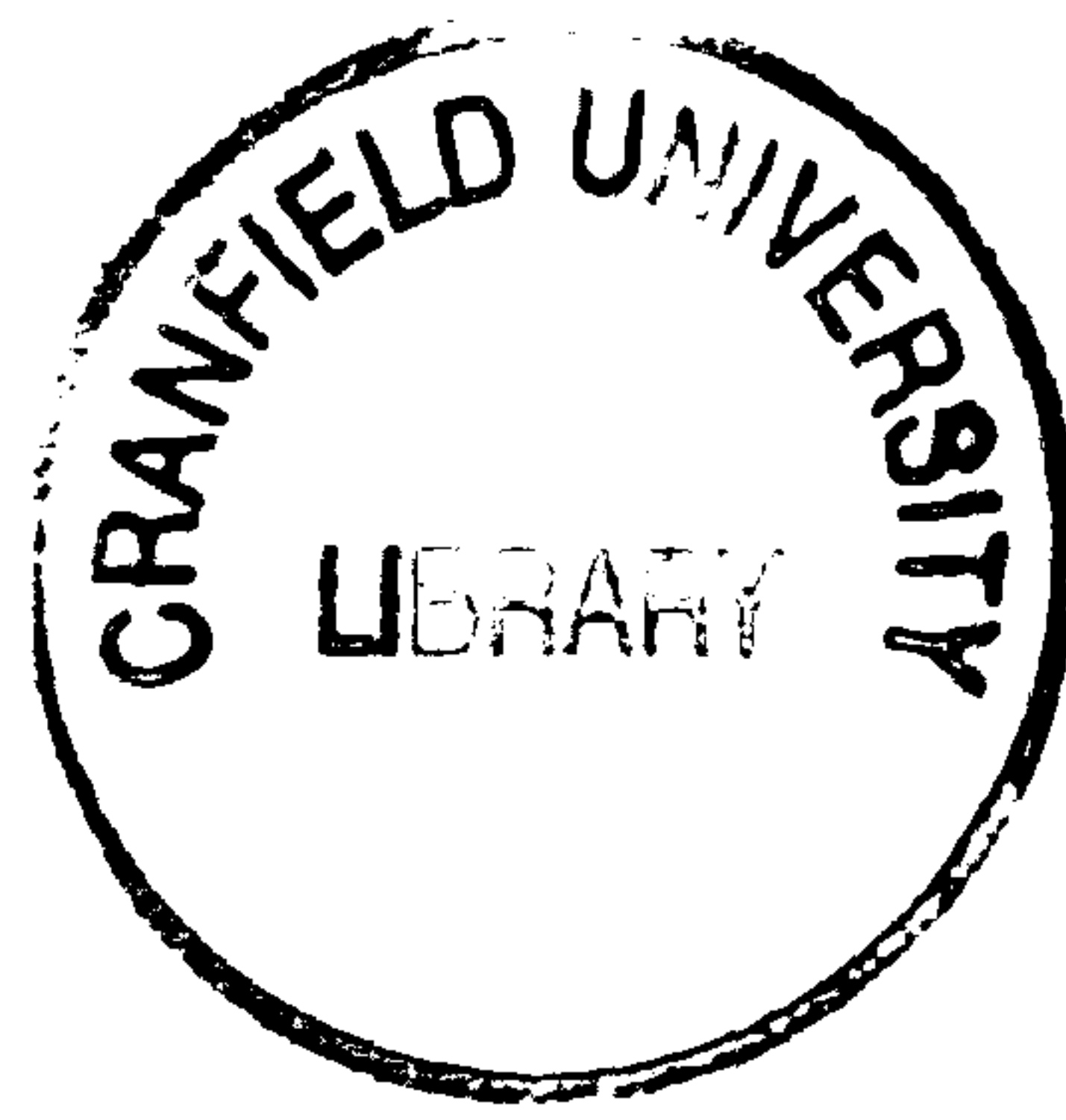




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**EXPERIMENTAL STUDY OF SLENDER VEHICLES
AT HYPERSONIC SPEEDS**

**COLLEGE OF AERONAUTICS
DEPARTMENT OF AEROSPACE SCIENCE**

PhD THESIS



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ABSTRACT

An experimental investigation of the hypersonic flow over (i) a wing-body configuration, (ii) a hemi-spherically blunted cone-cylinder body and (iii) a one-half-power-law body has been conducted for $M_\infty = 8.2$ and $Re_\infty = 9.35 \times 10^4$ per cm. The tests were performed at model incidences, $\alpha = 0, 5$ and 10° for flap deflection angles, $\beta = 0, 5, 15$, and 25° for the wing-body. The incidence ranged from -3 to 10° for the cone-cylinder and -5 to 15° for the power-law body.

