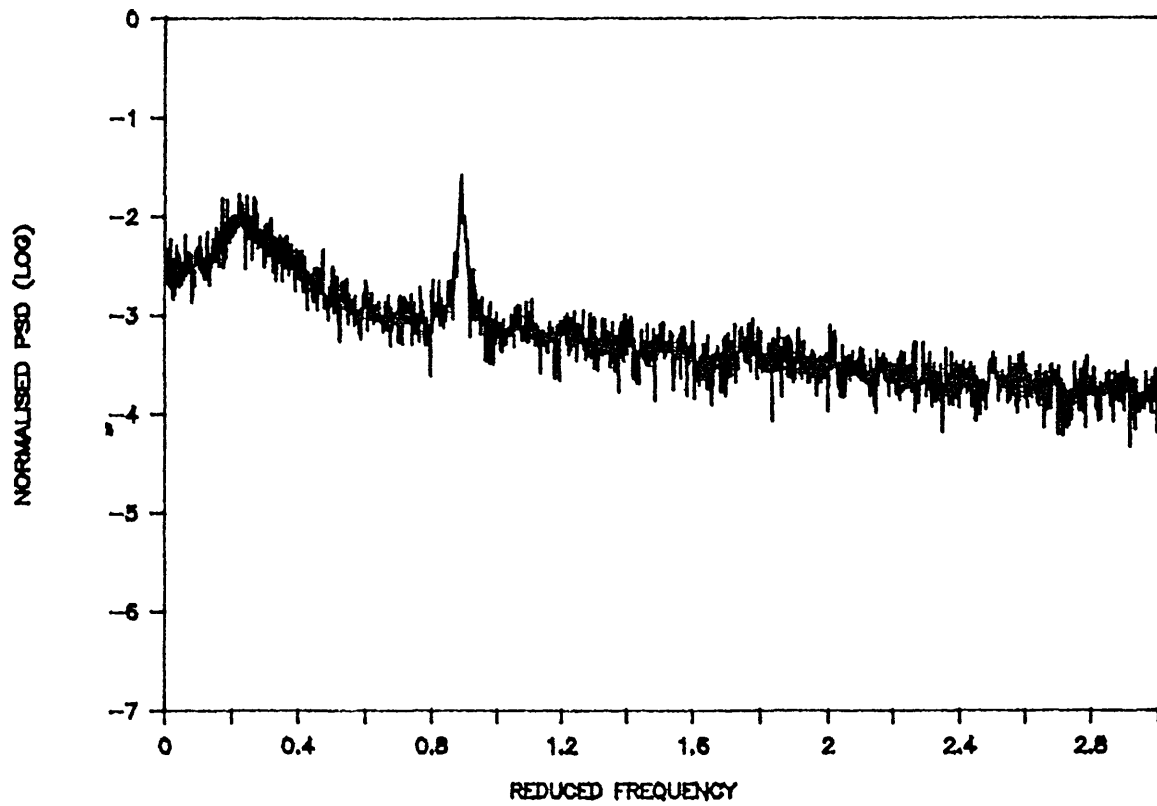


Fig 3.48a Power spectrum of C_{px} at $\alpha = -6^\circ$

PSD979B

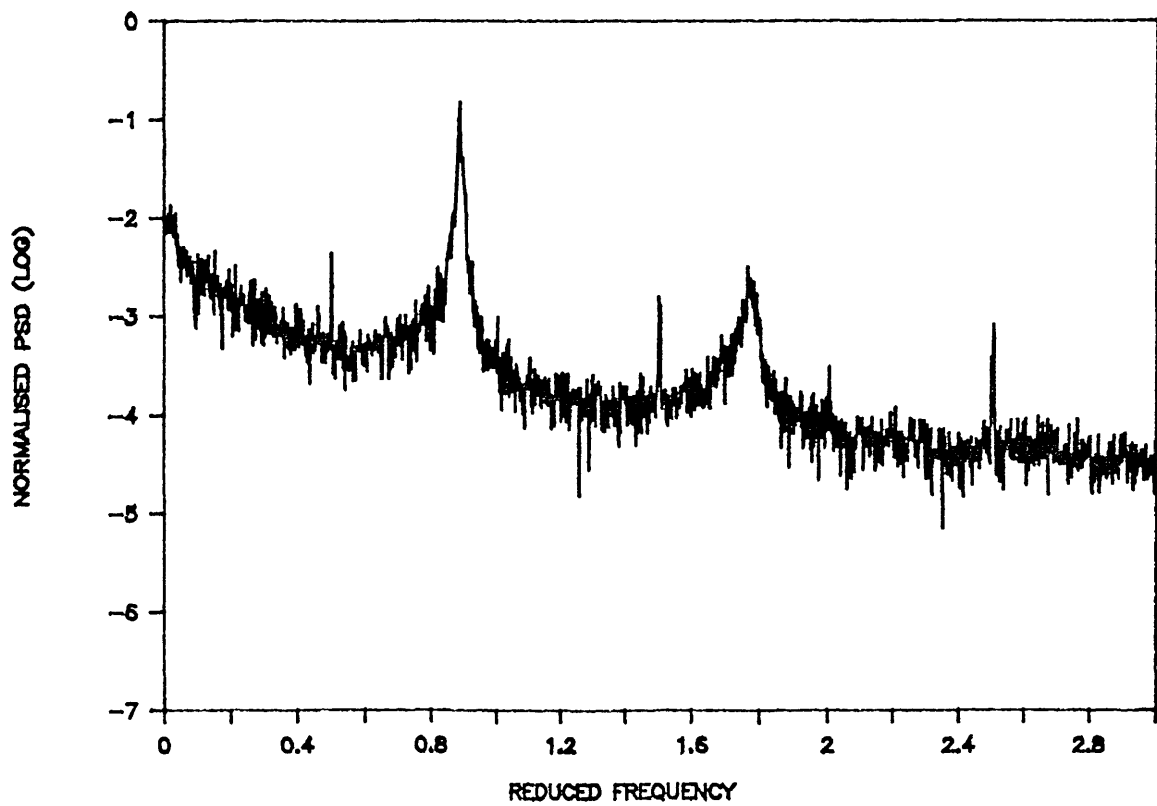


Fig 3.48b Power spectrum of C_{py} at $\alpha = 12^\circ$

PSD995A

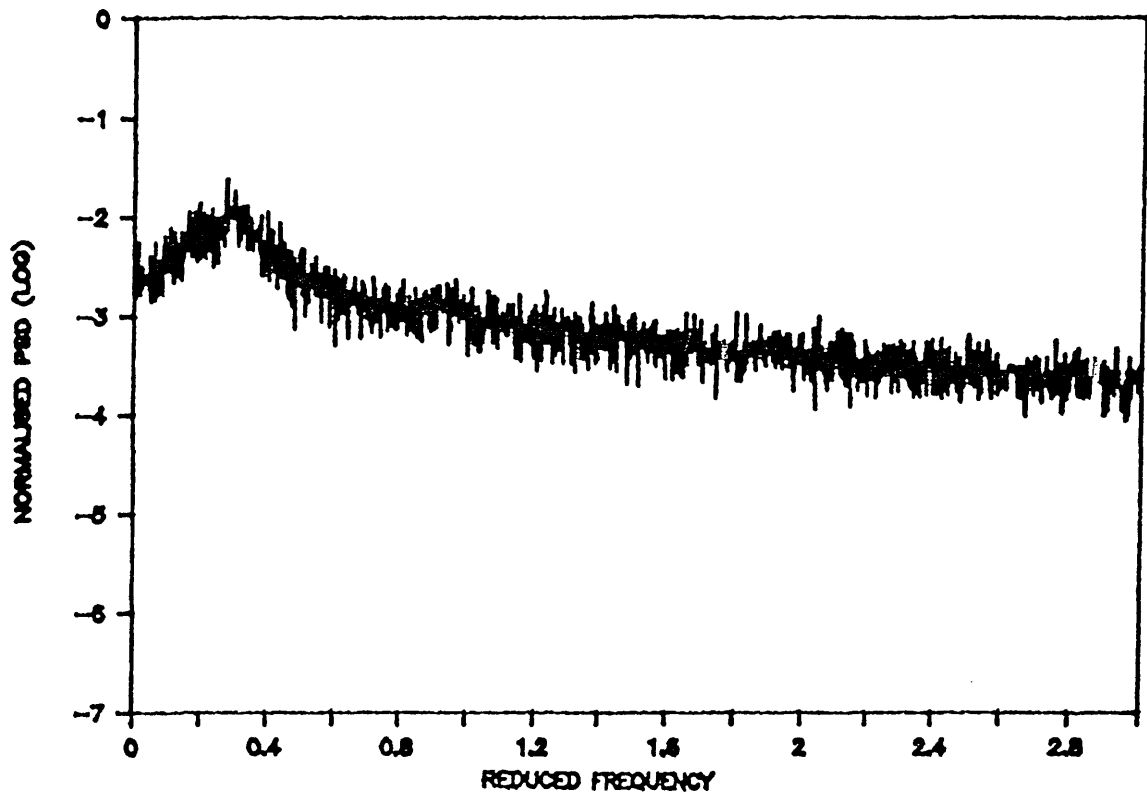


Fig 3.49a Power spectrum of C_{px} at $\alpha = -10^\circ$

PSD995B

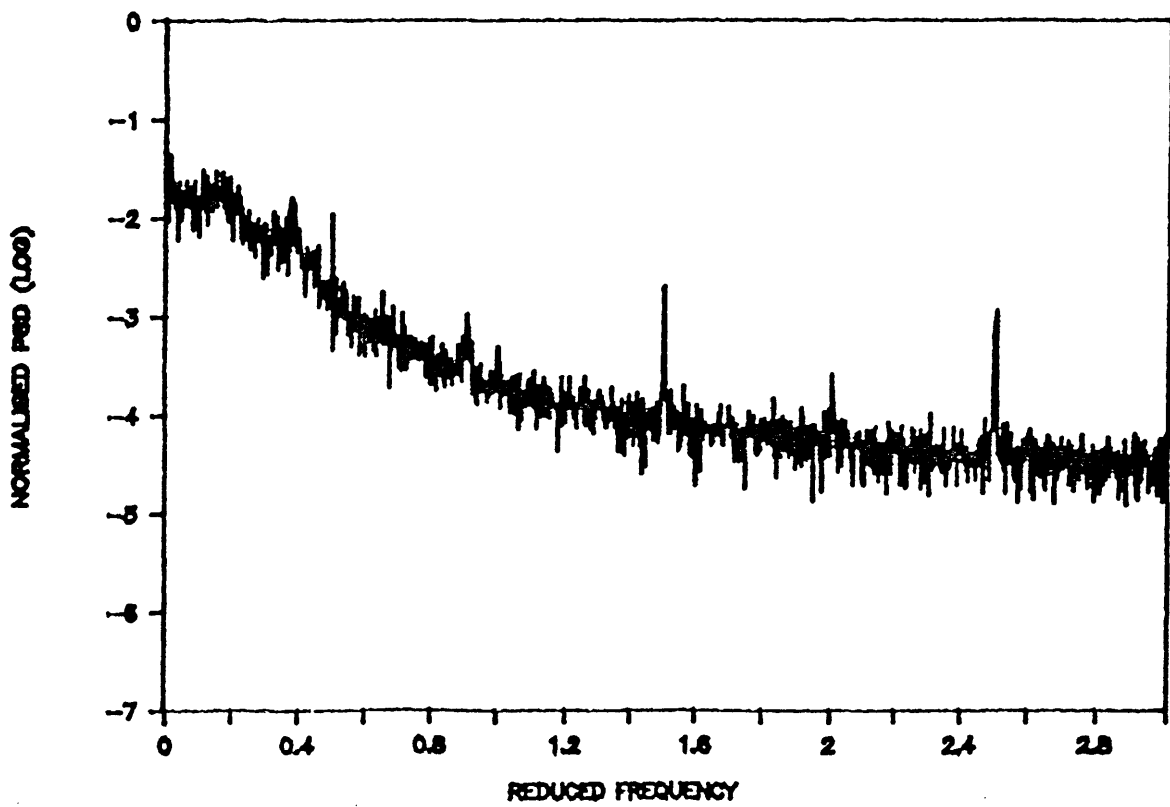


Fig 3.49b Power spectrum of C_{py} at $\alpha = -10^\circ$

PSD1003A

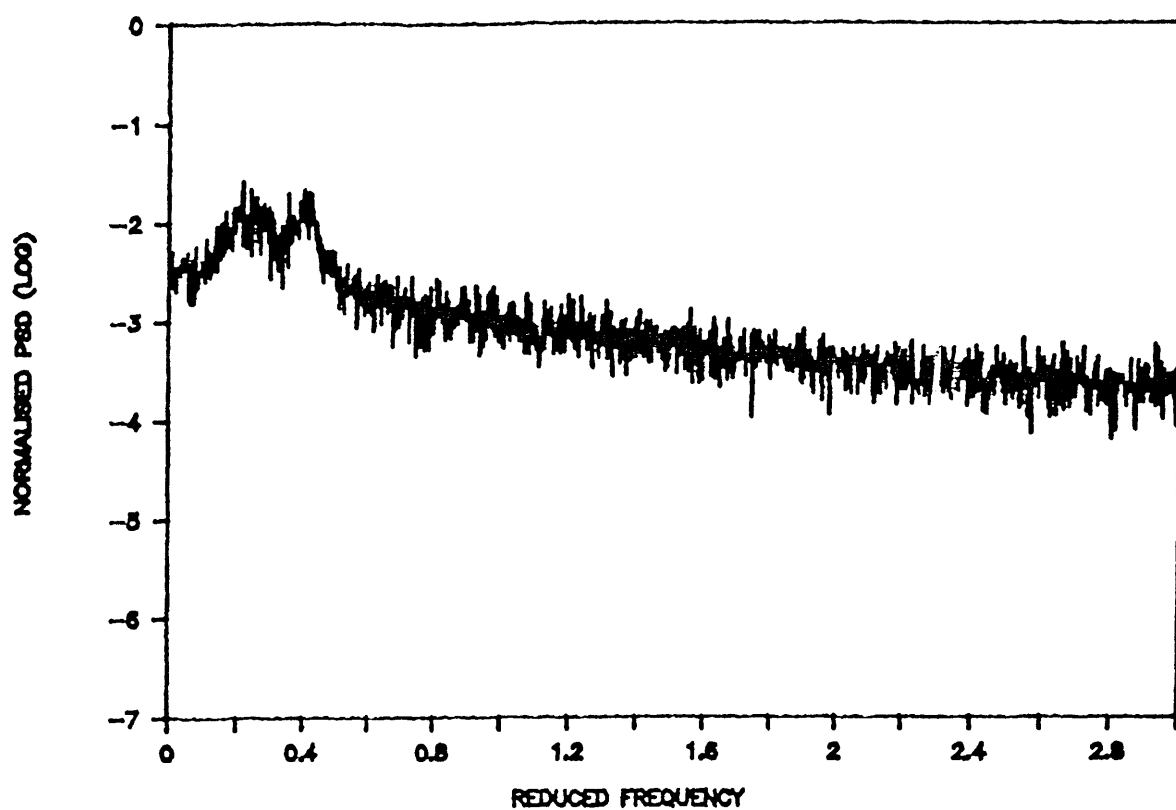


Fig 3.50a Power spectrum of C_{px} at $\alpha = -12^\circ$

PSD1003B

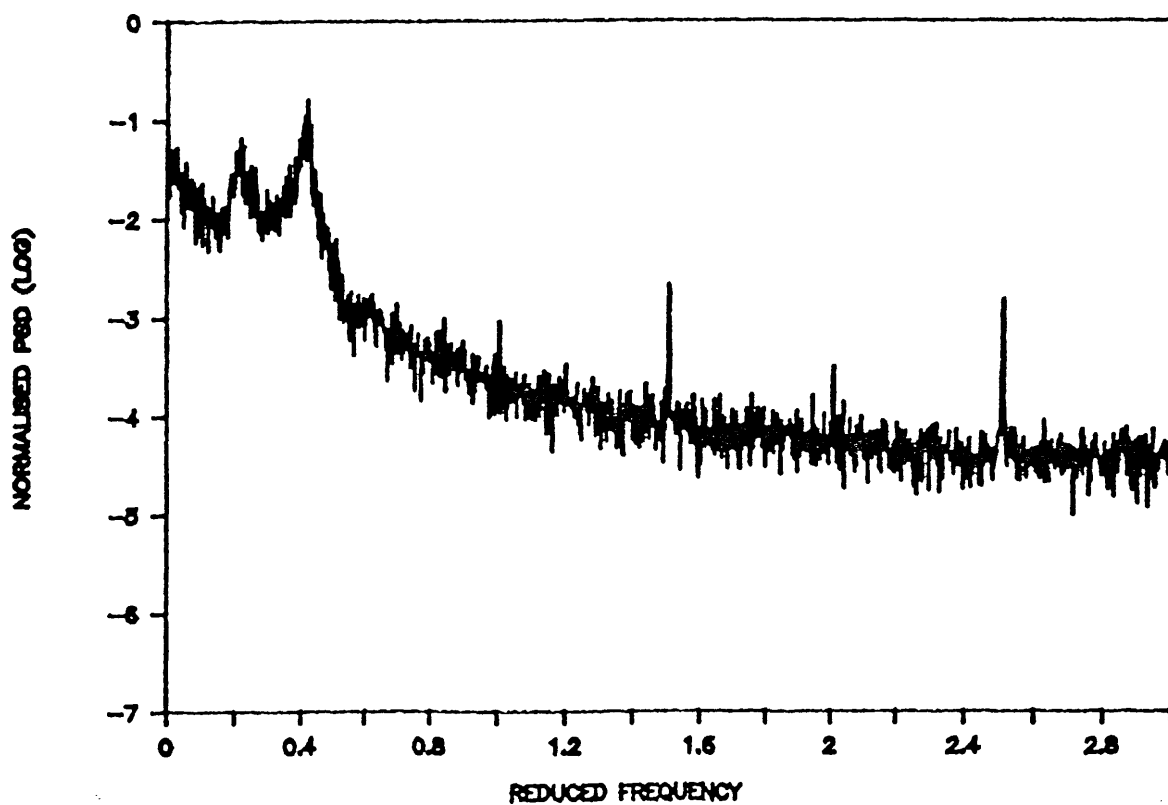


Fig 3.50b Power spectrum of C_{py} at $\alpha = -12^\circ$

PSD1011A

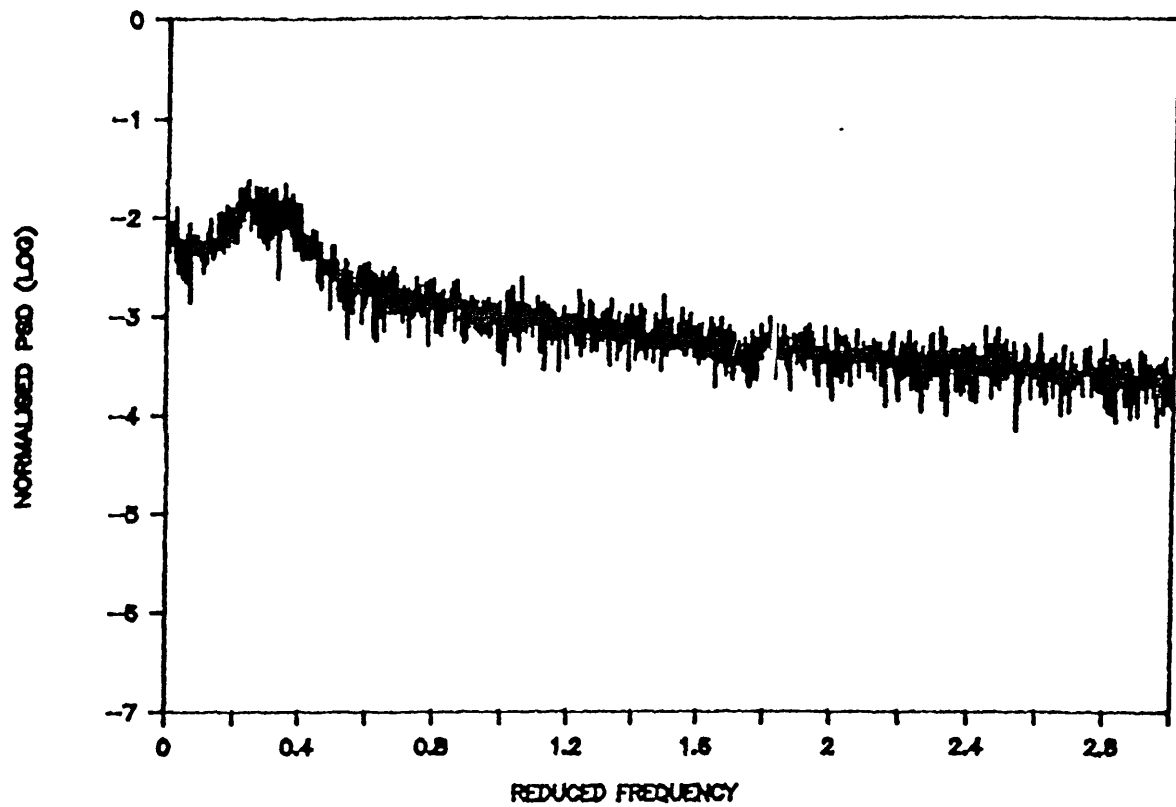


Fig 3.51a Power spectrum of C_{px} at $\alpha = -14^\circ$

PSD1011B

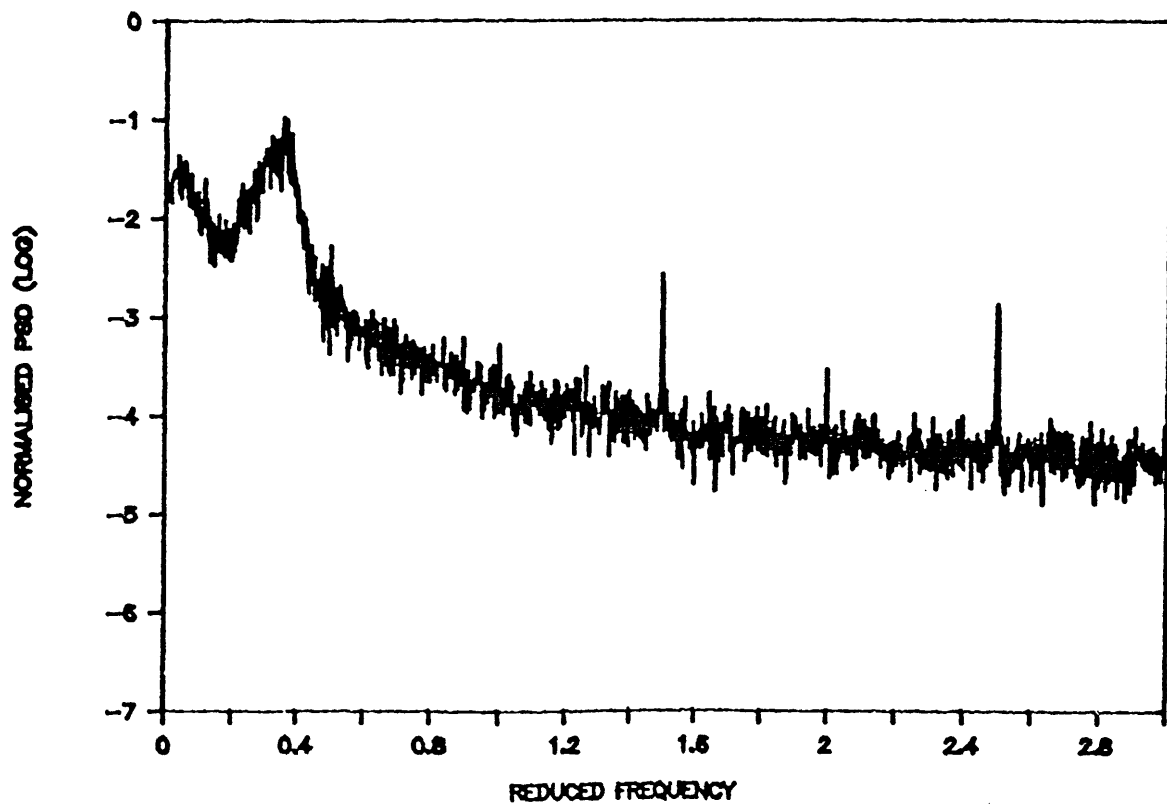


Fig 3.51b Power spectrum of C_{py} at $\alpha = -14^\circ$

Fig 4.29

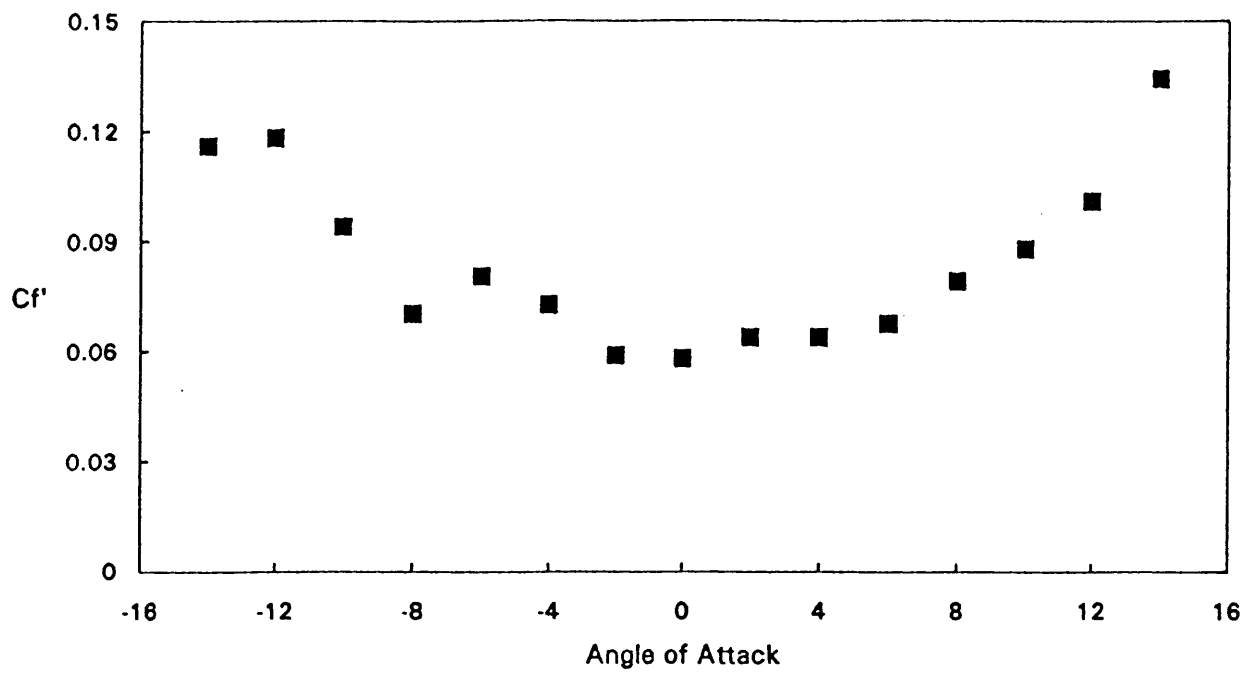


Fig 3.52 Variation of C_F' with α

PSD907

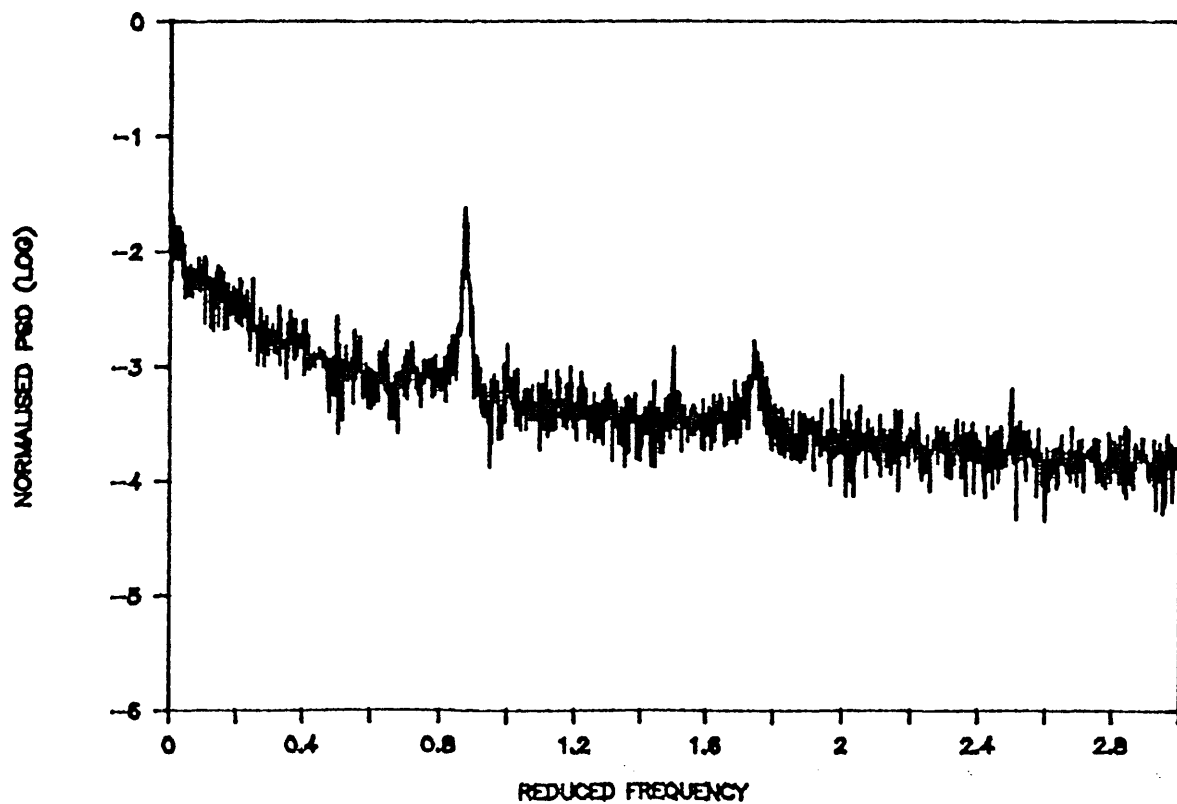


Fig3.53 Power spectrum of C_f at $\alpha = 0^\circ$

ANGLE 1780

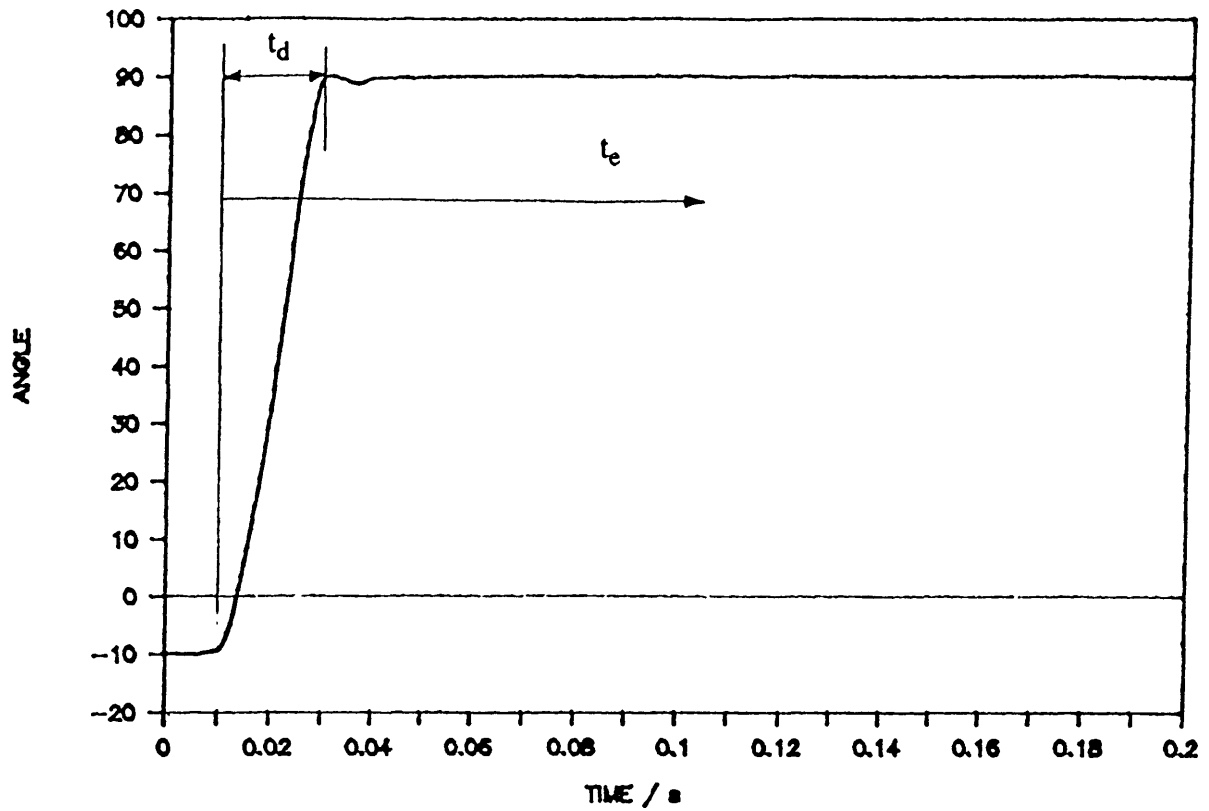


Fig 4.1a Typical time histories of δ for spoiler extension

CL1780

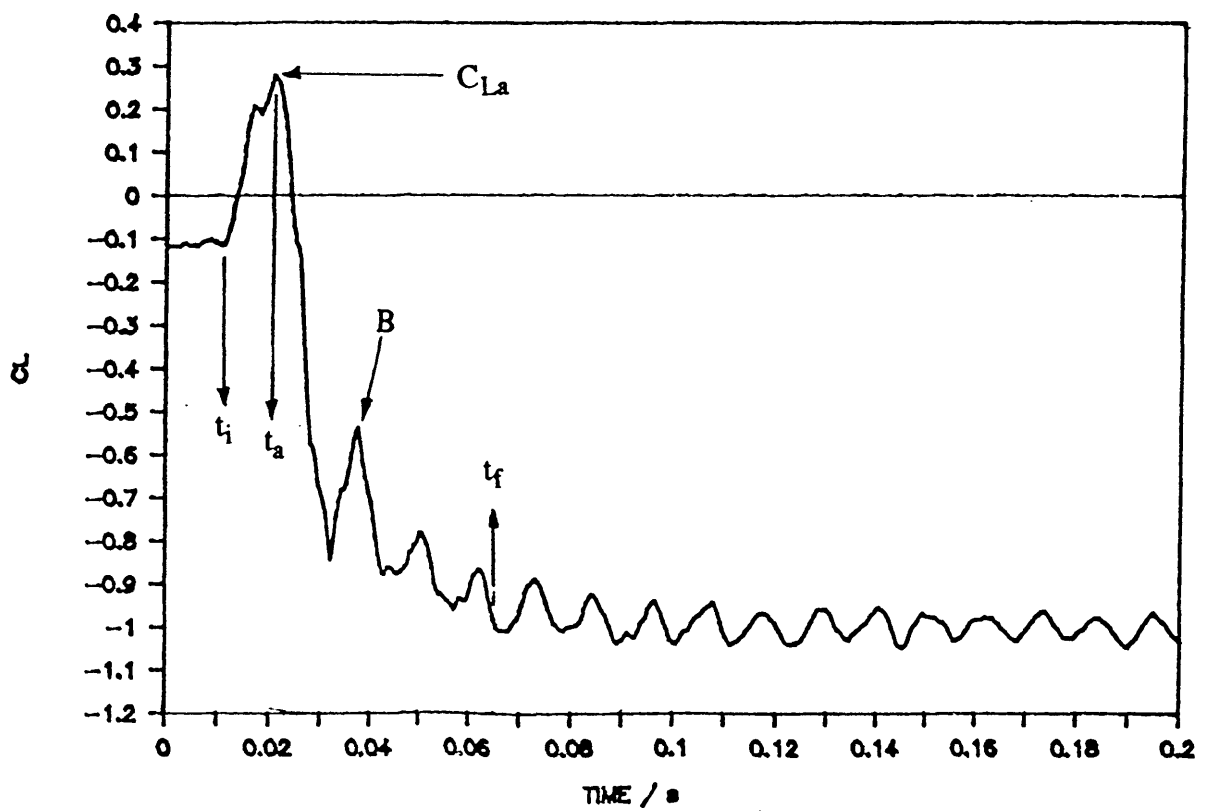


Fig 4.1b Typical time histories of C_L

CL1860

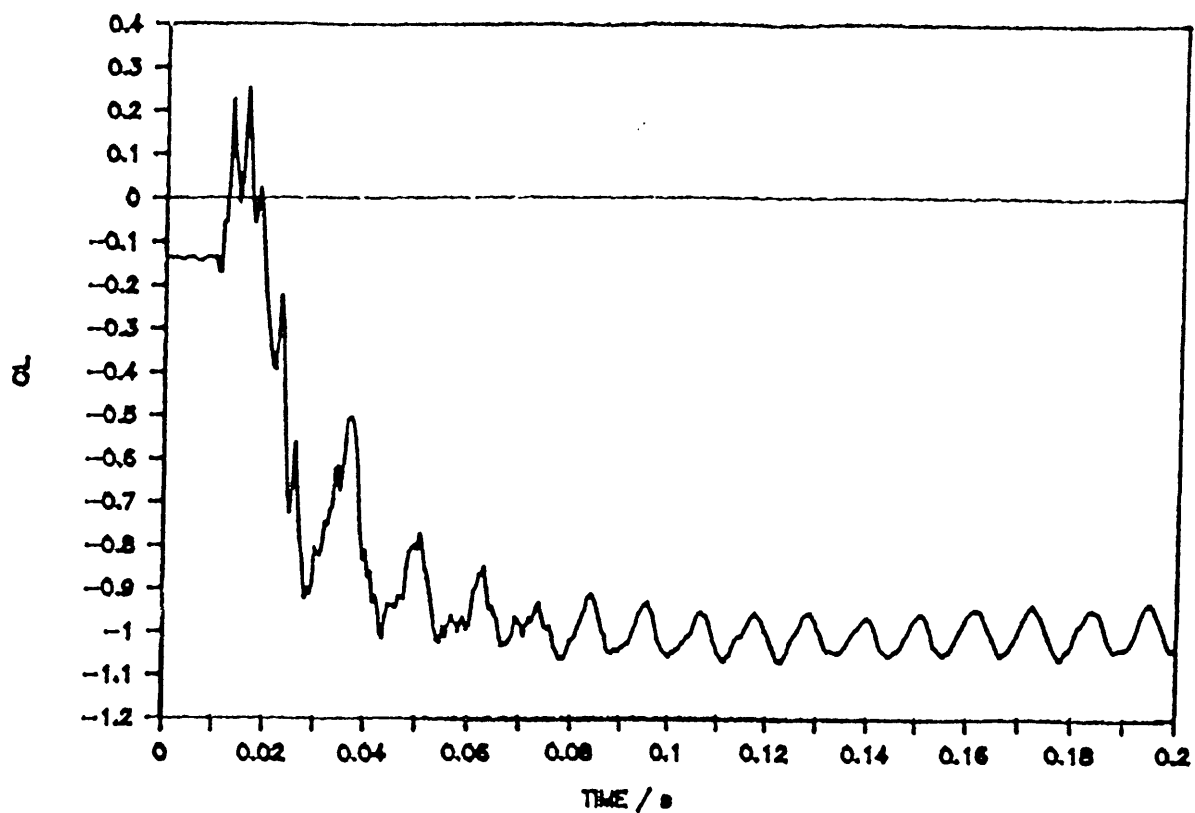


Fig 4.2a Time histories of C_L for $T = 1.1$, $\delta_{ss} = 90^\circ$

CL1130

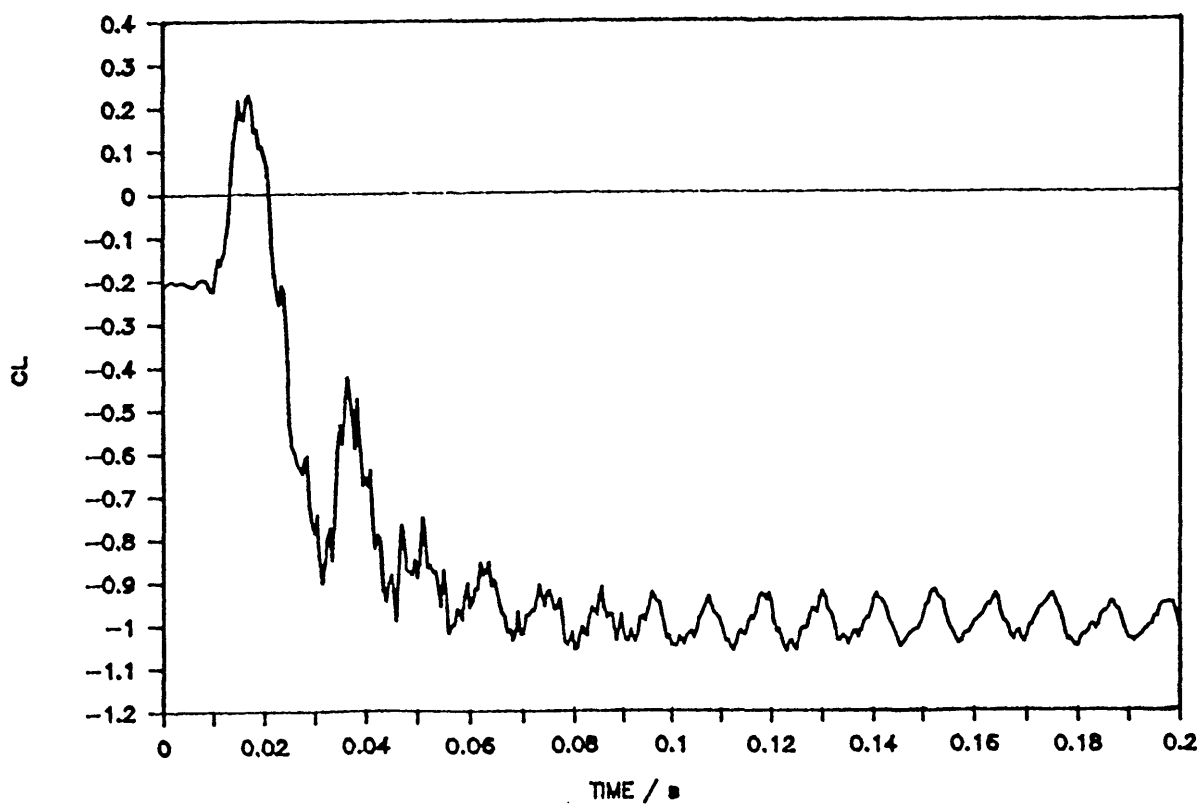


Fig 4.2b Time histories of C_L for $T = 1.5$, $\delta_{ss} = 90^\circ$

CL1780

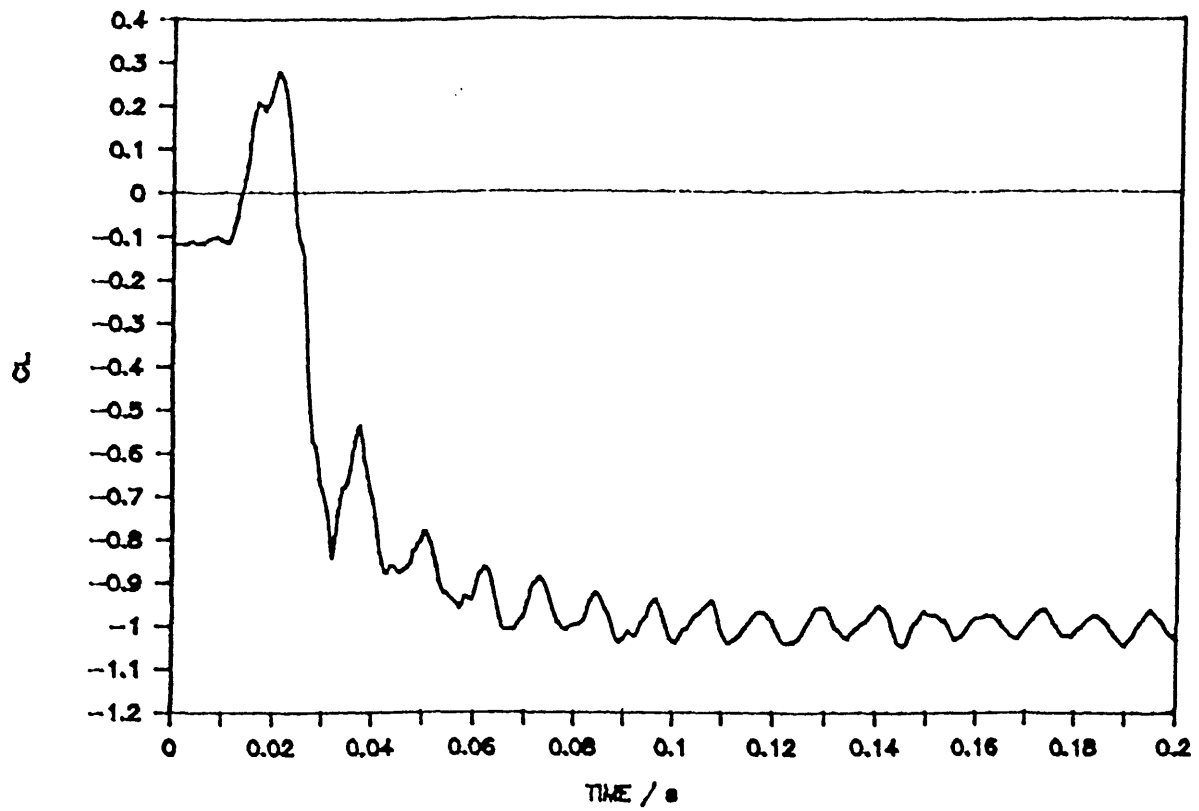


Fig 4.2c Time histories of C_L for $T = 2$, $\delta_{ss} = 90^\circ$

CL1790

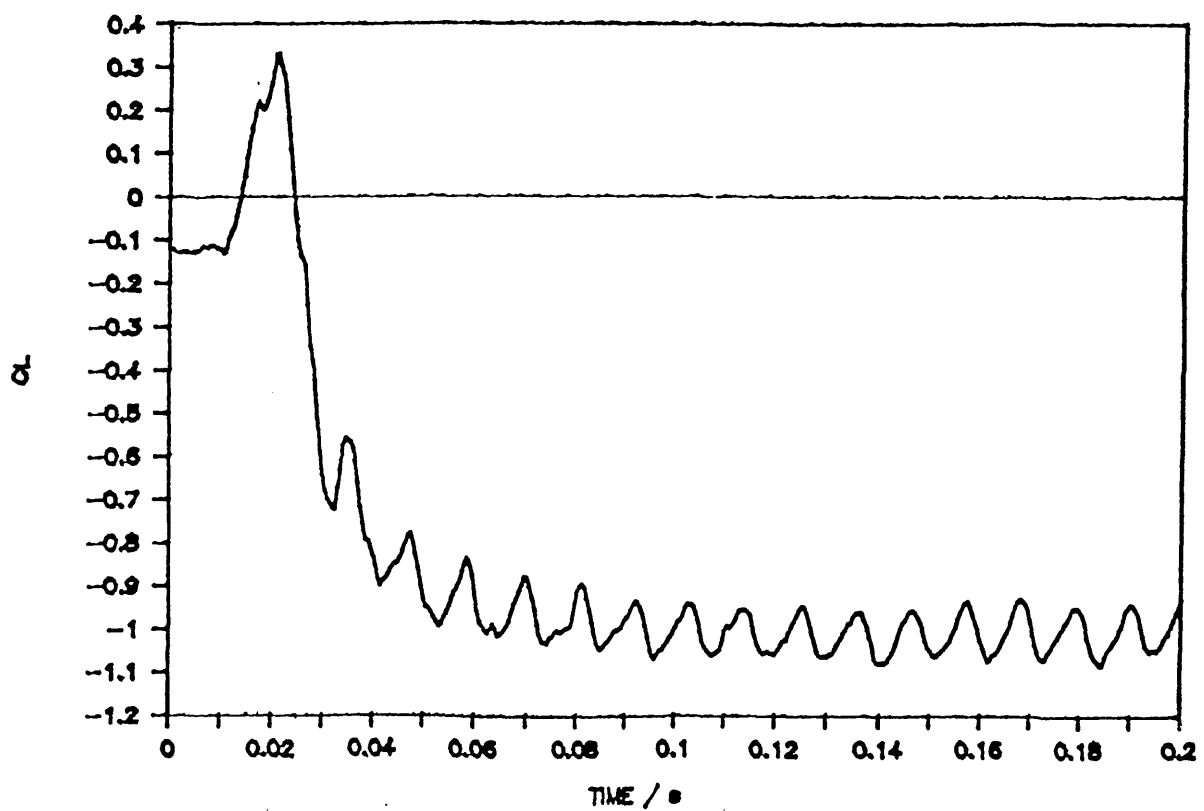


Fig 4.2d Time histories of C_L for $T = 2.3$, $\delta_{ss} = 90^\circ$

CL1760

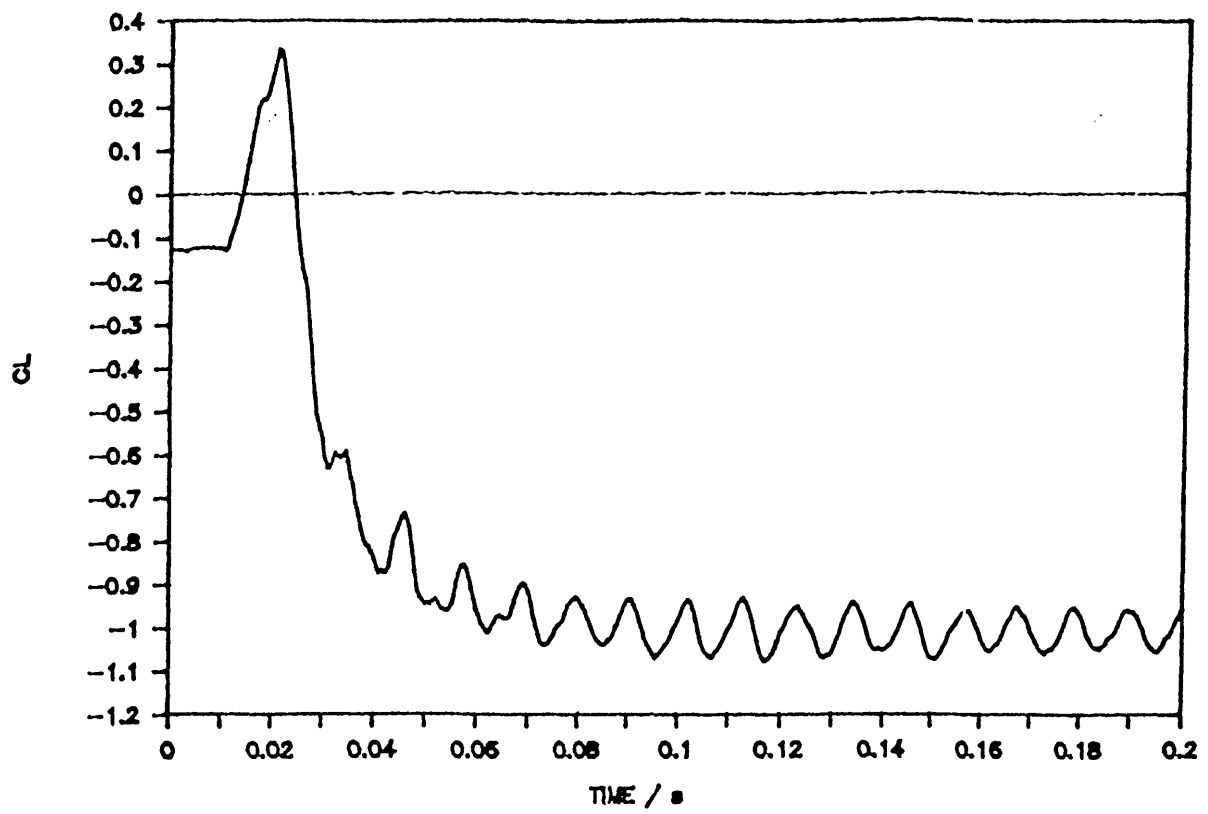


Fig 4.2e Time histories of C_L for $T = 2.6$, $\delta_{ss} = 90^\circ$

CL1800

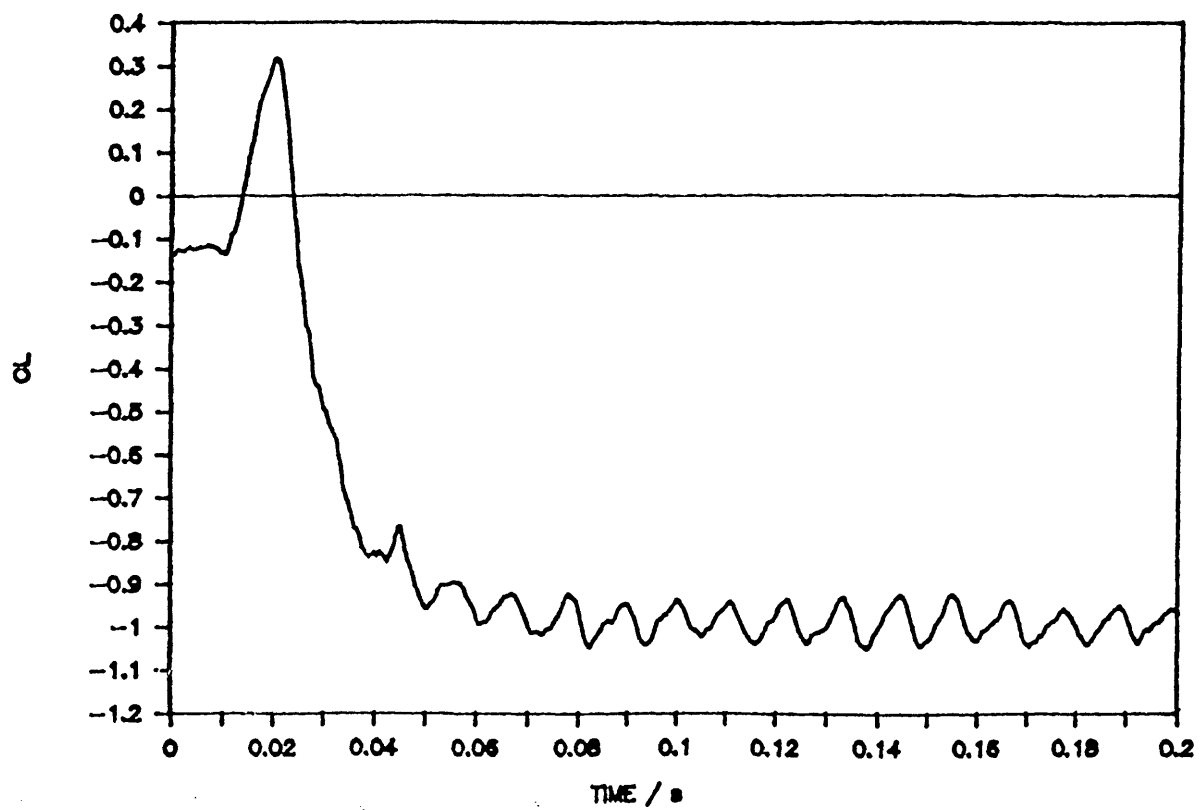


Fig 4.2f Time histories of C_L for $T = 2.8$, $\delta_{ss} = 90^\circ$

CL1810

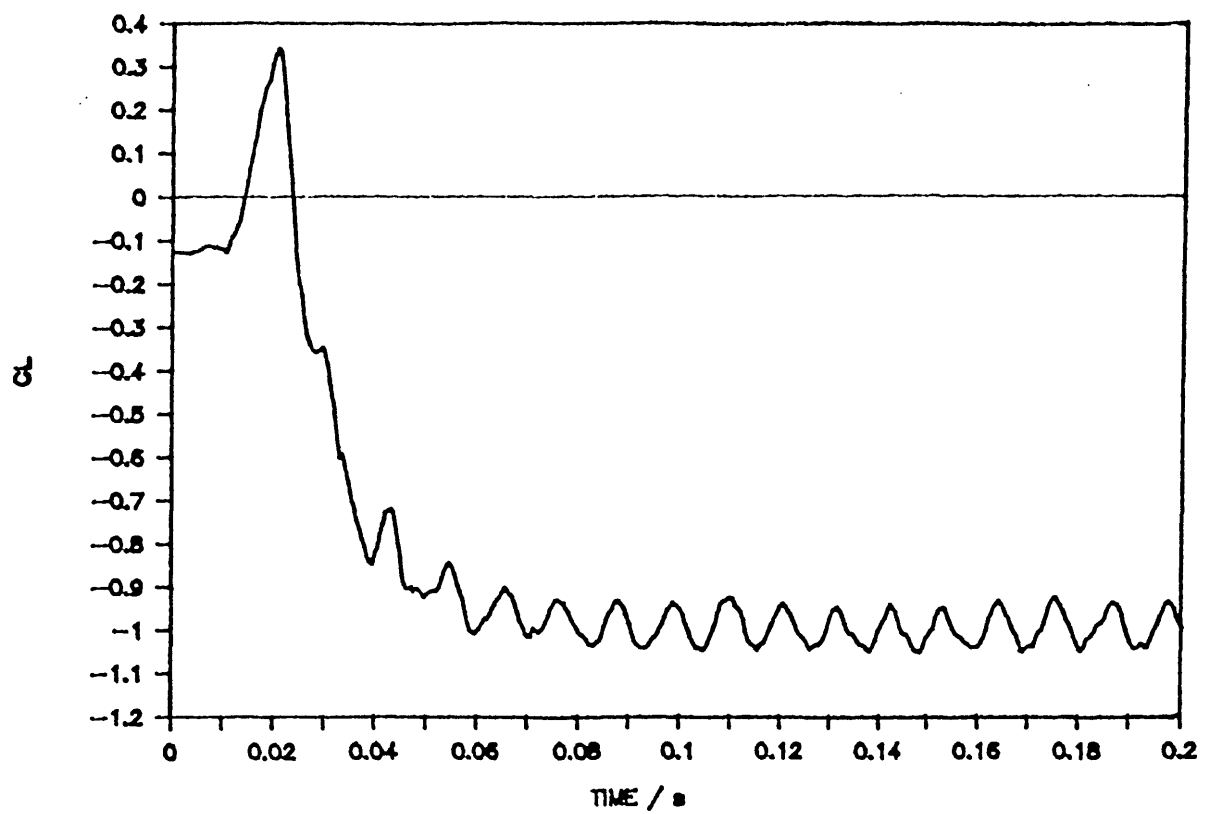


Fig 4.2g Time histories of C_L for $T = 3$, $\delta_{ss} = 90^\circ$

CL1770

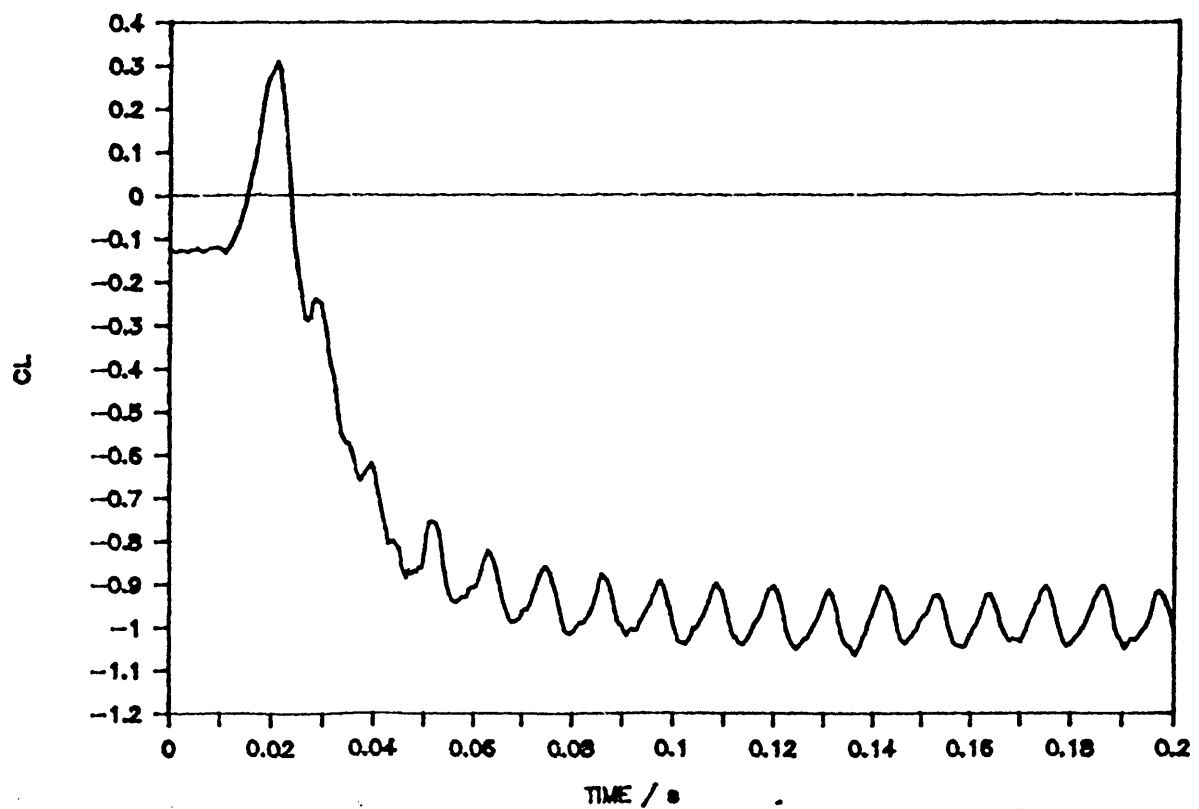