(i) The schlieren pictures showing top and side views of the model indicate that the body nose shock does not intersect the wing throughout the range of α under investigation. Detailed pressure measurements on the lower surface of the wing and flap along with the liquid crystal pictures suggest that the body nose shock does not strike the flap surfaces either. The wing leading edge shock is found to be attached at $\alpha = 0$ and 5° but detached at $\alpha = 10^\circ$.

The liquid crystal pictures and surface pressure measurements indicated attached flow on the lower surface of the wing and flap for $\beta = 0$ and 5° at all values of α under test. However at $\alpha = 0^\circ$, as the flap angle is increased to 15° the flow separates ahead of the hinge line. As incidence is increased the boundary layer becomes transitional giving rise to complex separation patterns around the flap hinge line.

The spherically blunted body nose causes strong entropy layer effects over the wing and the trailing edge flap. A Navier-Stokes solution indicated a thick entropy layer of approximately constant thickness all around the cylindrical section of the body at zero incidence. However, at an incidence of 10° the layer tapers and becomes thinner under the body. The surface pressure over the wing and the plateau pressure for separated flow was found to increase from the root to the tip. This is partly because of the decrease in local Reynolds number across the span, however in the present case, entropy layer effects also affected separation. The entropy layer effects were found to reduce the peak pressures obtainable on the flap. The peak pressures, over the portion of the flap unaffected by entropy layer effects, could be estimated assuming quasi two dimensional flow.

(ii) Force measurements were made for the blunted cone-cylinder alone as well as with the delta wing, with trailing-edge flap, attached to it. The lift, drag, and pitching moment characteristics for the cone-cylinder agree reasonably well with the modified Newtonian theory and the N-S results. The addition of a wing to the cone-cylinder body increases the lift as well as the drag coefficient but there is an overall increase in the lift/drag ratio. The deflection of a flap from 0° to 25° increases the lift and drag coefficients at all the incidences tested. However, the lift/drag ratio is reduced showing the affects of separation over the wing. The experimental results on the wing-body are compared with the theoretical estimates based upon two-dimensional shock-expansion theory.

(iii) The lift, and drag characteristics of a one-half-power-law body are compared with other existing results. The addition of strakes to the power-law body are found to improve its aerodynamic efficiency without any significant change in its pitching moment characteristics.

Dedicated

to my mother Gurdev Kaur

and father Bant Singh for their love and sacrifices

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NOMENCLATURE

a	Major length of an elliptical x-section
A	Axial force
b	Minor length of an elliptical x-section
C	balance calibration constant or aerodynamic coefficient
C_A	Axial force coefficient
C_D	Drag coefficient based on maximum x-sectional area of body for wing-body and hemi-spherically blunted cone-cylinder body
C_D	drag coefficient based on body planform area for power-law body
C_{exp}	Exposed wing mean aerodynamic chord, sketch on page 19.
C_f	Skin friction coefficient
C_H	Heat transfer coefficient
C_L	lift coefficient based on maximum x-sectional area of body for wing-body and hemi-spherically blunted cone-cylinder body
C_L	lift coefficient based on body planform area for power-law body
C_p	Pressure coefficient
C_m	Pitching moment coefficient about the balance moment-centre based on body length and maximum x-sectional area of body for wing-body and hemi-spherically blunted cone-cylinder body
C_m	pitching moment coefficient about the balance moment-centre based on body length and planform area for the power law body
C_N	Normal force coefficient based on maximum x-sectional area of body for wing-body and hemi-spherically blunted cone-cylinder body
C_N	Normal force coefficient based on body planform area for power-law body
C_r	Wing root chord
C_{tot}	Total wing mean aerodynamic chord, sketch on page 19
C^*	Chapman-Rubesin constant
C.G.	centre of gravity
d	diameter

D	Drag
G	amplifier gain used
H.L.	Hinge-line
k	Coefficient of thermal conductivity
K	Constant in Newton's $\sin^2$ law
l	Body length
l_c	Cylinder length
L	Lift
M	Mach number
mv	millivolt
n	Power-law exponent
Re	Reynolds number
R_b	Base radius
R_N	Nose radius
P	pressure
Pr	Prandtl number
T	temperature
Tr	transient recorder output
Re	Reynolds number
U	velocity
VFS	voltage full scale
X	Distance along body axis
X_{cp}	distance to centre of pressure from the nose
Y	Distance along radial or spanwise direction
α	incidence angle
α	Incipient separation angle due to glancing interaction
β	Flap deflection angle
β_i	Incipient separation angle due to flap deflection
γ	ratio of specific heat
θ	Shock angle
δ	Shock detachment distance

ρ	density
μ	coefficient of viscosity
v	Prandtl-Meyer function
χ	Viscous interaction parameter
Λ	Wing leading-edge sweep back angle

Subscripts

<i>h.l.</i>	hinge-line
1	static condition ahead of normal shock
01	reservoir condition ahead of normal shock
02	stagnation condition behind normal shock
∞	freestream condition
0∞	freestream stagnation condition

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INTRODUCTION

Ever since the emergence of human kind on planet Earth it has been striving for faster and faster means of transportation for common person and for man and material for a war. 1950s and 1960s saw some extensive and pioneering research in the aerodynamics of bodies flying at hypersonic speeds. The effort was fuelled by the need for intercontinental ballistic missiles and for manned space flight, probably in that order of urgency. The Space Shuttle program re-energised the interest in the hypersonic flight and mid 80s again saw many aerospace plane programs like the National Aerospace Plane (NASP) in the USA, (HOTOL) in UK, Sänger in Germany, and Hope in Japan on the drawing board. However, budget cuts as a result of slowing down of the economies of the countries involved and seemingly absence of any military threat has led to a slow progress of these projects. The fact that funding for hypersonic research has reduced does not deny the role that the hypersonic aerodynamics is going to play in the transportation systems of near future.

One of the main characteristics of hypersonic flow is that shock waves lie very close to the surface of the bodies. The region between the shock wave and the surface of the body is called the shock layer. The boundary layer thickness increases with increasing Mach number so that the interaction between a shock wave and a boundary

layer developing inside the shock layer is stronger at hypersonic speeds. However, for high Reynolds number flow the entire shock layer can be assumed to be inviscid for simple analysis. In the extreme case of zero shock layer thickness the flow approaches Newton's flow model and Impact theory provides reasonably accurate results.

Inviscid flow properties remain constant inside a shock layer for a sharp 2-dimensional wedge. However, the flow properties no longer remain constant if the leading edge is blunt due to the formation of an entropy layer next to the surface of the body. The detached shock wave formed at the blunt leading edge is found to be highly curved as a result of interaction between the shock wave and expansion waves created at the leading edge. Streamlines around the blunt wedge will therefore have different flow properties depending upon the part of the shock through which they have travelled. In particular, the streamlines passing through stronger (nearly normal) portion of the wave have higher entropy but lower stagnation pressure than those passing through the weaker portion of the wave.

Many aerospace vehicles use blunt noses and leading edges not just because it will be very difficult to keep them sharp in hypersonic flow but also because of the added advantage of reduced aerodynamic heating and increased internal space. In many cases the nose has to be blunt to satisfy other system requirements, for example to reduce radar or infra-red signal distortion. Aerodynamic performance of such a blunt nosed vehicle will be affected by the entropy layer generated by the nose. A wing trailing-edge-flap is one of the most common methods of pitch control for aerospace vehicles. The effectiveness of trailing-edge flaps is however affected by many parameters such as the extent of flow separation, shock-shock interactions and the entropy layer. The present study has focused attention on the effect of the entropy layer on trailing-edge-flap effectiveness.