Fig 4.2h Time histories of C_L for $T = 3.3$, $\delta_{ss} = 90^\circ$

CL1820

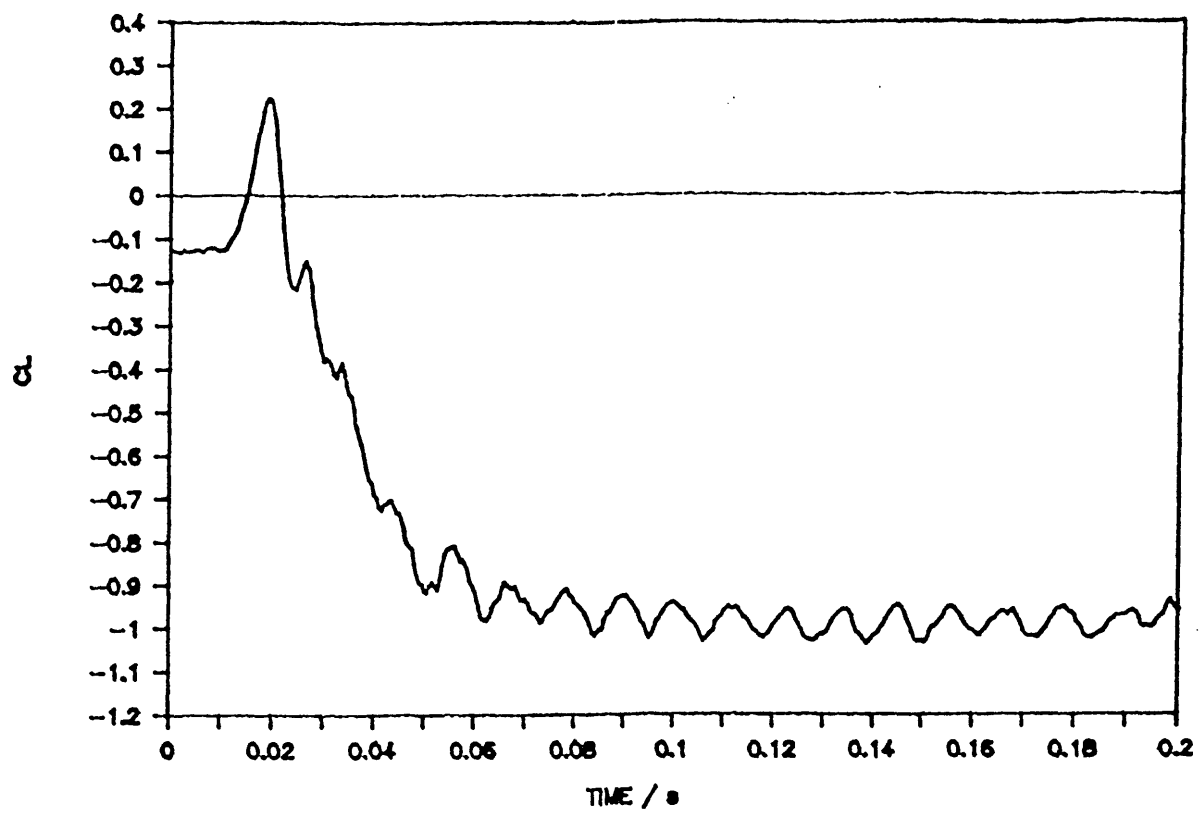


Fig 4.2i Time histories of C_L for $T = 4$, $\delta_{ss} = 90^\circ$

CL1850

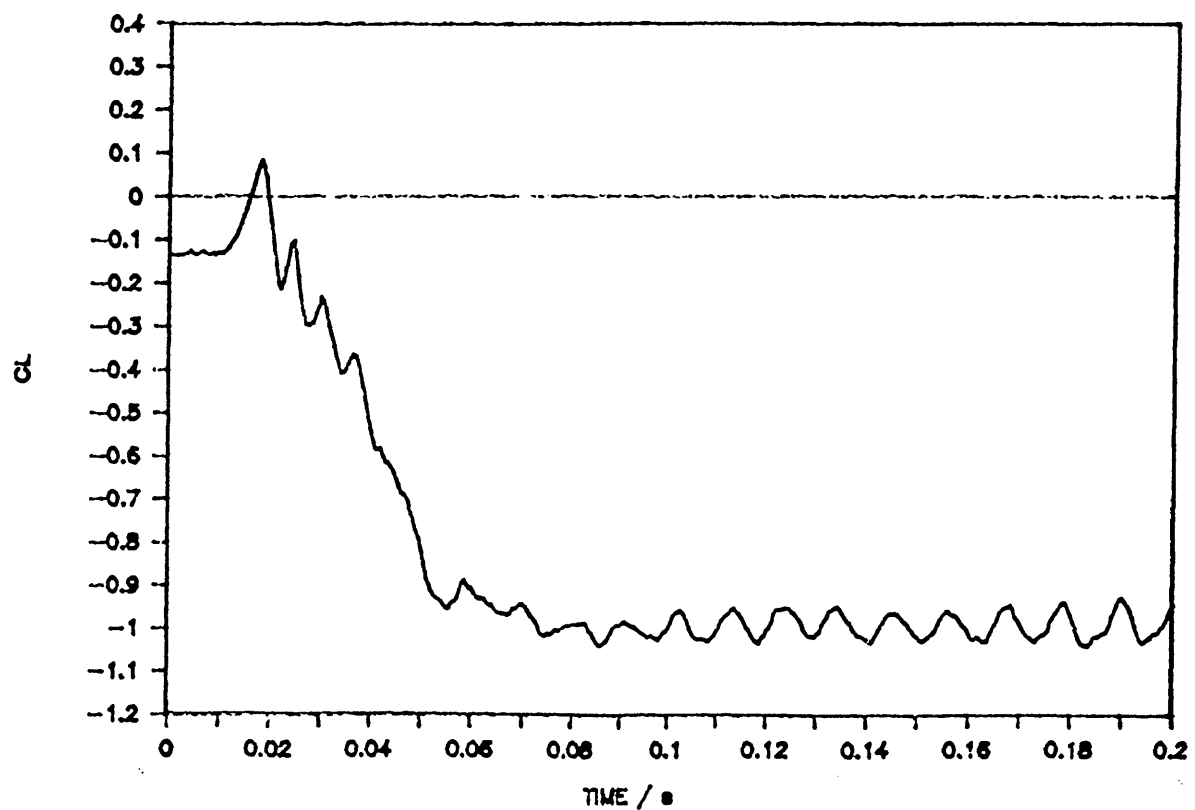


Fig 4.2j Time histories of C_L for $T = 4.6$, $\delta_{ss} = 90^\circ$

CL1830

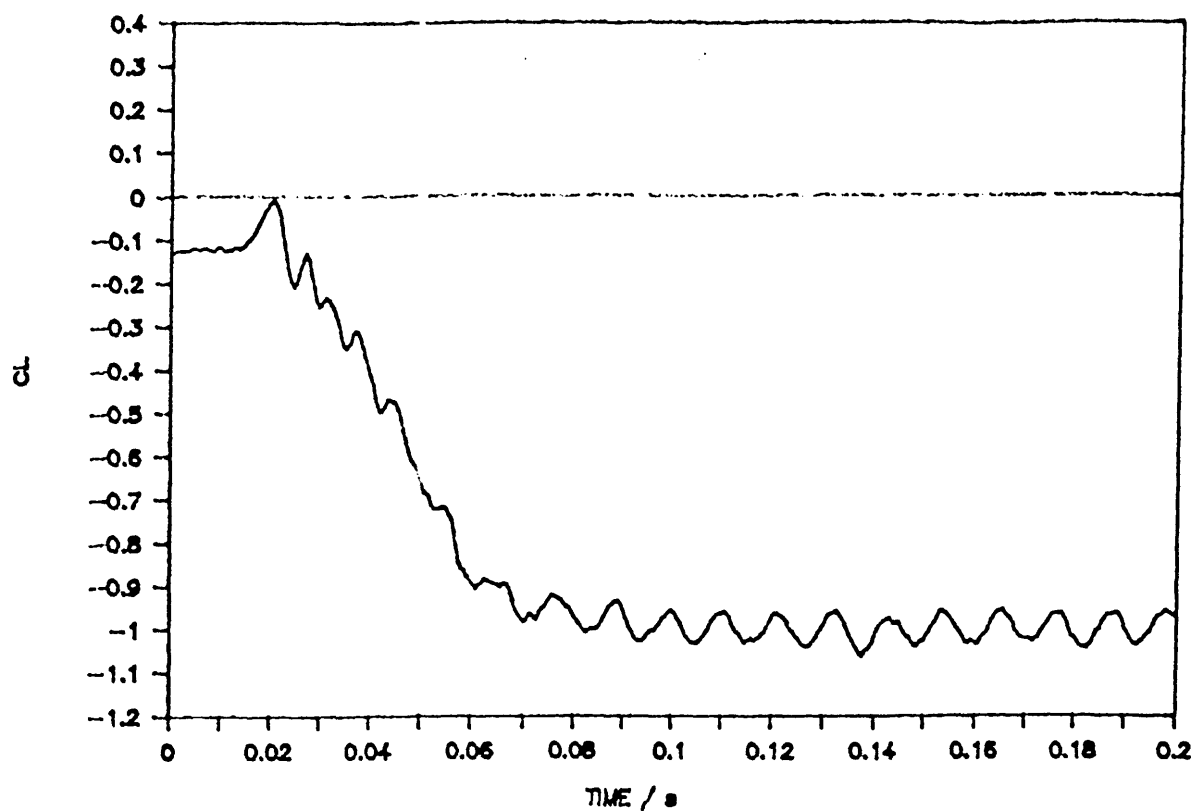


Fig 4.2k Time histories of C_L for $T = 5.5$, $\delta_{ss} = 90^\circ$

CL1840

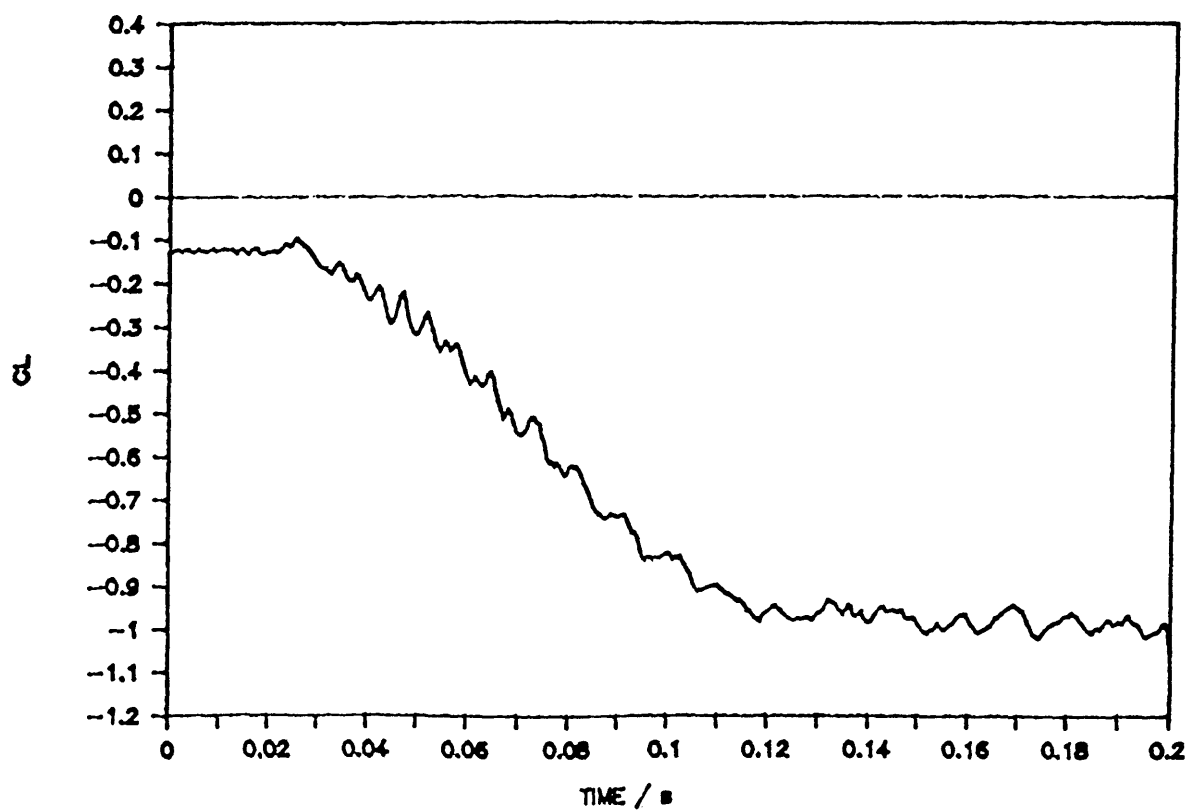


Fig 4.2l Time histories of C_L for $T = 11$, $\delta_{ss} = 90^\circ$

CL1910

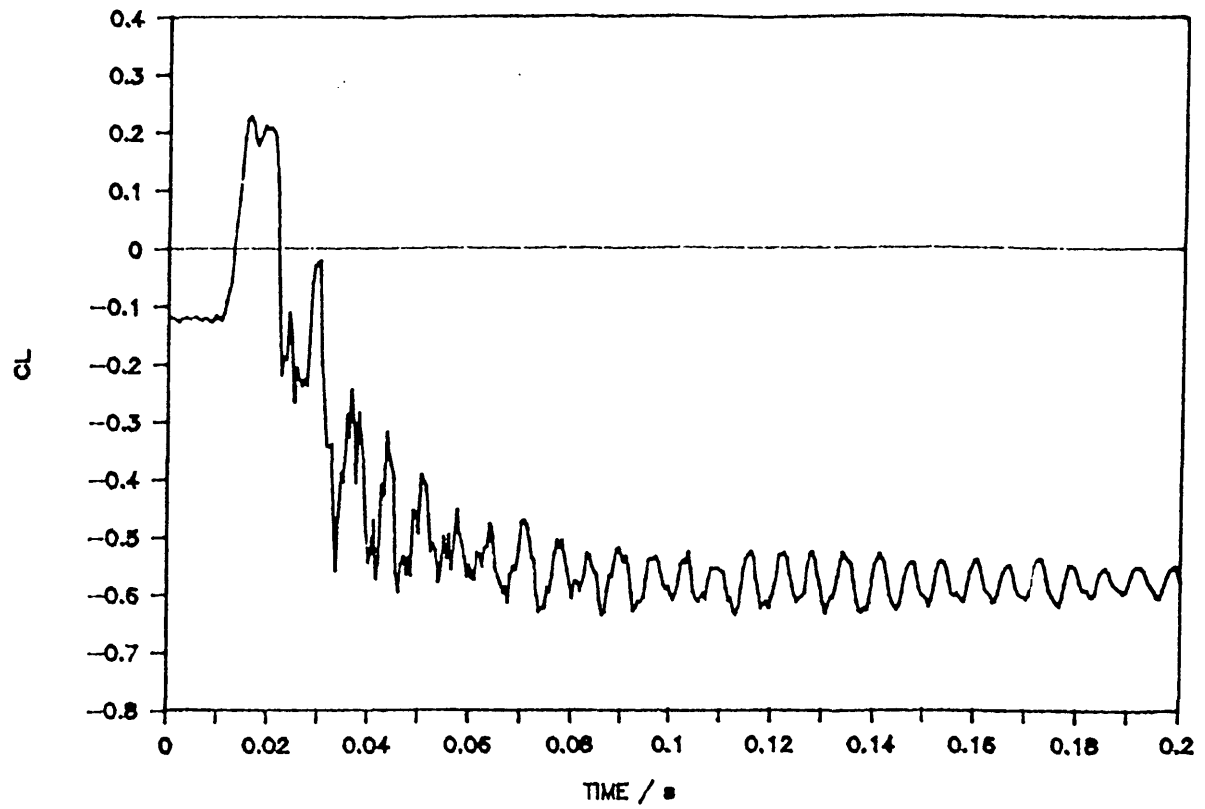


Fig 4.3a Time histories of C_L for $T = 1$, $\delta_{ss} = 30^\circ$

CL1900

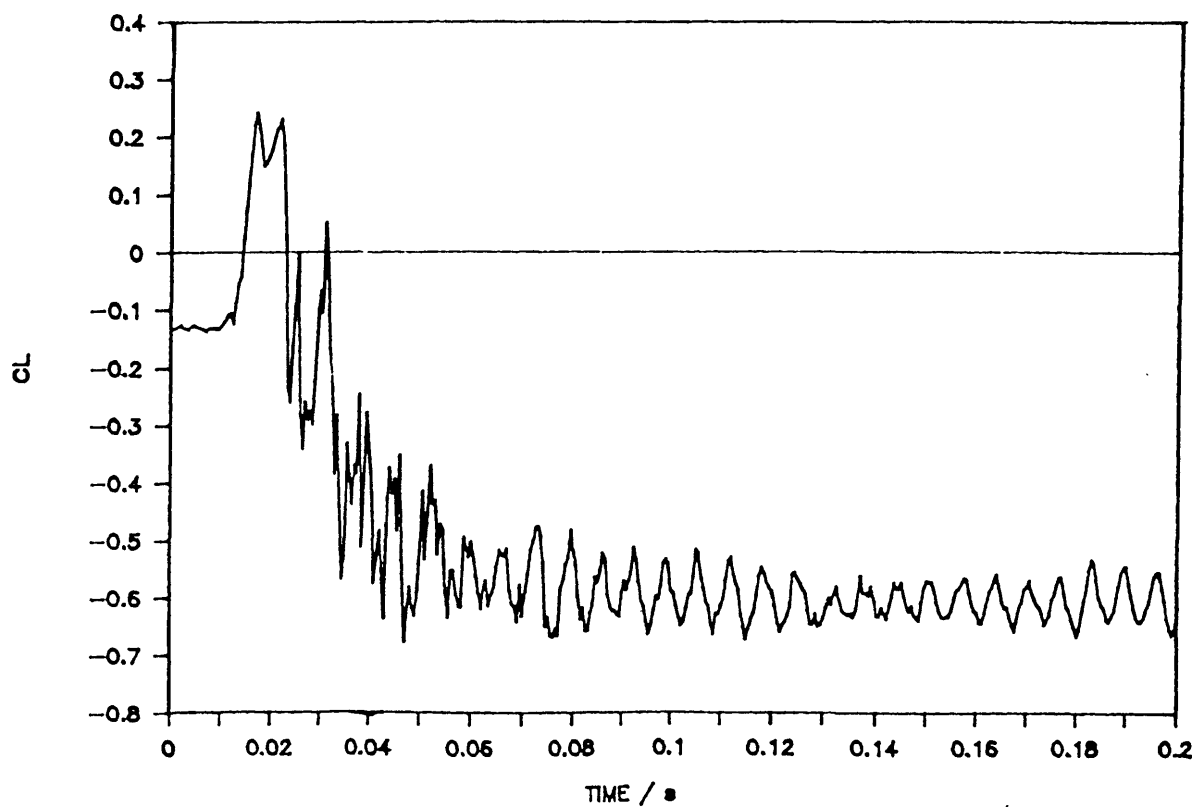


Fig 4.3b Time histories of C_L for $T = 1.1$, $\delta_{ss} = 30^\circ$

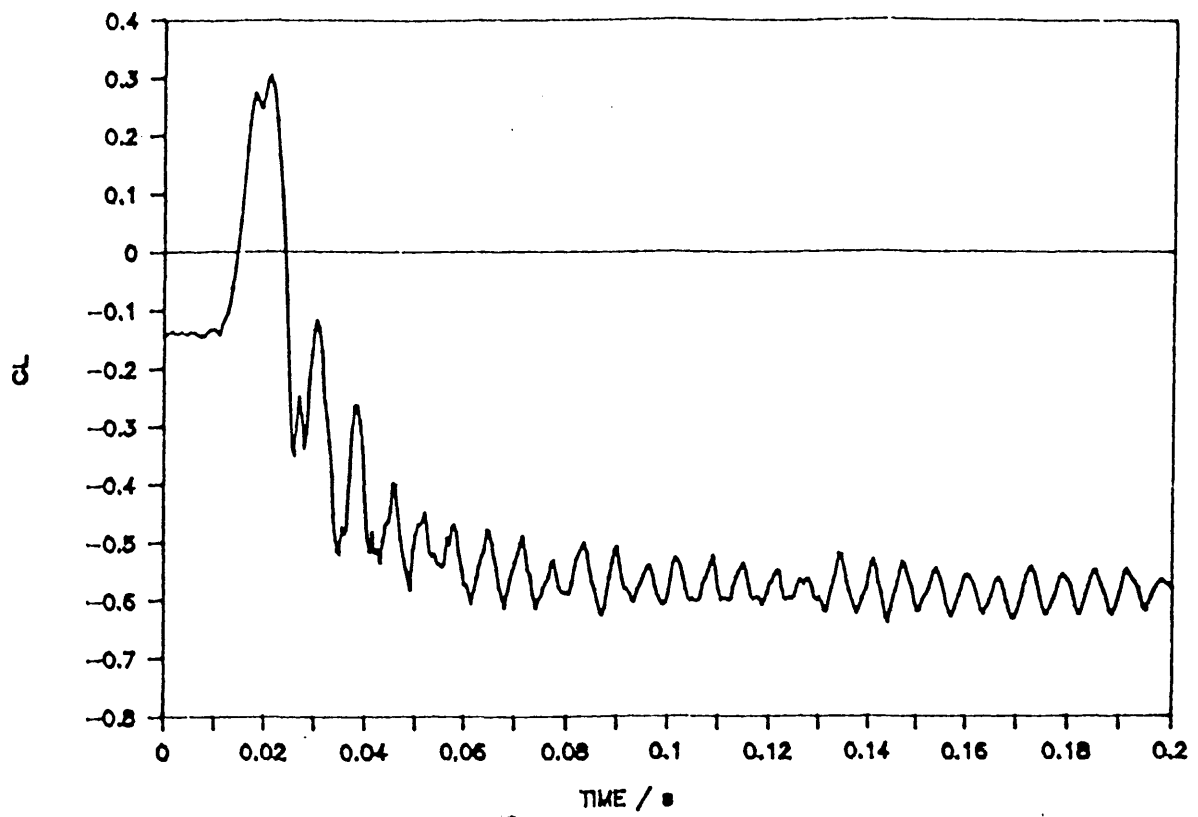


Fig 4.3c Time histories of C_L for $T = 1.35$, $\delta_{ss} = 30^\circ$

CL1930

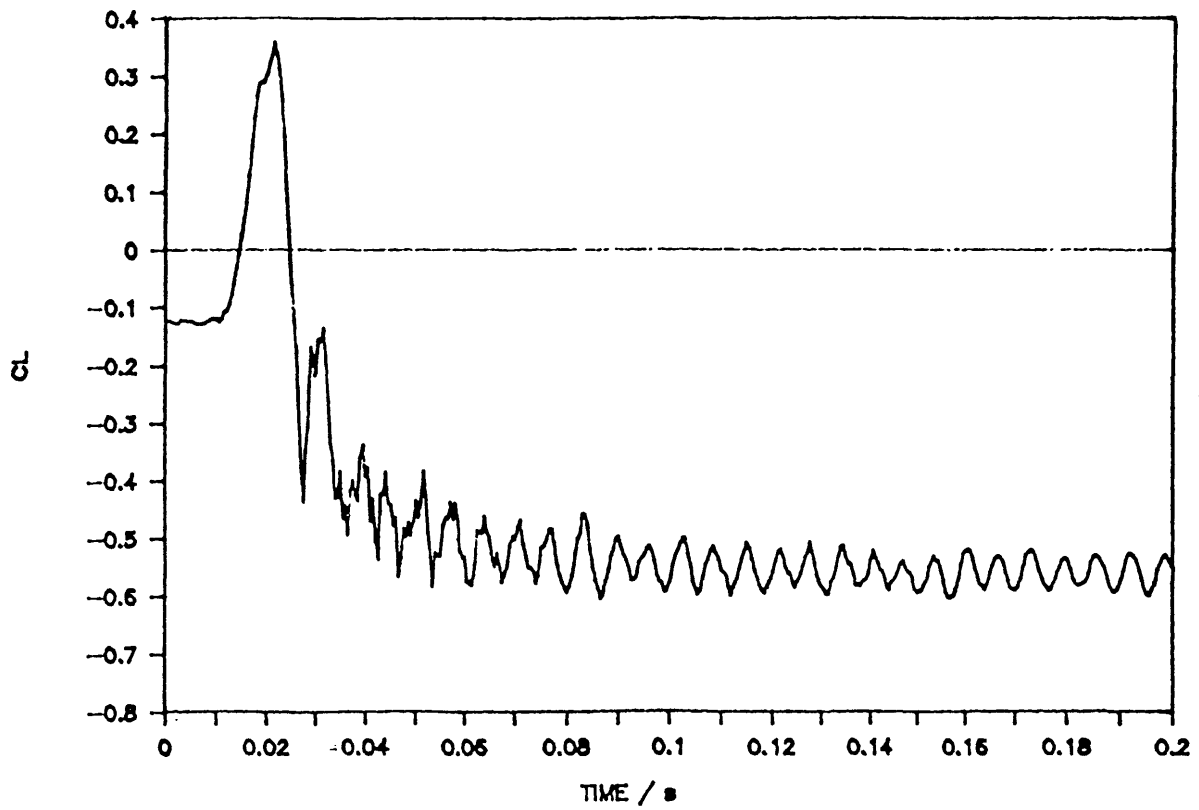


Fig 4.3d Time histories of C_L for $T = 1.5$, $\delta_{ss} = 30^\circ$

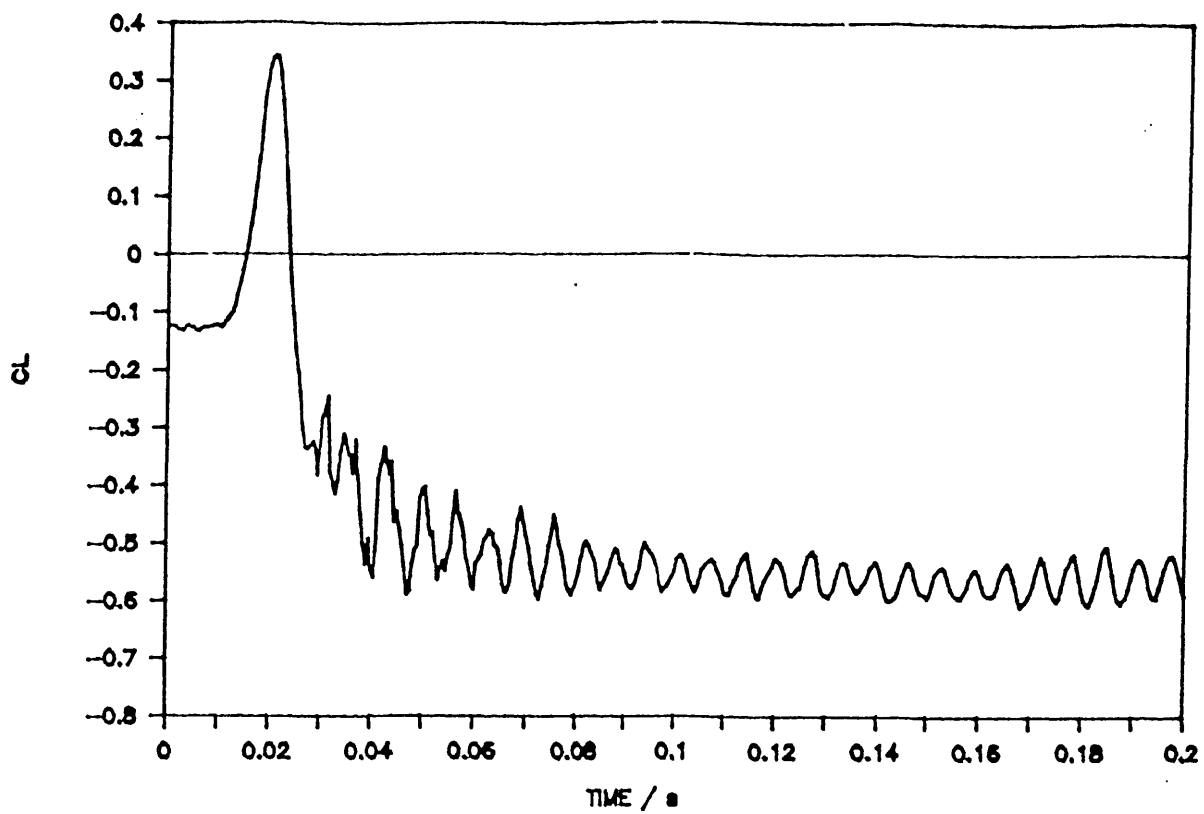


Fig 4.3e Time histories of C_L for $T = 1.75$, $\delta_{ss} = 30^\circ$

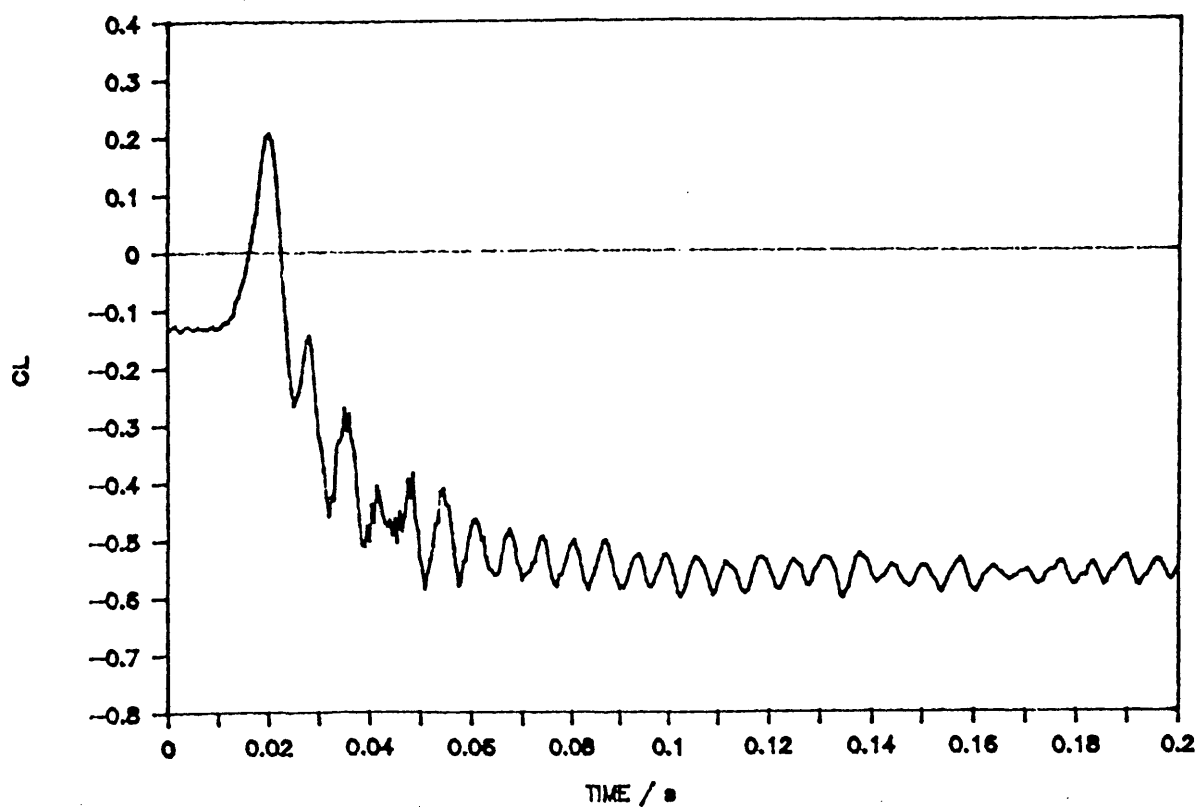


Fig 4.3f Time histories of C_L for $T = 2$, $\delta_{ss} = 30^\circ$

CL1950

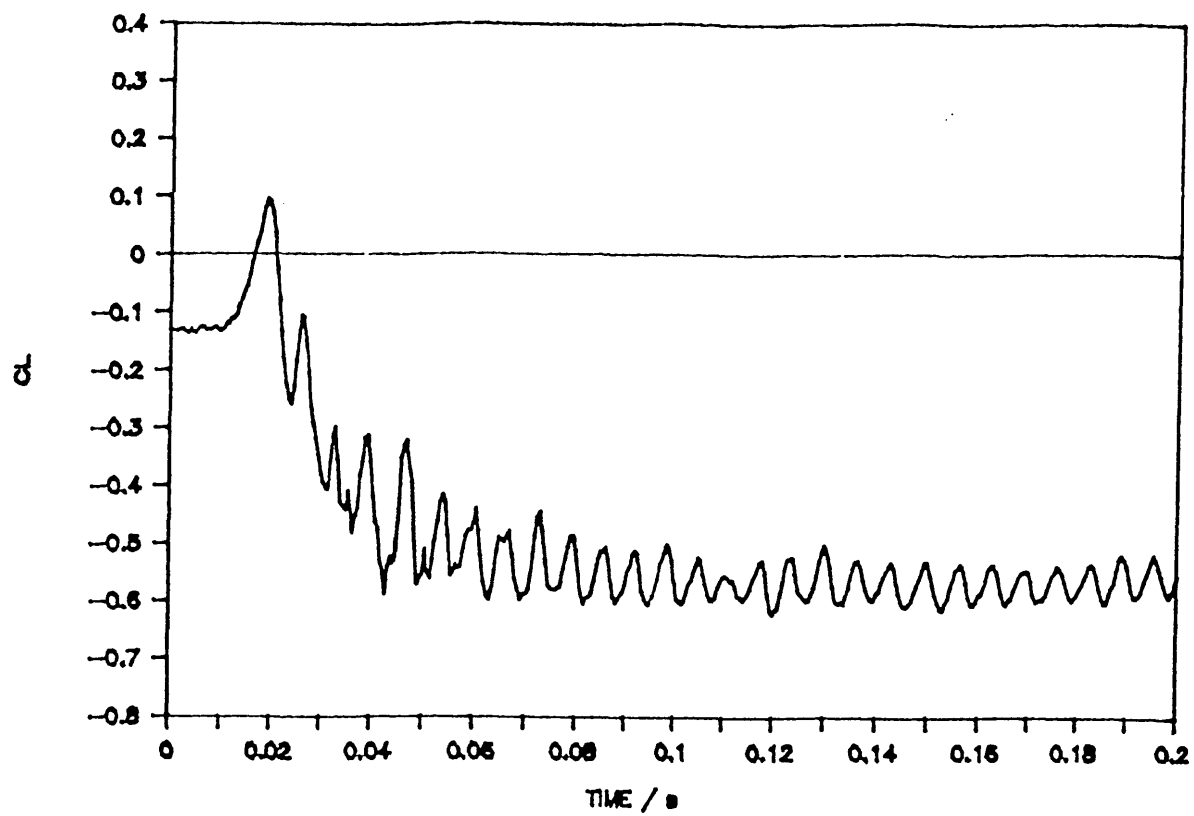


Fig 4.3g Time histories of C_L for $T = 2.2$, $\delta_{ss} = 30^\circ$

CL1960

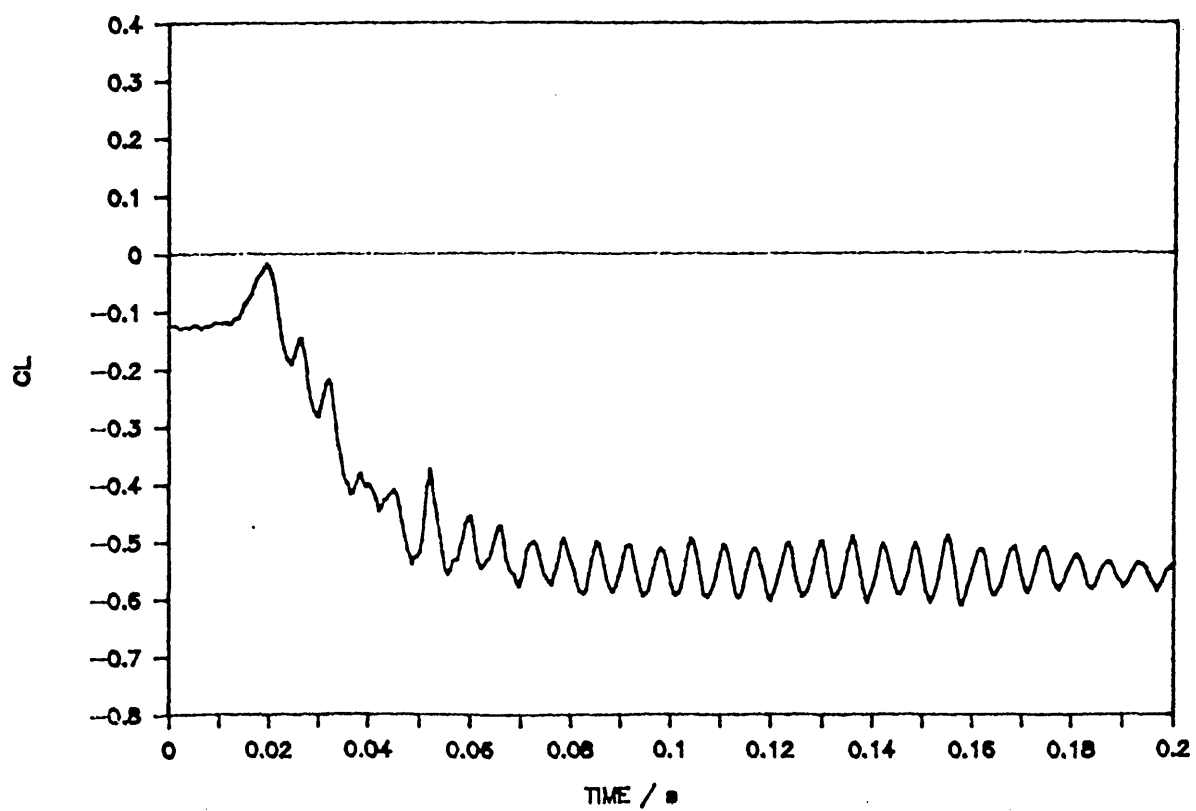


Fig 4.3h Time histories of C_L for $T = 2.75$, $\delta_{ss} = 30^\circ$

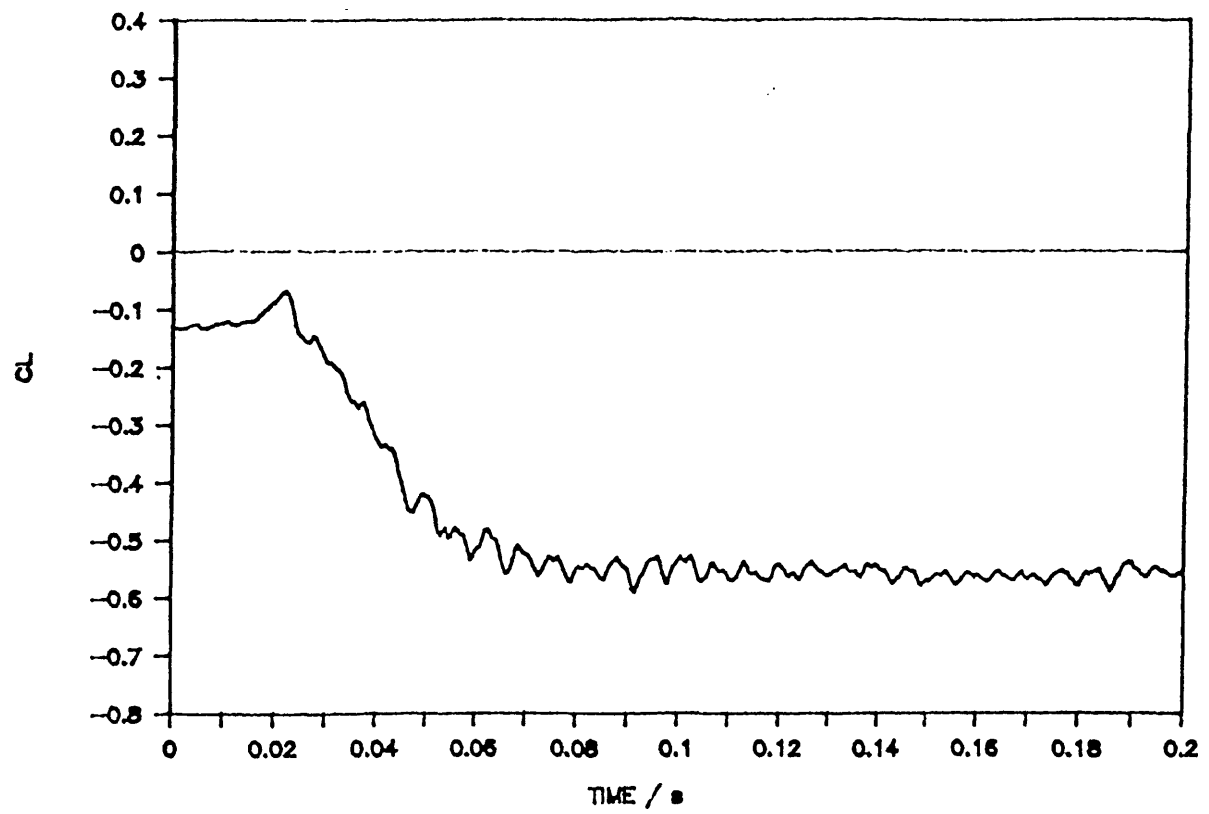


Fig 4.3i Time histories of C_L for $T = 3.8$, $\delta_{ss} = 30^\circ$

ANGLE 1180

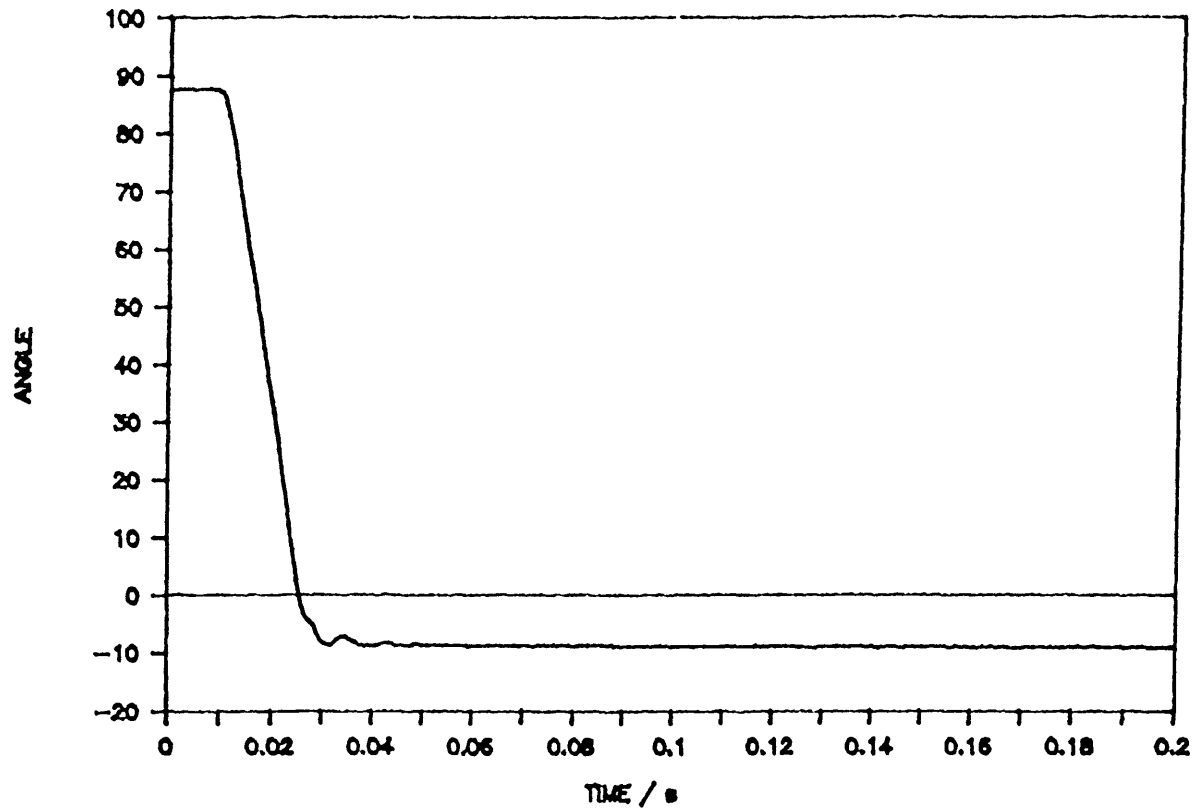


Fig 4.4a Typical time histories of δ for spoiler retraction

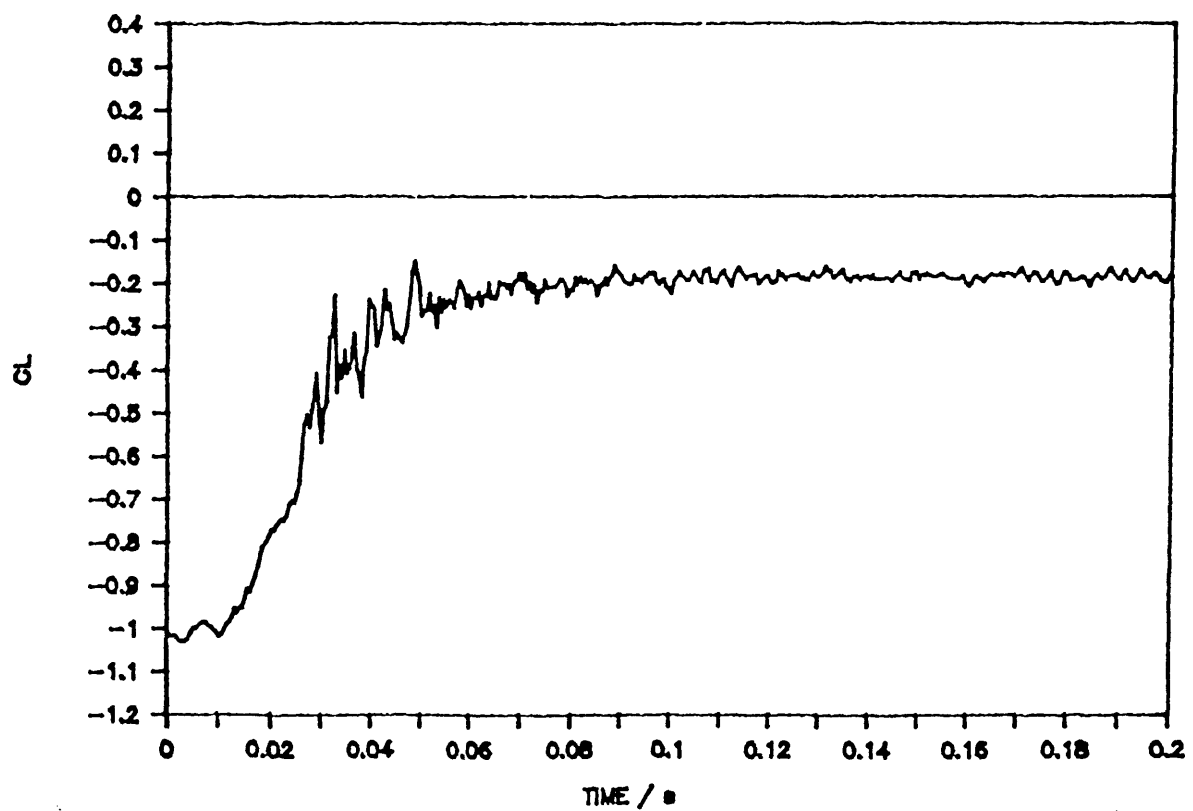


Fig 4.4b Typical time histories of C_L for spoiler retraction

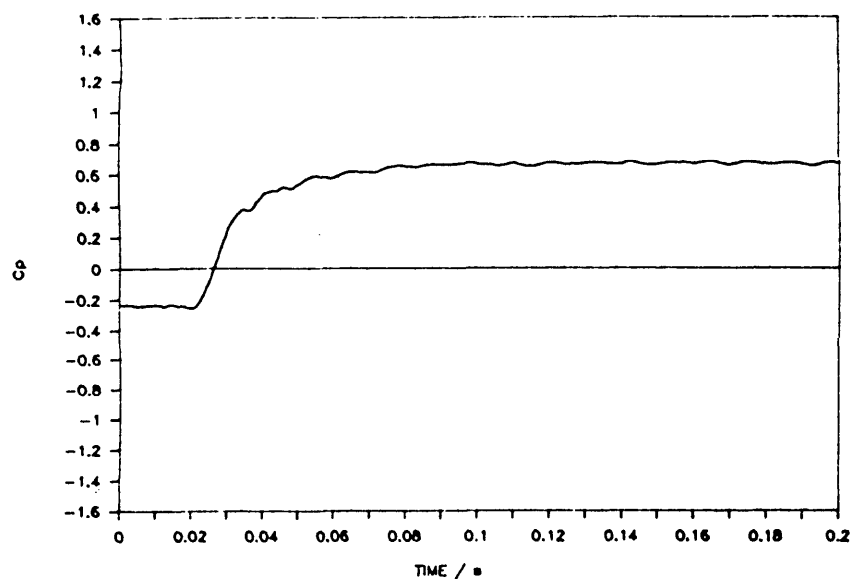


Fig 4.5a Time histories of C_{p1} , $x/c = 0.03$, $T = 2.3$, $\delta_{ss} = 90^\circ$

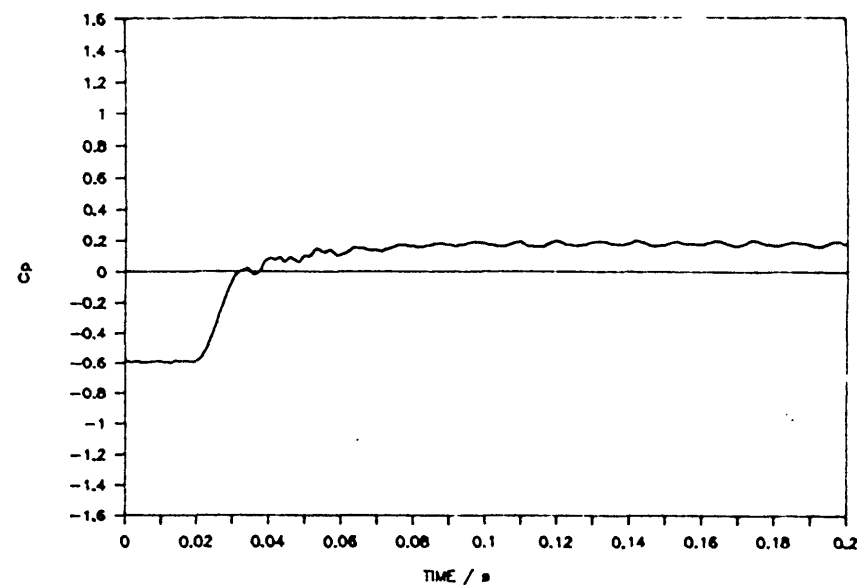


Fig 4.5c Time histories of C_{p3} , $x/c = 0.16$, $T = 2.3$, $\delta_{ss} = 90^\circ$

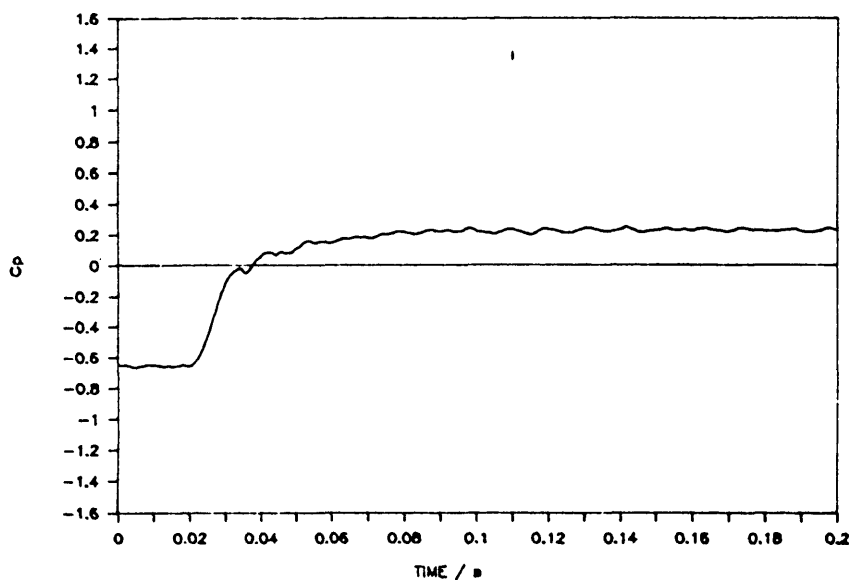


Fig 4.5b Time histories of C_{p2} , $x/c = 0.09$, $T = 2.3$, $\delta_{ss} = 90^\circ$