Slender elliptic planform cones are of interest as gun fired projectiles. Effect of strakes on such projectiles has been studied by measuring the lift, drag, and pitching moment characteristics of a one-half-power-law body with and without the strakes.

A wing-body model with trailing-edge flaps was chosen to represent a very blunt nosed vehicle. For small bluntness the nose shock tends to intersect the wing leading-edge. The intersection complicates the flow over the wing by changing its separation

and transition characteristics. However, the wing-body configuration chosen was designed to keep the wing within the nose shock layer. The surface flow over the wing-body was studied using oil-dot and liquid-crystal techniques. Some information regarding the shock layer was obtained from schlieren pictures but detailed flow properties in the shock layer were obtained from a Navier-Stokes solver. Computational solutions were obtained for the blunted body only. The effect of entropy layer on the flap effectiveness was deduced from detailed surface pressure measurements over the wing and the flap. Finally, lift, drag, and pitching moment characteristics for the body alone and for the wing-body combination were obtained using a three component strain-gauge balance.

This thesis is divided into five chapters. After an introduction in chapter 1, existing literature pertinent to the present work is reviewed in chapter 2. Experimental facilities and techniques used for the study are described in chapter 3. Results of the study are discussed in chapter 4. In chapter 4, analytical, computational and experimental results for the hemi-spherically blunted cone-cylinder body are first discussed in section 4.1. One-half-power-law body experimental results are given in section 4.2 followed by a brief discussion of flow visualisation results on a sharp leading-edge flat plate fitted with full span trailing-edge flaps in section 4.3. The wing-body results are discussed in section 4.4. The flow visualisation using schlieren, oil-dot flow, and liquid-crystals technique are presented in section 4.4.1. The pressure measurements are given in 4.4.2 and force measurements in 4.4.3. Finally, findings of the present study are concluded in chapter 5.

LITERATURE SURVEY

2.1 Slender bodies

The requirement of defence against ballistic missiles has led to a renewed interest in the aerodynamics of slender, blunt-nosed, conical bodies flying at hypersonic speeds. Slender blunted conical bodies exhibit the desired hypersonic aerodynamic characteristics. However, power-law bodies can be an alternative to the blunted cone configurations because a power-law body has a greater internal volume than a blunted cone of the same fineness ratio and secondly, because various theories predict that a power-law body represents the minimum drag case at hypersonic speeds. For example, the minimum-drag body, of given length and base diameter, obtained using the Newtonian theory has very nearly the same shape as the $3/4$ power law body⁽¹⁾. On the other hand the minimum-drag body, for a given fineness ratio, based upon the Newtonian-Busemann formulae or hypersonic small disturbance theory⁽²⁾ have shapes very much like the $2/3$ power-law body.

2.1.1 Slender circular bodies

Cleary⁽³⁾ conducted an experimental and theoretical investigation of the flow of a perfect gas over 15° and 30° half-angle spherically blunted cones at hypersonic speeds.

Numerical results based on the method of characteristics are presented for zero incidence to show the effects of cone angle, specific heat ratio, and Mach number on surface pressure distribution and pressure drag. Profiles of shock-layer properties for a 15° blunted cone are also shown. Experimental results to show the effect of incidence on the longitudinal and circumferential pressure distributions over the 15° and 30° half-angle blunted cones at Mach numbers of 5.25, 7.4, and 10.6 are presented. Experimental pitot traverses at two stations for a Mach number of 10.6 demonstrate the thinning of the entropy layer along the length of the body and the effect of a change in the incidence. Numerical and experimental results are compared to illustrate viscous effects and to verify the main features of the numerical solution.

Singh, Kumar, and Tiwari⁽⁴⁾ conducted a parametric study to determine the effects of nose bluntness on the entire flow-field over slender bodies under different hypersonic freestream conditions. The slender bodies considered are blunted cones and ogives. The analysis is carried out for air under perfect- and equilibrium-gas assumptions. The analyses range from a few simplified approaches to the solution of the complete Navier-Stokes equations. The numerical procedures are based on the solution of the Navier-Stokes and parabolized Navier-Stokes equations. Specific results obtained for spherically-blunted cones and ogives demonstrate that there are significant differences in flow-field and surface quantities between sharp and blunted bodies. Depending upon the flow conditions and geometry, the differences are found to persist as far as about 300 nose radii downstream.