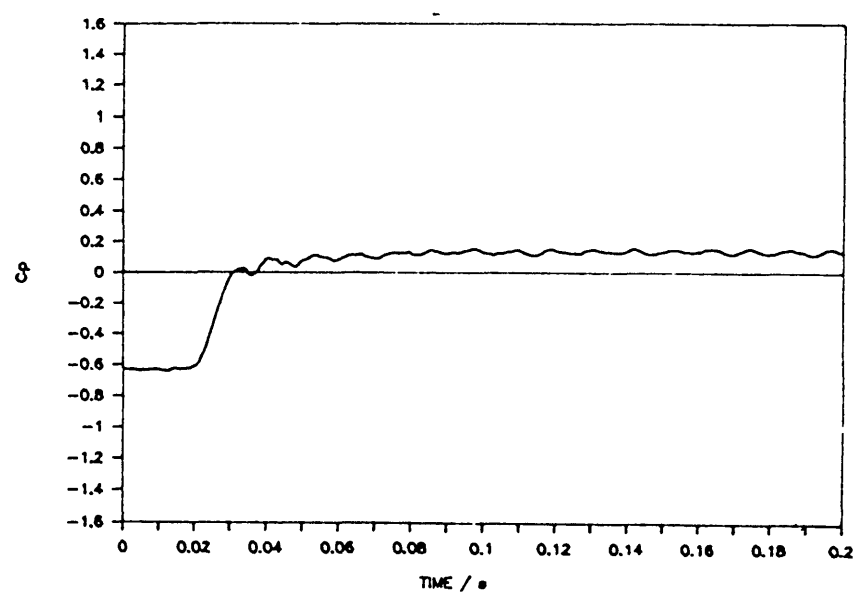


Fig 4.5d Time histories of C_{p4} , $x/c = 0.22$, $T = 2.3$, $\delta_{ss} = 90^\circ$

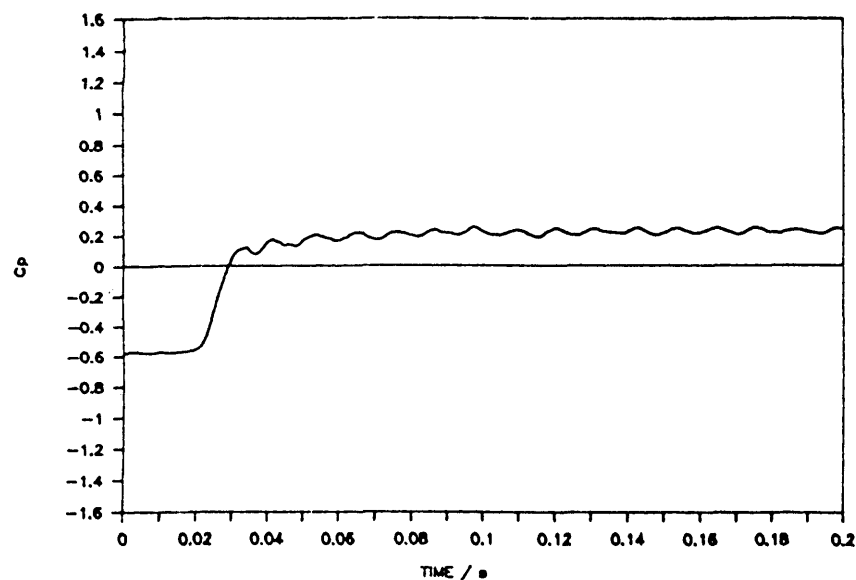


Fig 4.5e Time histories of C_{p5} , $x/c = 0.28$, $T = 2.3$, $\delta_{ss} = 90^\circ$

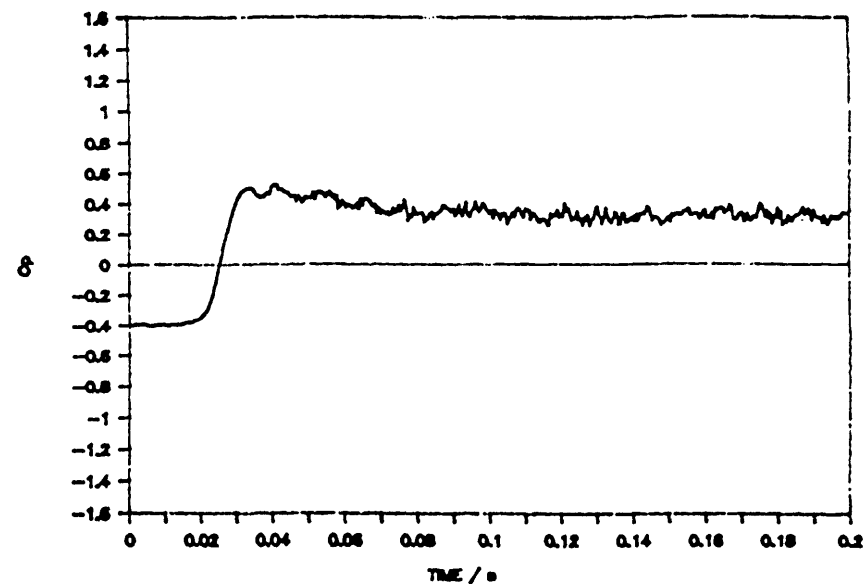


Fig 4.5g Time histories of C_{p7} , $x/c = 0.41$, $T = 2.3$, $\delta_{ss} = 90^\circ$

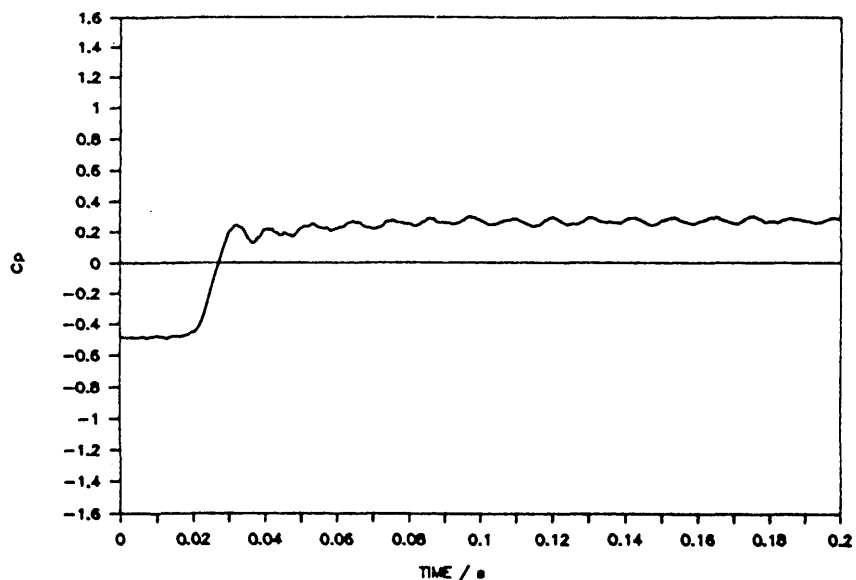


Fig 4.5f Time histories of C_{p6} , $x/c = 0.34$, $T = 2.3$, $\delta_{ss} = 90^\circ$

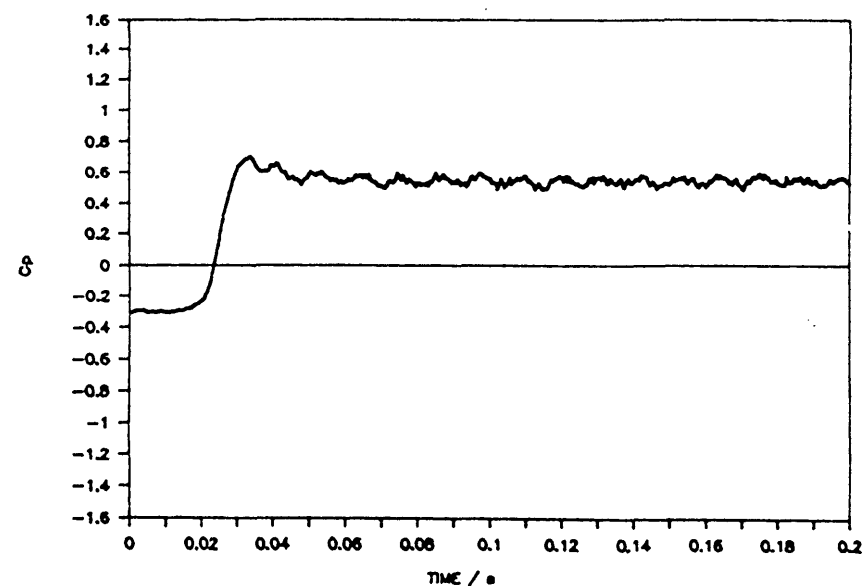


Fig 4.5h Time histories of C_{p8} , $x/c = 0.47$, $T = 2.3$, $\delta_{ss} = 90^\circ$

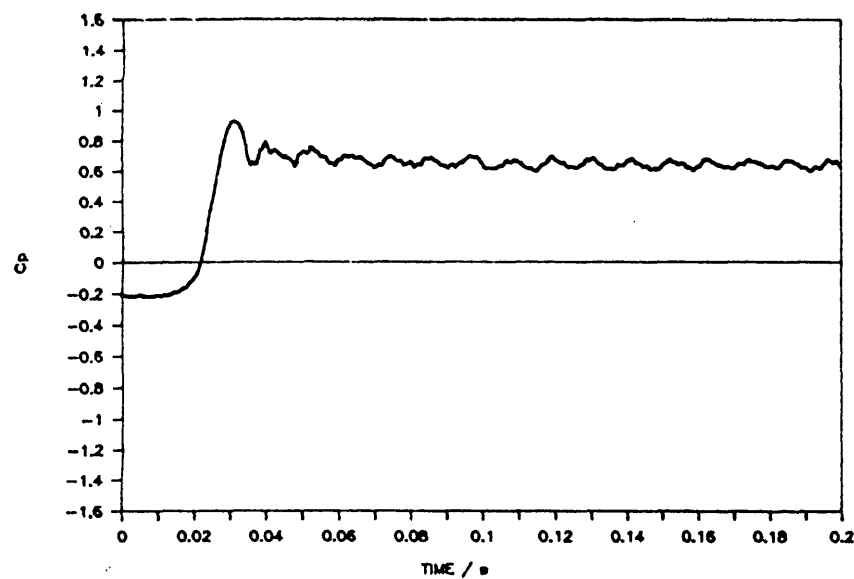


Fig 4.5i Time histories of C_{p9} , $x/c = 0.53$, $T = 2.3$, $\delta_{ss} = 90^\circ$

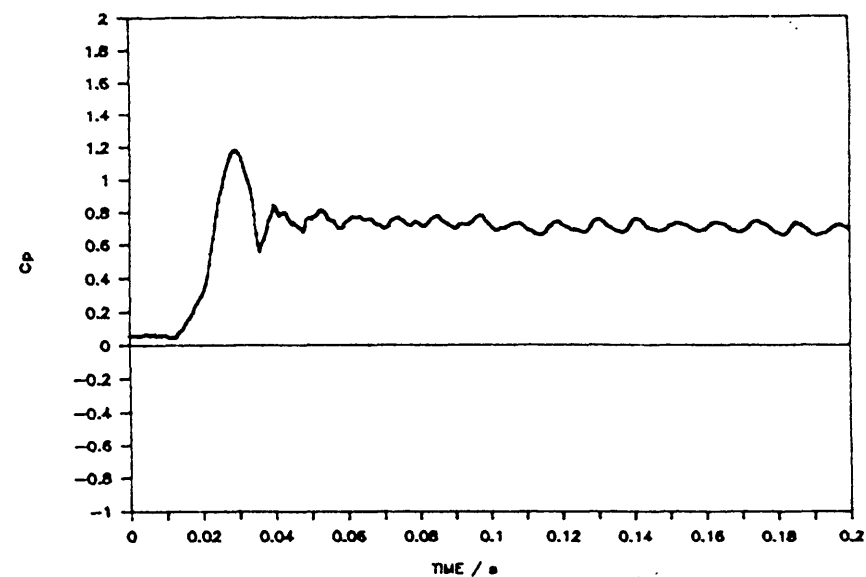


Fig 4.5k Time histories of C_{p11} , $x/c = 0.66$, $T = 2.3$, $\delta_{ss} = 90^\circ$

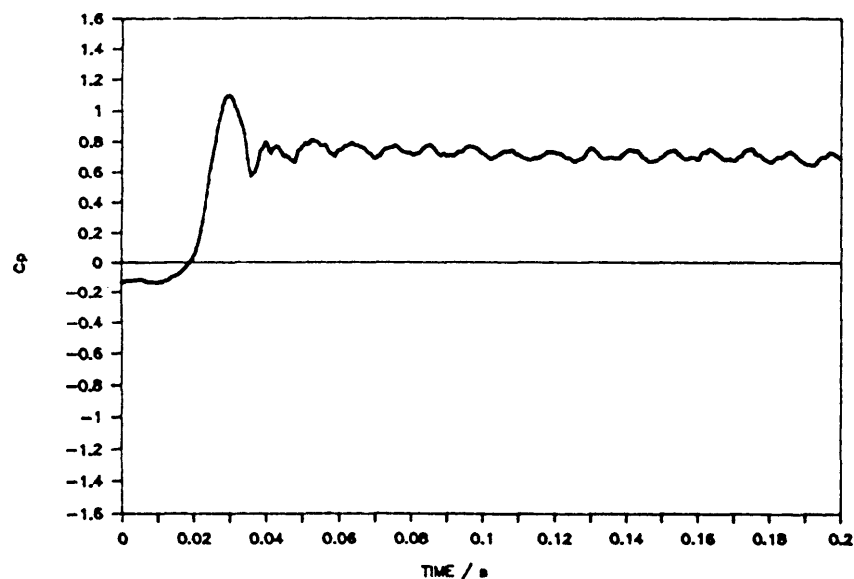


Fig 4.5j Time histories of C_{p10} , $x/c = 0.60$, $T = 2.3$, $\delta_{ss} = 90^\circ$

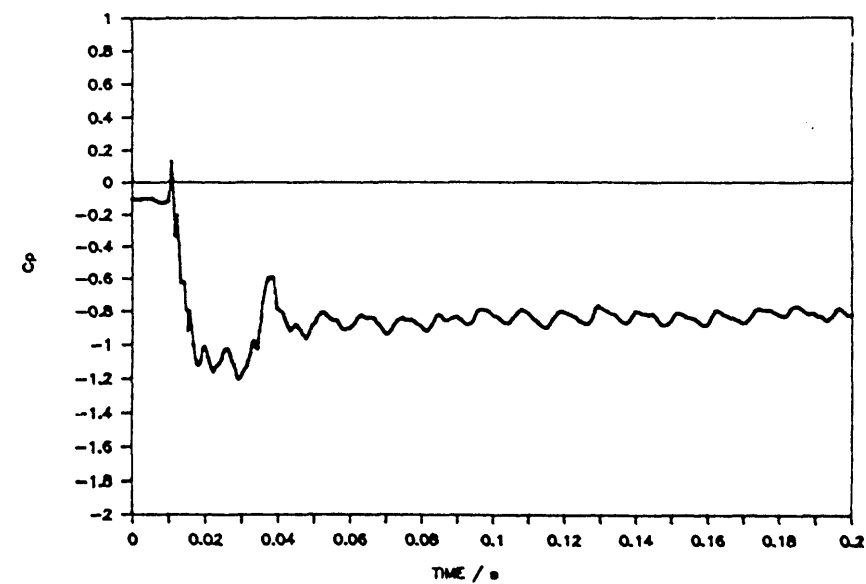


Fig 4.5l Time histories of C_{p12} , $x/c = 0.72$, $T = 2.3$, $\delta_{ss} = 90^\circ$

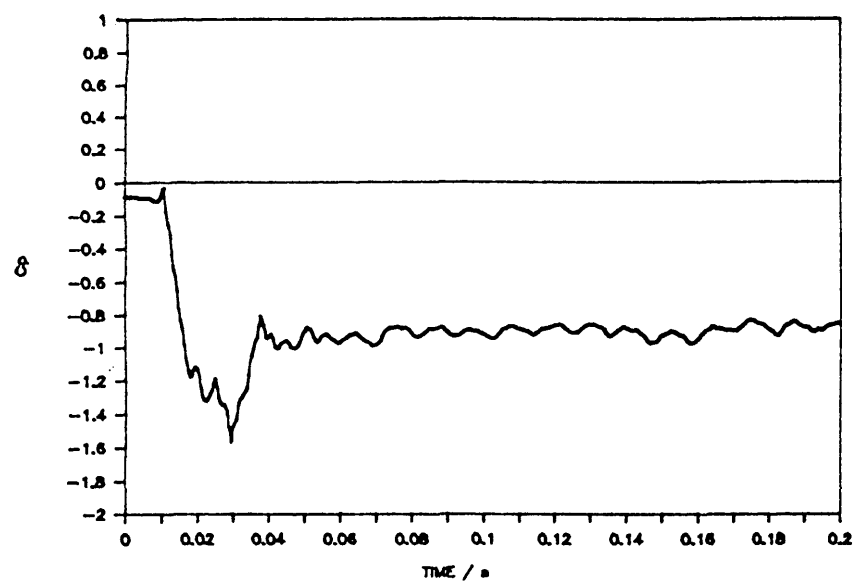


Fig 4.5m Time histories of C_{p13} , $x/c = 0.78$, $T = 2.3$, $\delta_{ss} = 90^\circ$

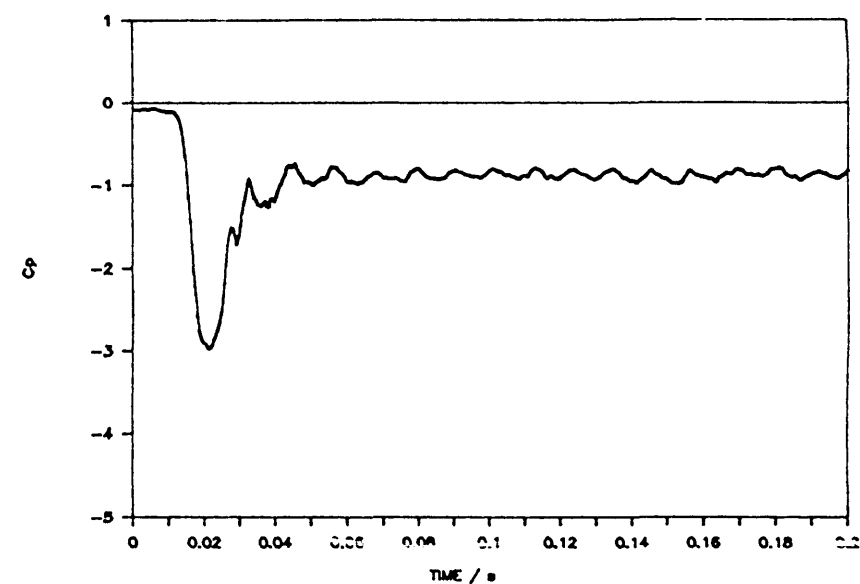


Fig 4.5o Time histories of C_{p15} , $x/c = 0.90$, $T = 2.3$, $\delta_{ss} = 90^\circ$

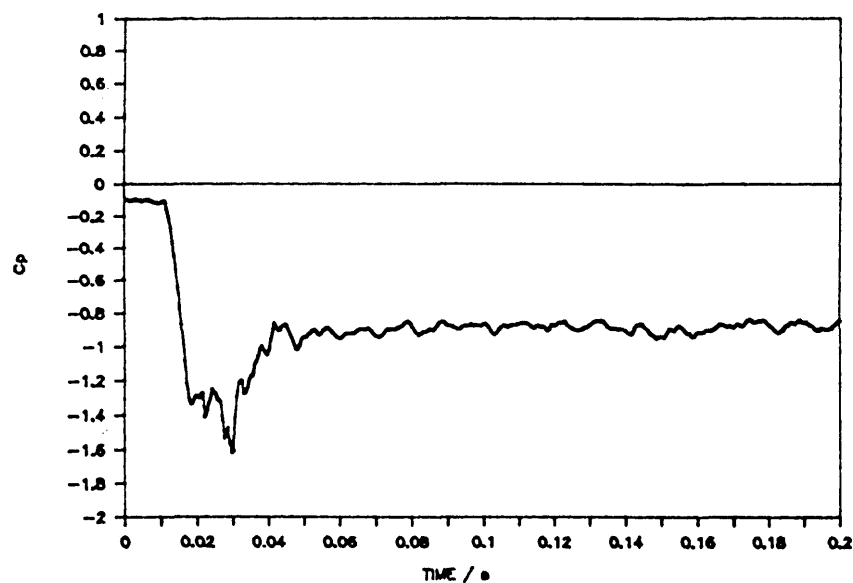


Fig 4.5n Time histories of C_{p14} , $x/c = 0.84$, $T = 2.3$, $\delta_{ss} = 90^\circ$

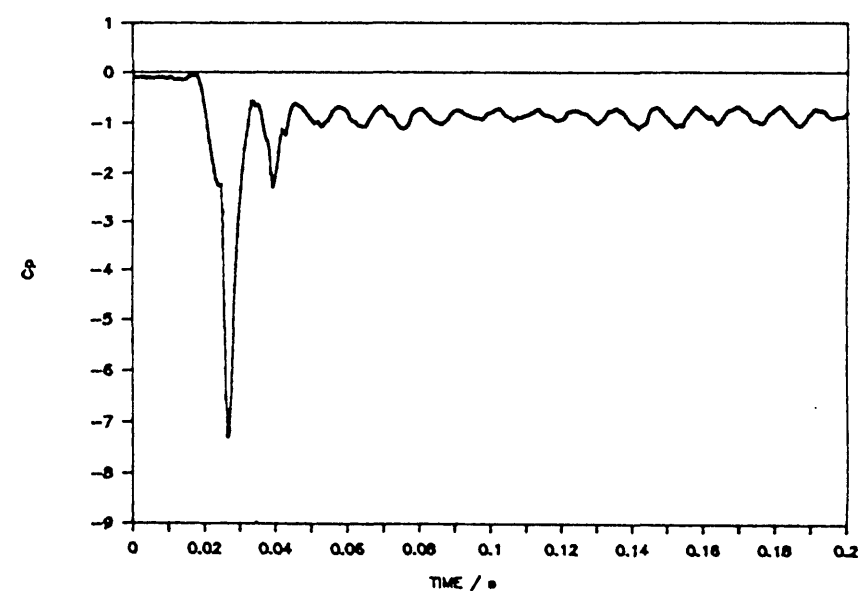


Fig 4.5p Time histories of C_{p16} , $x/c = 0.97$, $T = 2.3$, $\delta_{ss} = 90^\circ$

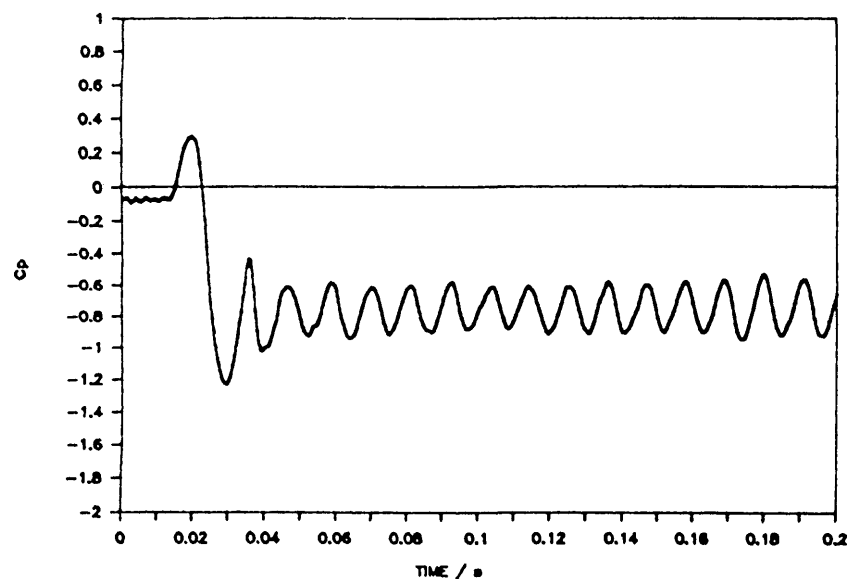


Fig 4.6a Time histories of C_{p17} , $x/c = 0.97$, $T = 2.3$, $\delta_{ss} = 90^\circ$

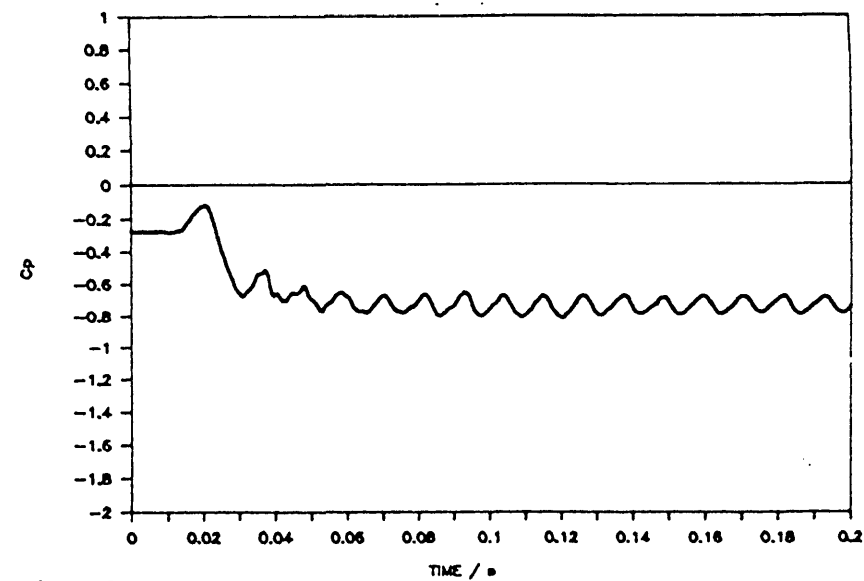


Fig 4.6c Time histories of C_{p21} , $x/c = 0.72$, $T = 2.3$, $\delta_{ss} = 90^\circ$

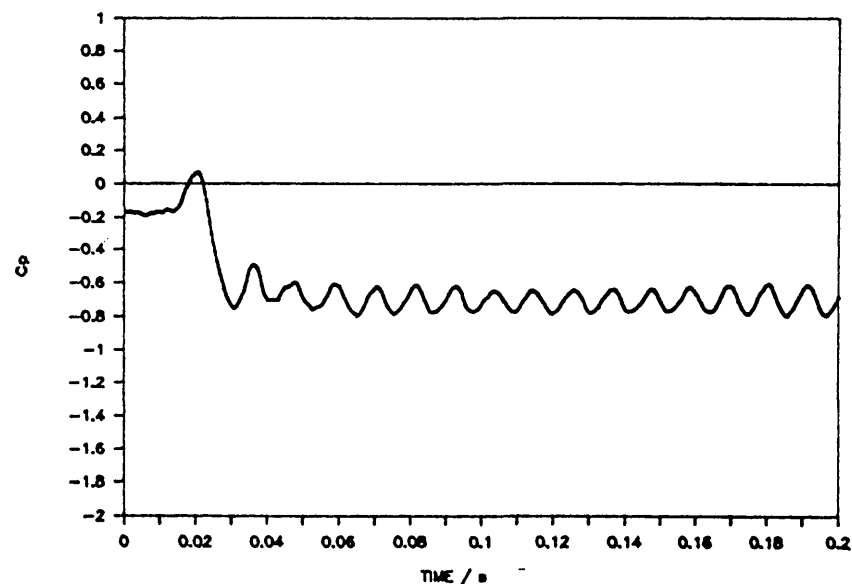


Fig 4.6b Time histories of C_{p19} , $x/c = 0.84$, $T = 2.3$, $\delta_{ss} = 90^\circ$

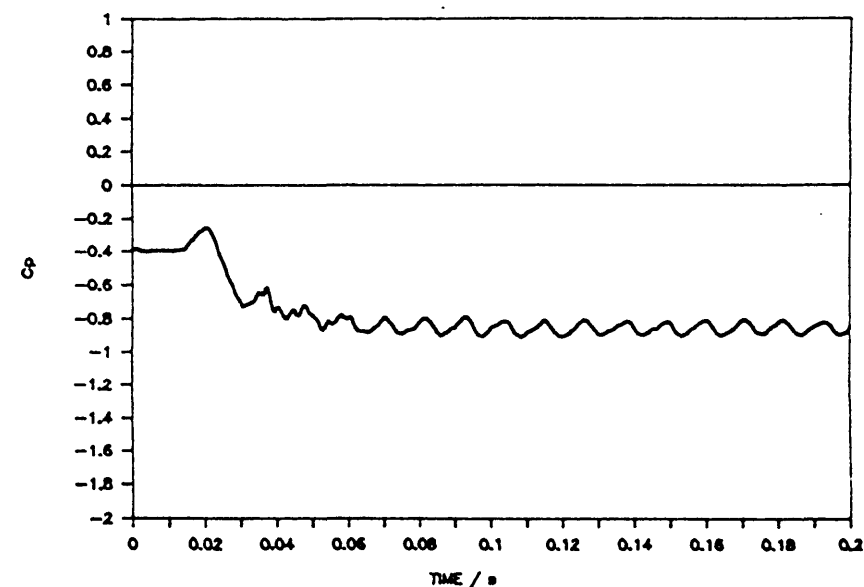


Fig 4.6d Time histories of C_{p23} , $x/c = 0.60$, $T = 2.3$, $\delta_{ss} = 90^\circ$

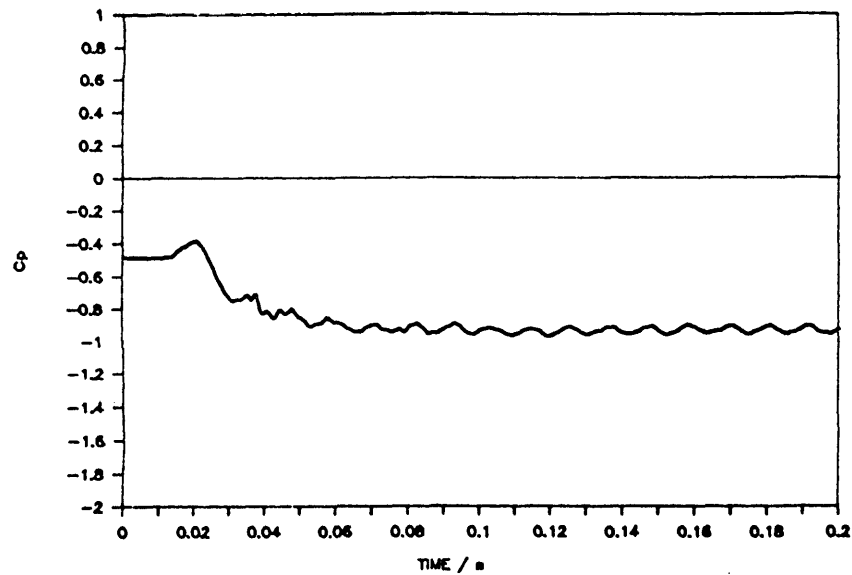


Fig 4.6e Time histories of C_{p25} , $x/c = 0.47$, $T = 2.3$, $\delta_{ss} = 90^\circ$

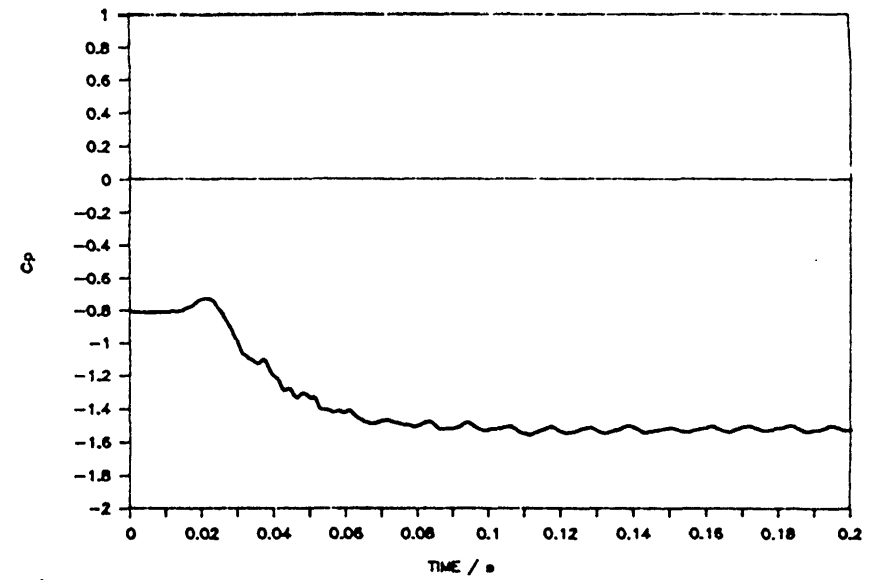


Fig 4.6g Time histories of C_{p30} , $x/c = 0.16$, $T = 2.3$, $\delta_{ss} = 90^\circ$

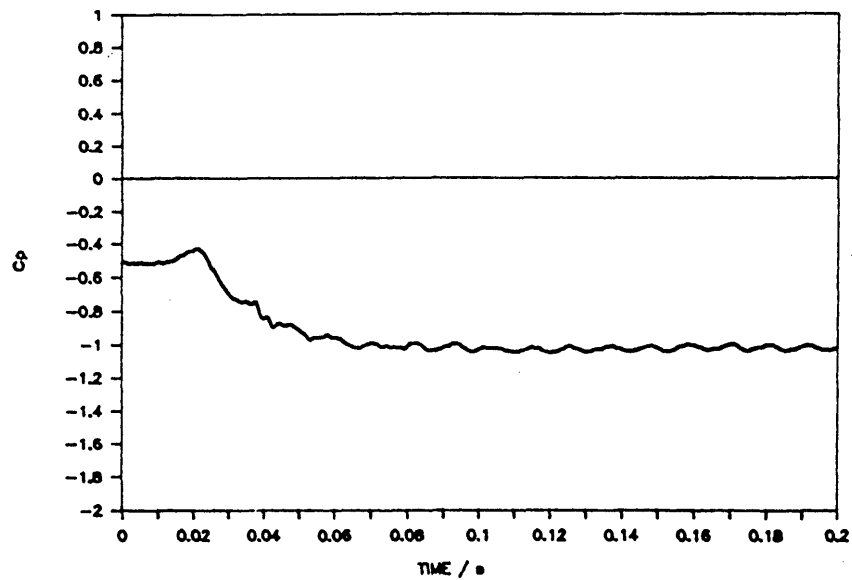


Fig 4.6f Time histories of C_{p28} , $x/c = 0.28$, $T = 2.3$, $\delta_{ss} = 90^\circ$

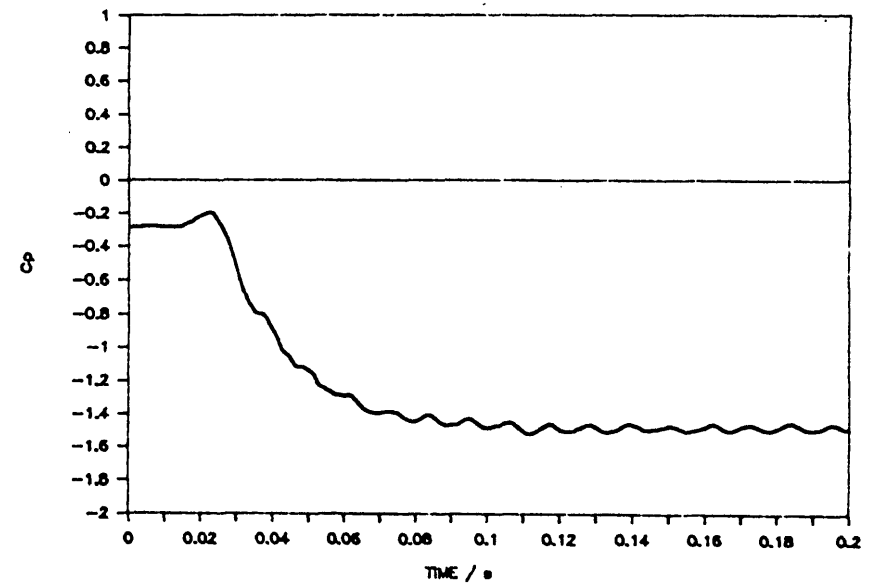


Fig 4.6h Time histories of C_{p32} , $x/c = 0.03$, $T = 2.3$, $\delta_{ss} = 90^\circ$

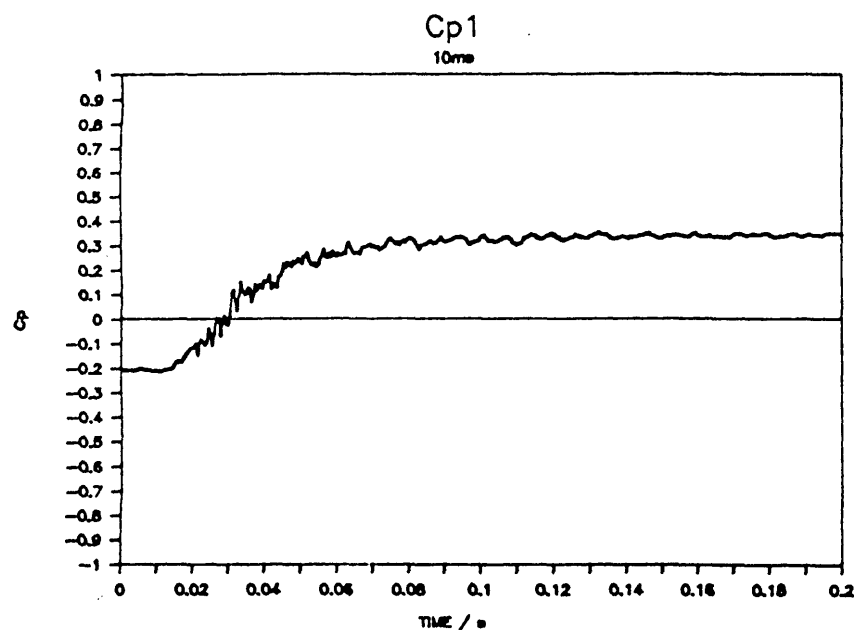


Fig 4.7a Time histories of C_{p1} , $x/c = 0.03$, $T = 1$, $\delta_{ss} = 30^\circ$

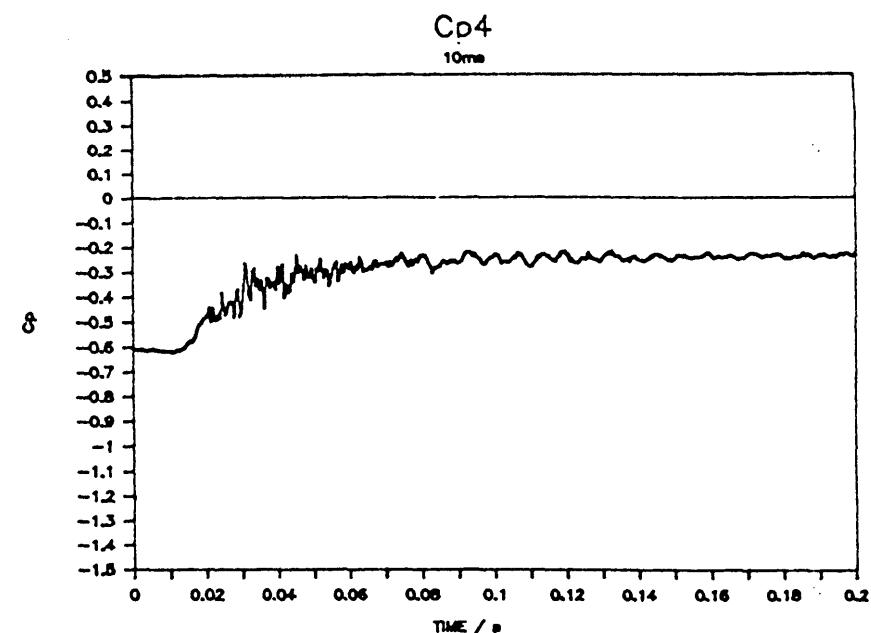


Fig 4.7c Time histories of C_{p4} , $x/c = 0.22$, $T = 1$, $\delta_{ss} = 30^\circ$

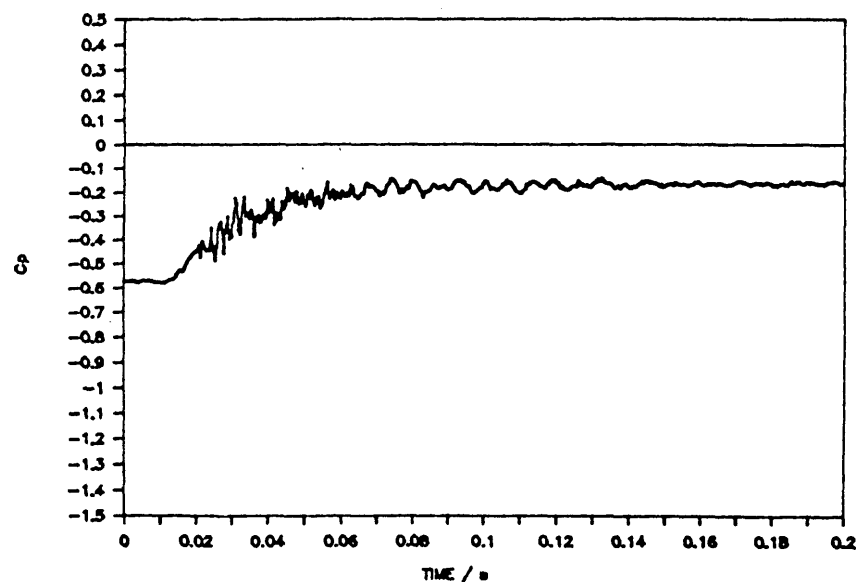


Fig 4.7b Time histories of C_{p3} , $x/c = 0.16$, $T = 1$, $\delta_{ss} = 30^\circ$

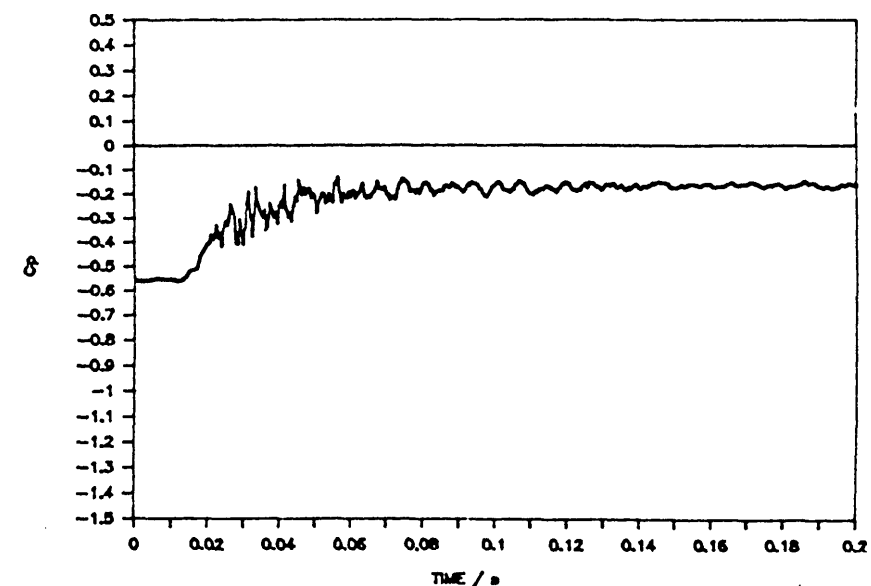


Fig 4.7d Time histories of C_{p5} , $x/c = 0.28$, $T = 1$, $\delta_{ss} = 30^\circ$

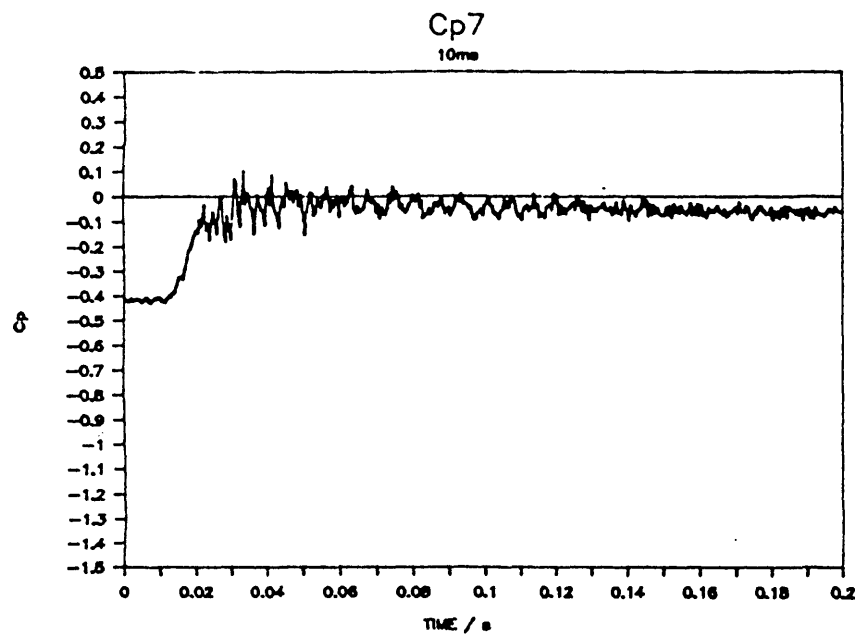


Fig 4.7e Time histories of C_{p7} , $x/c = 0.41$, $T = 1$, $\delta_{ss} = 30^\circ$

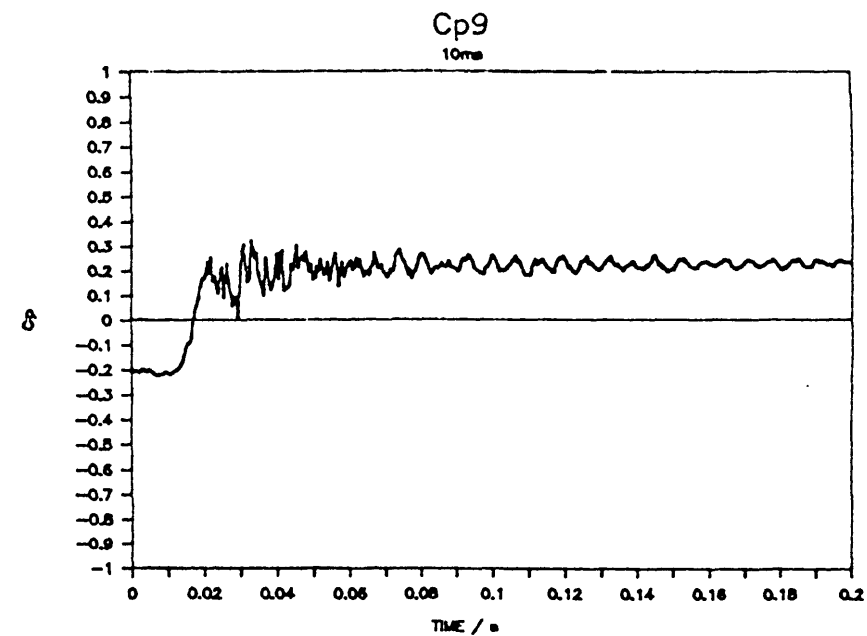


Fig 4.7g Time histories of C_{p9} , $x/c = 0.53$, $T = 1$, $\delta_{ss} = 30^\circ$

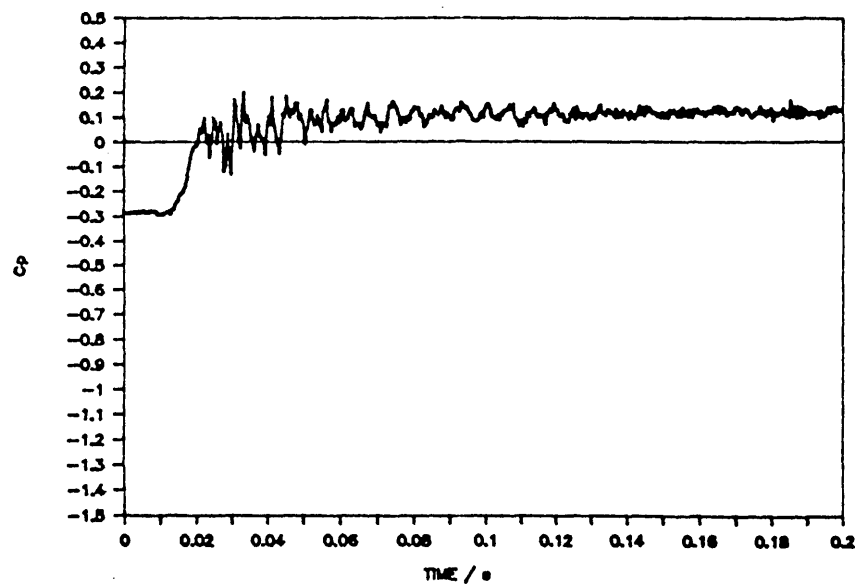


Fig 4.7f Time histories of C_{p8} , $x/c = 0.47$, $T = 1$, $\delta_{ss} = 30^\circ$

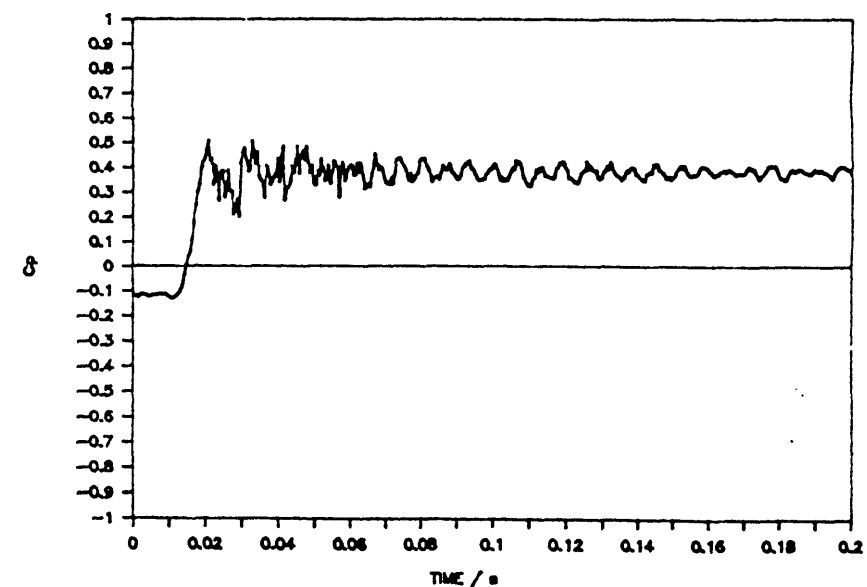


Fig 4.7h Time histories of C_{p10} , $x/c = 0.60$, $T = 1$, $\delta_{ss} = 30^\circ$

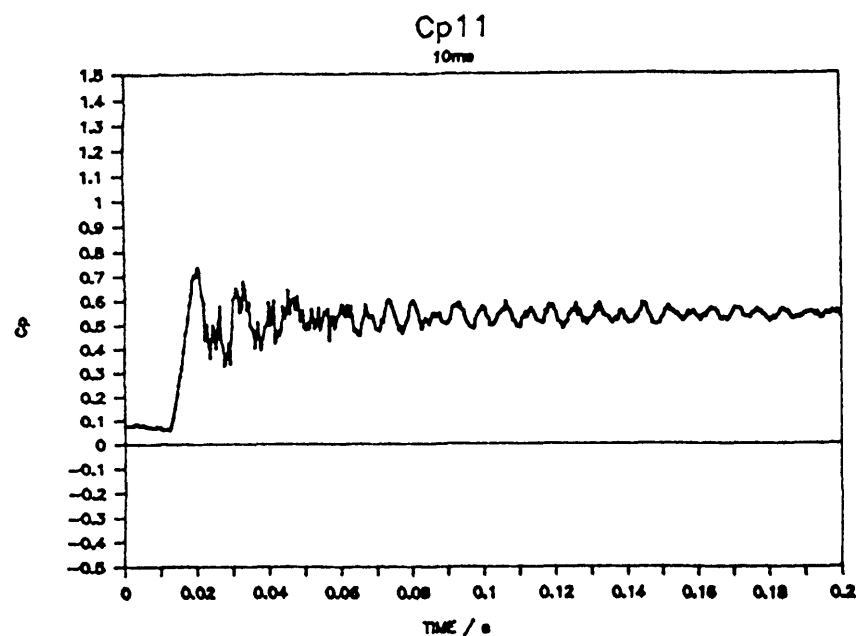


Fig 4.7i Time histories of C_{p11} , $x/c = 0.66$, $T = 1$, $\delta_{ss} = 30^\circ$

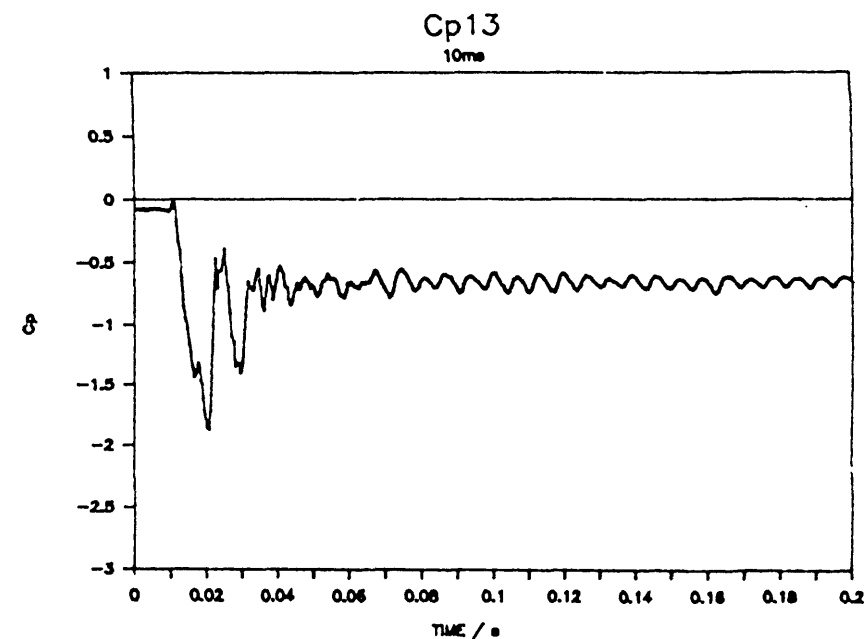


Fig 4.7k Time histories of C_{p13} , $x/c = 0.78$, $T = 1$, $\delta_{ss} = 30^\circ$

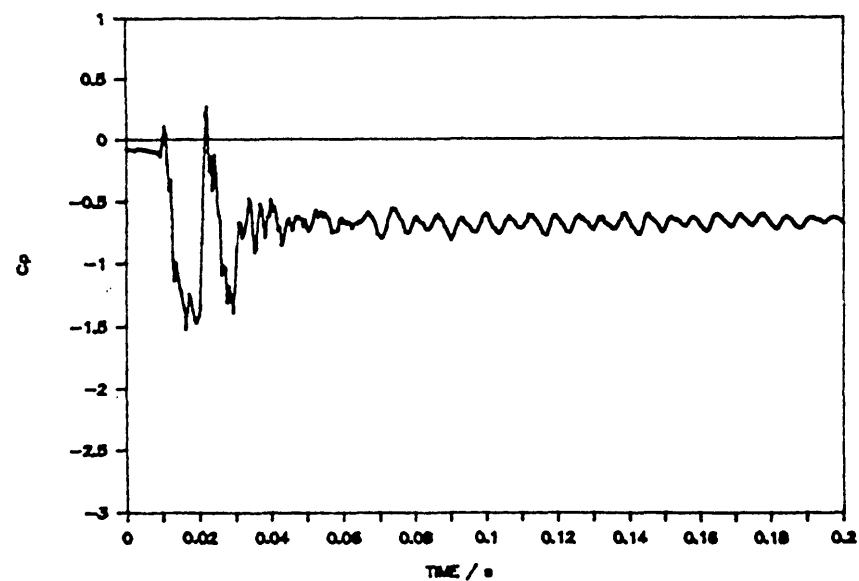


Fig 4.7j Time histories of C_{p12} , $x/c = 0.72$, $T = 1$, $\delta_{ss} = 30^\circ$

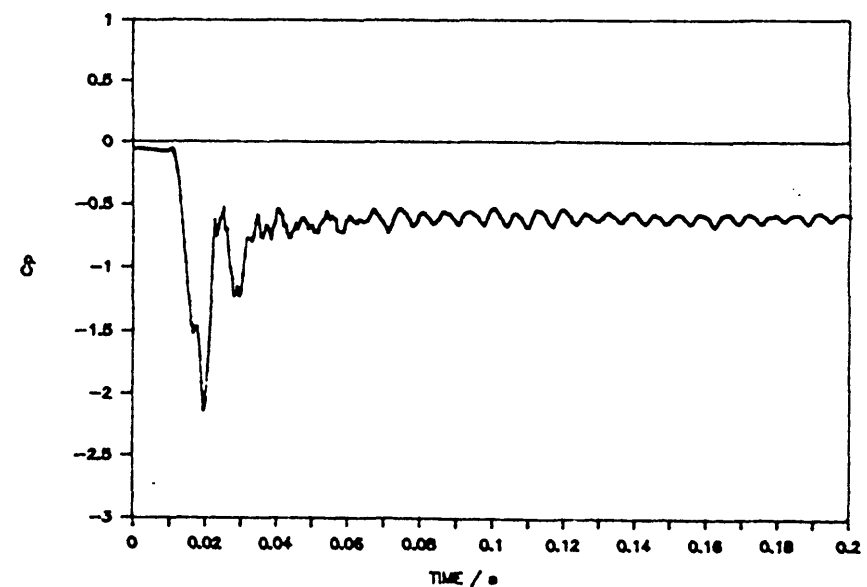


Fig 4.7l Time histories of C_{p14} , $x/c = 0.84$, $T = 1$, $\delta_{ss} = 30^\circ$

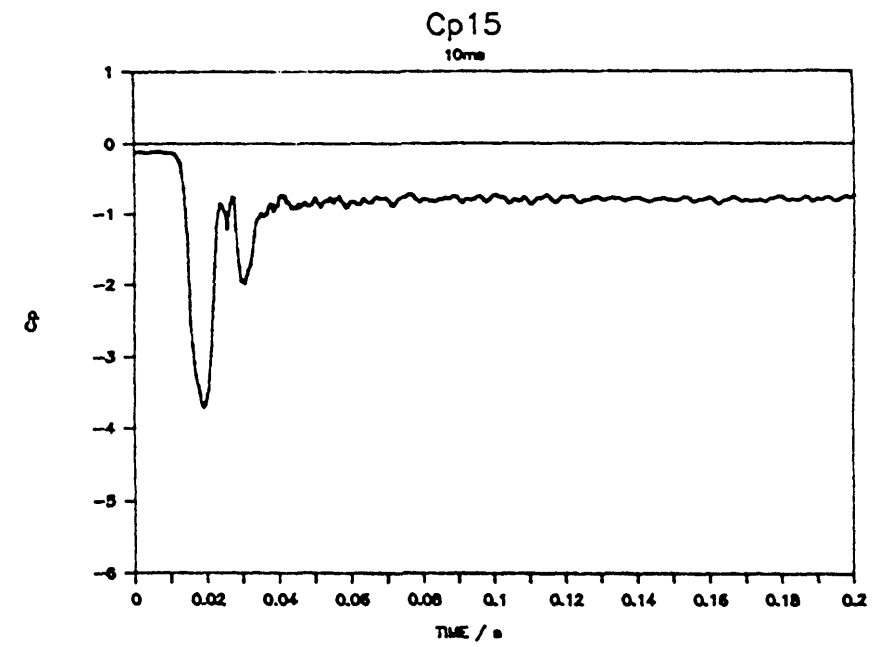


Fig 4.7m Time histories of C_{p15} , $x/c = 0.90$, $T = 1$, $\delta_{ss} = 30^\circ$

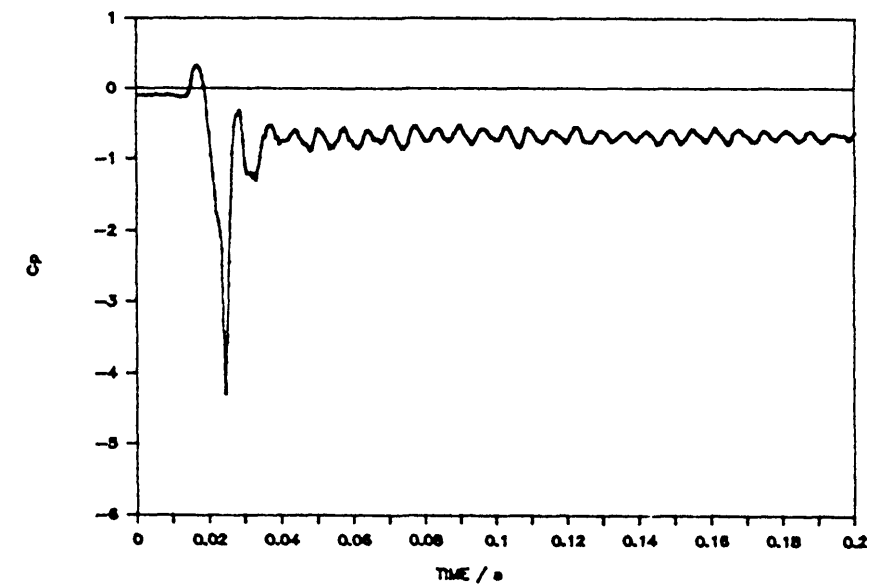


Fig 4.7n Time histories of C_{p16} , $x/c = 0.97$, $T = 1$, $\delta_{ss} = 30^\circ$

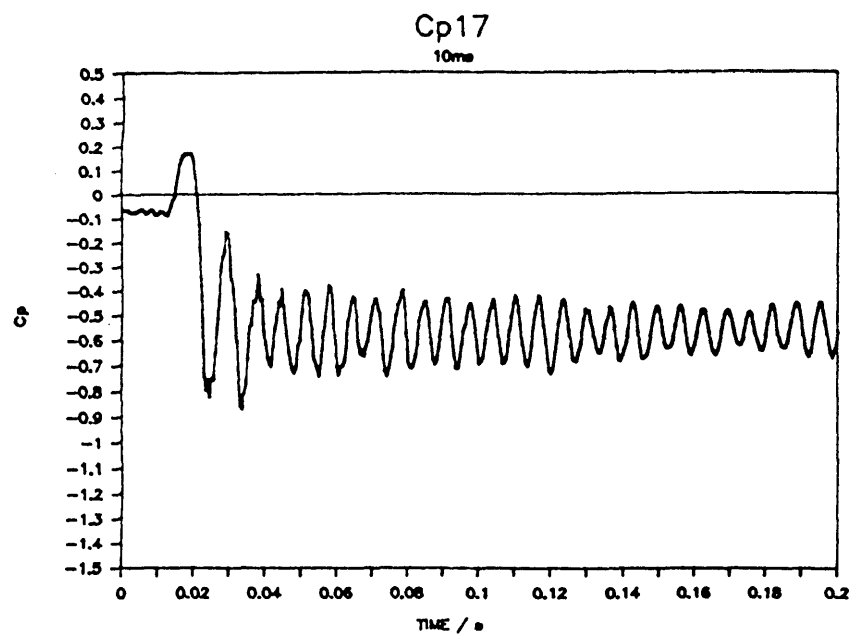


Fig 4.8a Time histories of C_{p17} , $x/c = 0.97$, $T = 1$, $\delta_{ss} = 30^\circ$

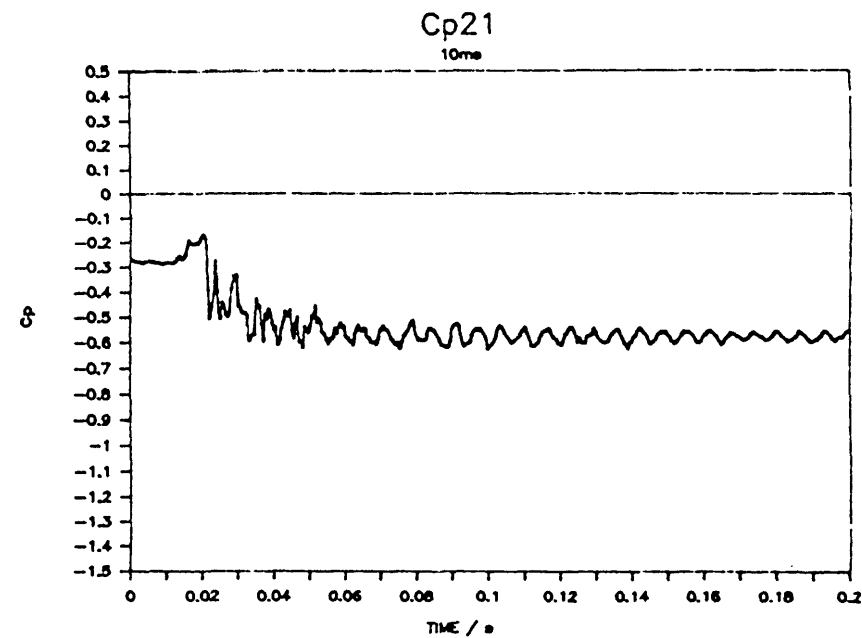


Fig 4.8c Time histories of C_{p21} , $x/c = 0.72$, $T = 1$, $\delta_{ss} = 30^\circ$

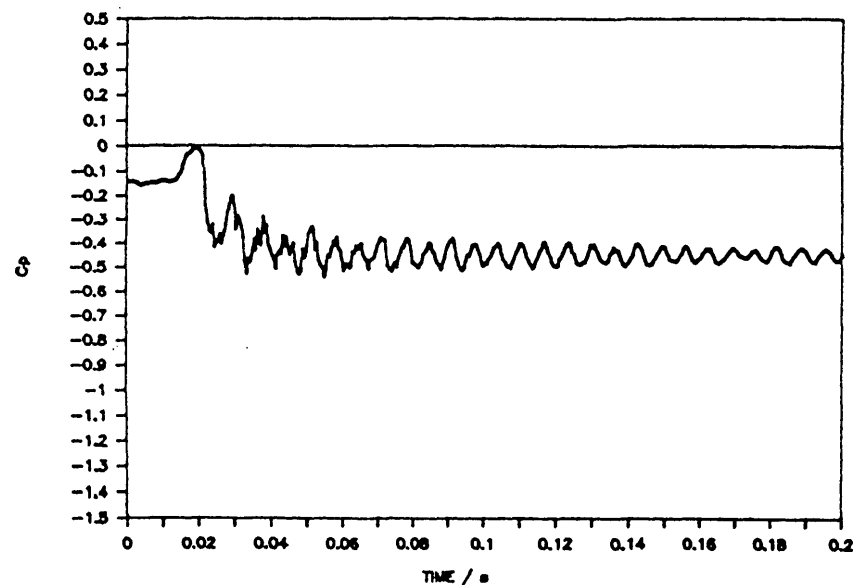


Fig 4.8b Time histories of C_{p19} , $x/c = 0.84$, $T = 1$, $\delta_{ss} = 30^\circ$

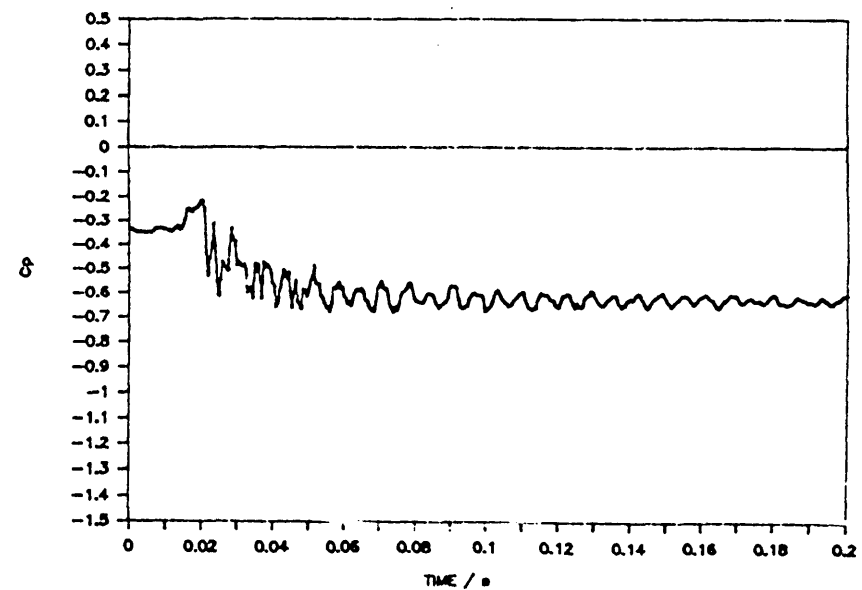


Fig 4.8d Time histories of C_{p23} , $x/c = 0.60$, $T = 1$, $\delta_{ss} = 30^\circ$

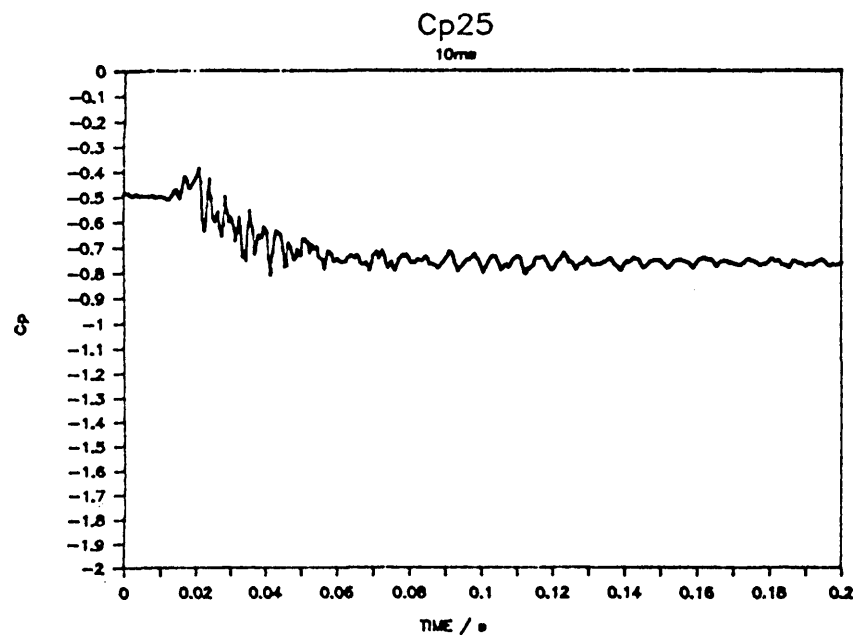


Fig 4.8e Time histories of C_{p25} , $x/c = 0.47$, $T = 1$, $\delta_{ss} = 30^\circ$

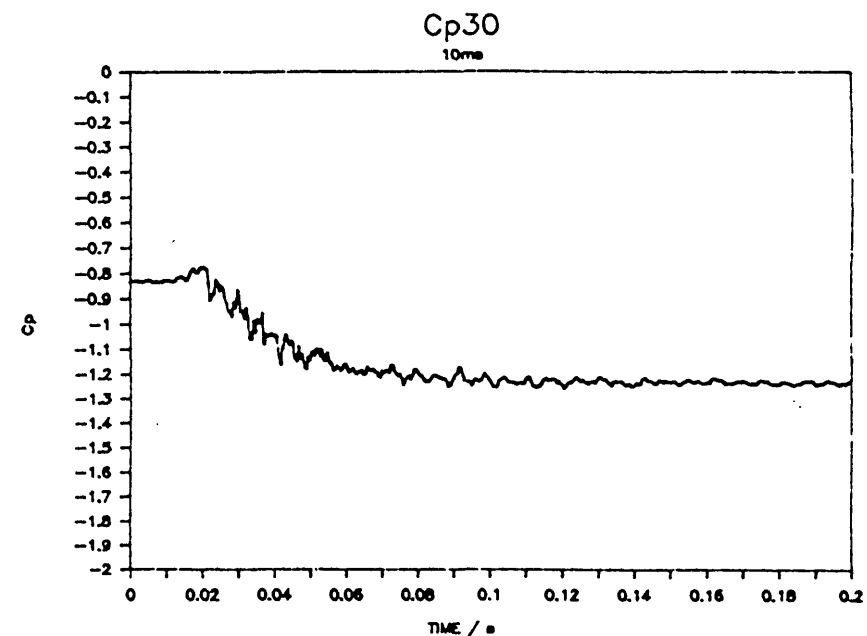


Fig 4.8g Time histories of C_{p30} , $x/c = 0.16$, $T = 1$, $\delta_{ss} = 30^\circ$

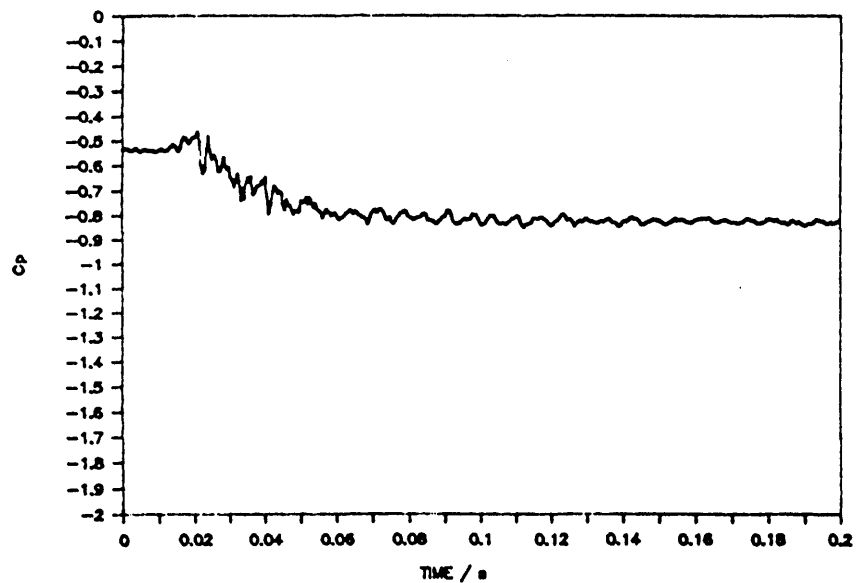


Fig 4.8f Time histories of C_{p28} , $x/c = 0.28$, $T = 1$, $\delta_{ss} = 30^\circ$

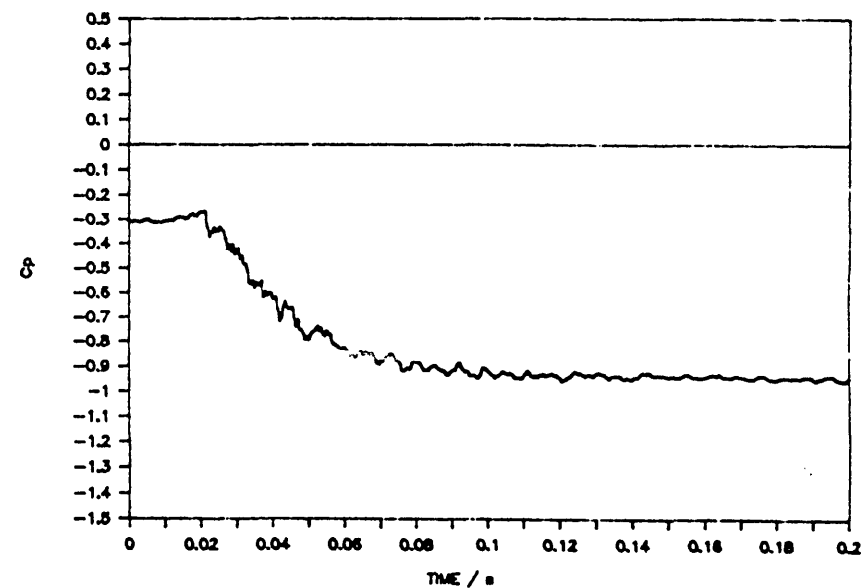


Fig 4.8h Time histories of C_{p32} , $x/c = 0.03$, $T = 1$, $\delta_{ss} = 30^\circ$

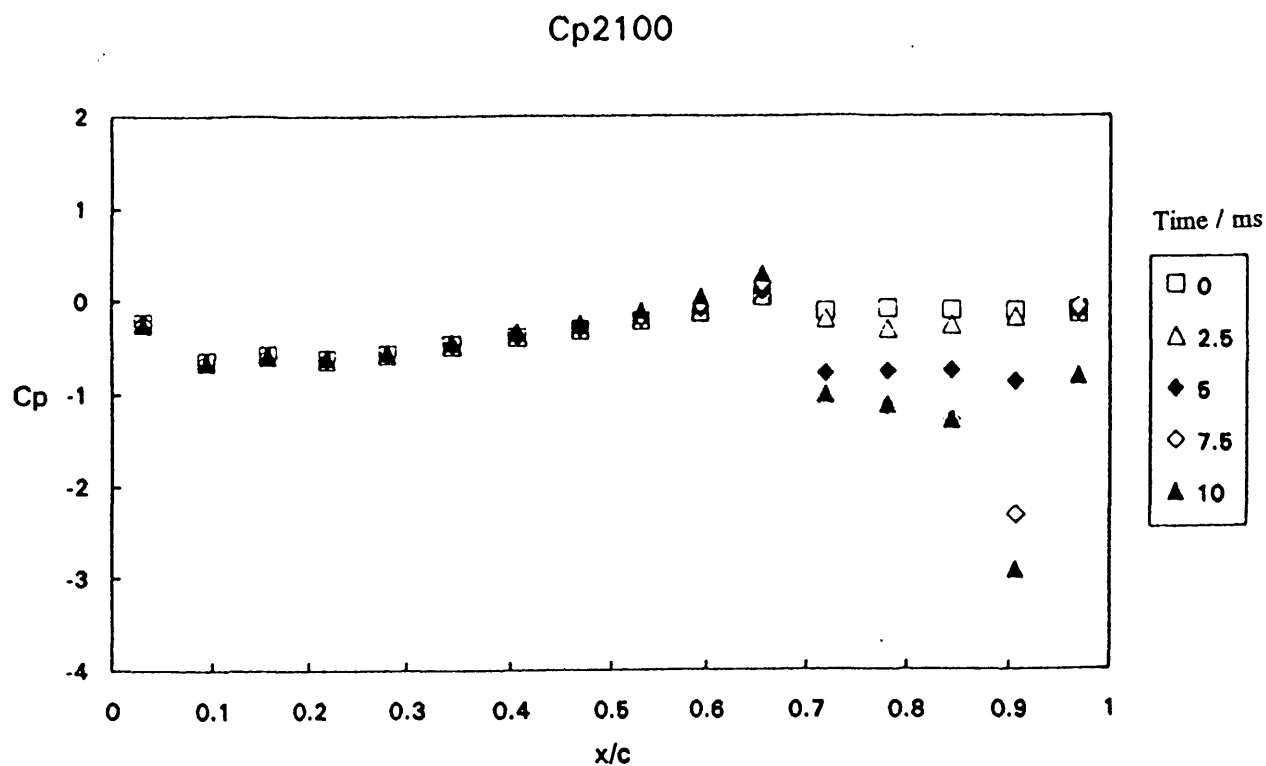


Fig 4.9a Distribution of C_p on aerofoil upper surface from $t_e = 0$ to 10 ms,
 $\delta_{ss} = 90^\circ$

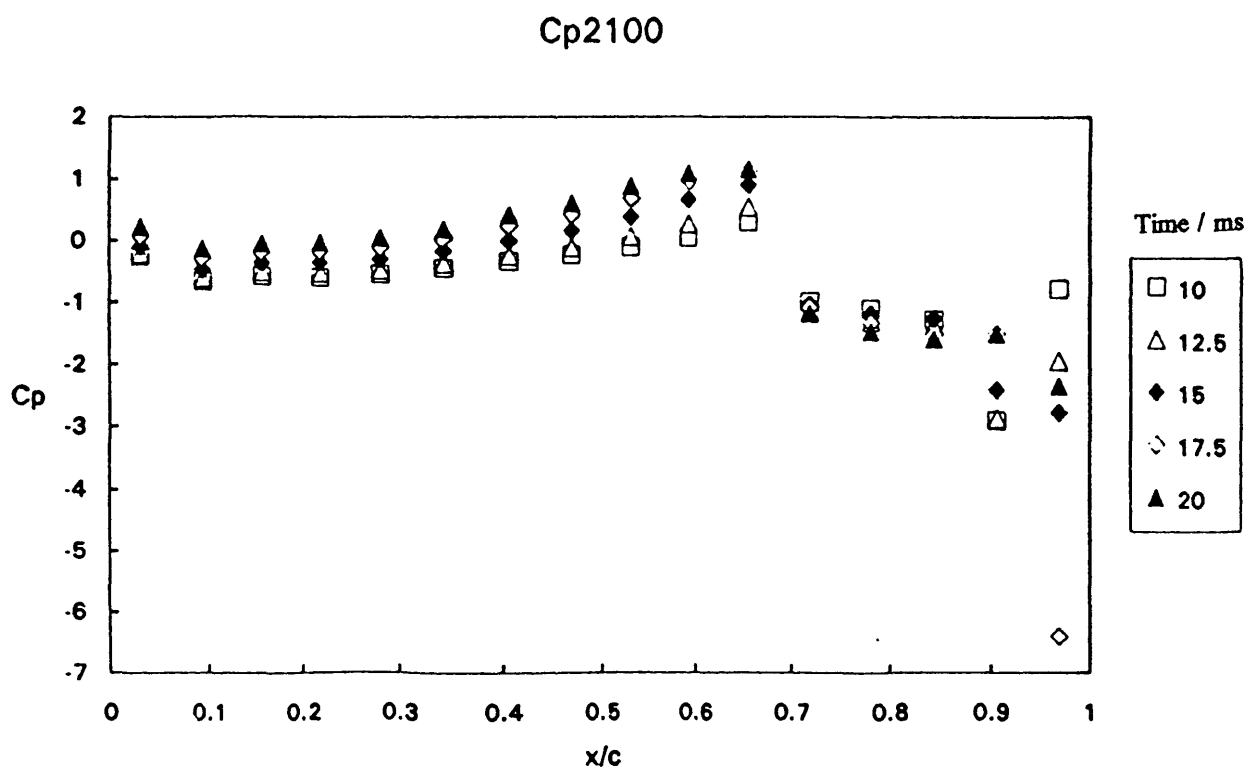


Fig 4.9b Distribution of C_p on aerofoil upper surface from $t_e = 10$ to 20 ms,
 $\delta_{ss} = 90^\circ$

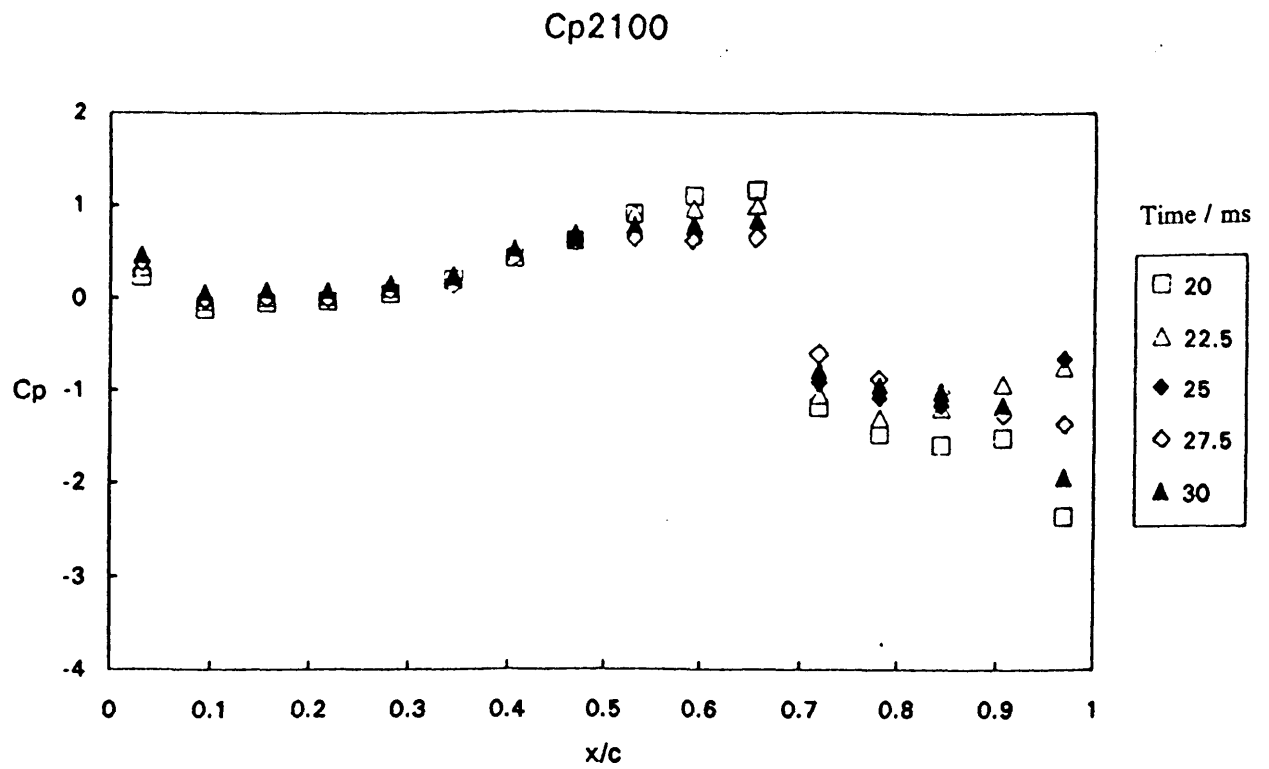


Fig 4.9c Distribution of C_p on aerofoil upper surface from $t_e = 20$ to 30 ms,
 $\delta_{ss} = 90^\circ$

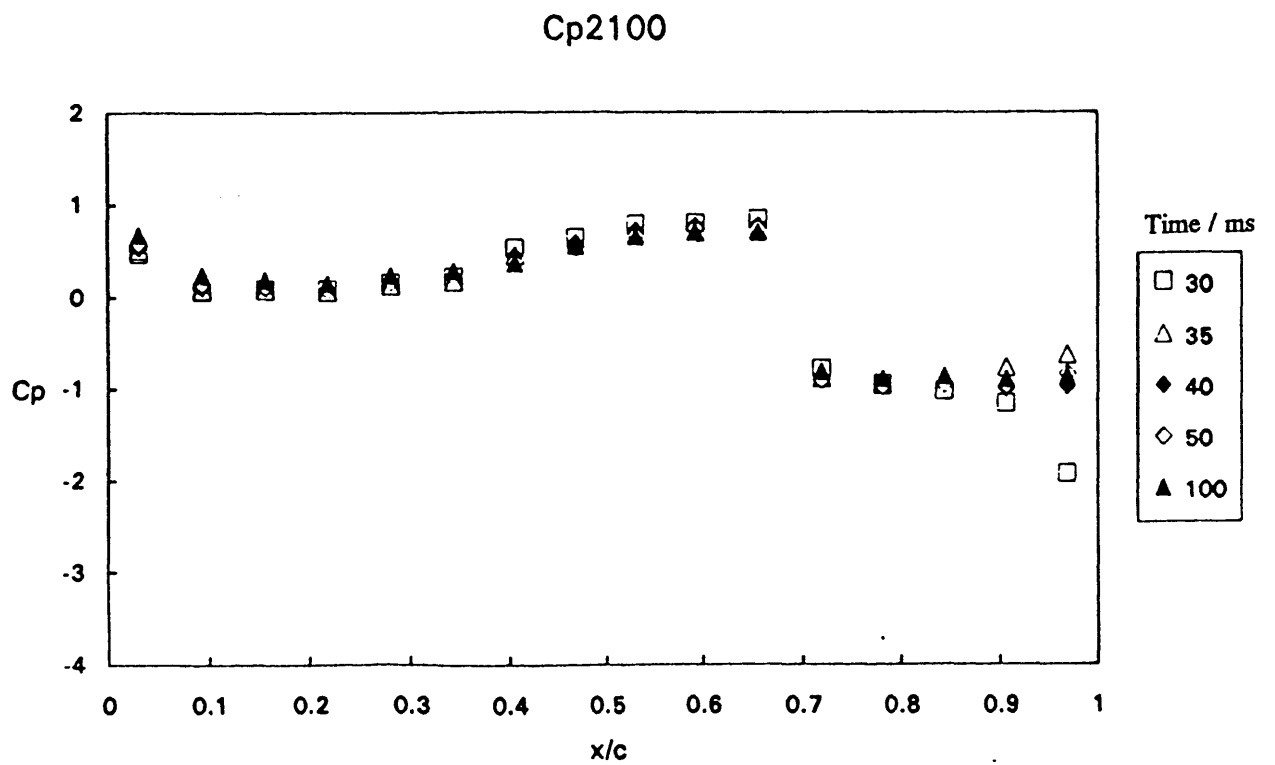


Fig 4.9d Distribution of C_p on aerofoil lower surface from $t_e = 30$ to 100
ms,
 $\delta_{ss} = 90^\circ$

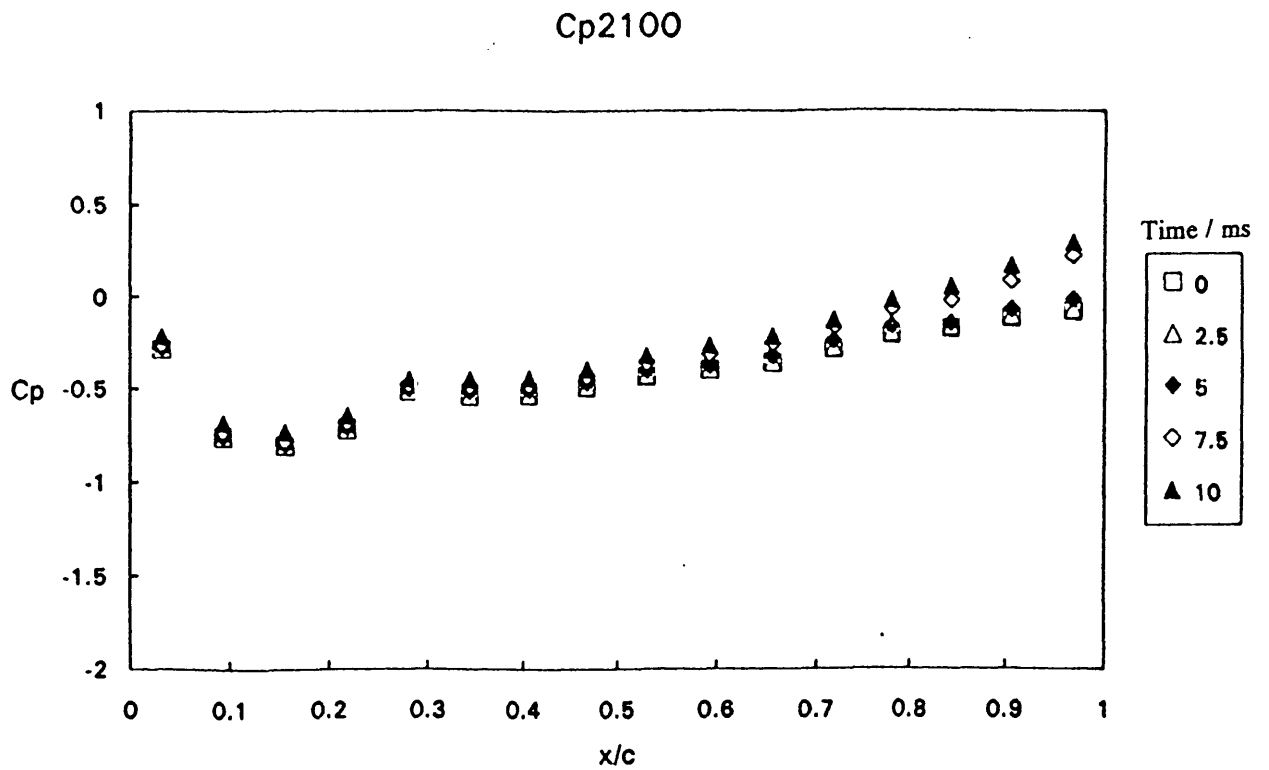


Fig 4.10a Distribution of C_p on aerofoil lower surface from $t_e = 0$ to 10 ms,
 $\delta_{ss} = 30^\circ$

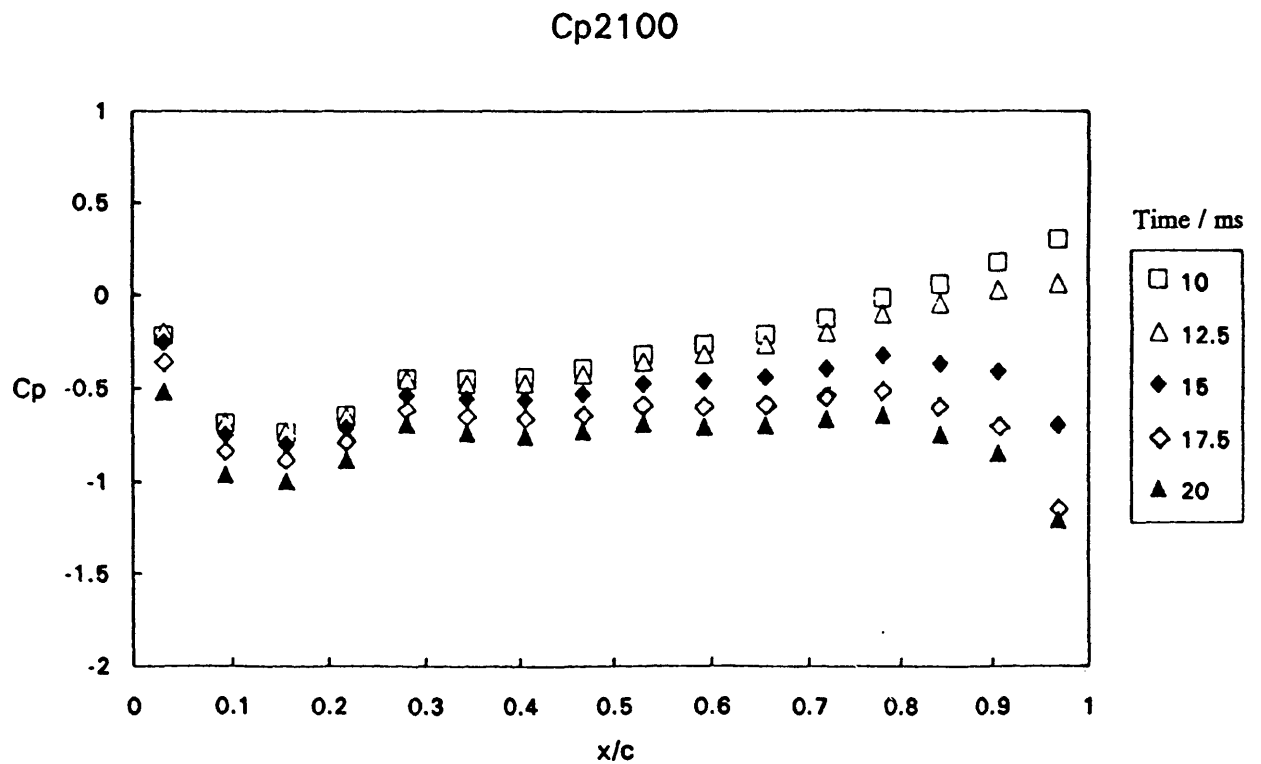


Fig 4.10b Distribution of C_p on aerofoil lower surface from $t_e = 10$ to 20 ms,
 $\delta_{ss} = 30^\circ$

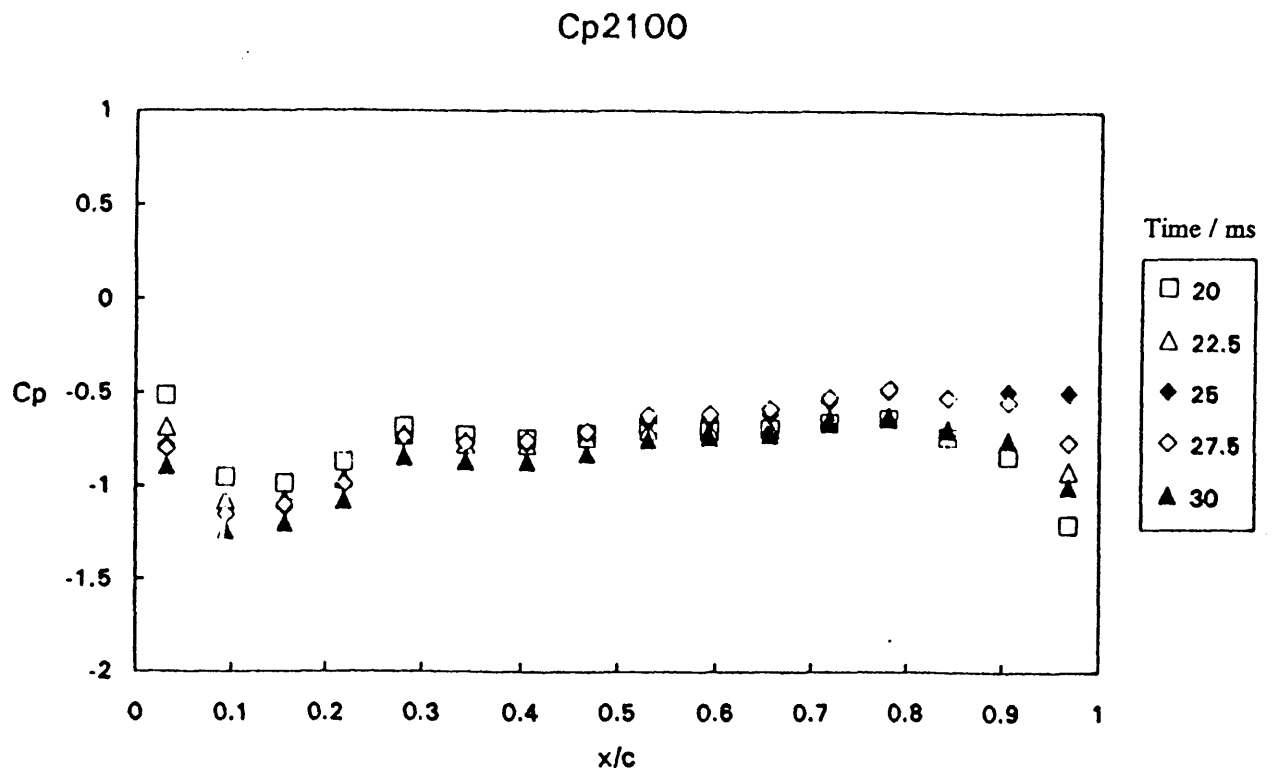


Fig 4.10c Distribution of C_p on aerofoil lower surface from $t_e = 20$ to 30 ms,
 $\delta_{ss} = 30^\circ$

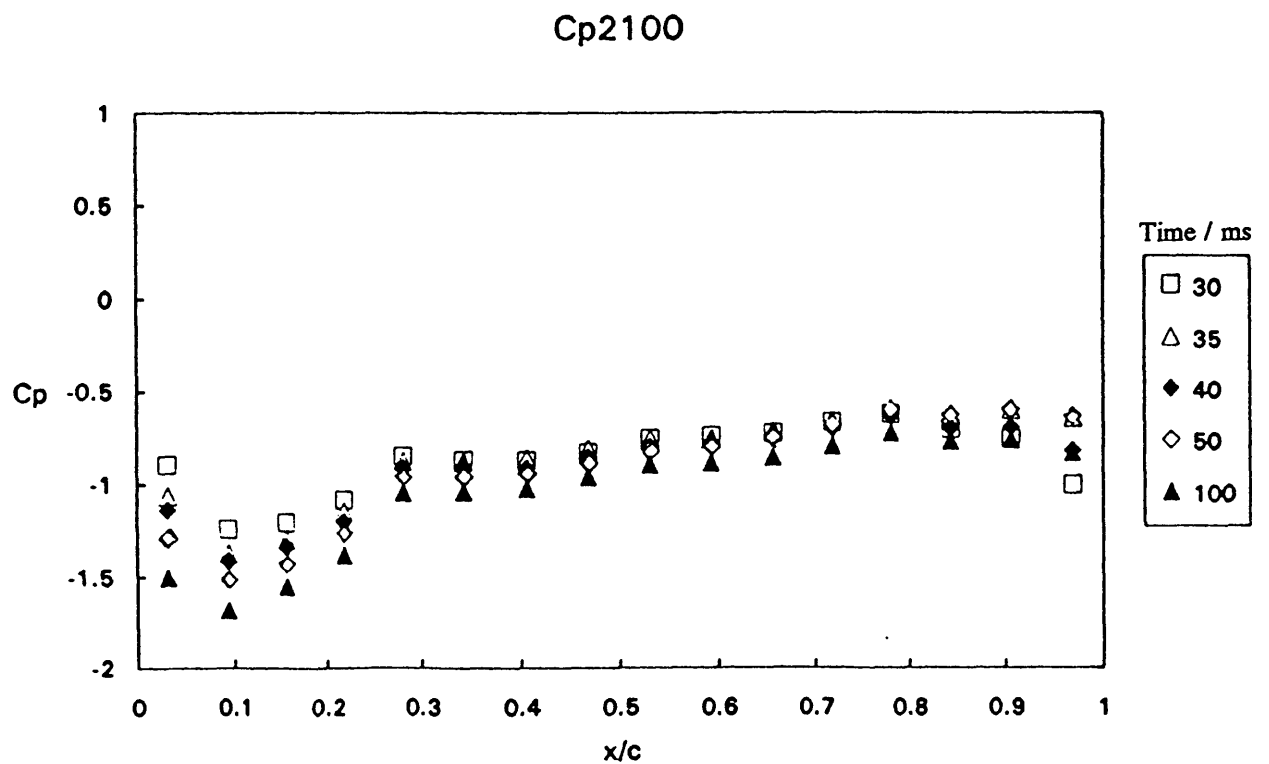
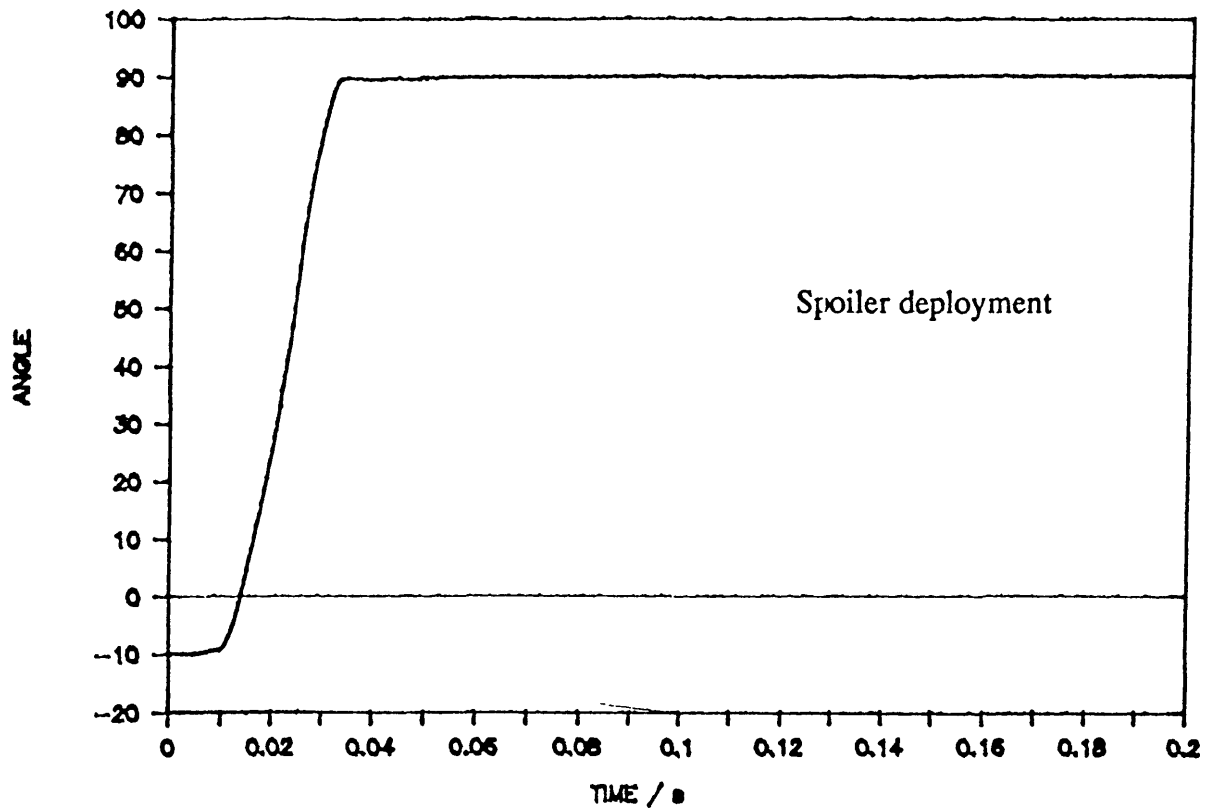


Fig 4.10d Distribution of C_p on aerofoil lower surface from $t_e = 30$ to 100 ms,
 $\delta_{ss} = 30^\circ$

ANGLE 1790



CL1790

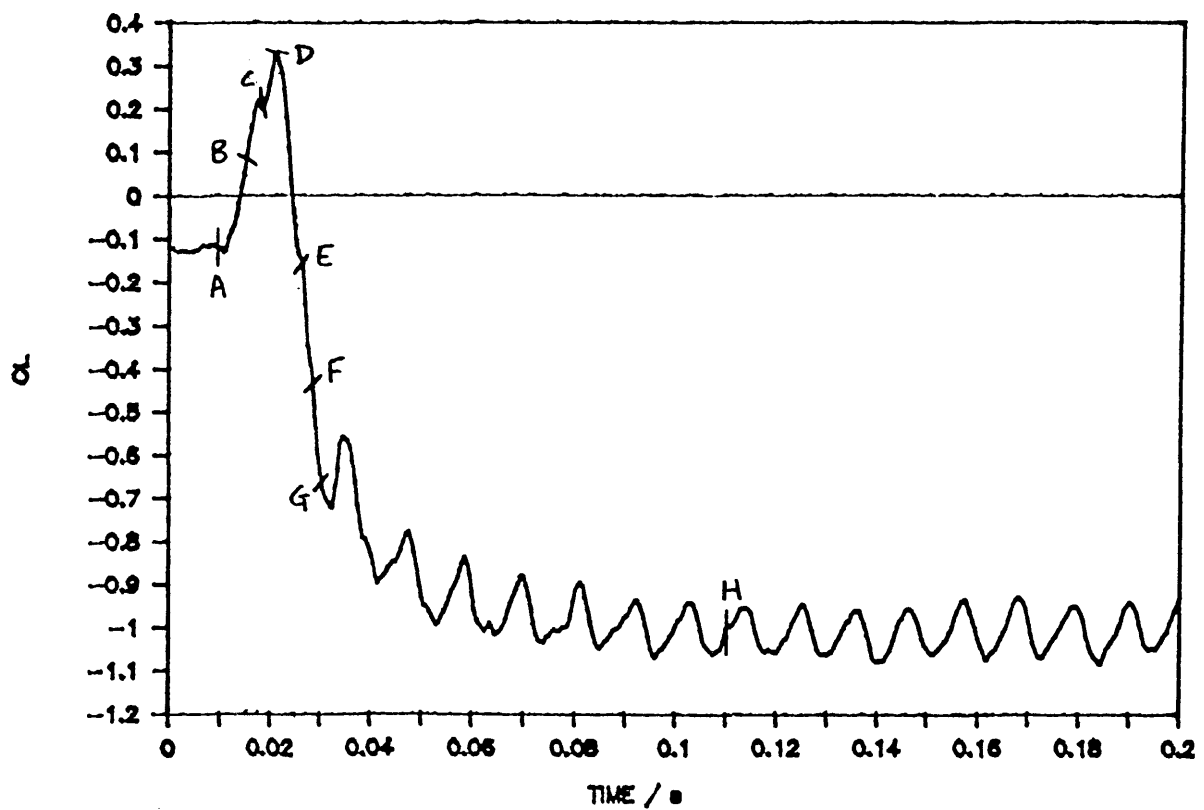


Fig 4.11 Time history of C_L for $T = 2.3$, $\delta_{ss} = 90^\circ$

Pressure Distribution at $t = 0$ ms

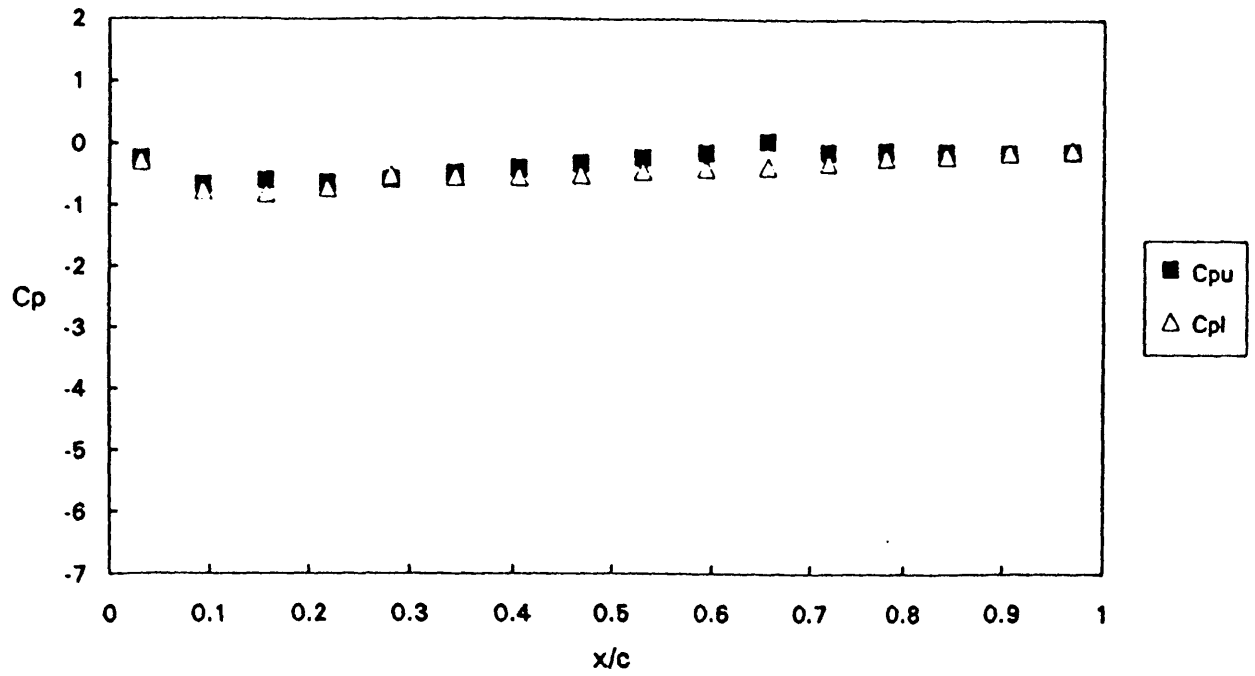


Fig 4.12a Pressure distribution on aerofoil at $t_e = 0$ ms for $T = 2.3$, $\delta_{ss} = 90^\circ$

Pressure Distribution at $t = 5$ ms

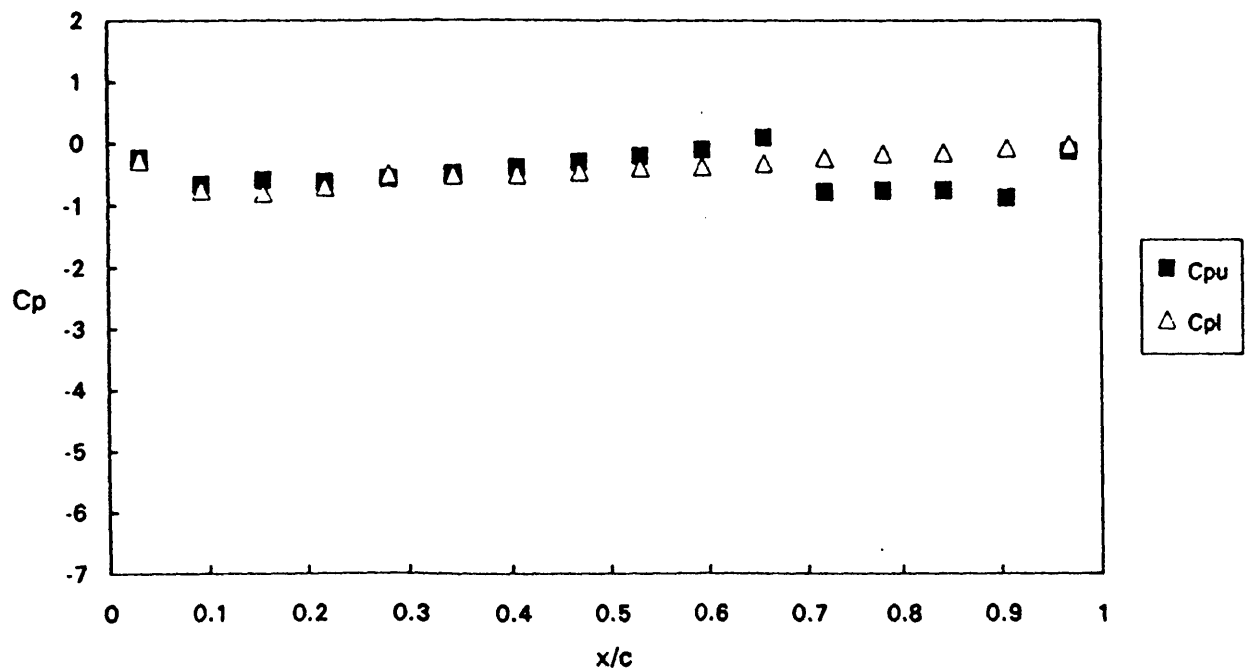


Fig 4.12b Pressure distribution on aerofoil at $t_e = 5$ ms for $T = 2.3$, $\delta_{ss} = 90^\circ$