Tiwari, Singh, and Sehgal⁽⁵⁾ calculated hypersonic flows over cones and straight biconic configurations for a wide range of freestream conditions in which the gas behind the shock is treated as perfect. Effect of angle of attack and nose bluntness on these slender cones in air is studied extensively. The numerical procedures are based on the solution of the complete Navier-Stokes equations over the nose section and parabolized Navier-Stokes equations further downstream. The flow field variables and surface quantities show significant differences when the angle of attack and nose bluntness are varied. The complete flow field is thoroughly analysed with respect to velocity, temperature, pressure, and entropy profiles. The post shock flow field is studied in detail

from the contour plots of Mach number, density, pressure, and temperature. The effect of nose bluntness for slender cones persists as far as 200 nose radii downstream.

Ashby and Cary⁽⁶⁾ conducted tests to determine the effect of nose shape, cylinder length, flare angle, and flare length on the longitudinal aerodynamic characteristics of nose-cylinder-flare bodies at a Mach number of 6.0. The two nose shapes used were a 22.5° conical and a hemispherical body. The two cylinders used measured four diameters and one diameter in length, and the flares used varied from 0° to 30° in angle and 0.61 to 3 cylinder diameters in length. The most important observation, from the present work point of view, was that both the nose shapes were found to be comparatively blunt. These forebody shapes introduced a sizeable variation in the local dynamic pressure between the nose shock and the body. Therefore, comparison of flare effectiveness on the basis of constant length, surface area, or diameter depends on the size of the flares. With the flare sizes used in this investigation, flare effectiveness increased with flare angle when the flare length was held constant, and also to a lesser extent with constant flare surface area. A part of this increase is expected from the geometric considerations, but the effectiveness of the larger flares was also influenced by the strong entropy gradient region. On the other hand for the constant diameter comparison all the flares were embedded in the low total pressure region so that only the axial force coefficient was affected by the flare angle. The flow field properties were calculated for an incidence of 0° using the method of characteristics.

Gray⁽⁷⁾ investigated laminar- and transitional-flow separation induced by flares and ramps of different angles and over a broad range of Reynolds numbers and at Mach numbers of 3.0, 5.0, and 7.0. Surface pressure distributions, schlieren and shadowgraph pictures, and the oil-film technique were used to determine the effect of transition during flow reattachment, on the scale of laminar separation. It was concluded that flow deflection angles less than 10° are required for investigation of laminar reattaching flows at similar test conditions because the transition is always triggered prematurely by the reattachment pressure gradient and by the relative instability of separated shear layer. The separation length increased with increasing Reynolds number, and the pressure distribution upstream of the flare was characterised by the absence of any

plateau whenever the flow was laminar through the reattachment zone. Nose blunting reduced the extent of such separation.

Ericsson ⁽⁸⁾ analysed the well documented effect of nose bluntness on the hypersonic aerodynamics of slender cones. It is shown that the combined effect of nose bluntness and semi-cone angle can be represented by a scaling parameter, and that the scaling concept can be extended to include the effects of moderate angles of attack. Using this generalised scaling concept it can be demonstrated that the approximation on which it is based introduces errors that are substantially smaller than the differences between tests in different wind tunnels. Thus, the author concludes that further parametric investigations of the effect of nose bluntness on hypersonic slender cone aerodynamics are not needed.

Raju and Reddy ⁽⁹⁾ measured the aerodynamic forces over a blunt nosed cone-cylinder body with and without flares and fins. They found good agreement between the modified Newtonian theory and the experimental results for all the test cases. These results are surprising because the formation of thick entropy layer should affect the performance of the flares as well as the fins. The modified Newtonian theory ignores entropy layer effects. Noticeable decreases in measured lift and pitching moment coefficients are obtained at $M_\infty = 9.15$ compared to those at $M_\infty = 3.85$. The dependence of drag coefficient on the freestream Mach number seems to be insignificant.