Pressure Distribution at $t = 7.5$ ms

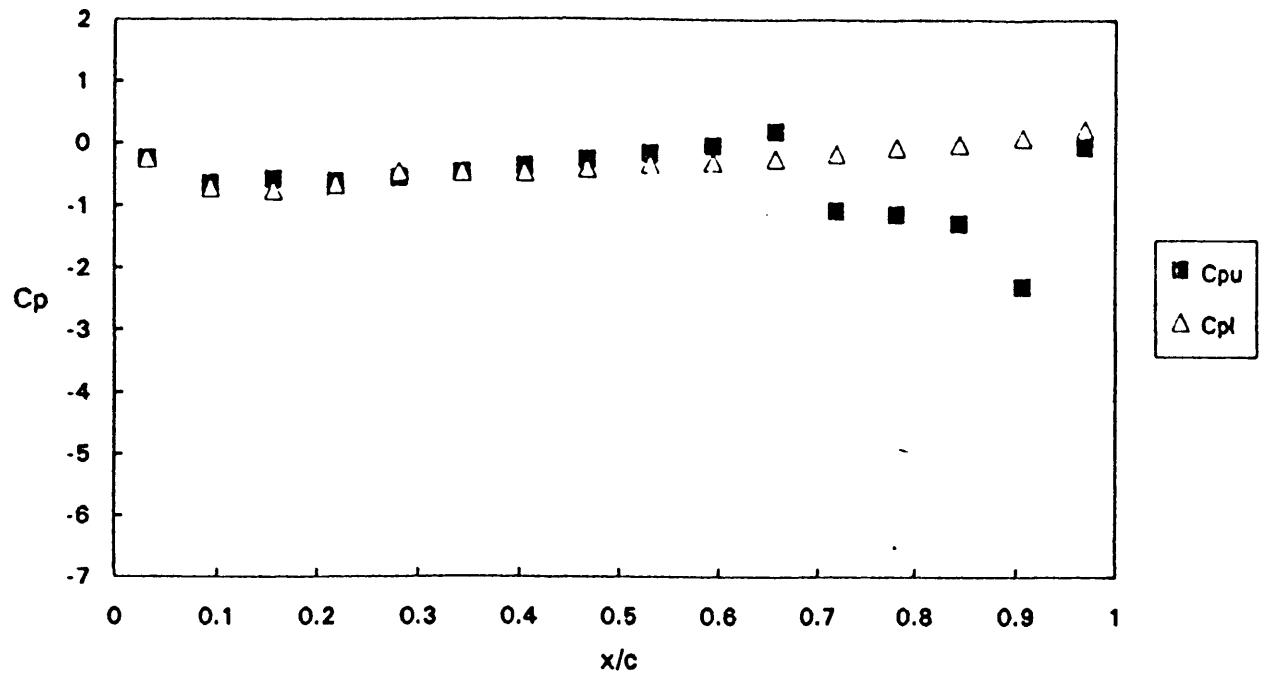


Fig 4.12c Pressure distribution on aerofoil at $t_e = 7.5$ ms for $T = 2.3$, $\delta_{ss} = 90^\circ$

Pressure Distribution at $t = 10$ ms

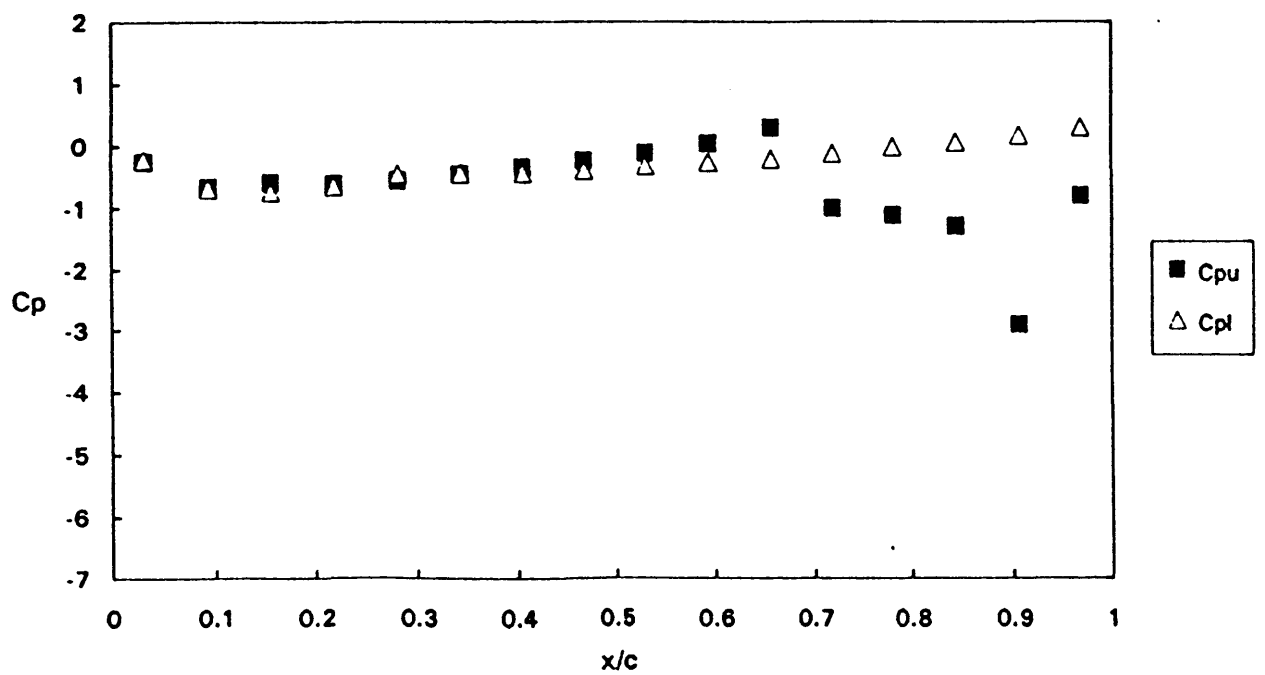


Fig 4.12d Pressure distribution on aerofoil at $t_e = 10$ ms for $T = 2.3$, $\delta_{ss} = 90^\circ$

Pressure Distribution at $t = 15$ ms

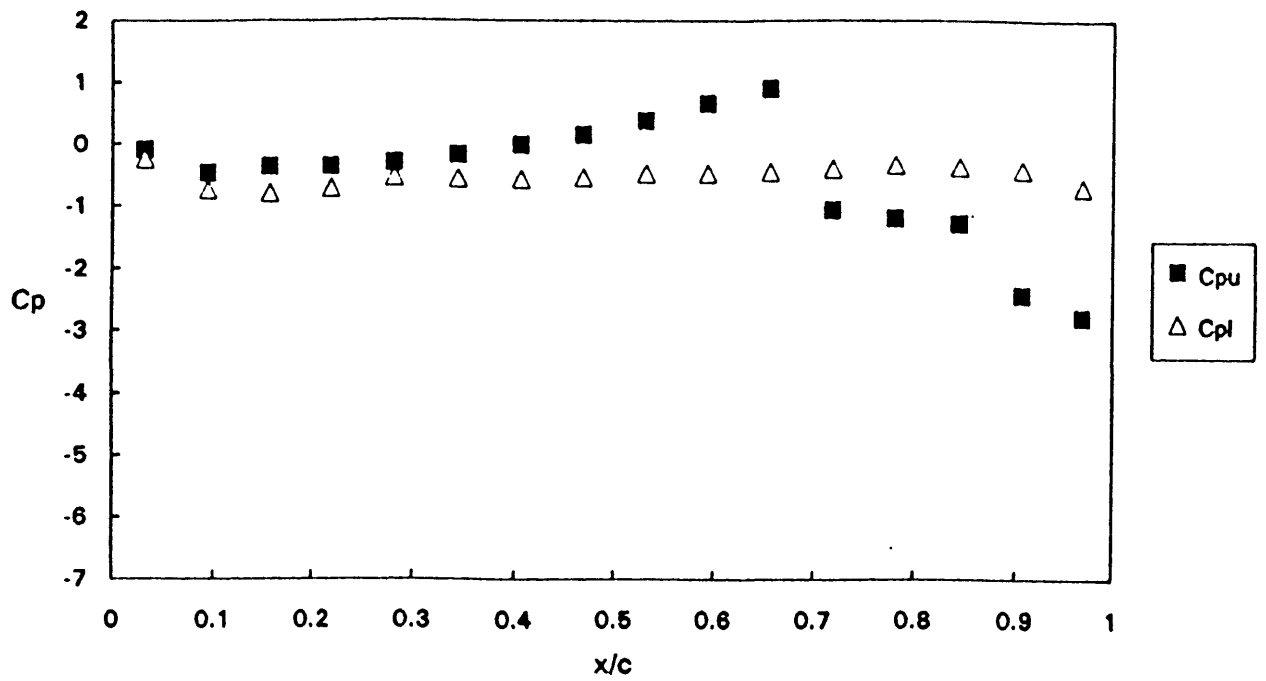


Fig 4.12e Pressure distribution on aerofoil at $t_e = 15$ ms for $T = 2.3$, $\delta_{ss} = 90^\circ$

Pressure Distribution at $t = 17.5$ ms

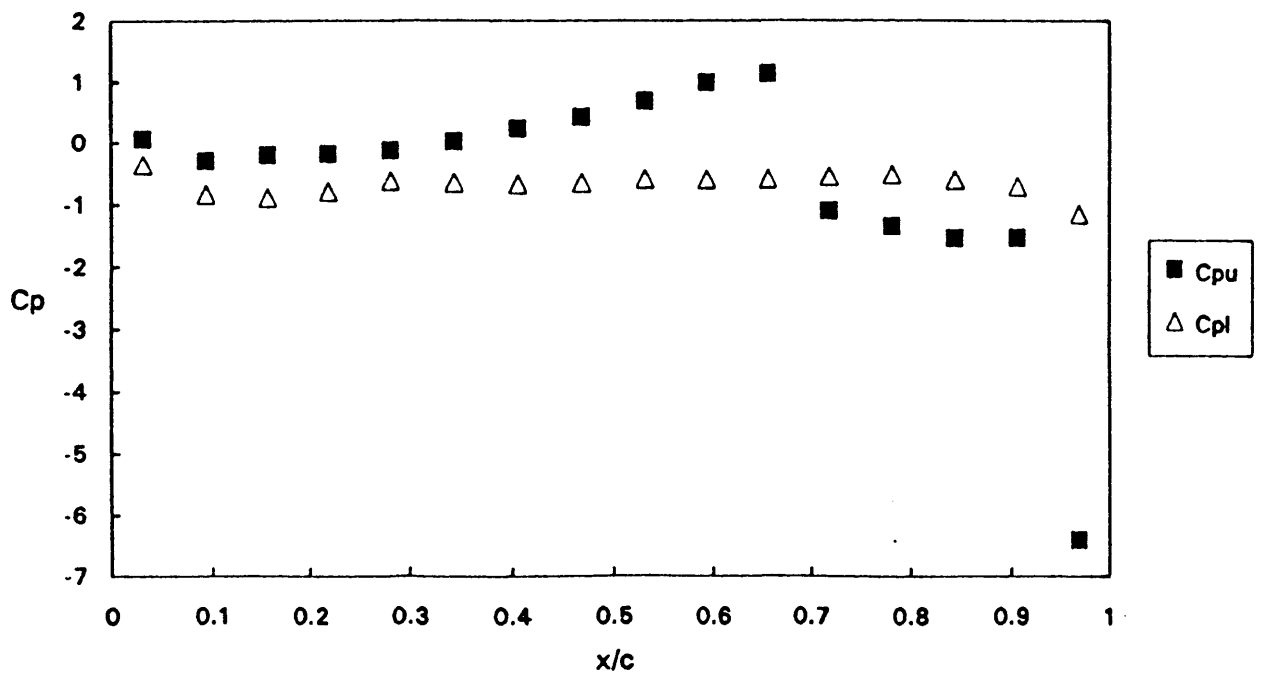


Fig 4.12f Pressure distribution on aerofoil at $t_e = 17.5$ ms for $T = 2.3$, $\delta_{ss} = 90^\circ$

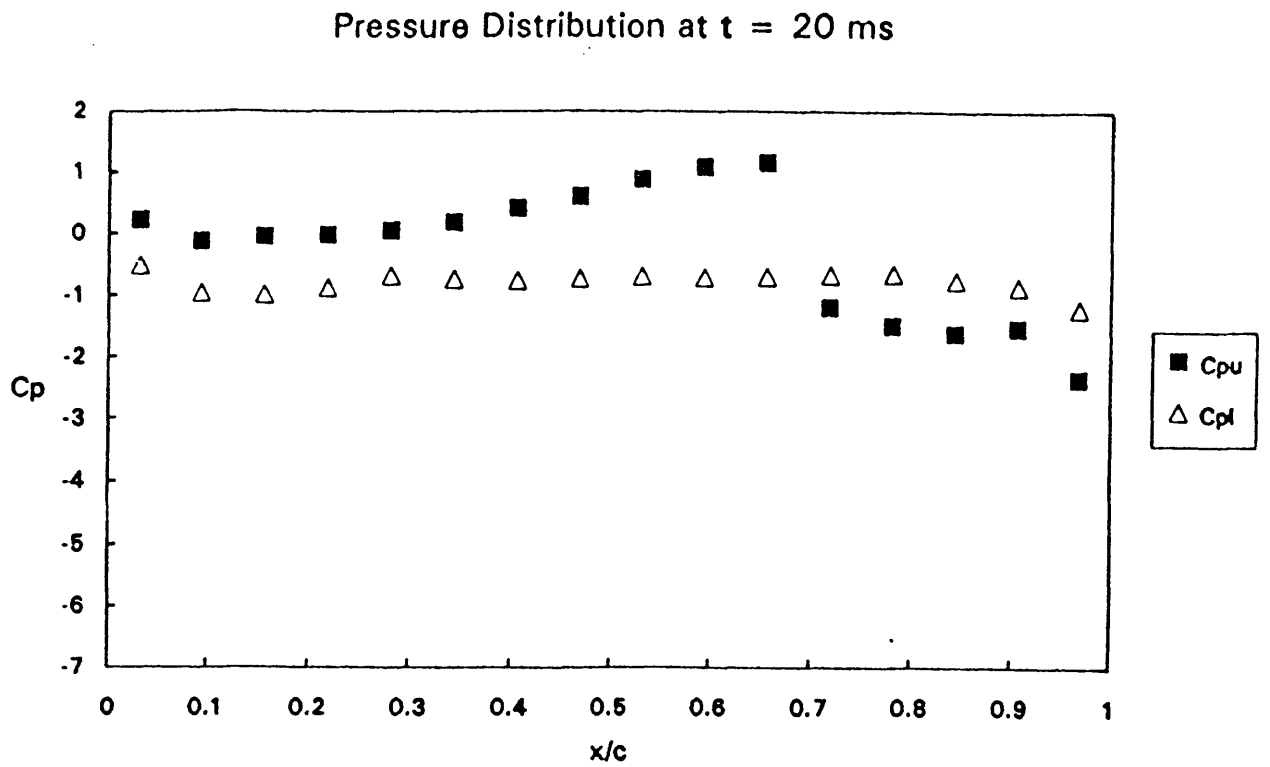


Fig 4.12g Pressure distribution on aerofoil at $t_e = 20$ ms for $T = 2.3$, $\delta_{ss} = 90^\circ$

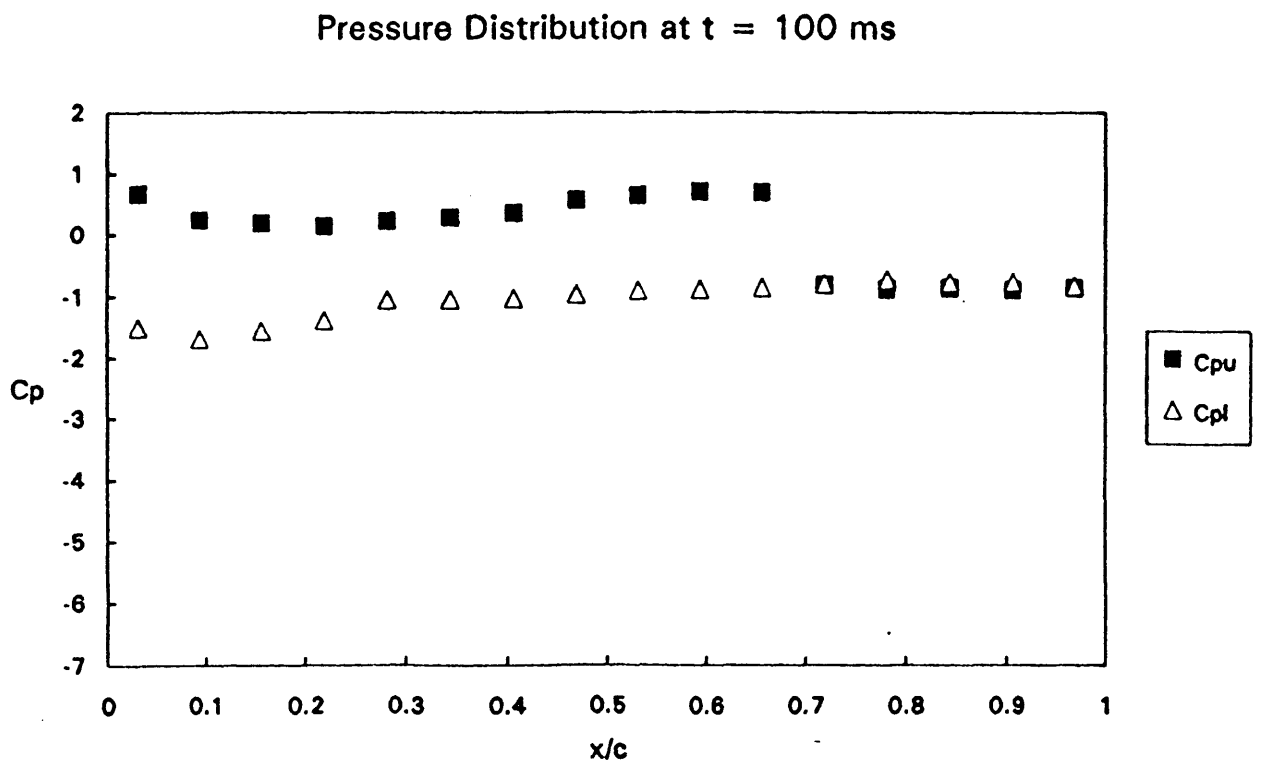


Fig 4.12h Pressure distribution on aerofoil at $t_e = 100$ ms for $T = 2.3$, $\delta_{ss} = 90^\circ$

Pressure Distribution at $t = 5$ ms

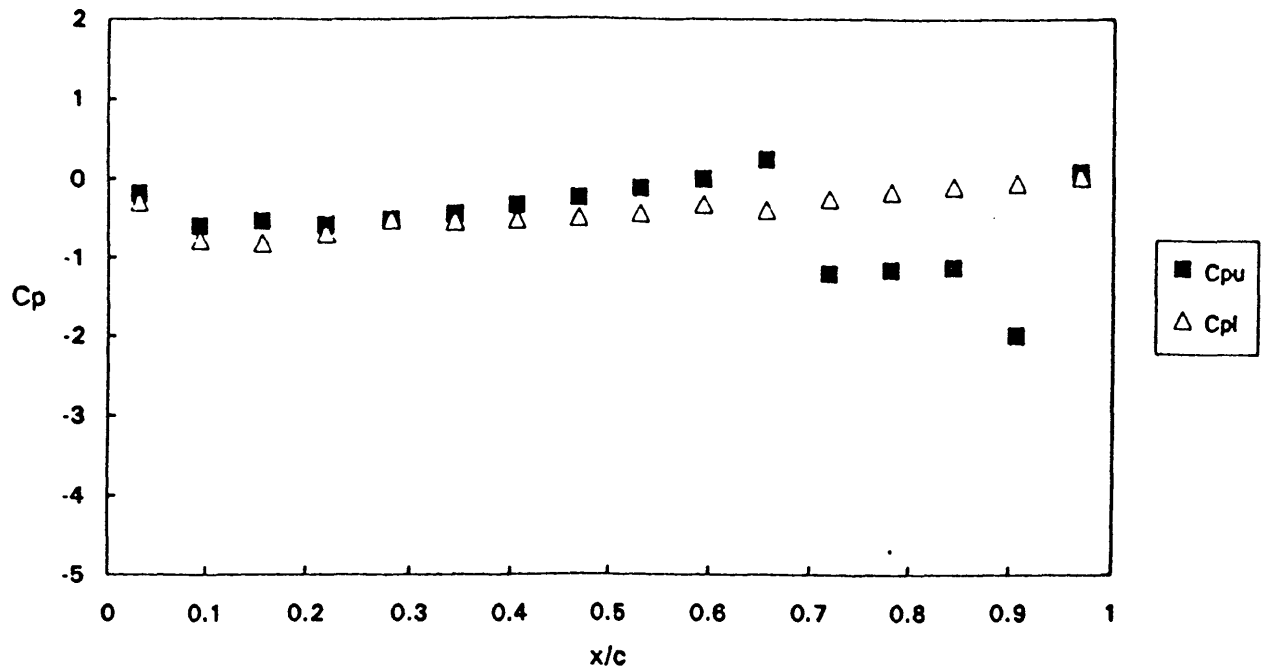


Fig 4.13a Pressure distribution on aerofoil at $t_e = 5$ ms for $T = 1$, $\delta_{ss} = 30^\circ$

Pressure Distribution at $t = 7.5$ ms

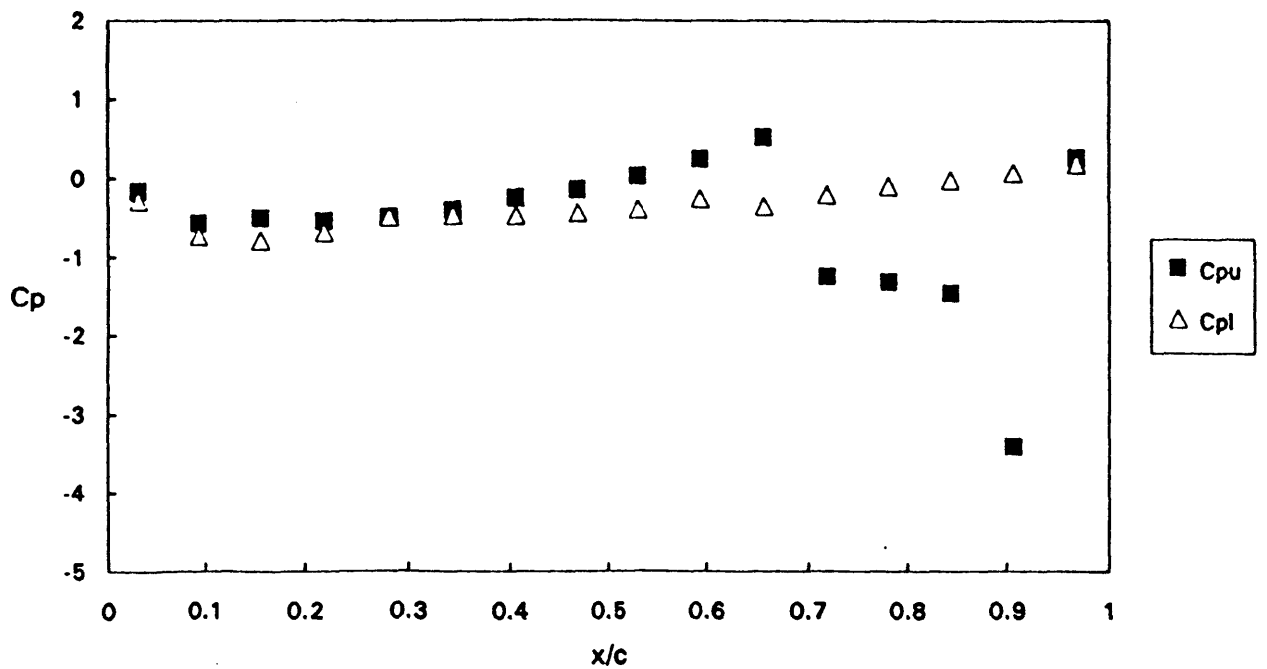


Fig 4.13b Pressure distribution on aerofoil at $t_e = 7.5$ ms for $T = 1$, $\delta_{ss} = 30^\circ$

Pressure Distribution at $t = 15$ ms

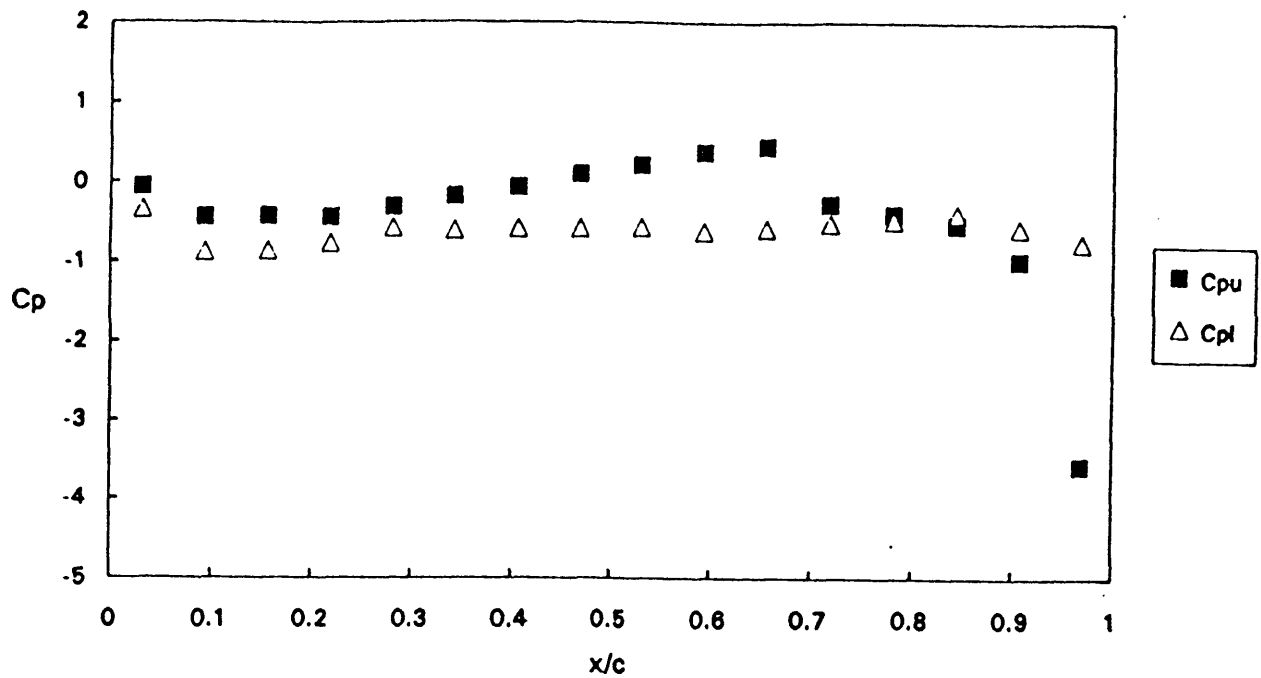


Fig 4.13c Pressure distribution on aerofoil at $t_e = 15$ ms for $T = 1$, $\delta_{ss} = 30^\circ$

Pressure Distribution at $t = 100$ ms

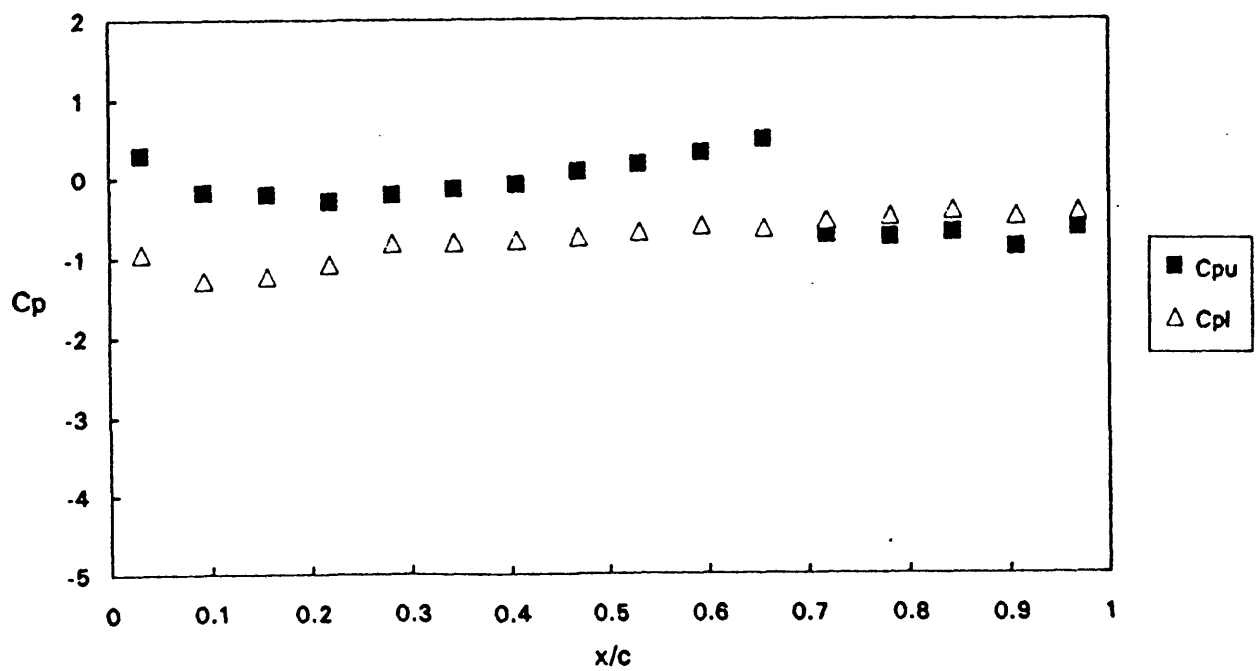
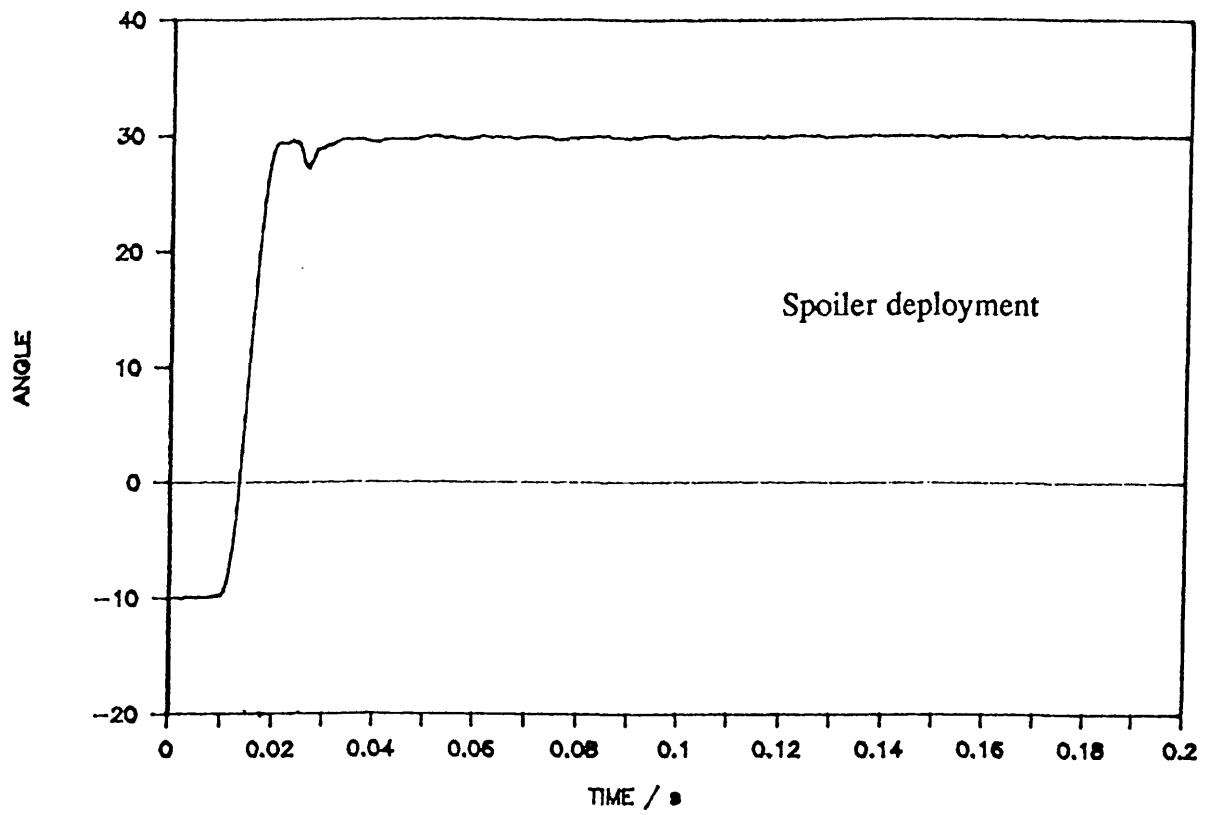


Fig 4.13d Pressure distribution on aerofoil at $t_e = 100$ ms for $T = 1$, $\delta_{ss} = 30^\circ$

ANGLE 1910



CL1910

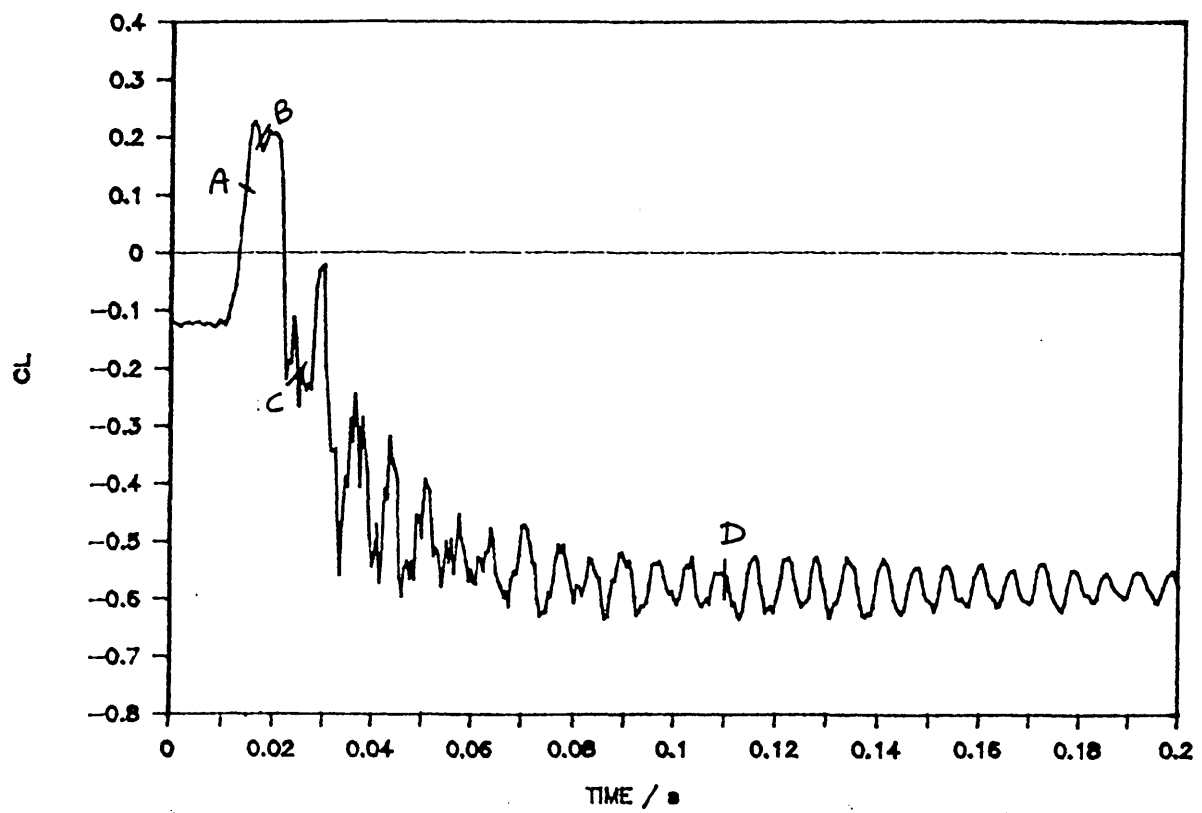


Fig 4.14 Time history of C_L for $T = 1$, $\delta_{ss} = 30^\circ$

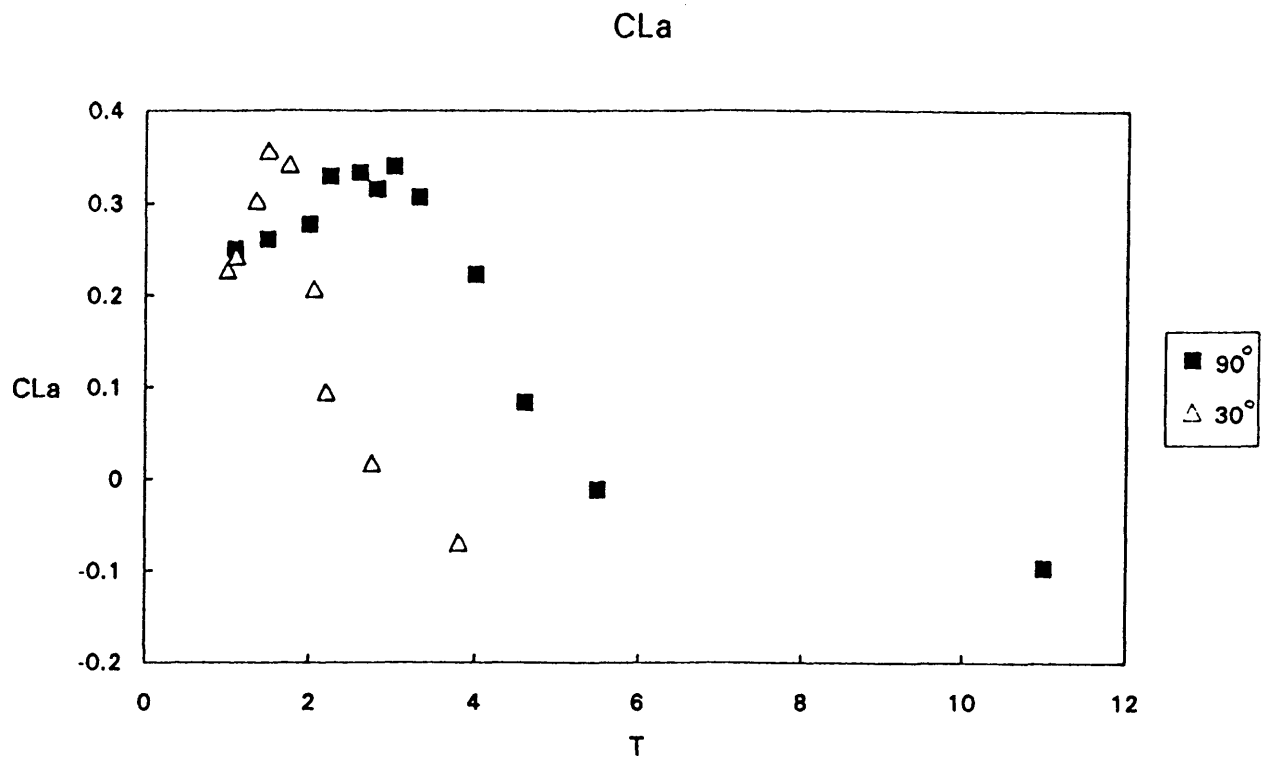


Fig 4.15 Variation of C_{L_a} with T

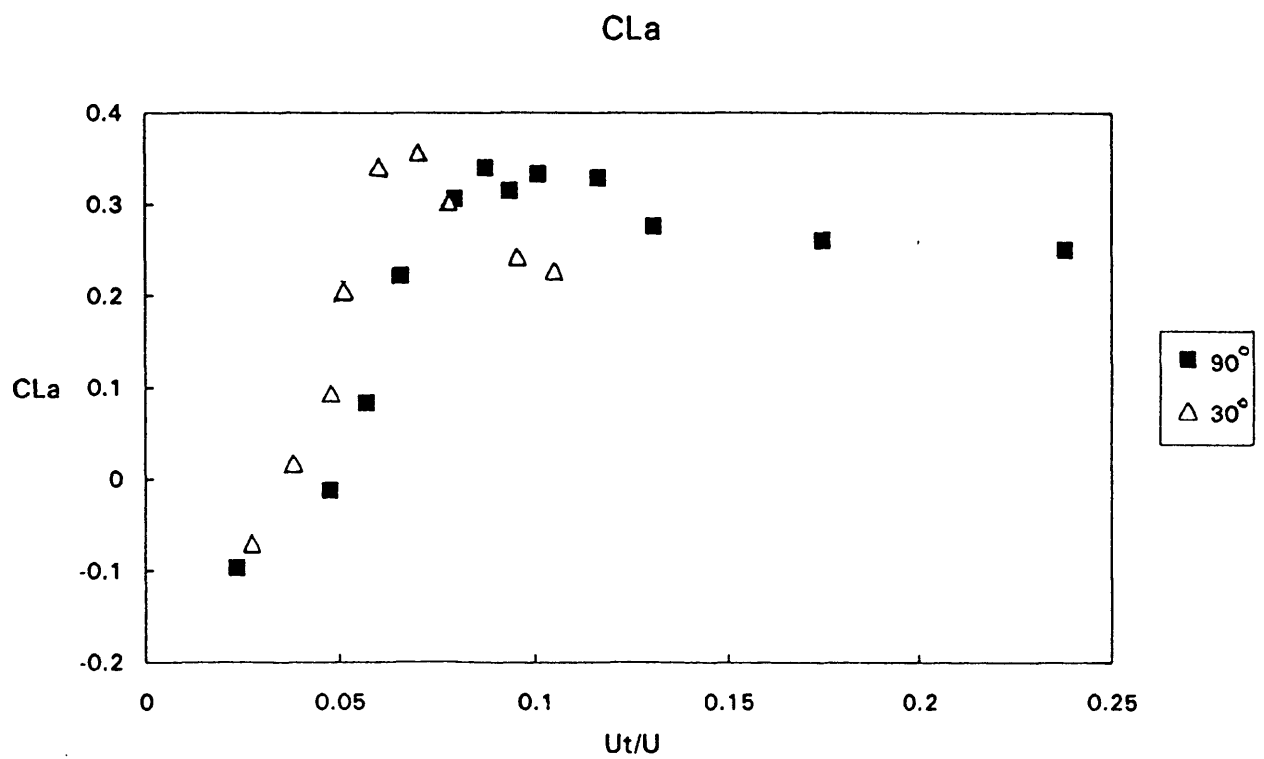


Fig 4.16 Variation of C_{L_a} with $\frac{U_{tip}}{U_o}$

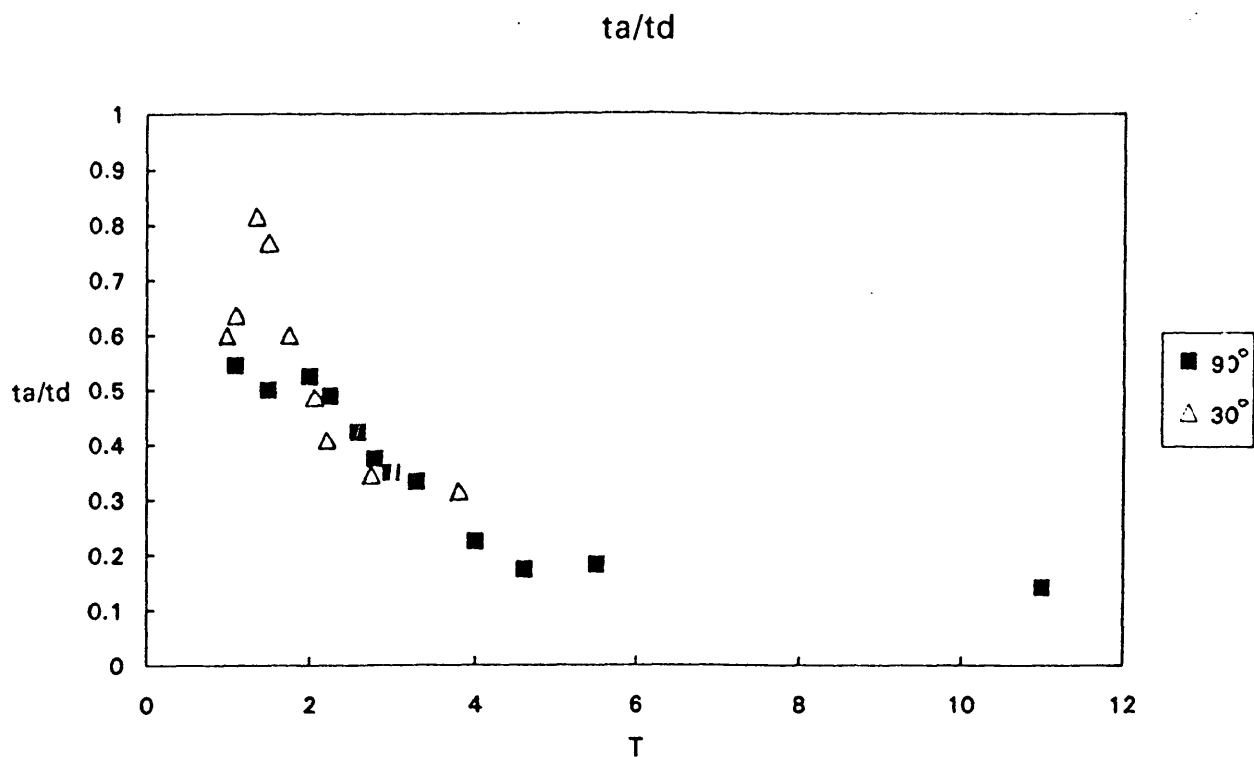


Fig 4.17 Variation of t_a/t_d with T

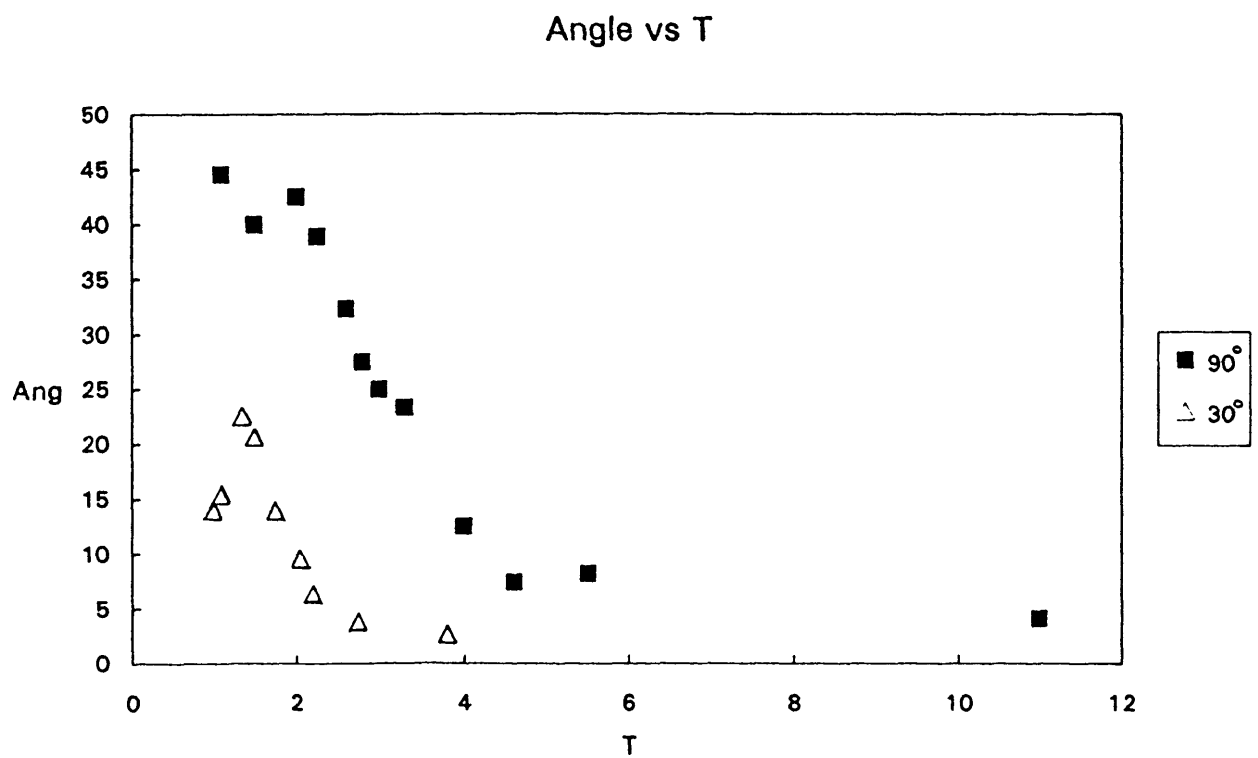


Fig 4.18 Variation of δ_a with T

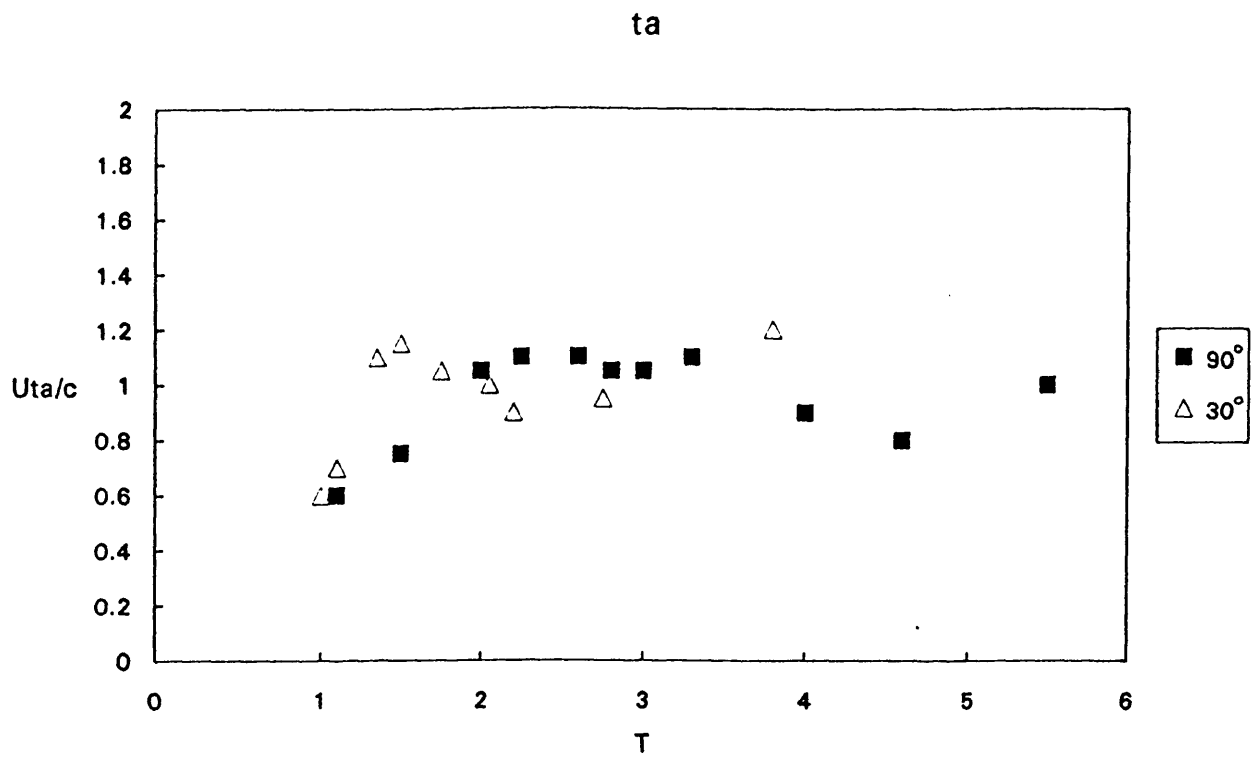


Fig 4.19 Variation of $\frac{U_{ot_a}}{c}$ with T

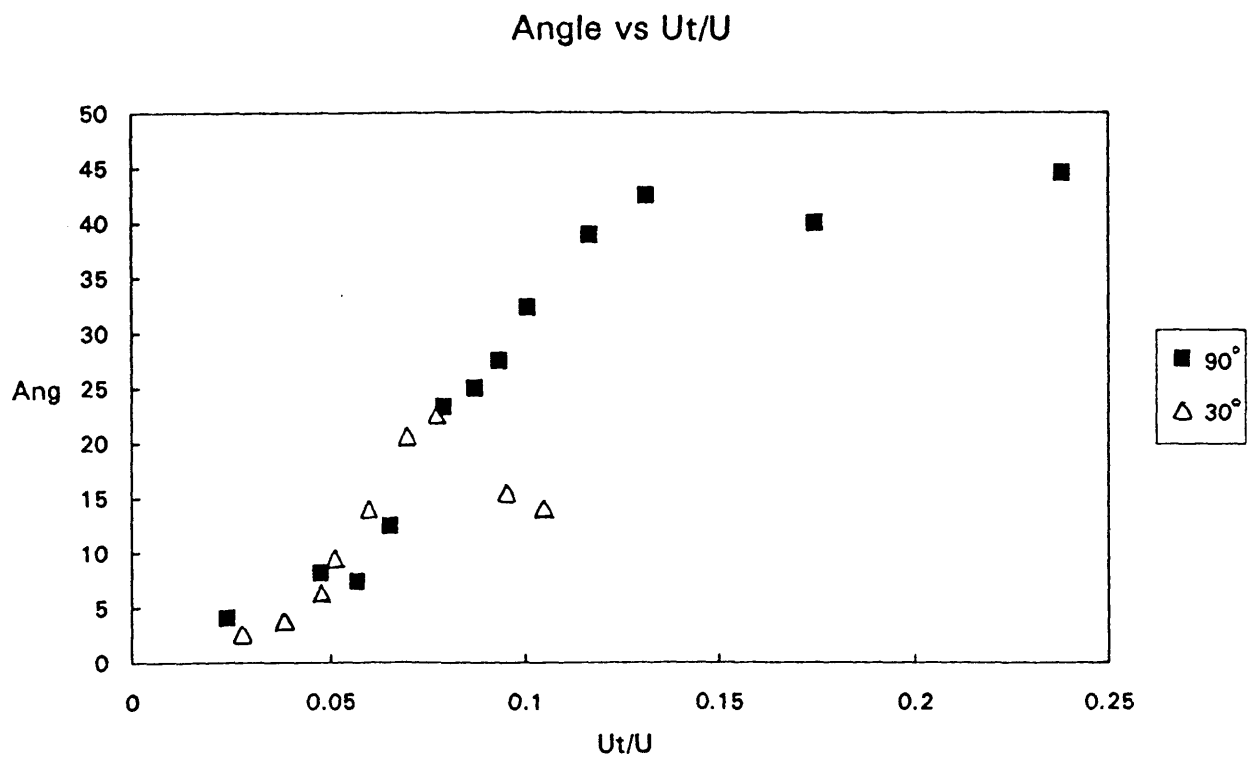


Fig 4.20 Variation of δ_a with $\frac{U_{tip}}{U_o}$

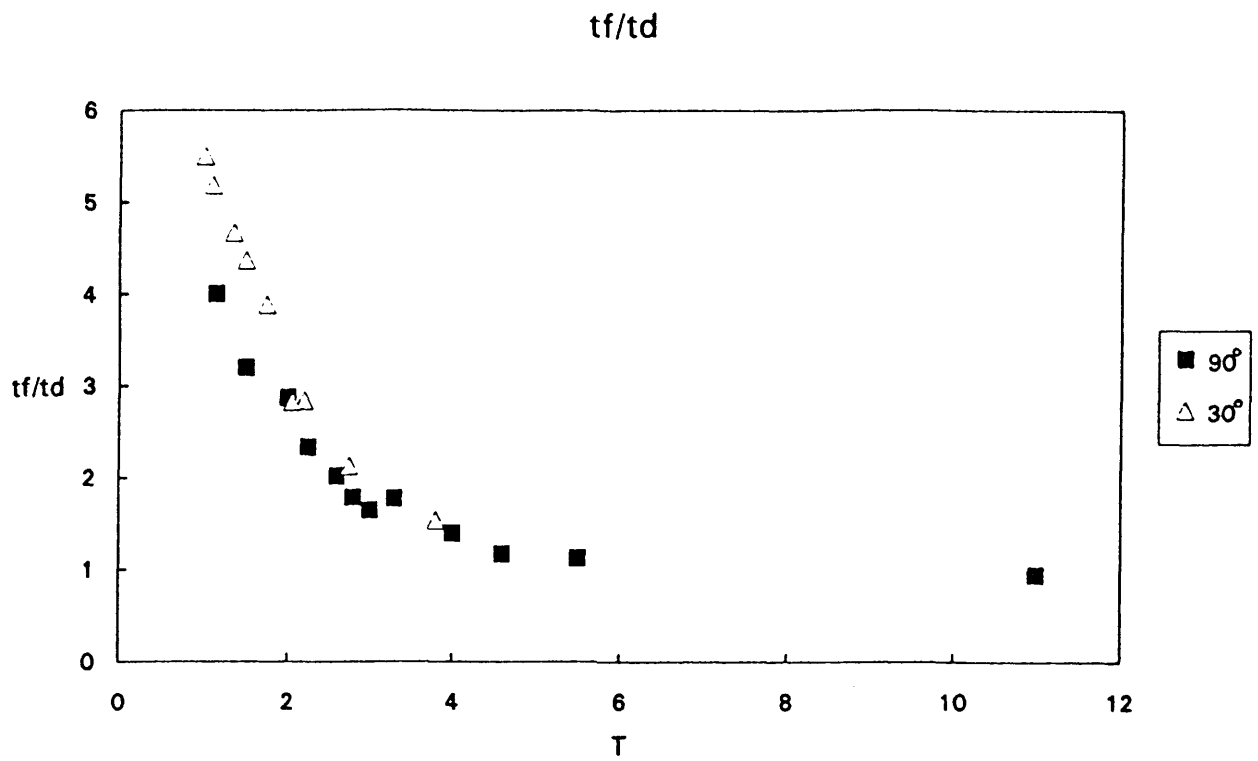


Fig 4.21 Variation of t_f/t_d with T

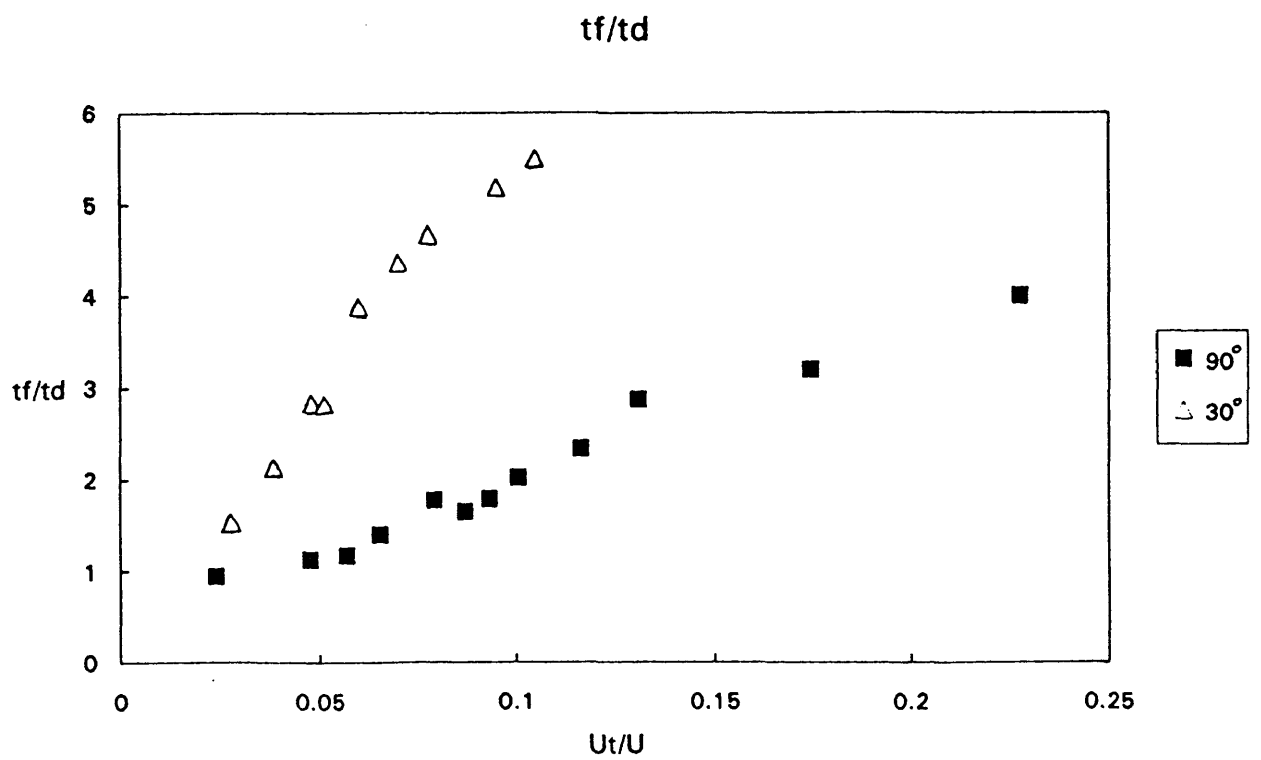


Fig 4.22 Variation of t_f/t_d with $\frac{U_{tip}}{U_o}$

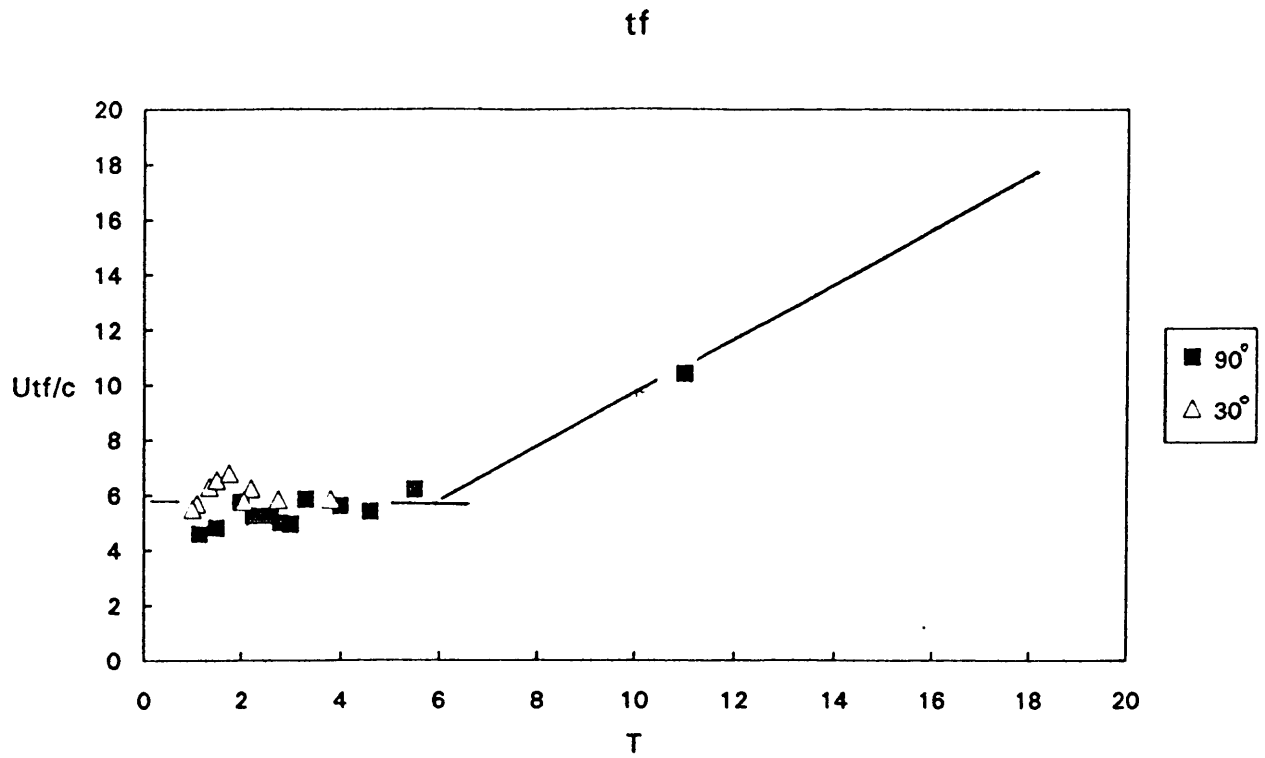


Fig 4.23a Variation of $\frac{U_{0t}f}{c}$ with T

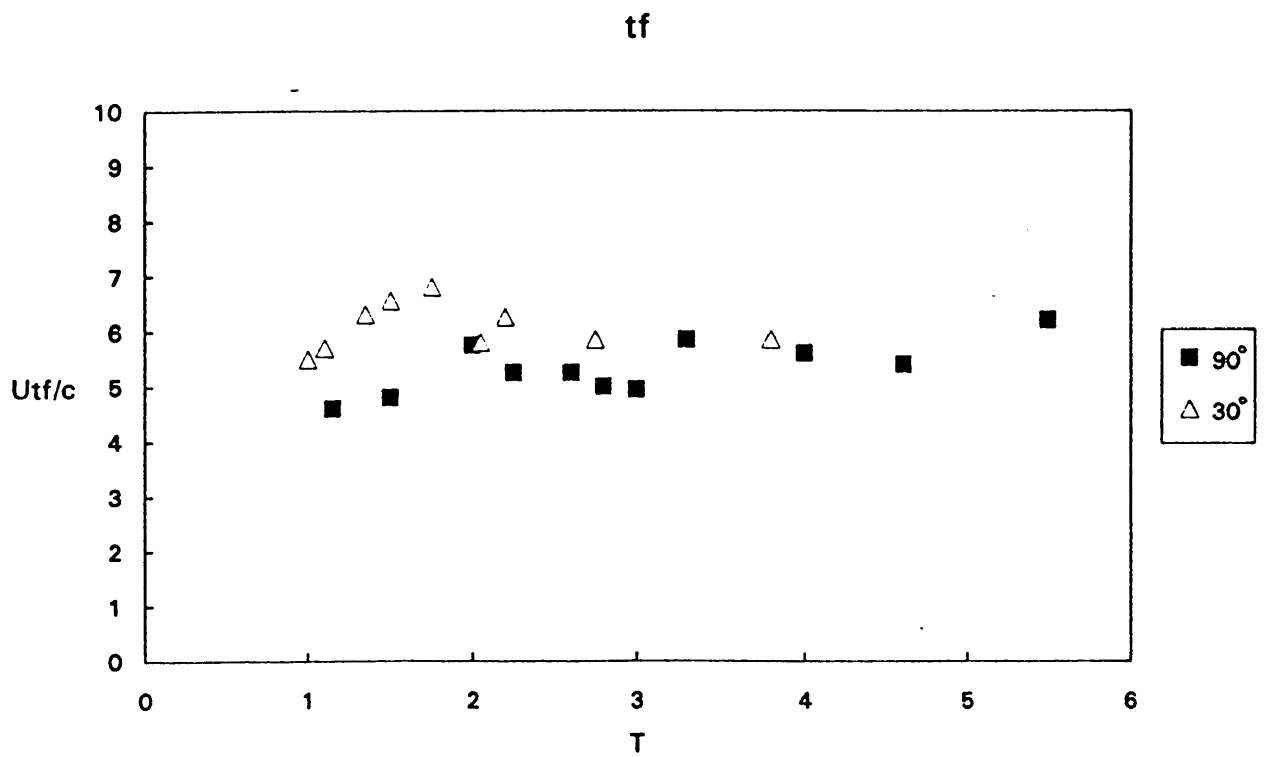
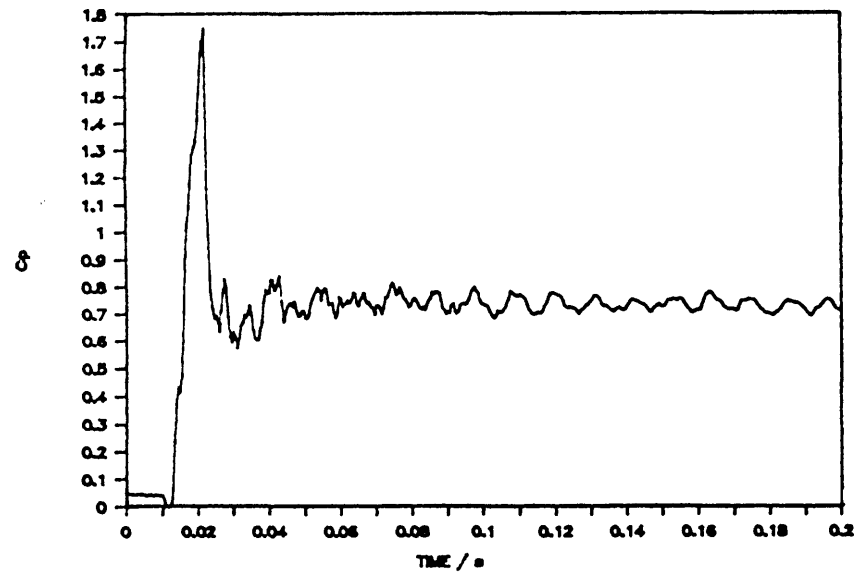
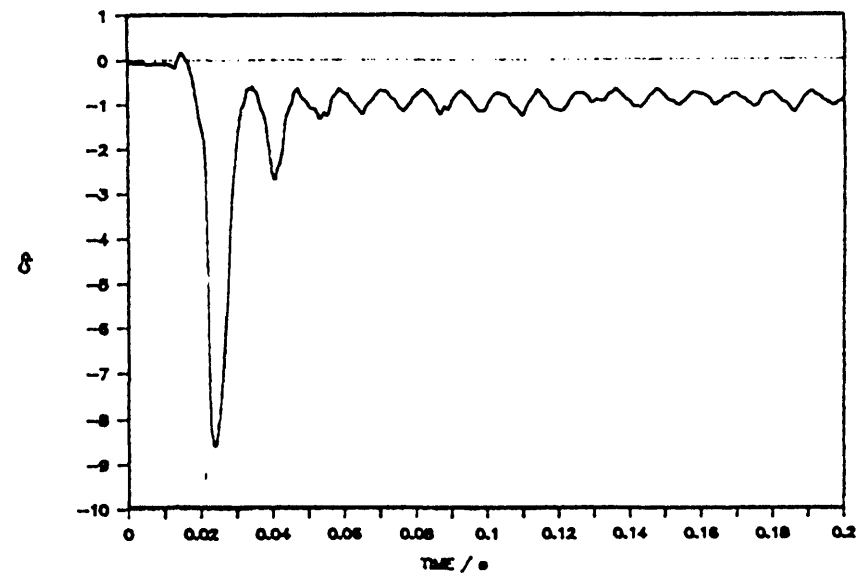
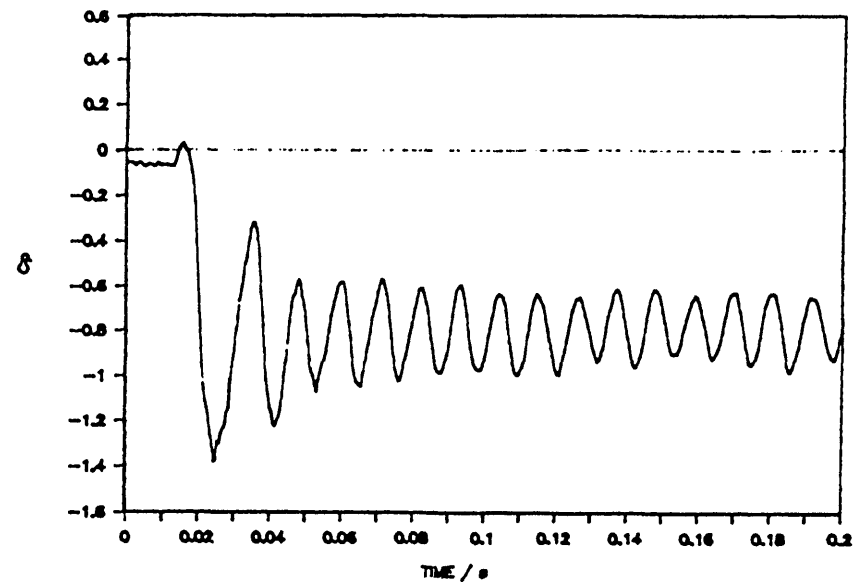


Fig 4.23b Variation of $\frac{U_{0t}f}{c}$ with T for $T = 0$ to 6

CpC2260

Fig 4.24a Time histories of C_{p11} for $T = 1.1$, $\delta_{ss} = 90^\circ$

CpA2260

Fig 4.24b Time histories of C_{p16} for $T = 1.1$, $\delta_{ss} = 90^\circ$ Fig 4.24c Time histories of C_{p17} for $T = 1.1$, $\delta_{ss} = 90^\circ$

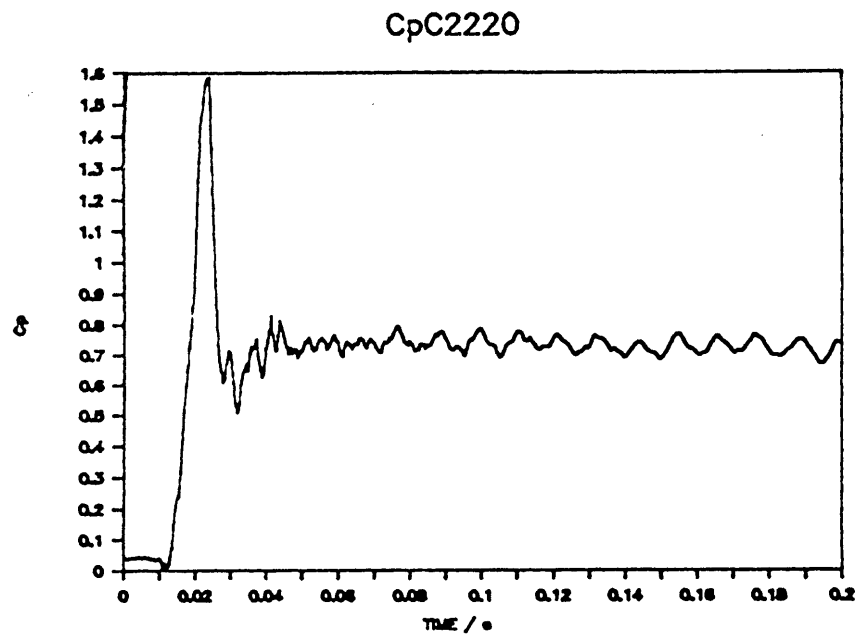


Fig 4.25a Time histories of C_{p11} for $T = 1.5$, $\delta_{ss} = 90^\circ$

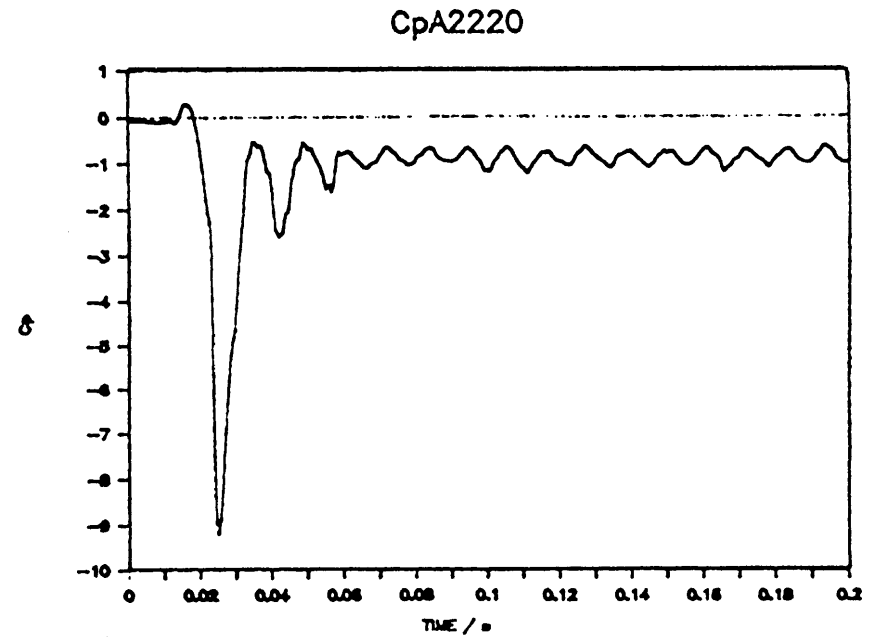


Fig 4.25b Time histories of C_{p16} for $T = 1.5$, $\delta_{ss} = 90^\circ$

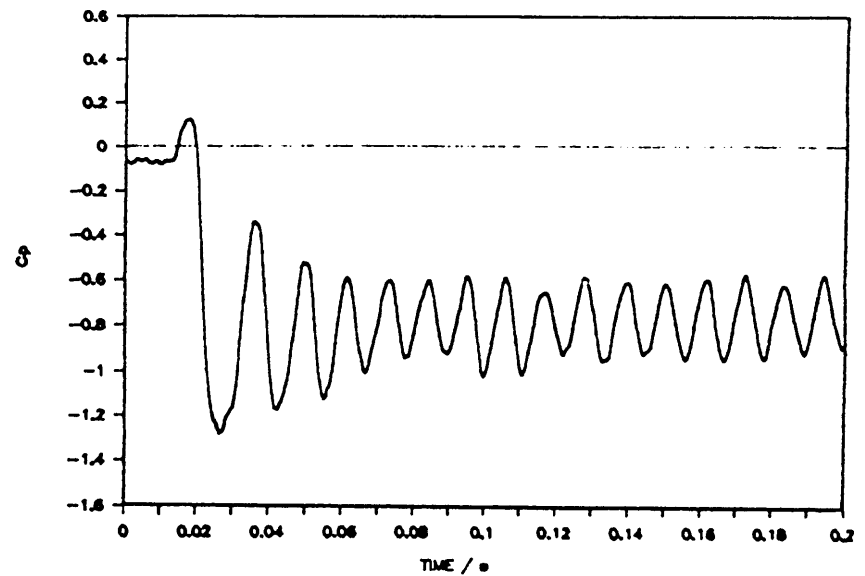
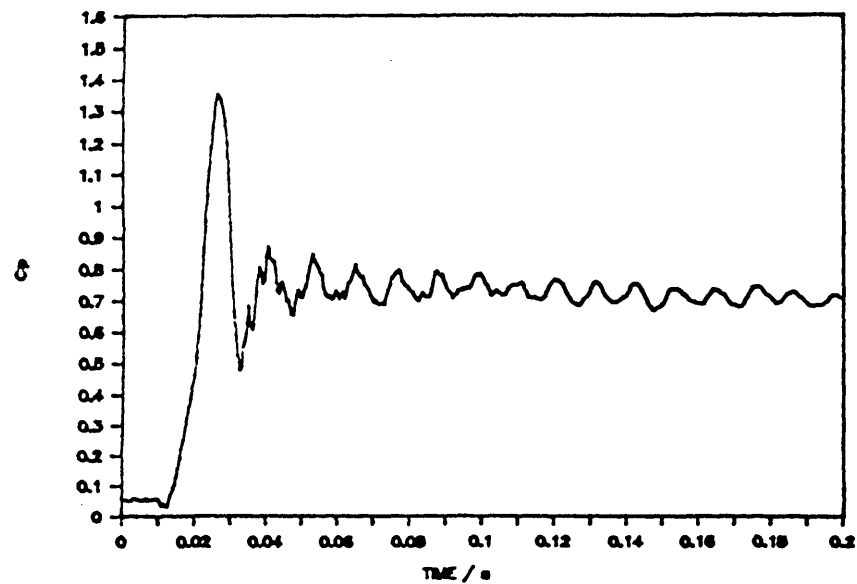
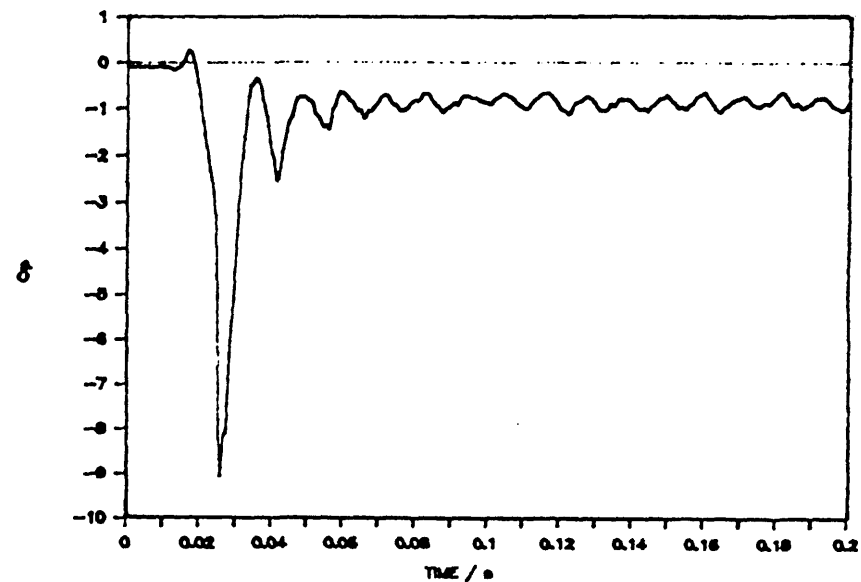
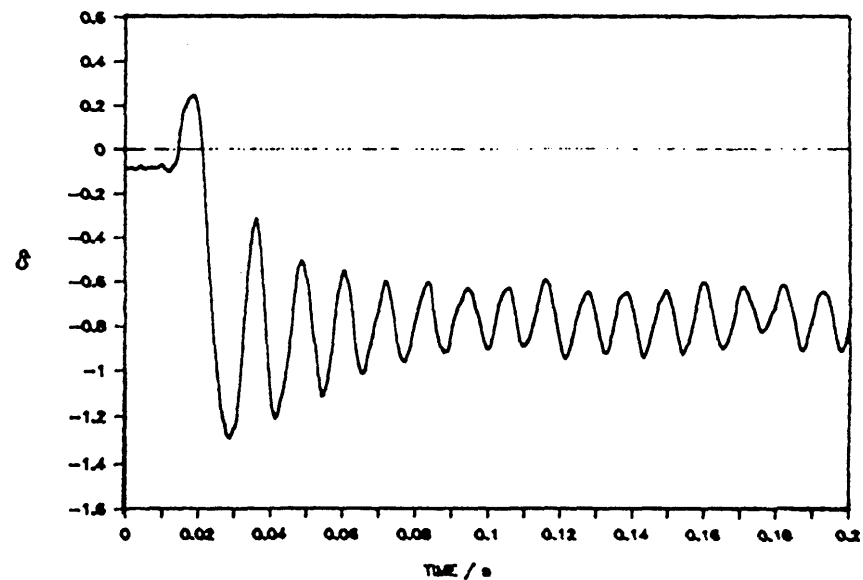


Fig 4.25c Time histories of C_{p17} for $T = 1.5$, $\delta_{ss} = 90^\circ$

CpC2230

Fig 4.26a Time histories of C_{p11} for $T = 2$, $\delta_{ss} = 90^\circ$

CpA2230

Fig 4.26b Time histories of C_{p16} for $T = 2$, $\delta_{ss} = 90^\circ$ Fig 4.26c Time histories of C_{p17} for $T = 2$, $\delta_{ss} = 90^\circ$

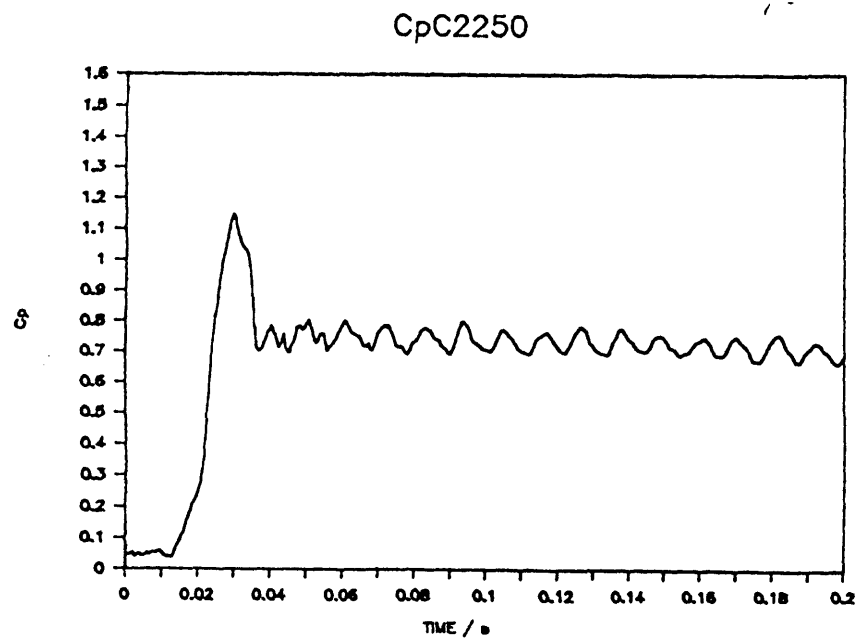


Fig 4.27a Time histories of C_{p11} for $T = 2.5$, $\delta_{ss} = 90^\circ$

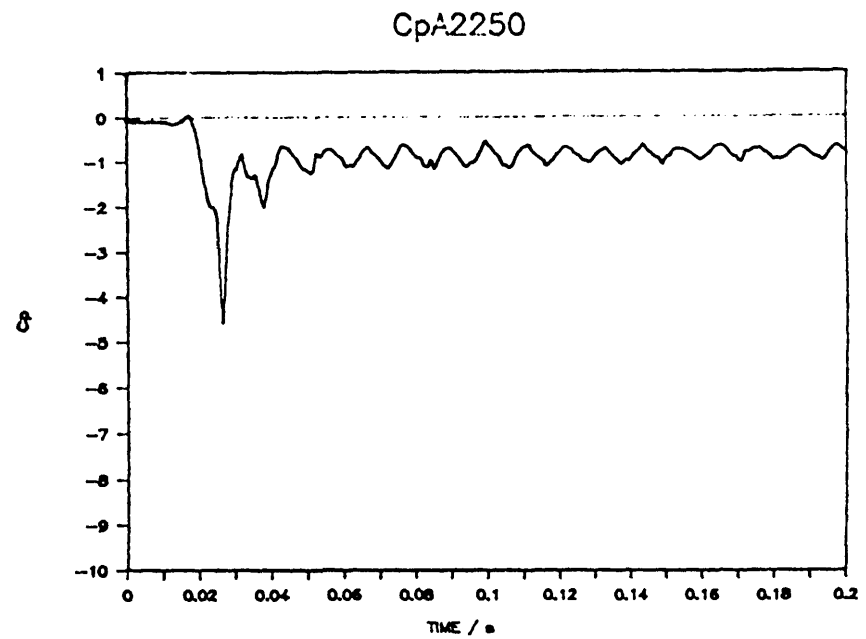


Fig 4.27b Time histories of C_{p16} for $T = 2.5$, $\delta_{ss} = 90^\circ$

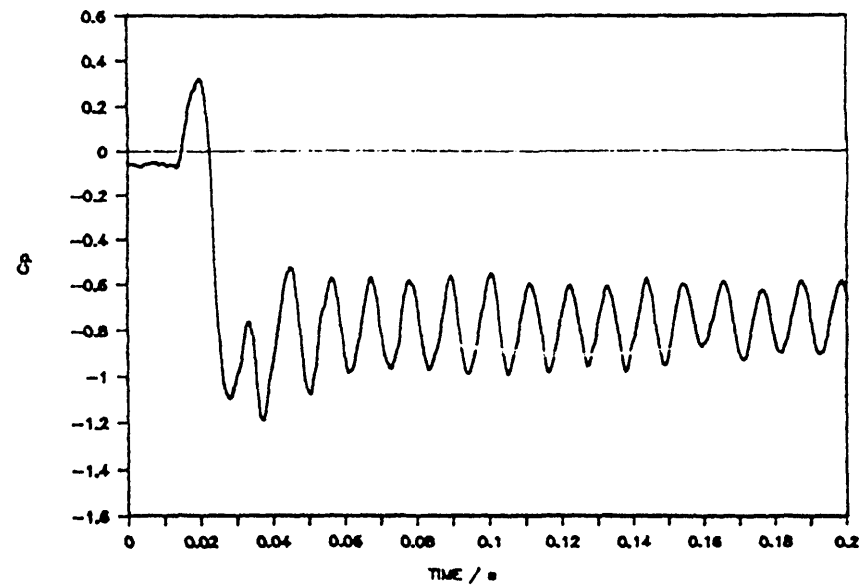


Fig 4.27c Time histories of C_{p17} for $T = 2.5$, $\delta_{ss} = 90^\circ$

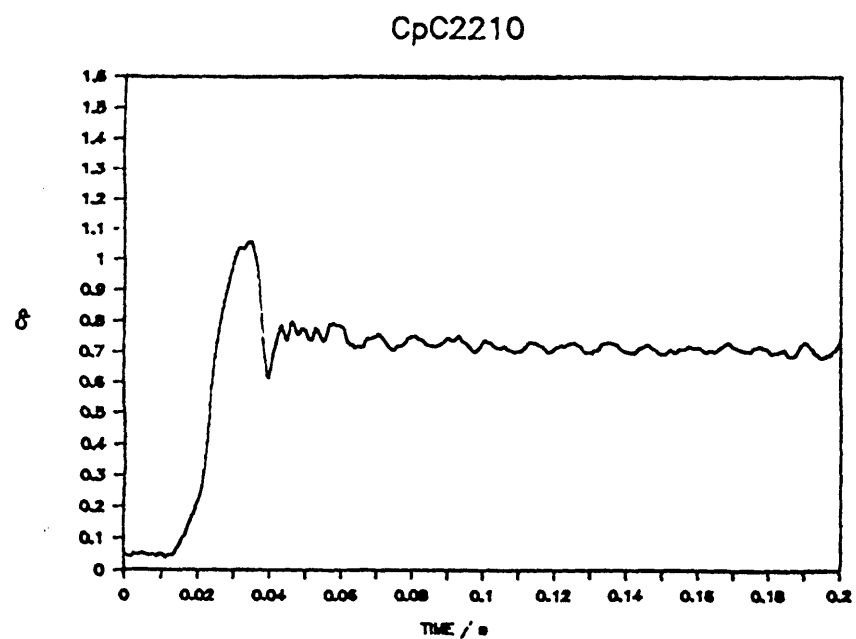


Fig 4.28a Time histories of C_{p11} for $T = 2.8$, $\delta_{ss} = 90^\circ$

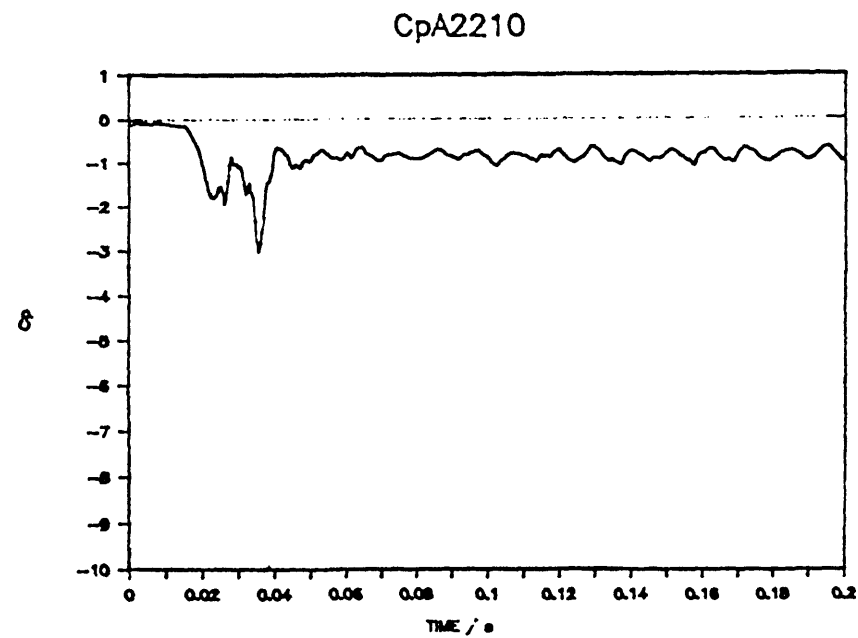


Fig 4.28b Time histories of C_{p16} for $T = 2.8$, $\delta_{ss} = 90^\circ$

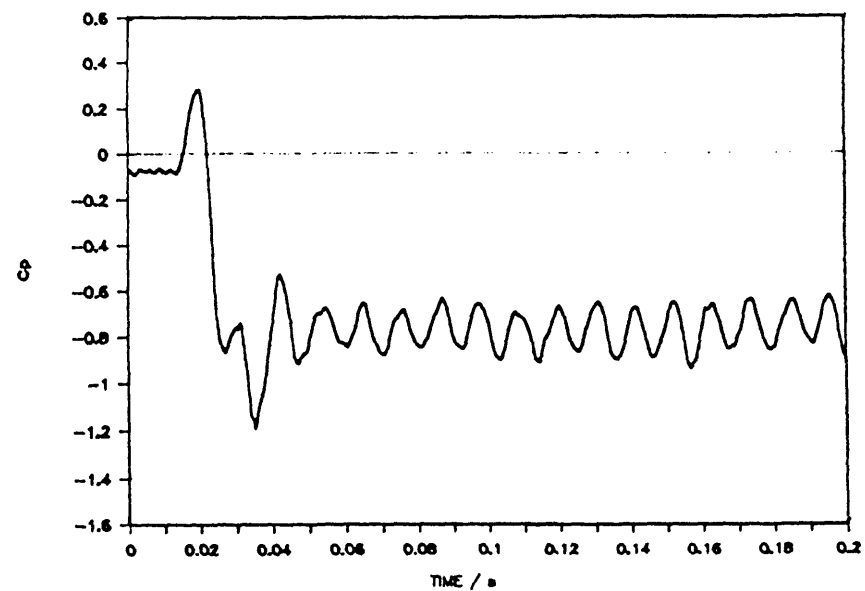


Fig 4.28c Time histories of C_{p17} for $T = 2.8$, $\delta_{ss} = 90^\circ$

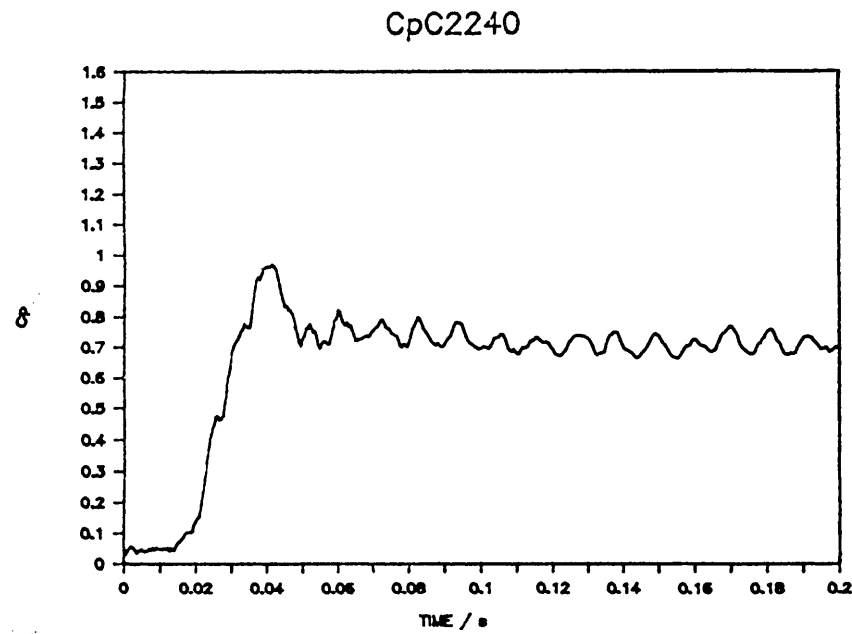


Fig 4.29a Time histories of C_{p11} for $T = 4$, $\delta_{ss} = 90^\circ$

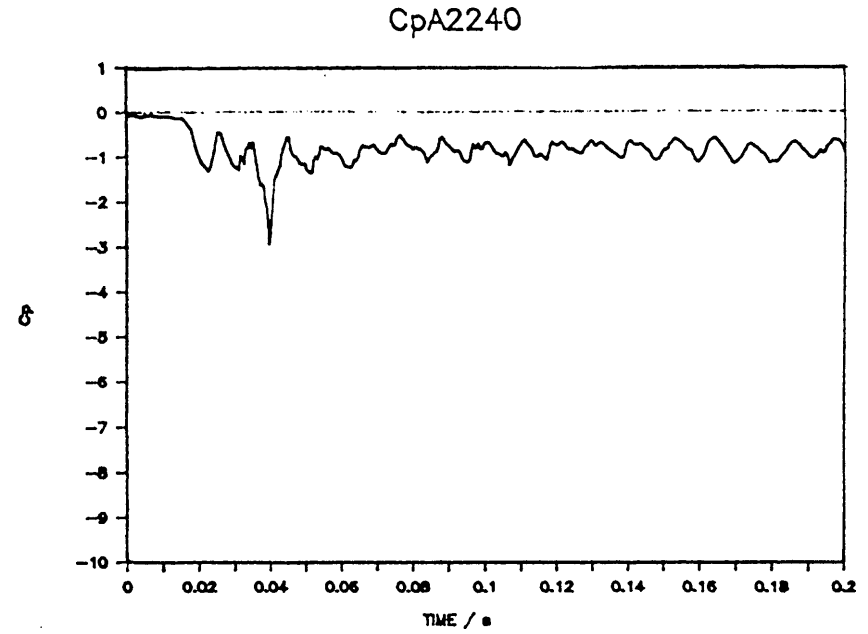


Fig 4.29b Time histories of C_{p16} for $T = 4$, $\delta_{ss} = 90^\circ$

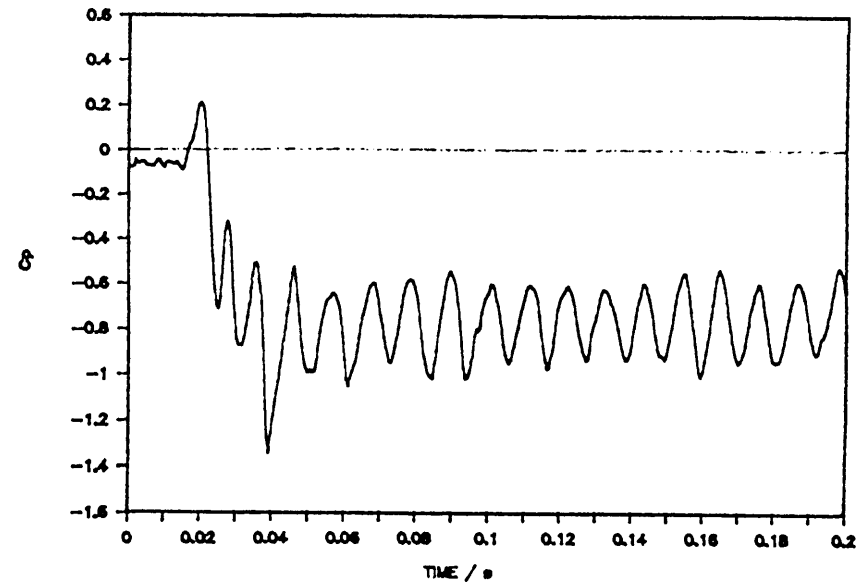


Fig 4.29c Time histories of C_{p17} for $T = 4$, $\delta_{ss} = 90^\circ$

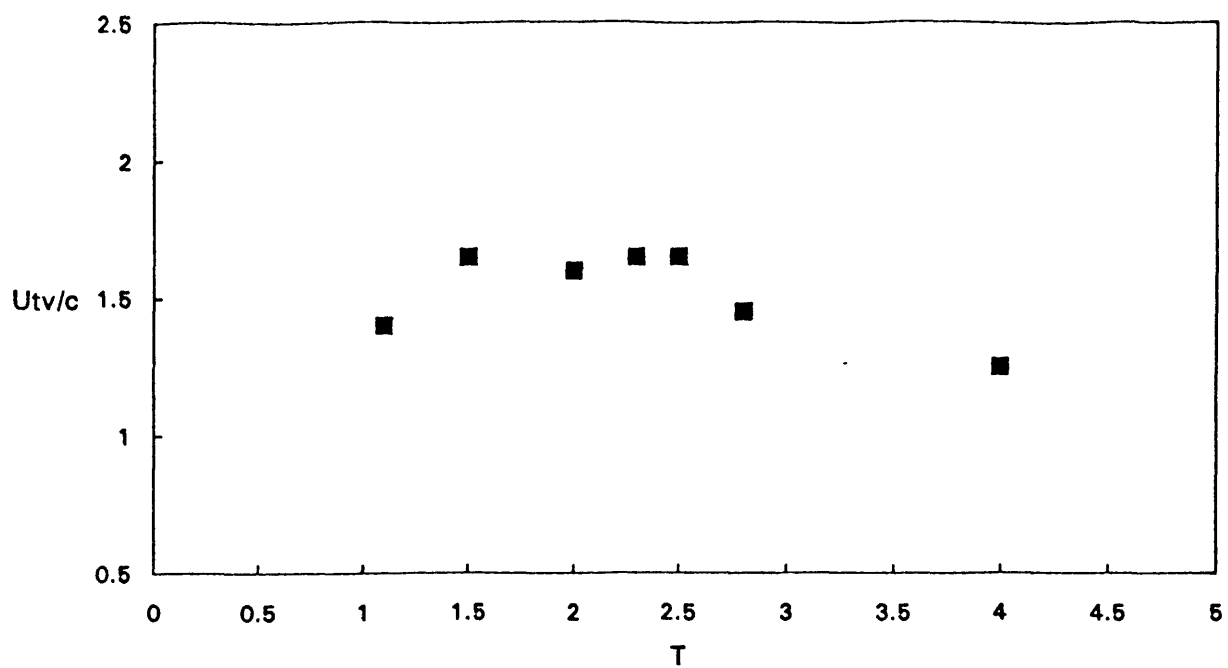


Fig 4.30 Variation of $U_0 t_v / c$ with T

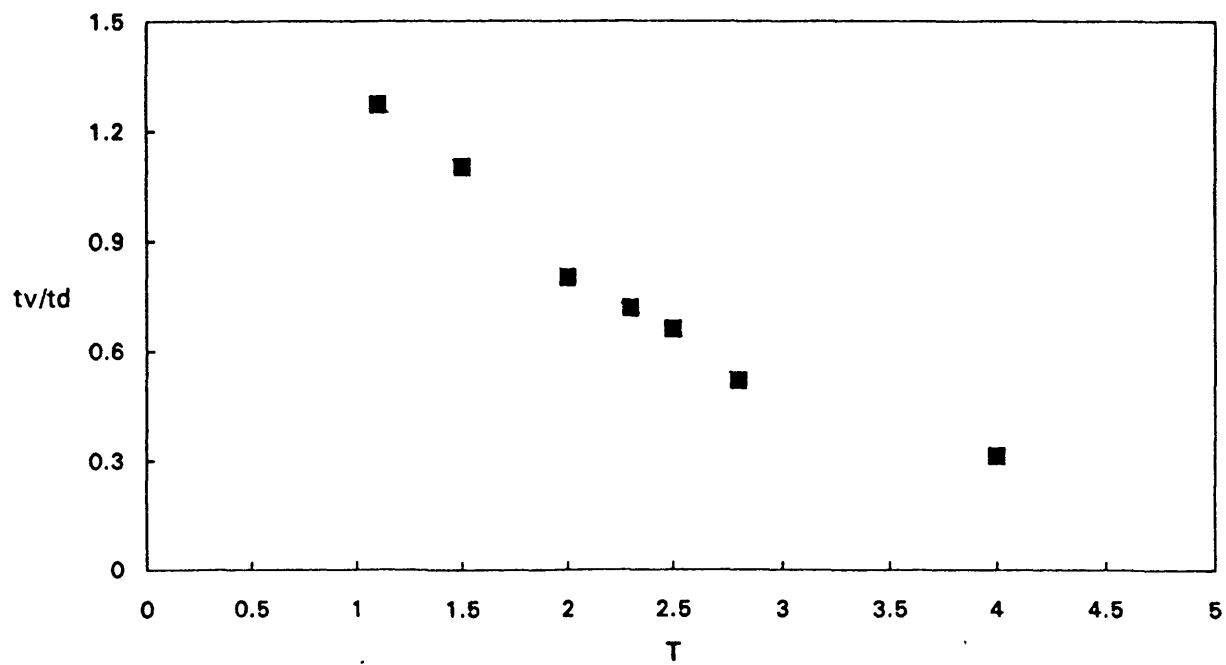


Fig 4.31 Variation of t_v / t_d with T

U - Velocity Profile

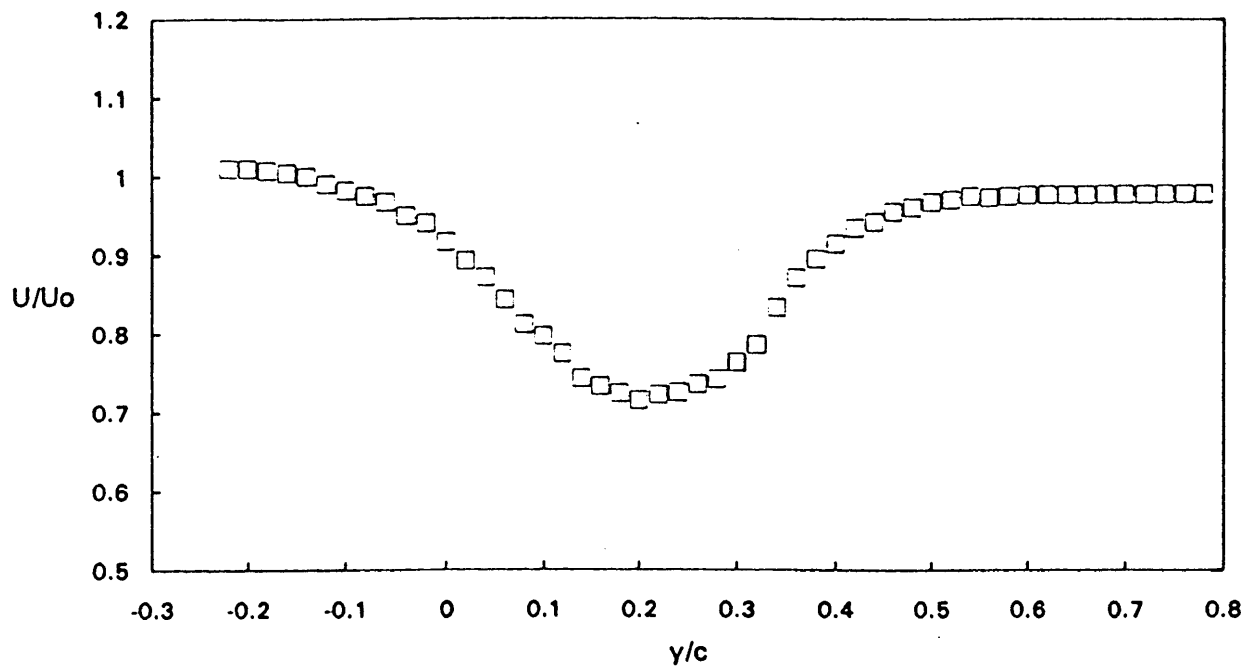


Fig 5.1a Variation of u-velocity with y/c

V- Velocity Profile

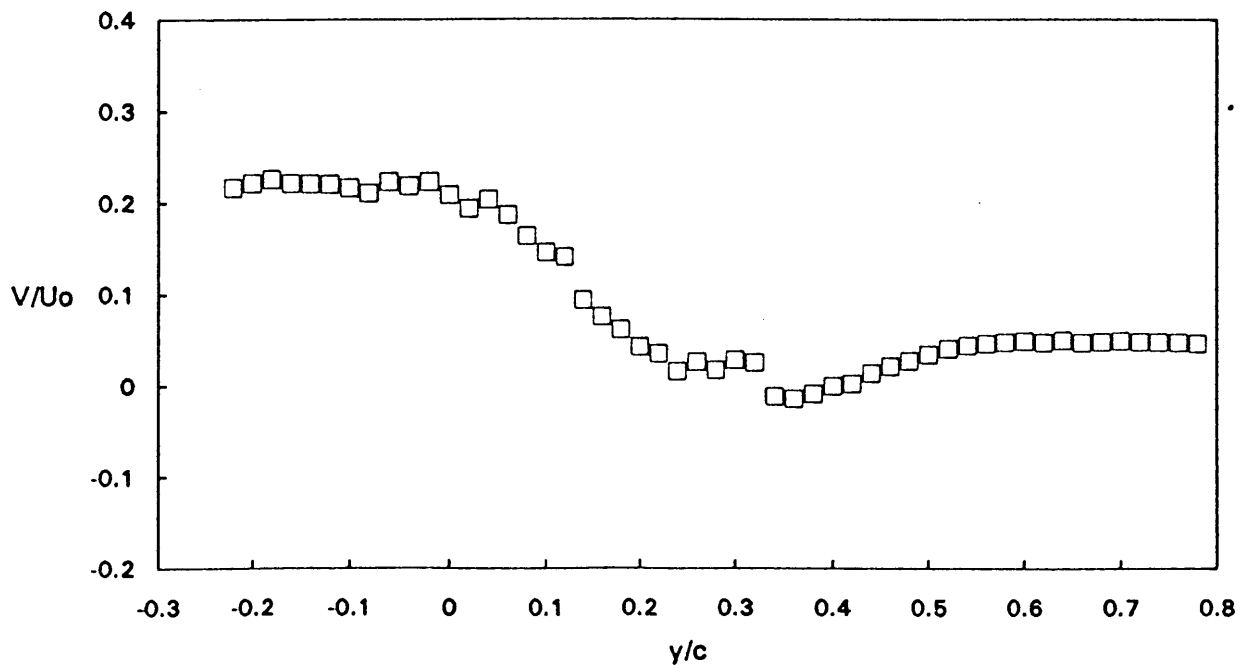


Fig 5.1b Variation of v-velocity with y/c

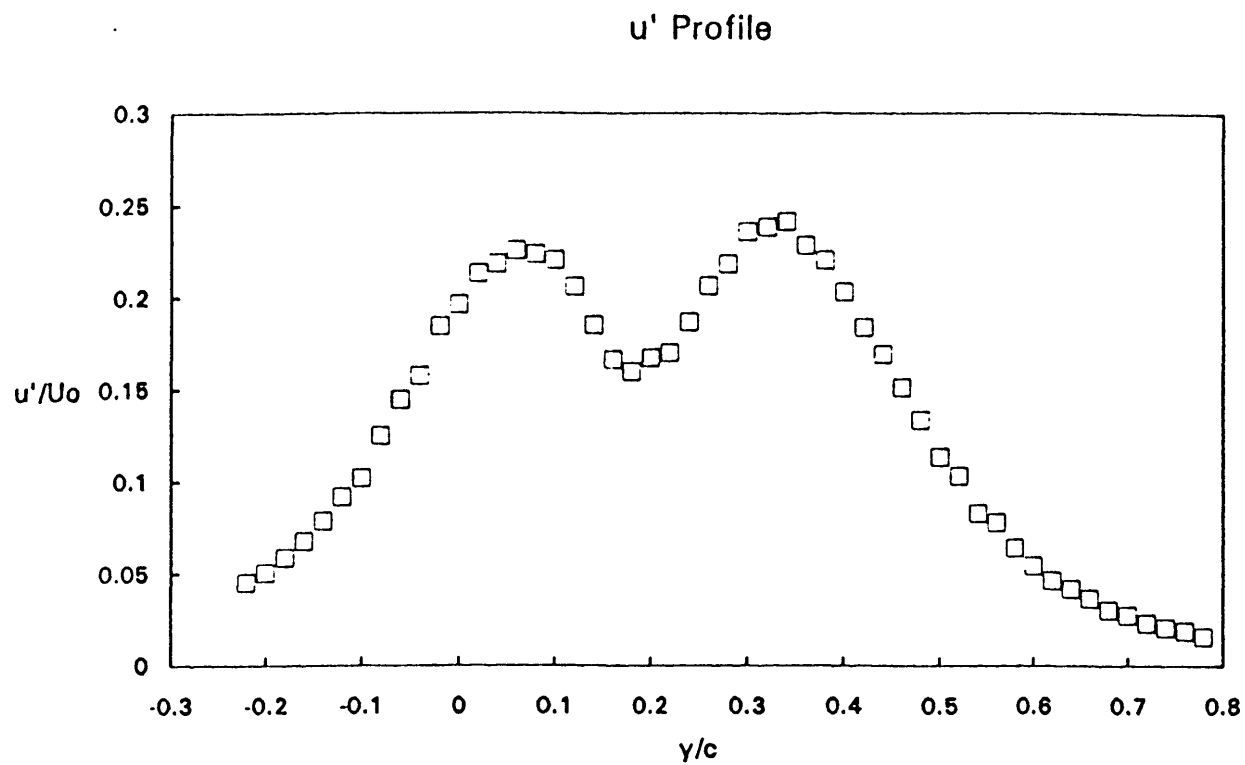


Fig 5.2a Variation of u' -velocity with y/c

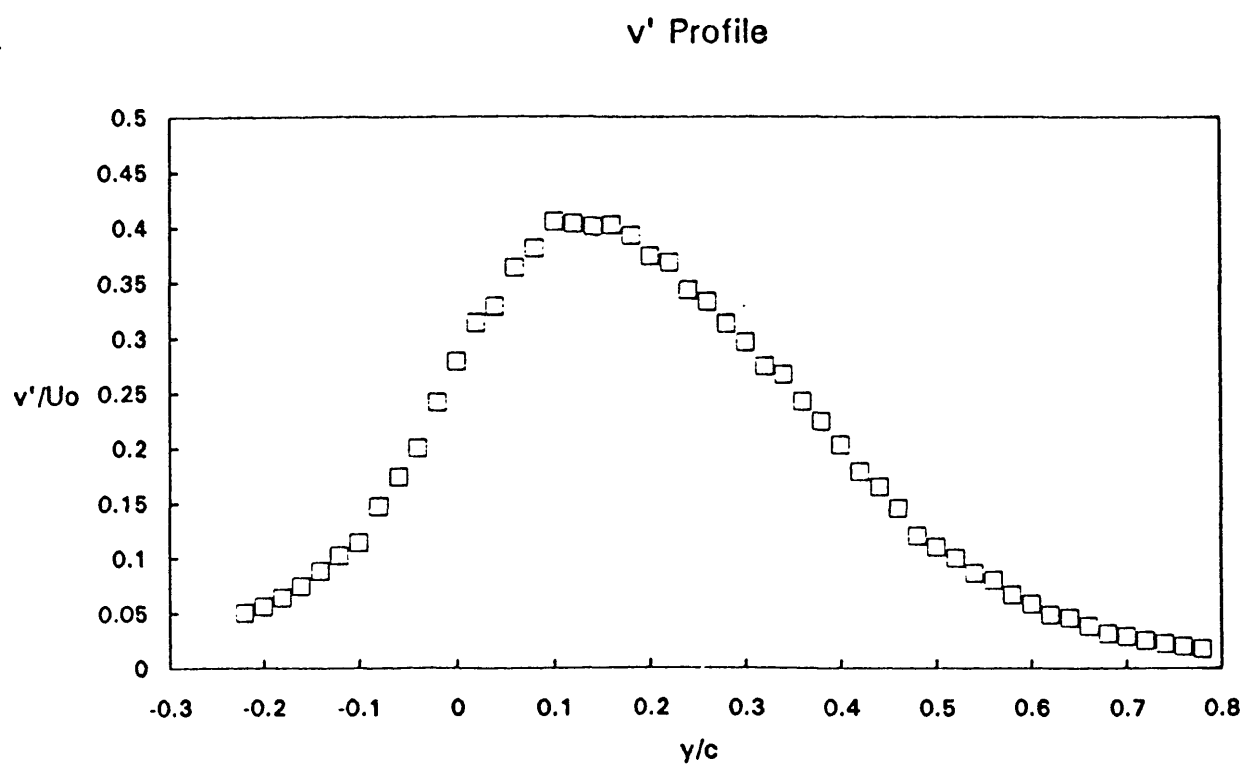


Fig 5.2b Variation of v' -velocity with y/c

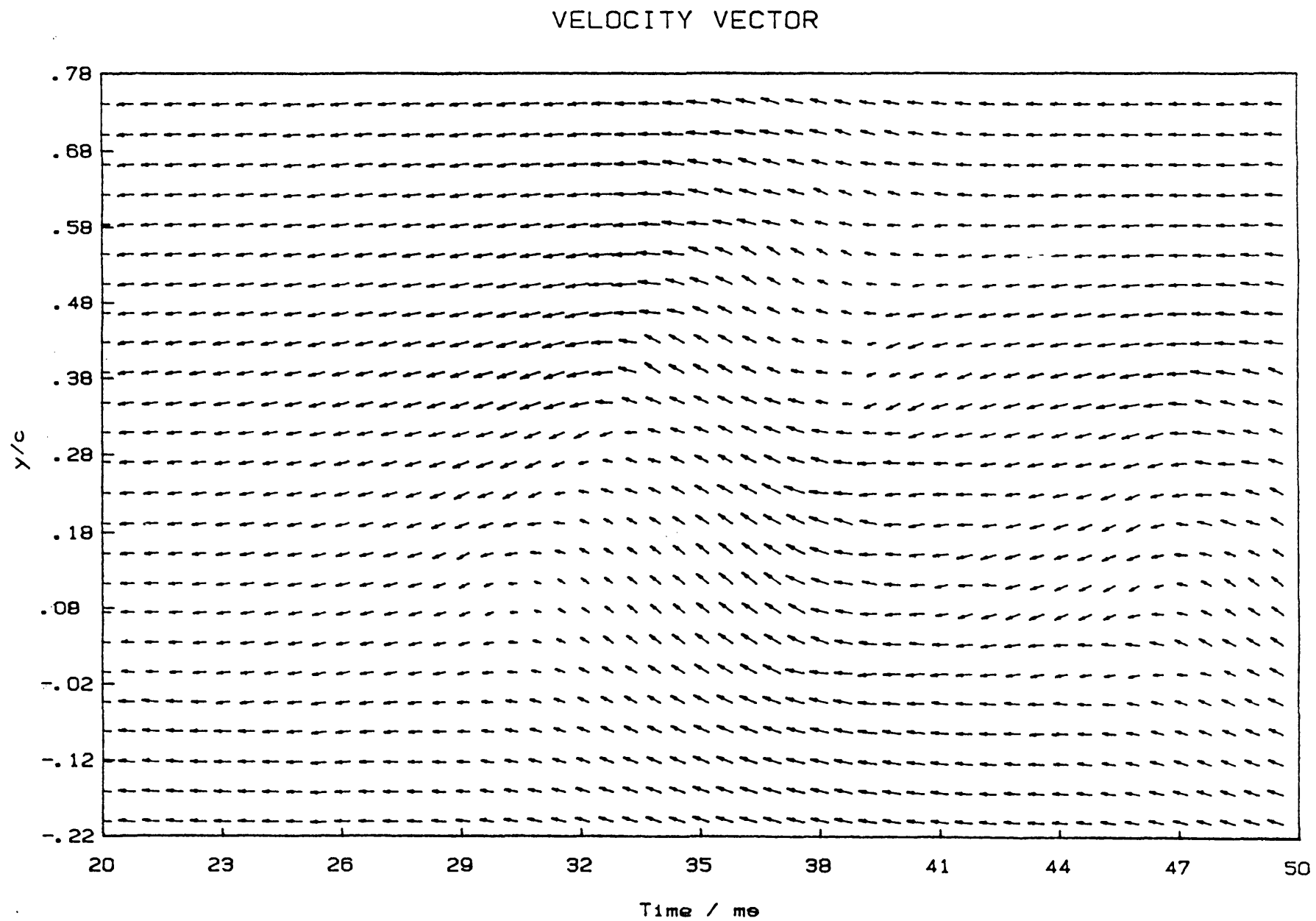


Fig 5.3a Velocity vector field for $T = 1.1$, from $t = 20$ to 50 ms

VELOCITY VECTOR

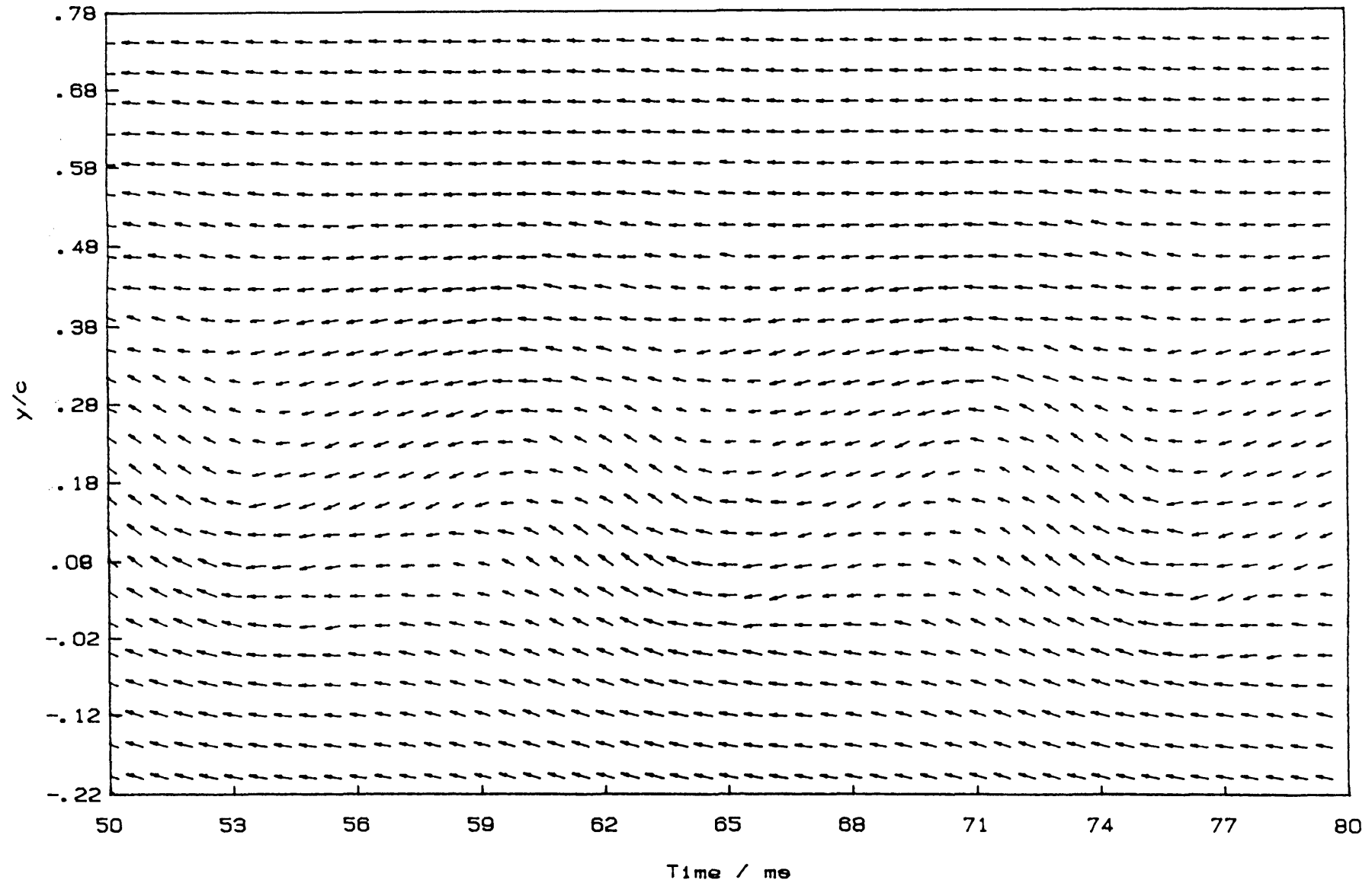


Fig 5.3b Velocity vector field for $T = 1.1$, from $t = 50$ to 80 ms

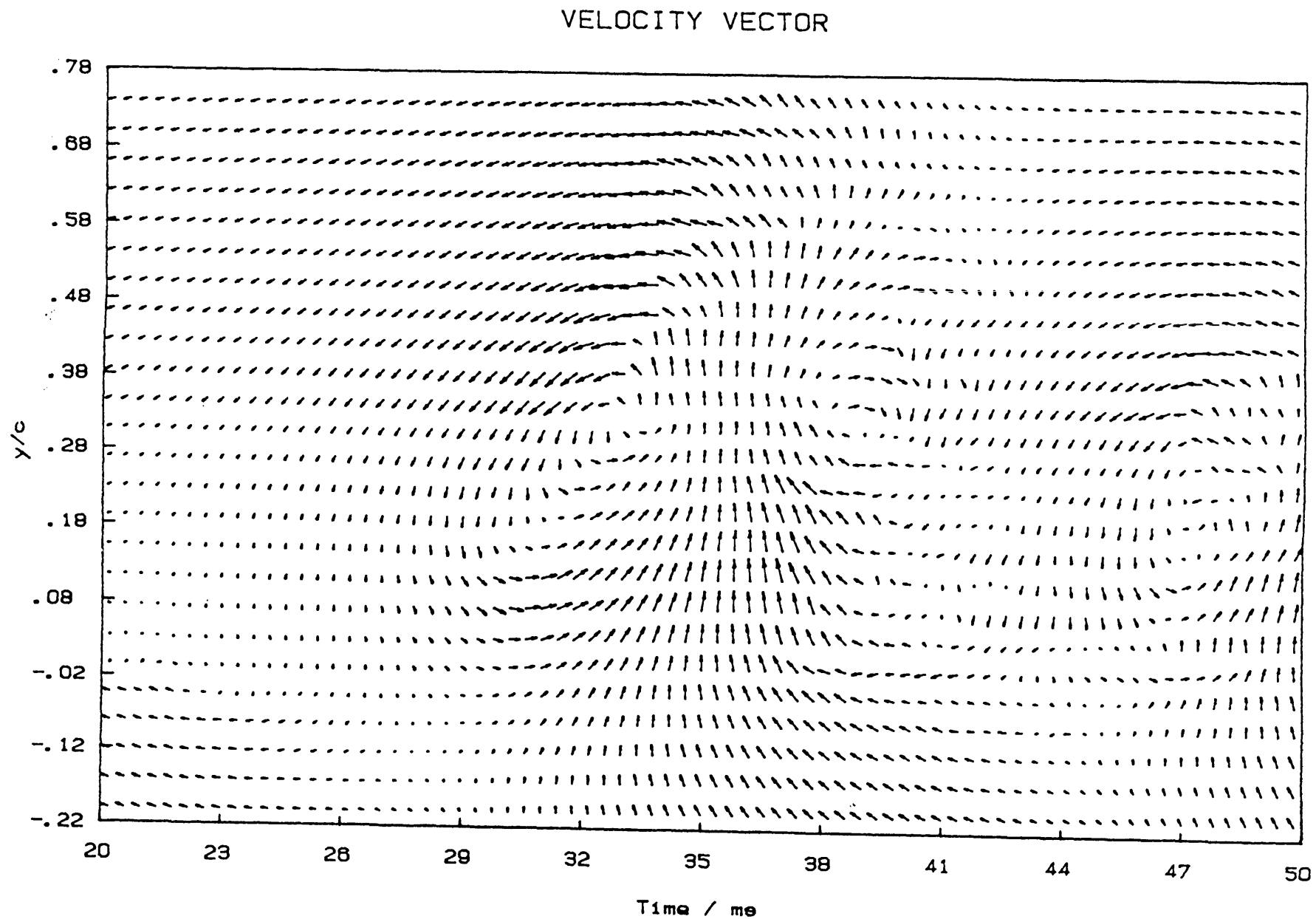


Fig 5.4a Velocity vector field for $T = 1.1$, from $t = 20$ to 50 ms, with U_{ref} removed

VELOCITY VECTOR

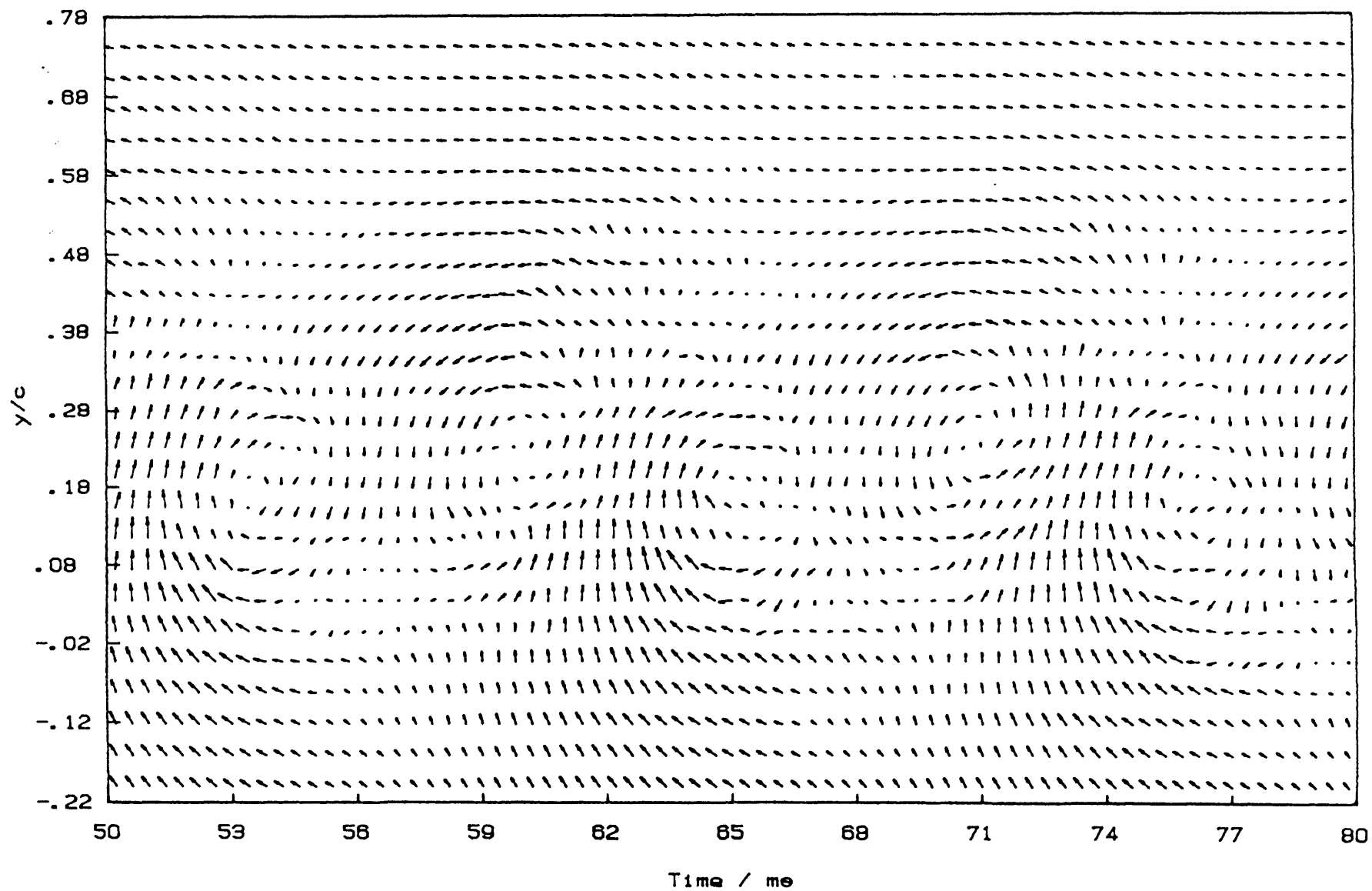


Fig 5.4b Velocity vector field for $T = 1.1$, from $t = 50$ to 80 ms, with U_{ref} removed

VELOCITY VECTOR

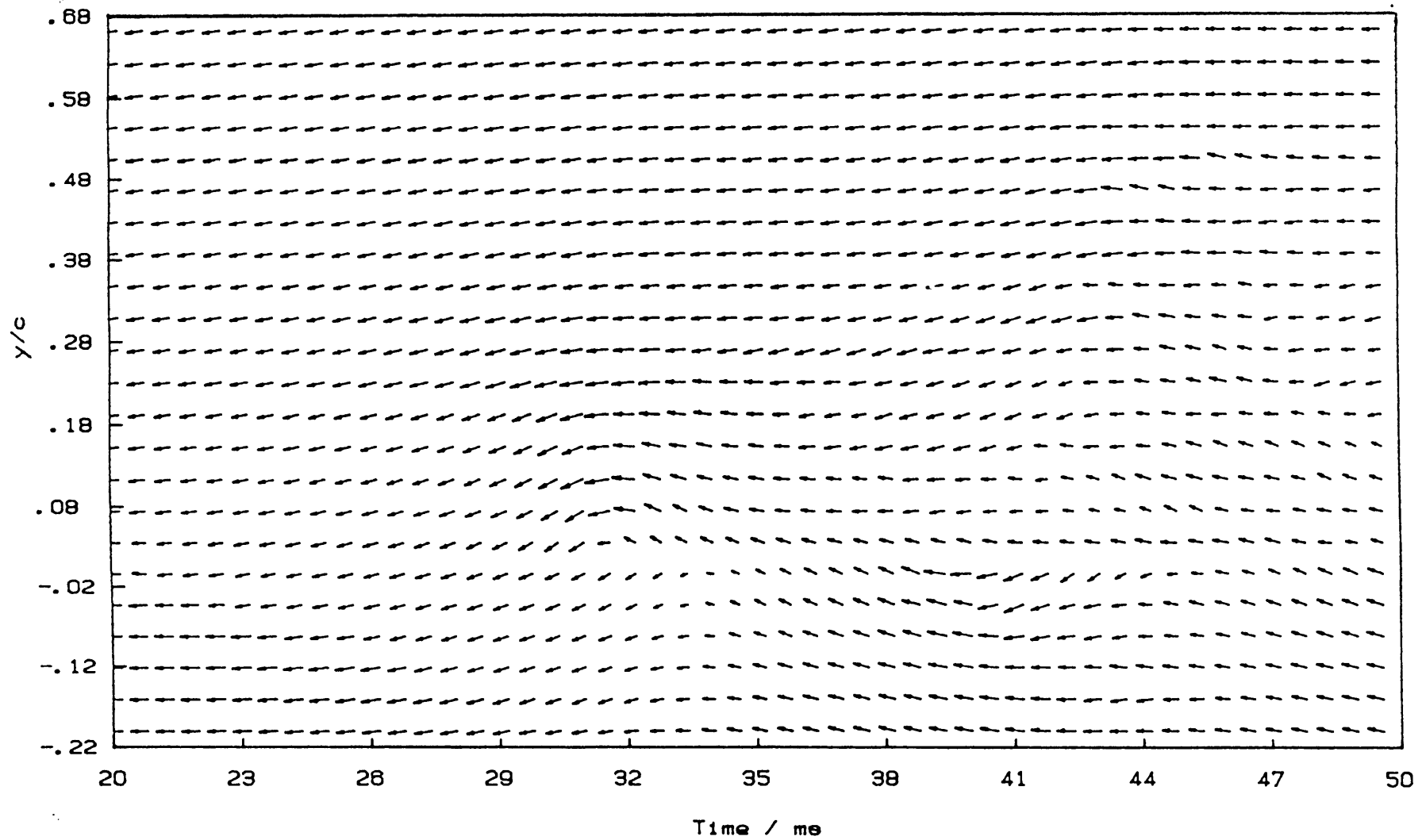


Fig 5.5 Velocity vector field for $T = 2.3$, from $t = 20$ to 50 ms

VELOCITY VECTOR

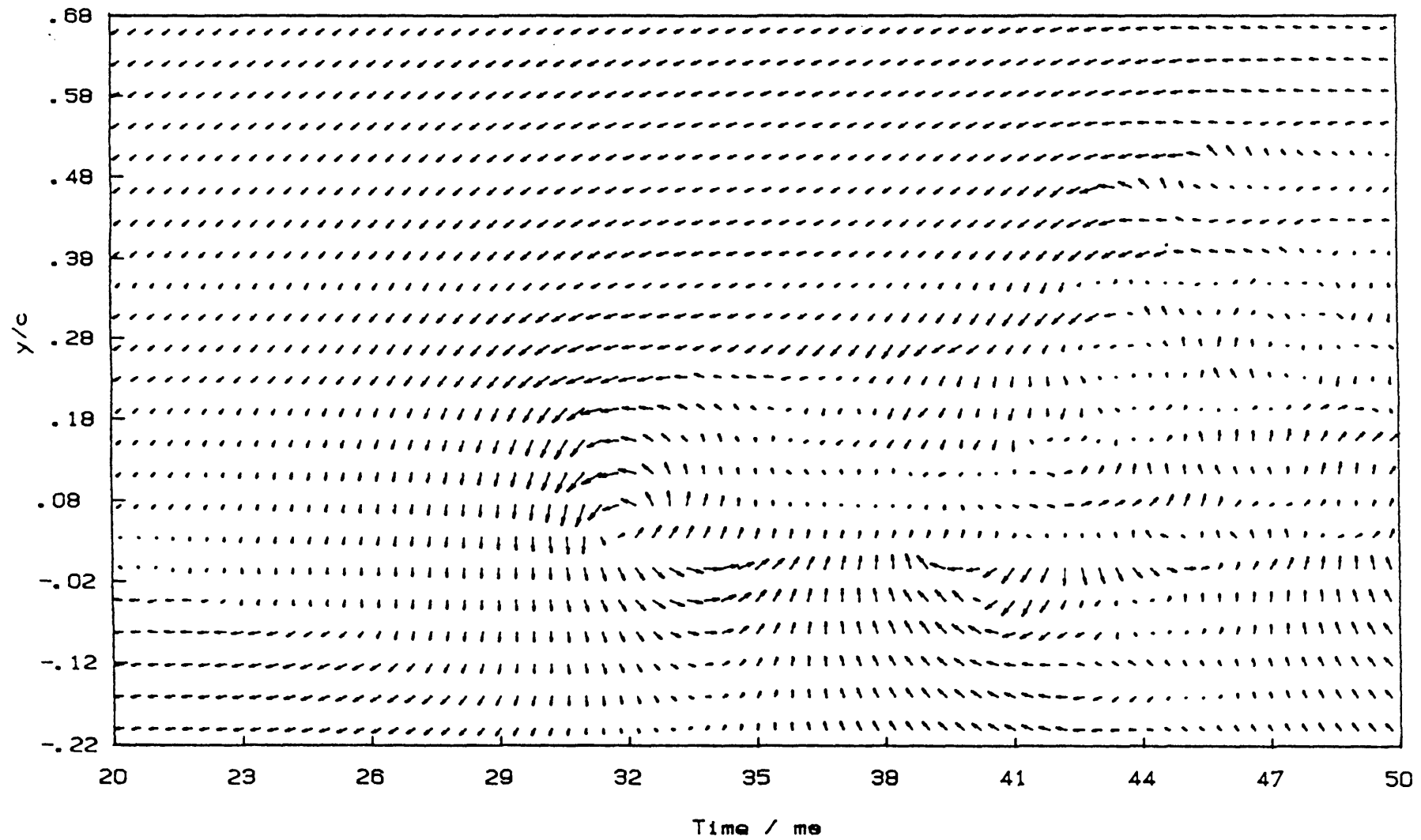


Fig 5.6 Velocity vector field for $T = 2.3$, from $t = 20$ to 50 ms, with U_{ref} removed

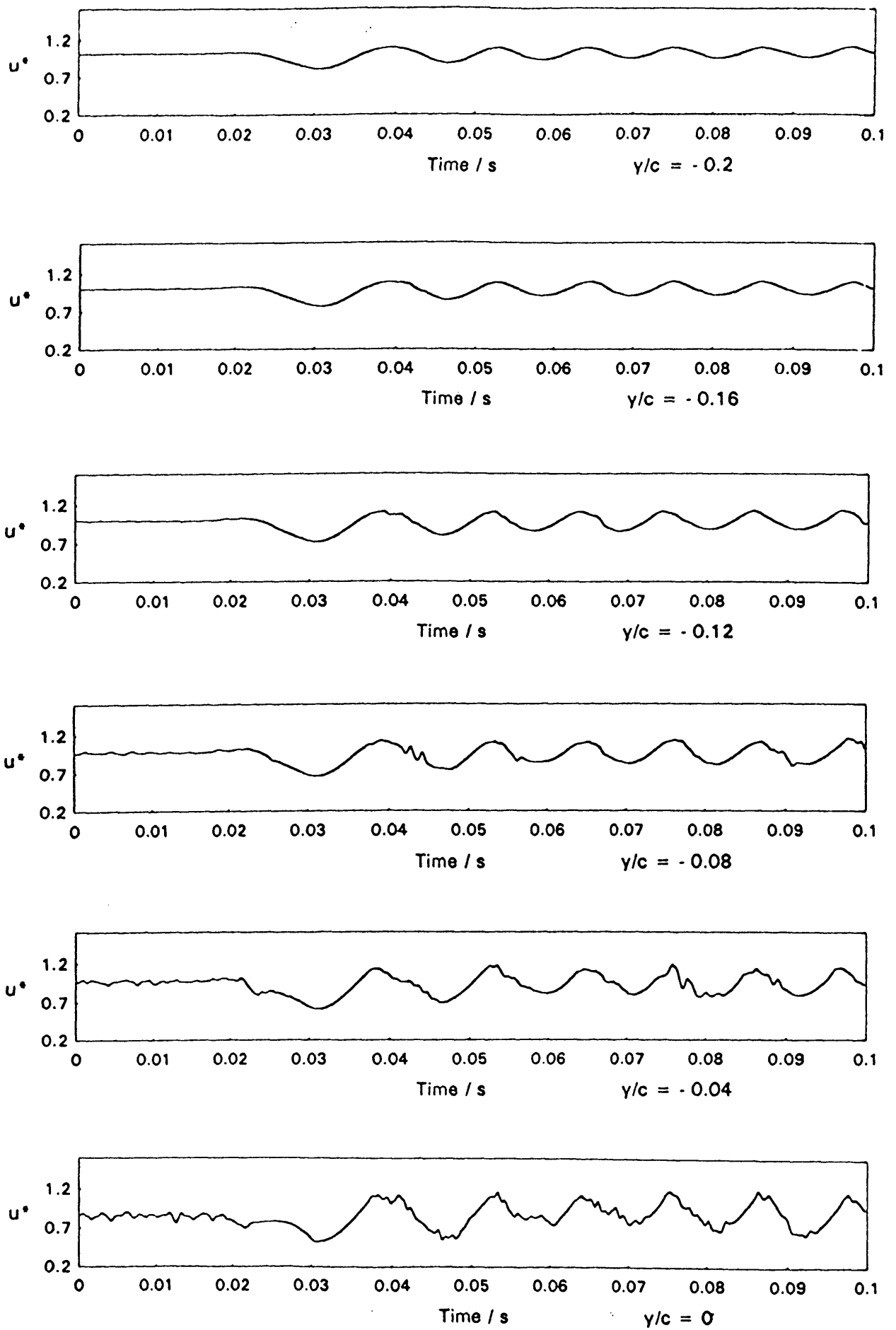


Fig 5.7 Traces of u^* for $T = 1.1$, $\delta_{ss} = 90^\circ$

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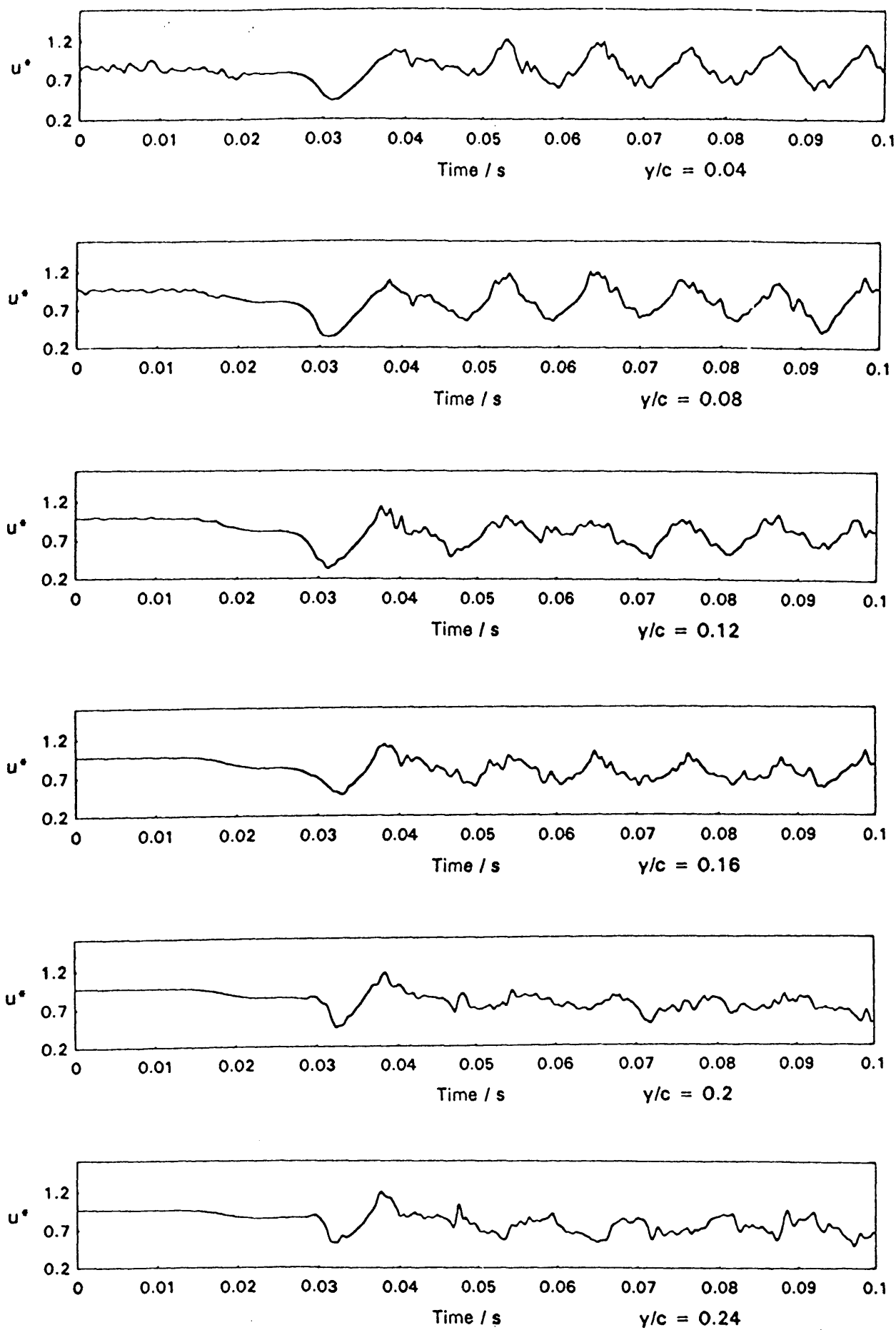


Fig 5.7 Traces of u^* for $T = 1.1$, $\delta_{ss} = 90^\circ$

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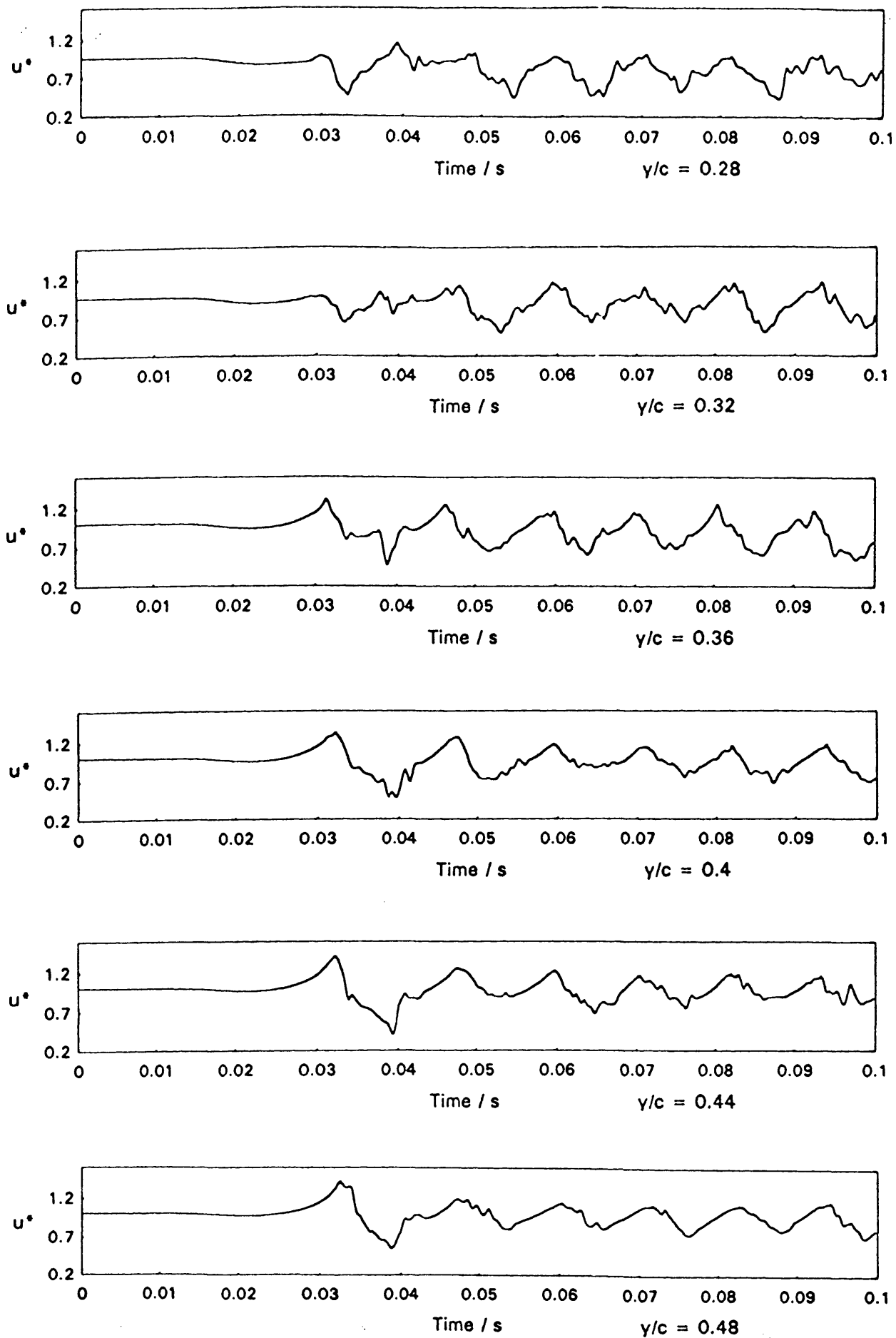


Fig 5.7 Traces of u^* for $T = 1.1$, $\delta_{ss} = 90^\circ$

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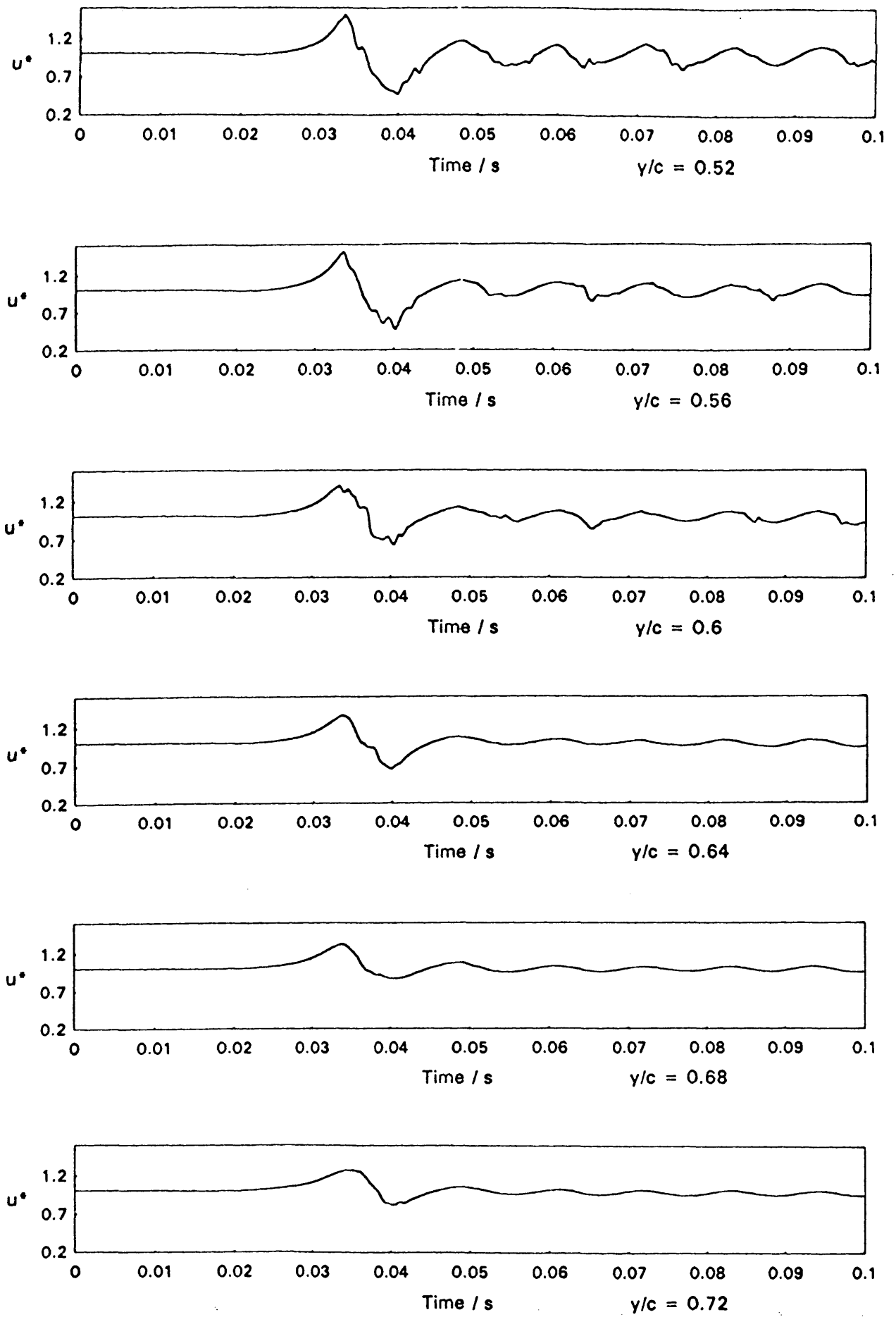


Fig 5.7 Traces of u^* for $T = 1.1$, $\delta_{ss} = 90^\circ$

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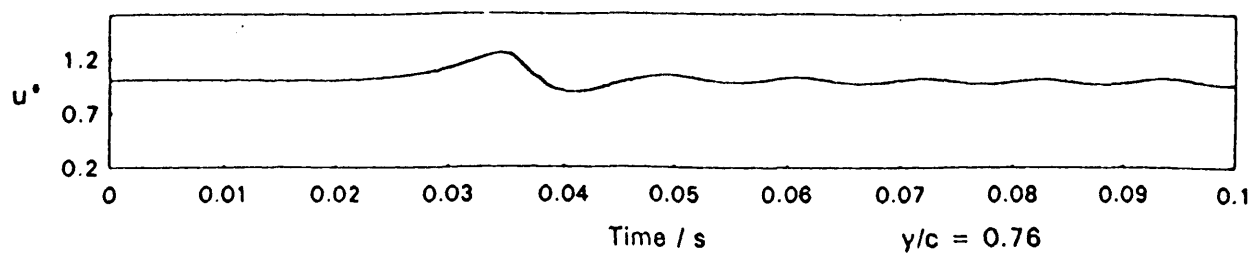


Fig 5.7 Traces of u^* for $T = 1.1$, $\delta_{ss} = 90^\circ$

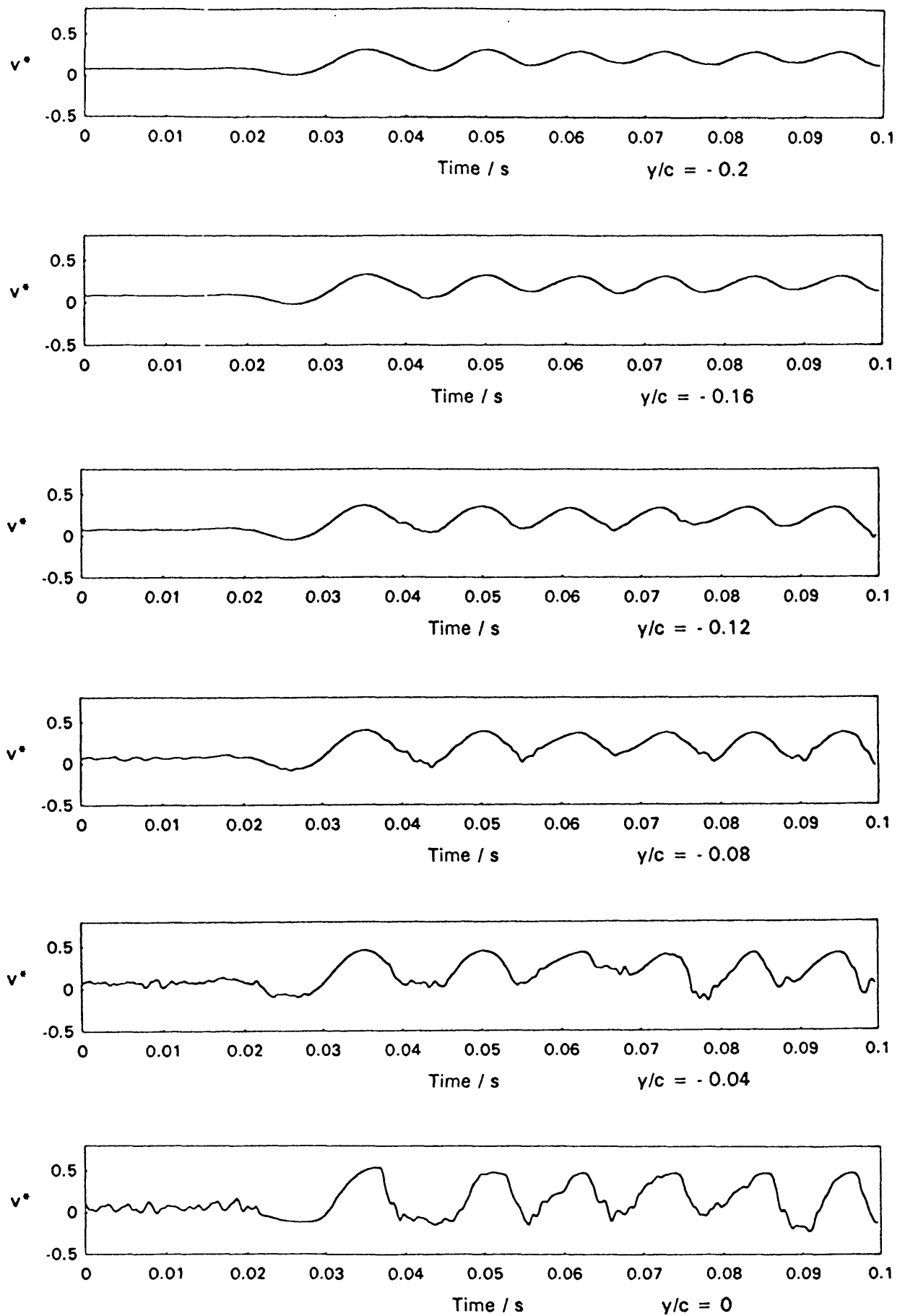


Fig 5.8 Traces of v^* for $T = 1.1$, $\delta_{ss} = 90^\circ$

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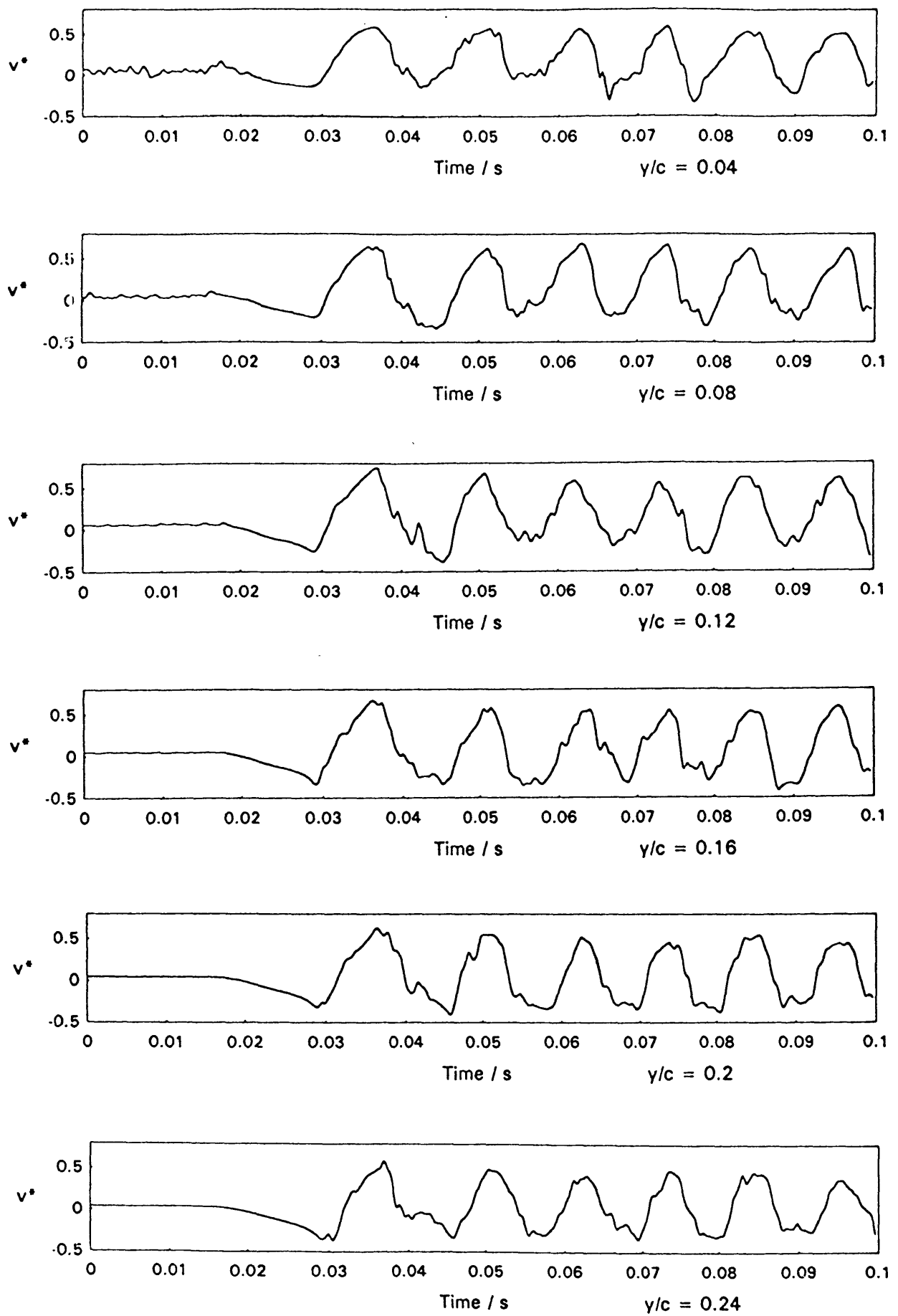


Fig 5.8 Traces of v^* for $T = 1.1$, $\delta_{ss} = 90^\circ$

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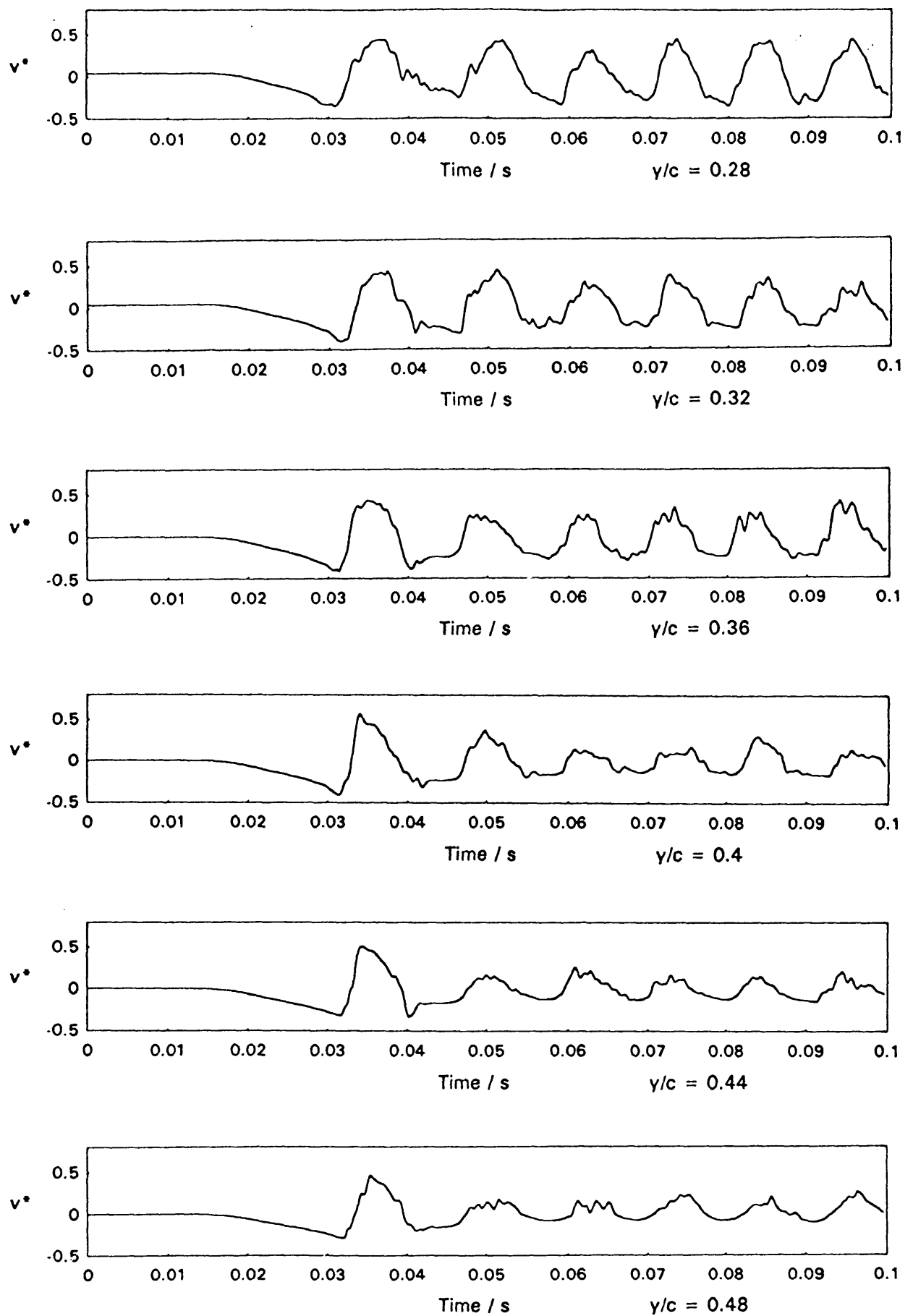


Fig 5.8 Traces of v^* for $T = 1.1$, $\delta_{ss} = 90^\circ$

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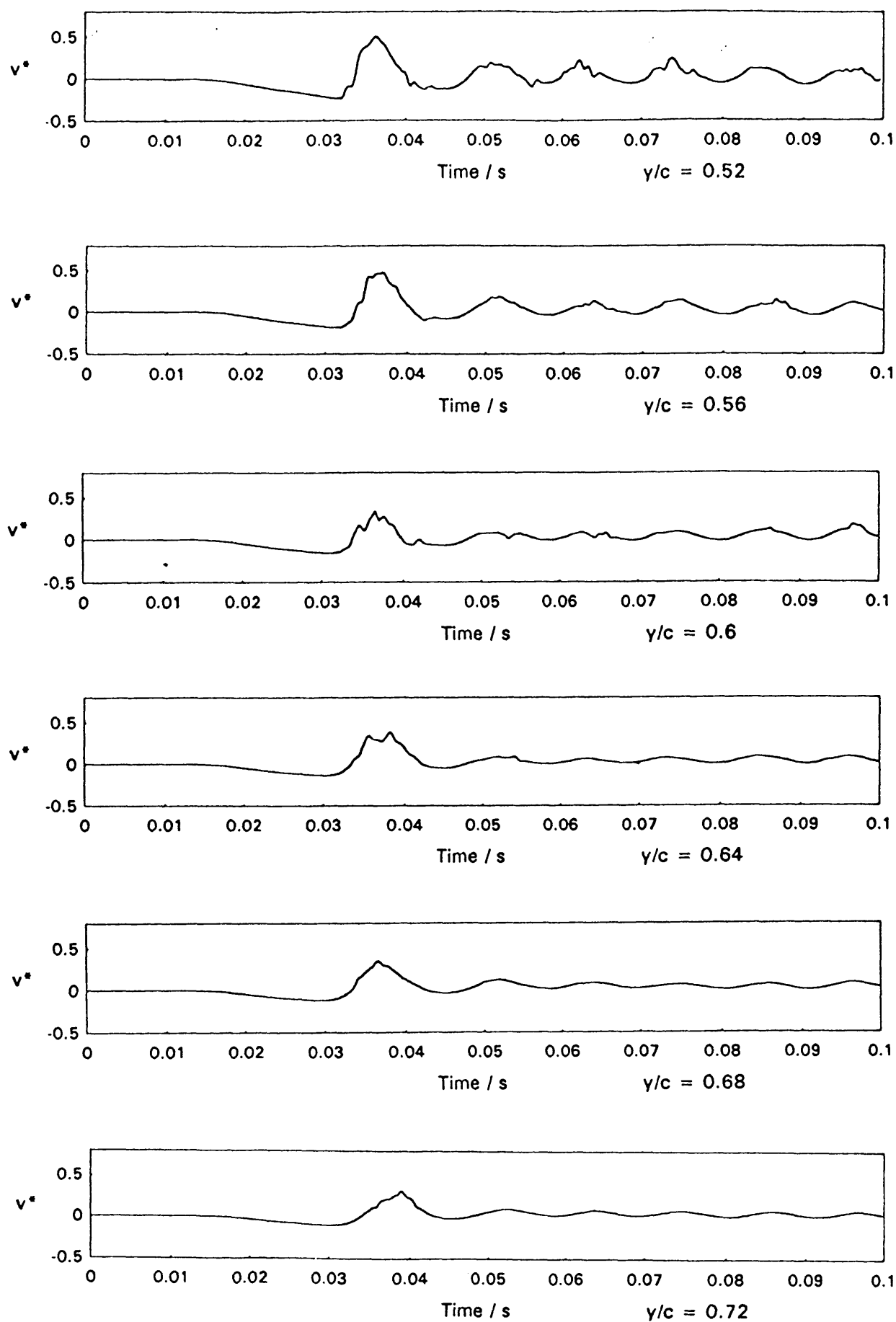


Fig 5.8 Traces of v^* for $T = 1.1$, $\delta_{ss} = 90^\circ$

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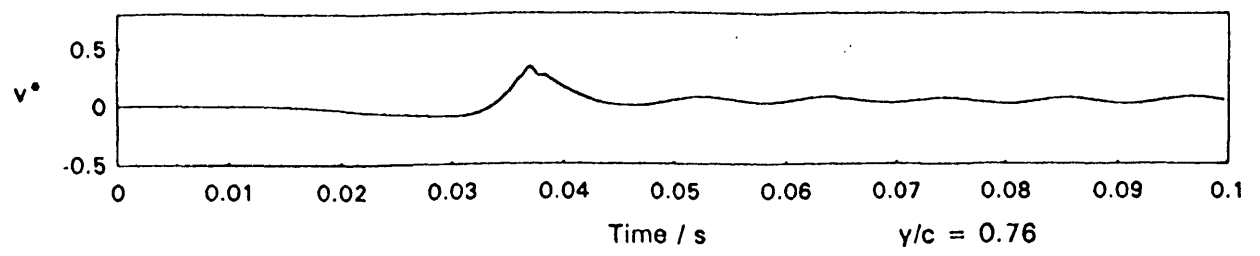


Fig 5.8 Traces of v^* for $T = 1.1$, $\delta_{ss} = 90^\circ$

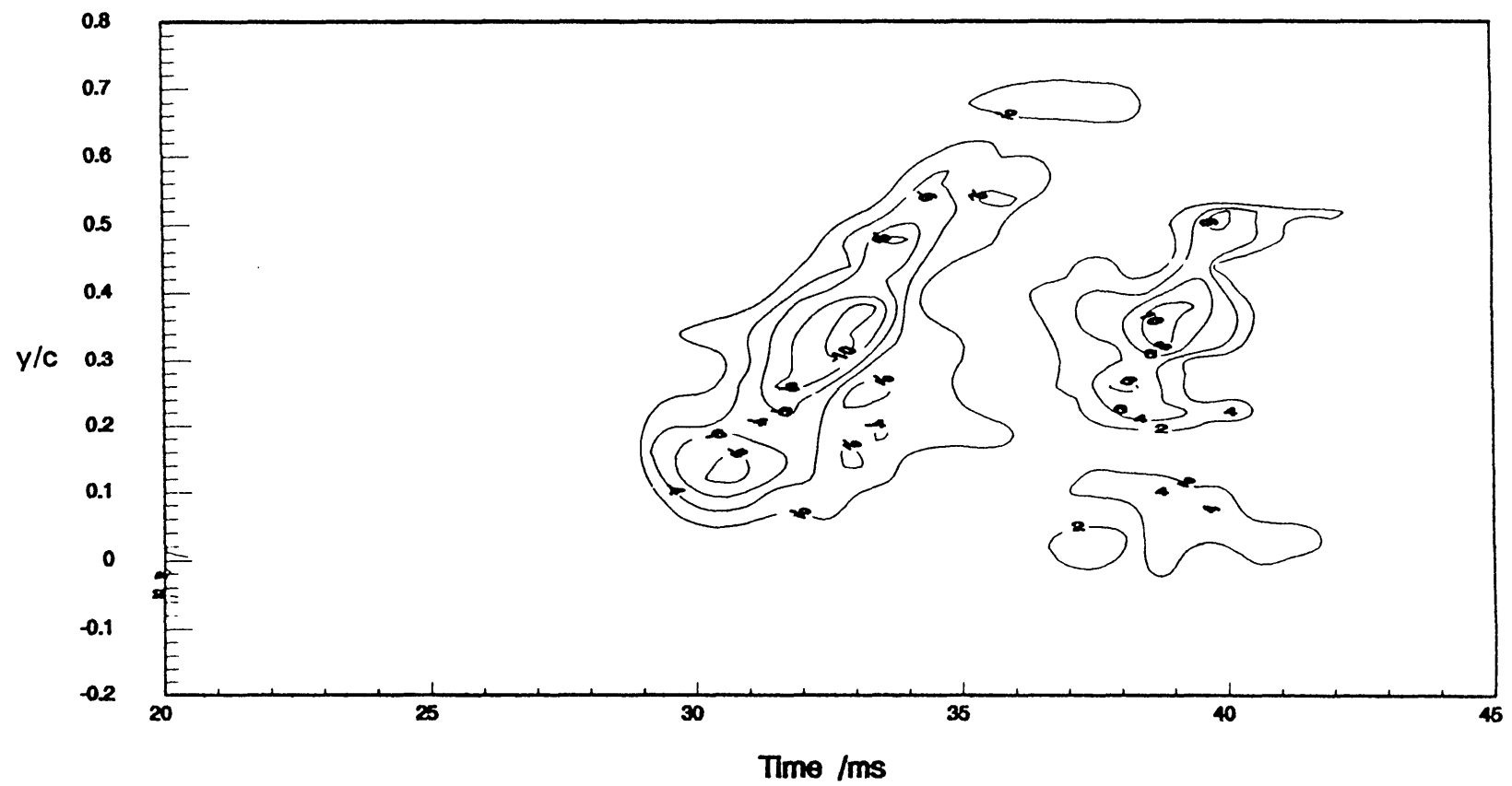


Fig 5.9 Distribution of Vorticity for $T = 1.1$, $\delta_{ss} = 90^\circ$, from $t = 20$ to 45 ms

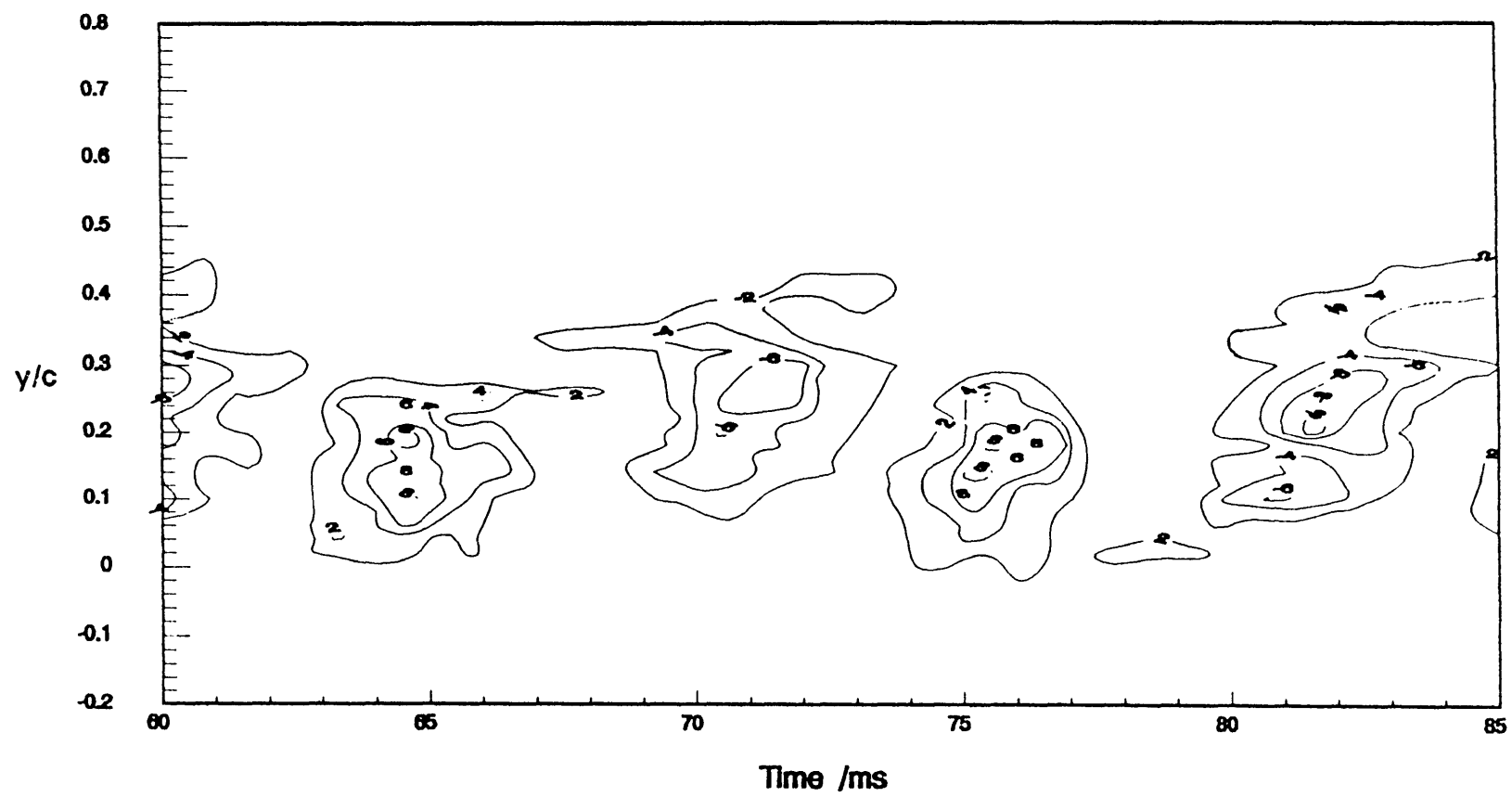


Fig 5.10 Distribution of Vorticity for $T = 1.1$, $\delta_{ss} = 90^\circ$, from $t = 60$ to 85 ms

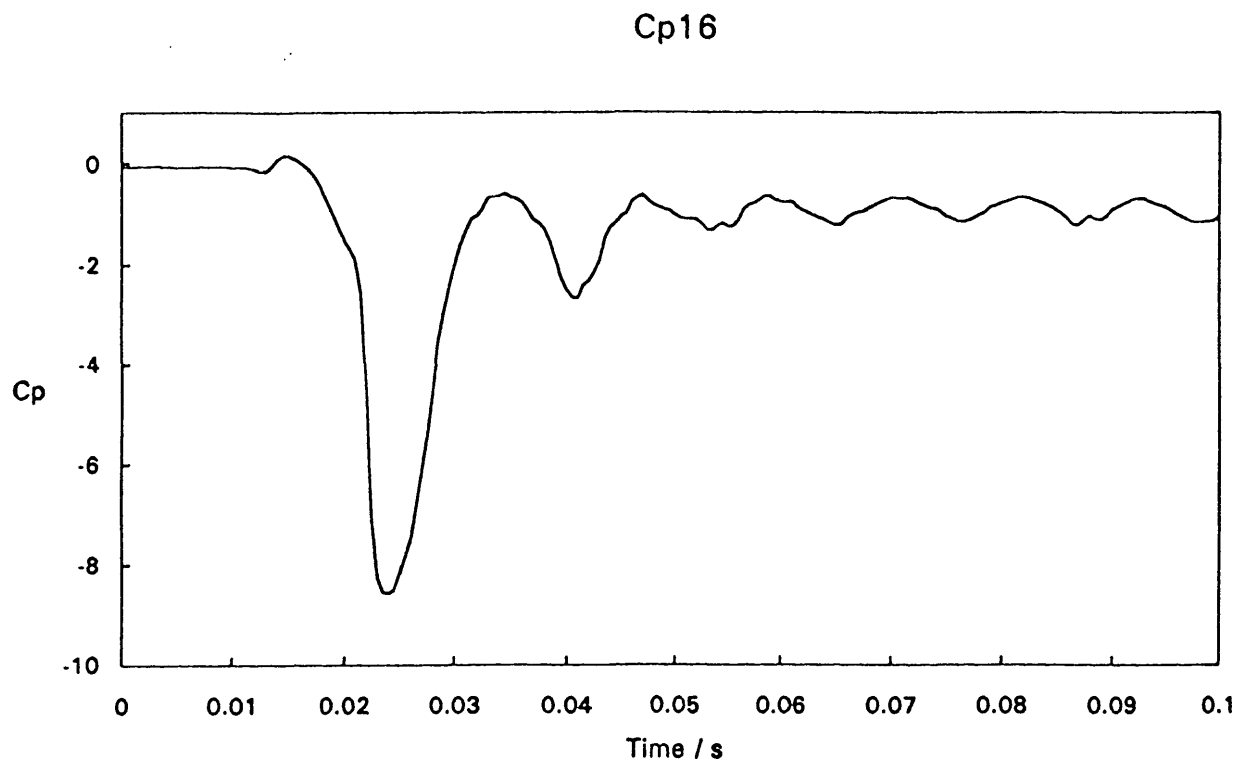


Fig 5.11 Time histories of C_{p16} for $T = 1.1$, $\delta_{ss} = 90^\circ$

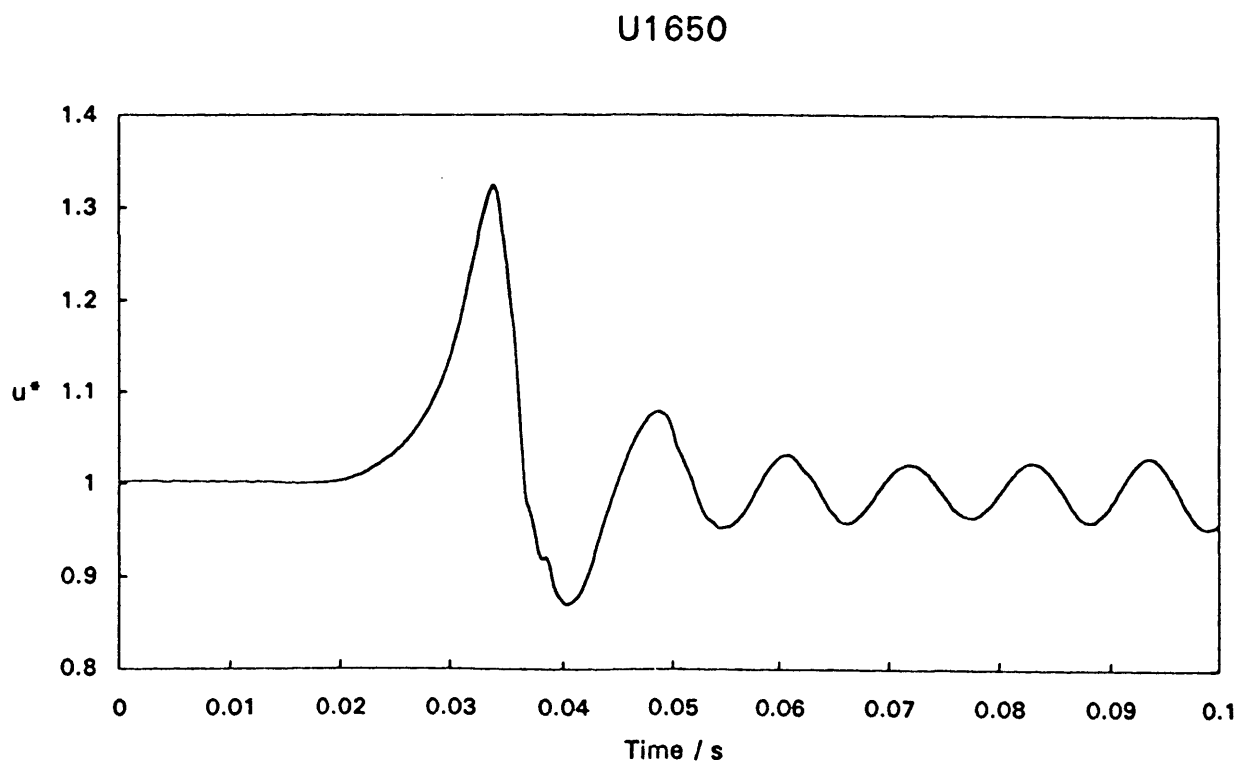


Fig 5.12 Time histories of u^* at $y/c = 0.72$ for $T = 1.1$, $\delta_{ss} = 90^\circ$

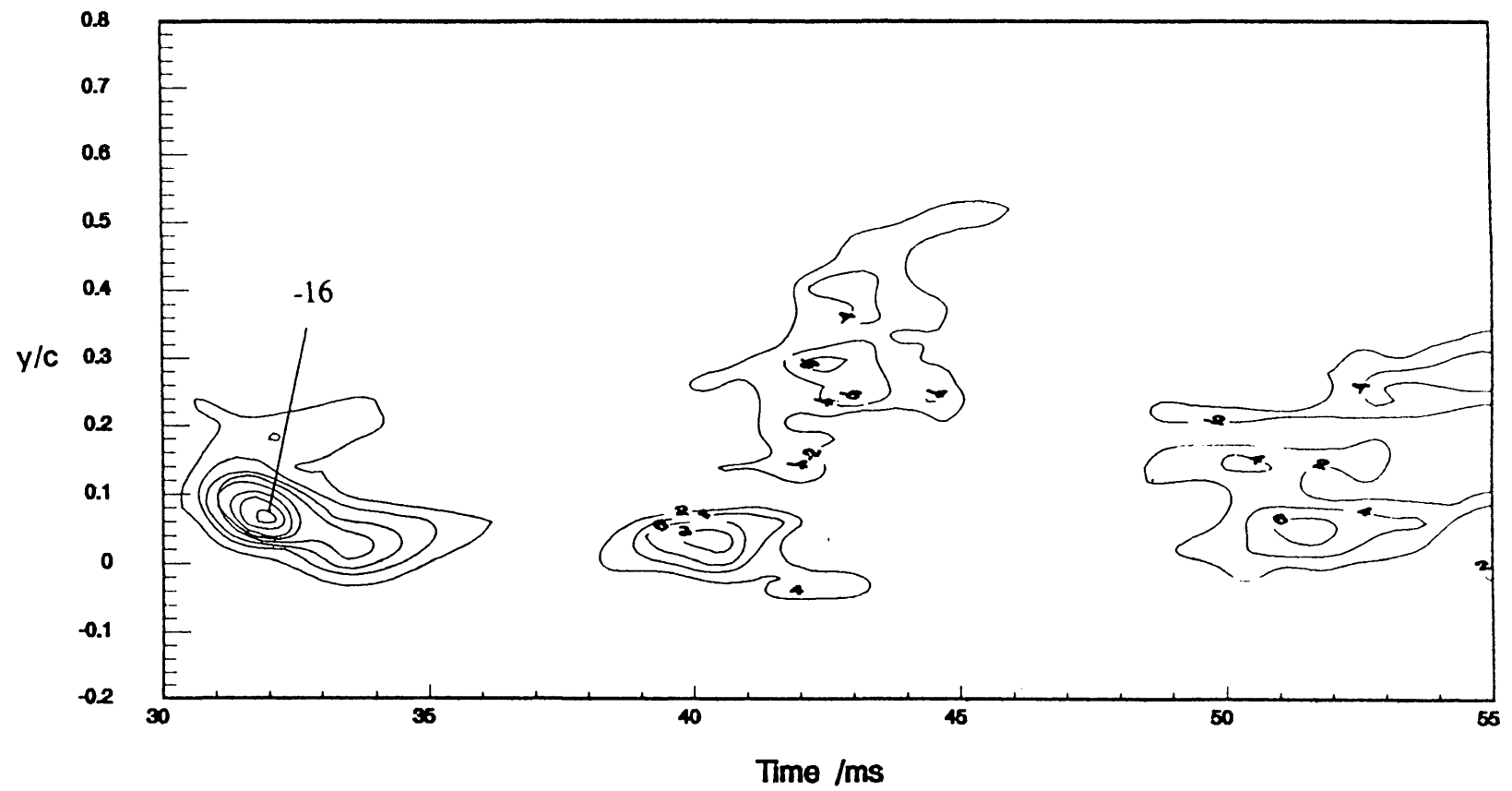


Fig 5.13 Distribution of Vorticity for $T = 2.3$, $\delta_{ss} = 90^\circ$, from $t = 30$ to 55 ms

U2440

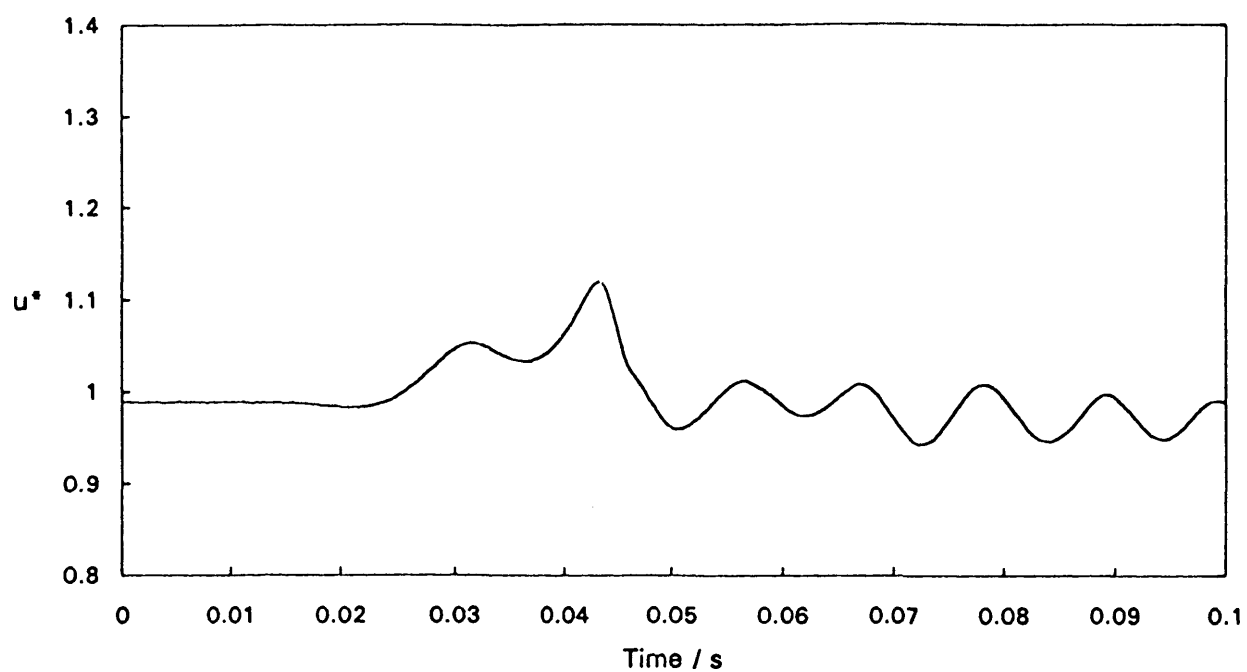


Fig 5.14 Time histories of u^* at $y/c = 0.72$ for $T = 2.3$, $\delta_{ss} = 90^\circ$

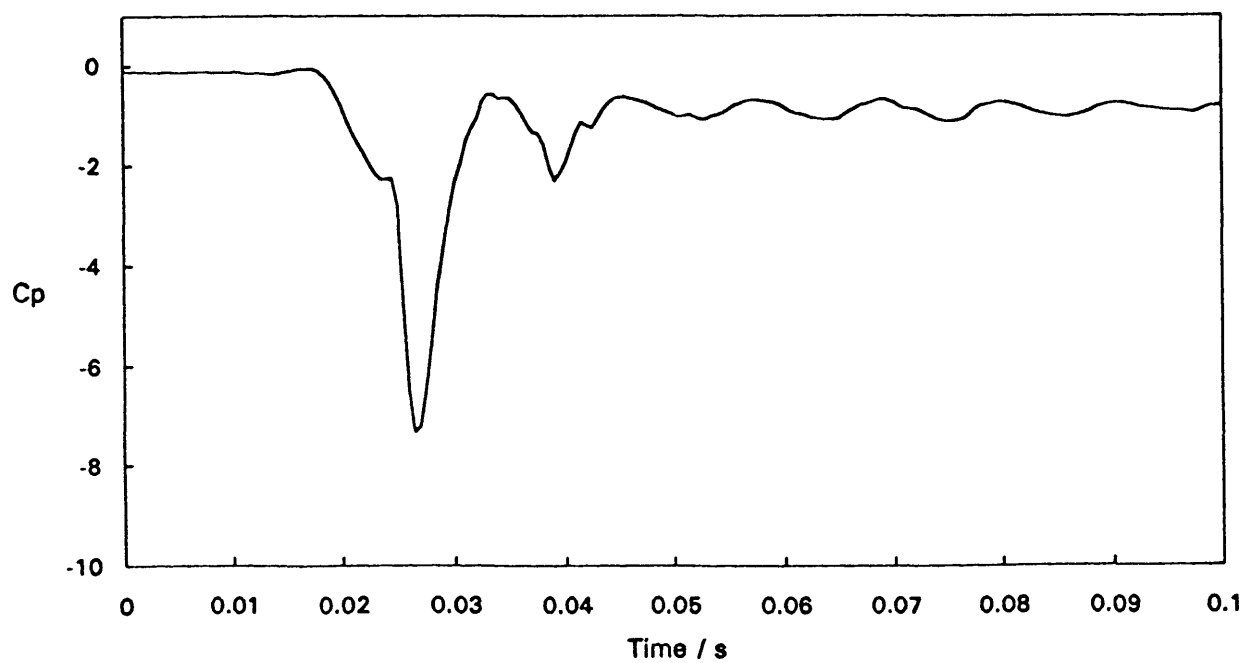


Fig 5.15 Time histories of C_{p16} for $T = 2.3$, $\delta_{ss} = 90^\circ$

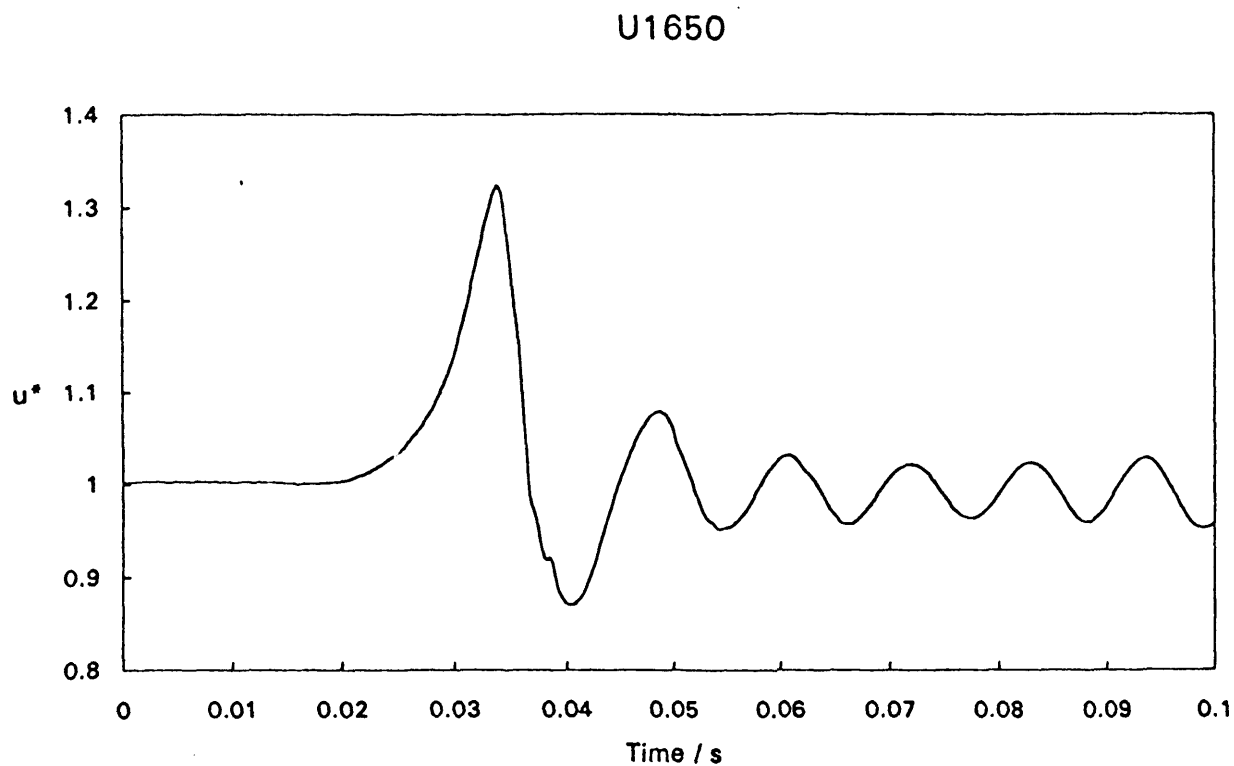


Fig 5.16a Time histories of u^* at $y/c = 0.72$ for $T = 1.1$, $\delta_{ss} = 90^\circ$

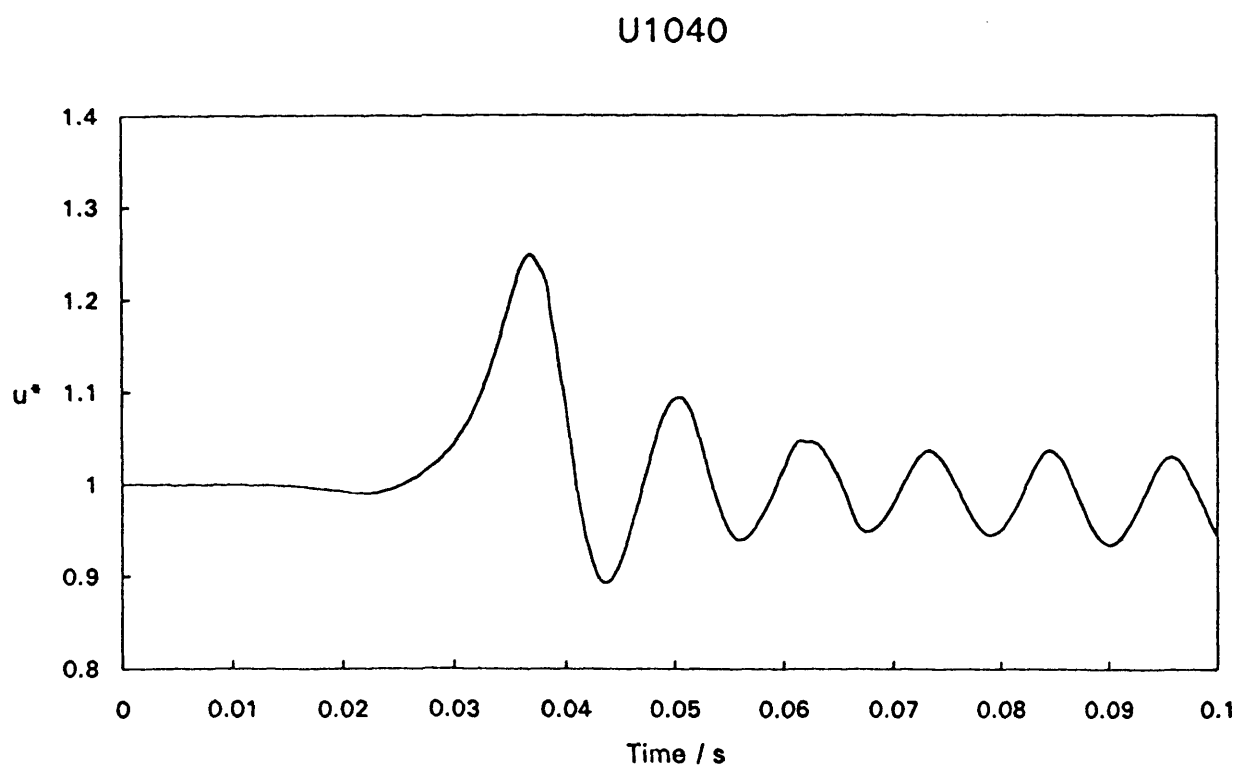


Fig 5.16b Time histories of u^* at $y/c = 0.72$ for $T = 2$, $\delta_{ss} = 90^\circ$

U2440

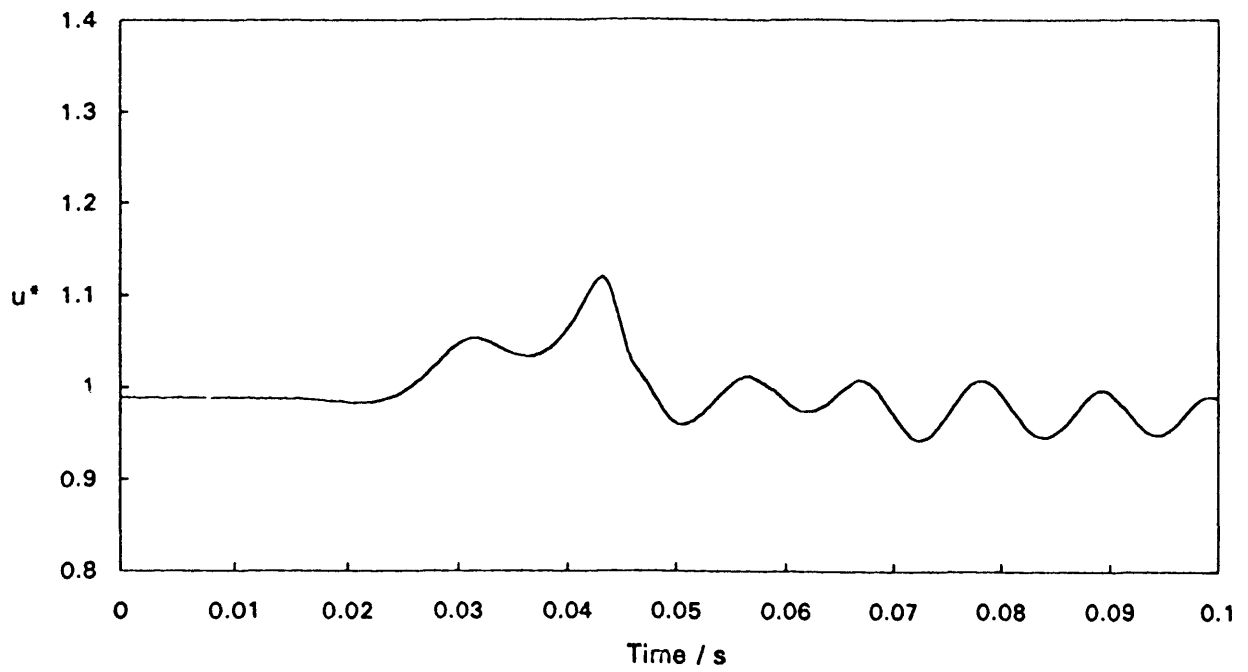


Fig 5.16c Time histories of u^* at $y/c = 0.72$ for $T = 2.3$, $\delta_{ss} = 90^\circ$

U1050

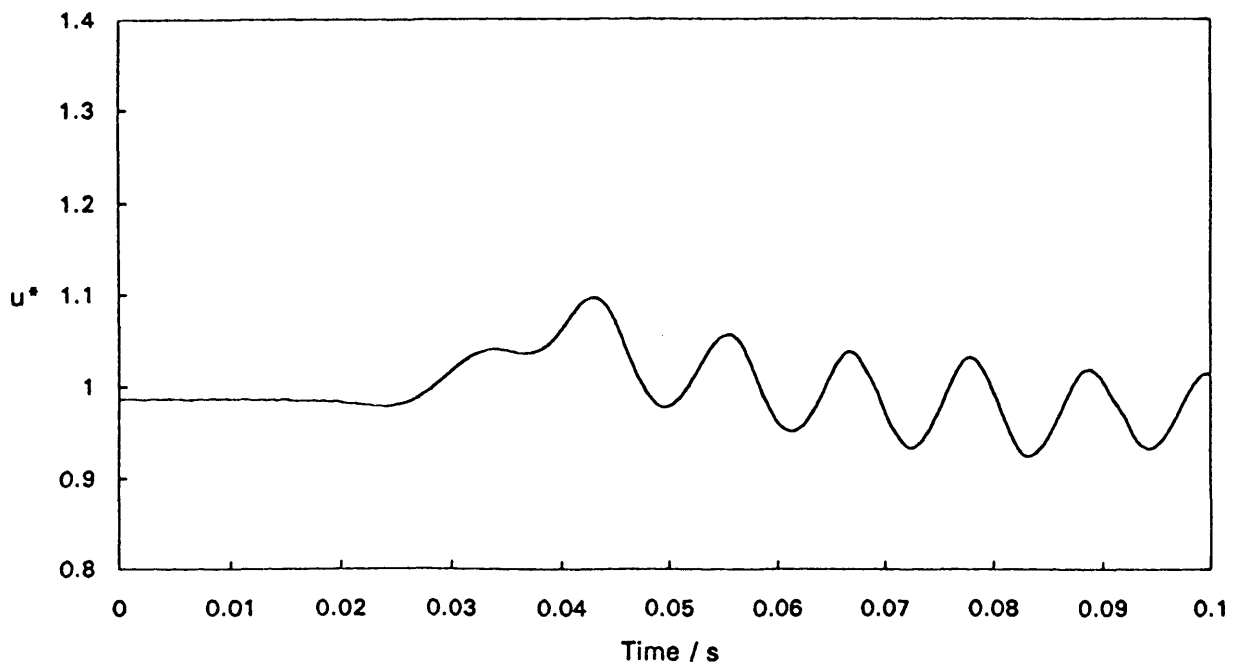


Fig 5.16d Time histories of u^* at $y/c = 0.72$ for $T = 2.8$, $\delta_{ss} = 90^\circ$

U1080

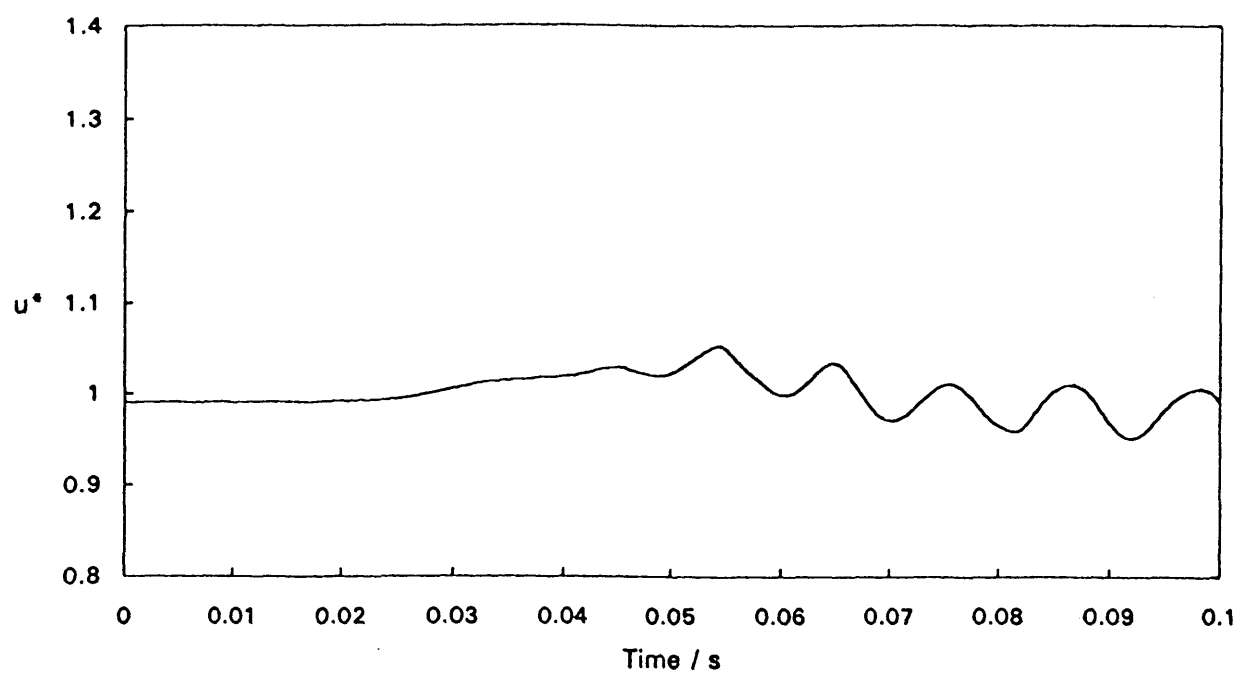


Fig 5.16e Time histories of u^* at $y/c = 0.72$ for $T = 4$, $\delta_{ss} = 90^\circ$

CpA2260

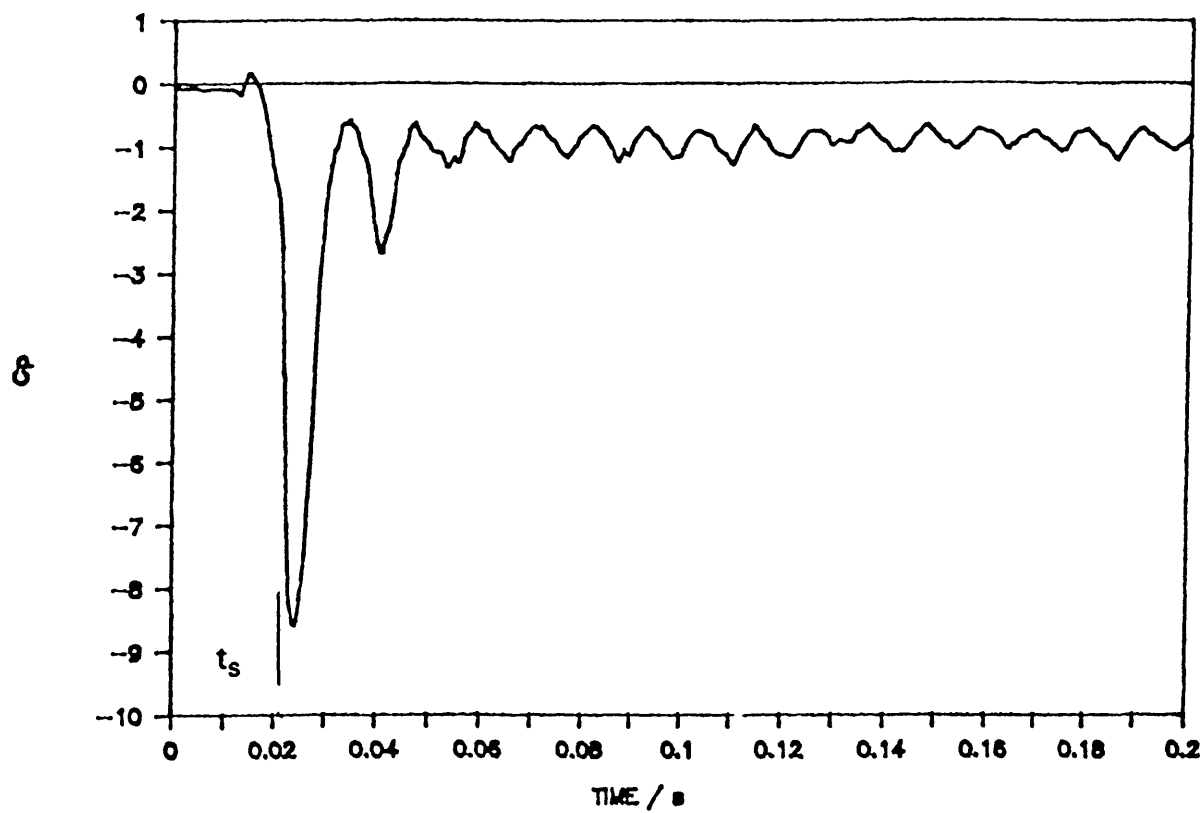


Fig 5.17a Time histories of C_{p16} for $T = 1.1$, $\delta_{ss} = 90^\circ$

CpA2230

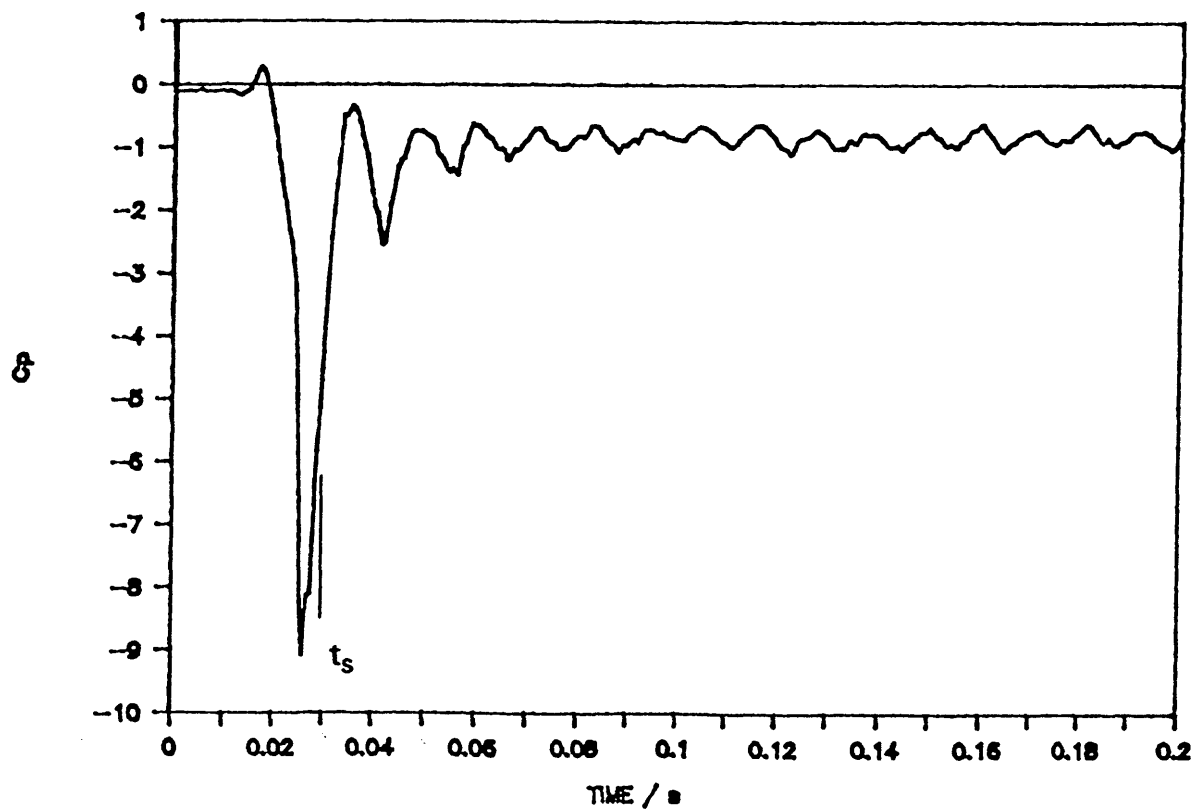


Fig 5.17b Time histories of C_{p16} for $T = 2$, $\delta_{ss} = 90^\circ$

CpA2200

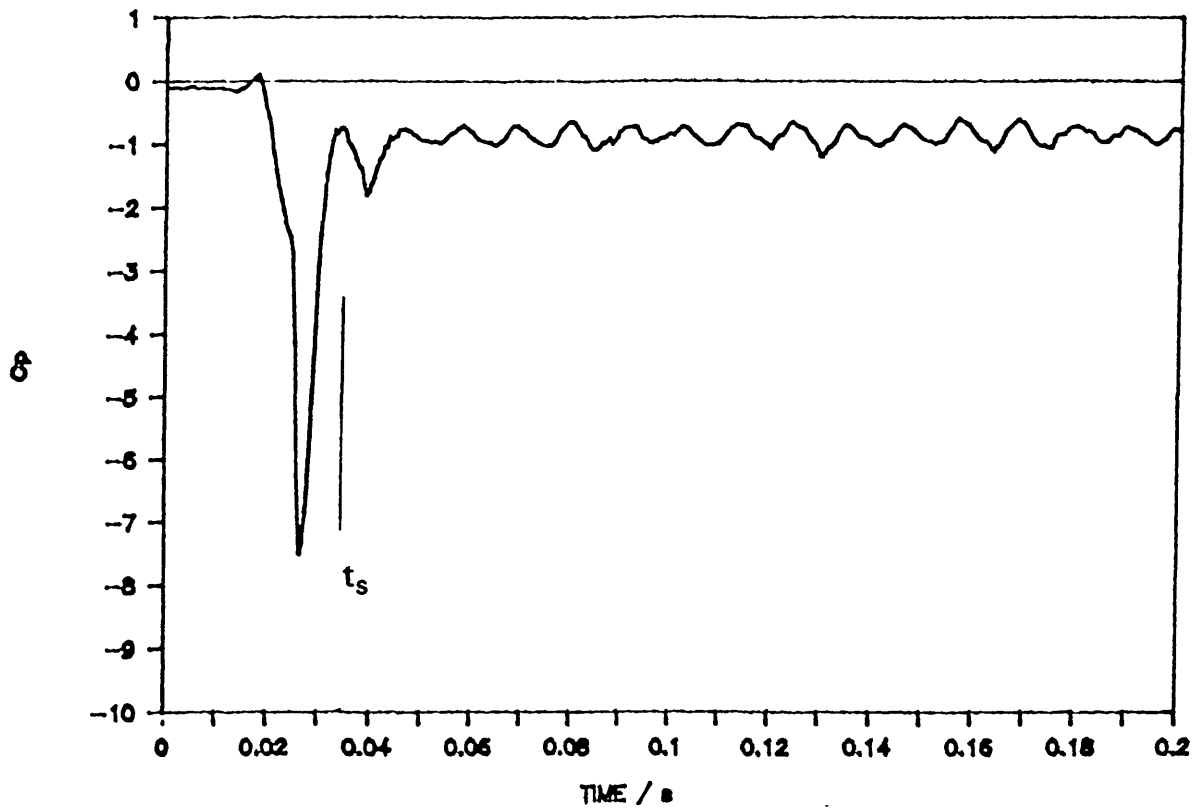


Fig 5.17c Time histories of C_{p16} for $T = 2.3$, $\delta_{ss} = 90^\circ$

CpA2210

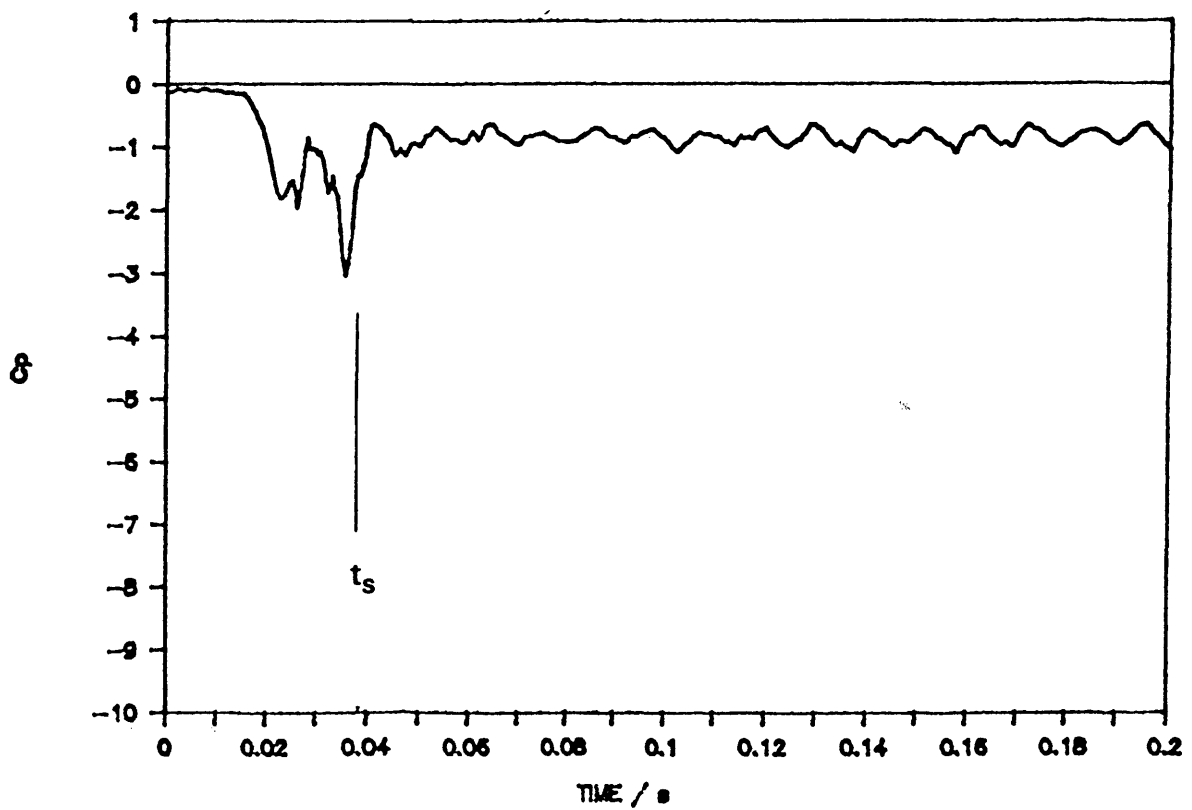


Fig 5.17d Time histories of C_{p16} for $T = 2.8$, $\delta_{ss} = 90^\circ$

CpA2240

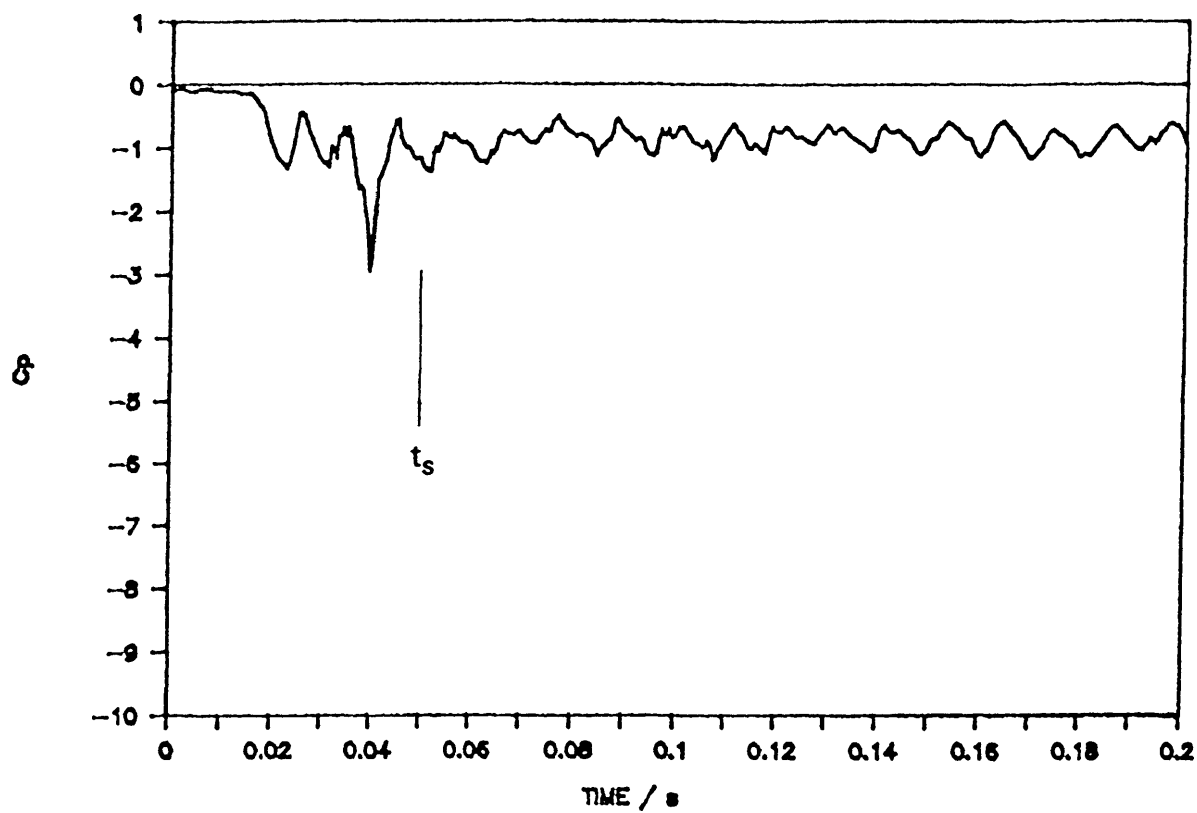


Fig 5.17e Time histories of C_{p16} for $T = 4$, $\delta_{ss} = 90^\circ$