2.1.2 Slender elliptic bodies

Spencer and Fox ⁽¹⁰⁾ compared the aerodynamic characteristics of power-law bodies, $Y/R_b = (X/l)^{0.5}$, of circular and elliptic cross-section with that of a body of minimum-wave drag shape determined under the constraint of prescribed body length and volume. The power-law bodies are found to exhibit minimum zero-lift-drag as well as maximum lift-drag ratio for $n = 0.66$ for both the circular as well as the elliptical cross-section. However, the zero-lift drag of the theoretical minimum-wave-drag body is slightly lower and the resultant maximum lift-drag ratio is a little higher than that for the $n = 0.66$ power-law body. The increase in the ellipticity ratio for a given power-law or the theoretical minimum-wave-drag body results in an almost constant incremental increase in maximum lift-drag ratio, independent of body longitudinal contour. The

increase in ellipticity slightly reduces the minimum-drag coefficient, and increases the lift-curve slope for each body. Although the conical bodies ($n = 1.0$) have the highest lift-curve slope and minimum wetted area, the minimum-drag- or the $n = 0.66$ power-law-bodies have 25-30% higher lift-drag ratio. The centre of pressure location moves rearward from 53% for $n = 0.25$ to 67% of body length from the nose for $n = 1.0$.

Ashby⁽¹¹⁾ obtained experimental data for two series of bodies at Mach 6 and Reynolds numbers, based on model length, from 1.4 million to 9.5 million. One series consisted of axisymmetric power-law bodies geometrically constrained for constant length and base diameter with the exponent $n = 0.25, 0.5, 0.6, 0.667, 0.75$, and 1.0 . These models had a fineness ratio of 6.63 and were tested at incidences from -4° to 16° . It was found that the Reynolds number effects on drag and performance are significant for power-law bodies from $n = 0.5$ to 1.0 . At the higher Reynolds number the variation of boundary layer transition location with nose bluntness causes the drag to be a minimum for the $n \approx 0.6$ instead of the $n = 0.667$ body. At the highest Reynolds number, the power-law body for minimum drag is blunter (exponent n lower) than predicted by inviscid theory (n approximately 0.6 instead of $n = 0.667$); however, the peak value of lift-drag ratio occurs at $n = 0.667$.

Westby and Regan⁽¹²⁾ conducted an experimental investigation of the longitudinal aerodynamic characteristics of power-law bodies of revolution at Mach 12.8 in a gun-tunnel. The model length and the base diameter were kept constant at 127 mm and 63.5 mm respectively. The Reynolds number based on the body base diameter was 3.5×10^5 so that the boundary layer was likely to be laminar along the whole length of the model. Tests were conducted on four models with the power-law exponent, $n = 0.5, 0.667, 0.75$, and 1.0 at incidences from 0° to 20° . The axial force coefficient, C_A , was found to be minimum for the body with $n = 0.75$, however, C_A per unit volume was found to be minimum for the body with $n = 0.667$. The body with $n = 0.5$ exhibited maximum C_A at all the incidences tested. On the other hand the normal force coefficient, C_N , was found to be a minimum for the power-law body with $n = 0.5$ although the values for the other body shapes were found to be very close to each other. Finally the position of the centre of pressure was found to vary with the body shape. The centre of pressure moves further upstream with decreasing values of n from 1.0 to

0.5, as expected because of increased planform area near the nose of the blunter body shapes.

Mason and Lee ⁽¹³⁾ conducted a study of minimum-drag body shapes over a Mach number range from 3 to 12. Numerical results show that the power-law bodies earlier found to be minimum drag bodies (n equals 0.75 or 0.66, depending on the particular form of the theory) are really not so. Numerical results indicate that the power $n = 0.69$ ($l/d = 3$) or $n = 0.70$ ($l/d = 5$) shapes have lower drag than the theoretical minimum results (i.e. for $n = 0.75$ or 0.66). To evaluate the results, a numerical analysis was made, including viscous and real gas effects. None of these considerations altered the conclusions. The Hayes minimum-drag body was also analysed and had a higher drag than the optimum power-law body obtained from the numerical analysis.

Jorgensen ⁽¹⁴⁾ investigated the aerodynamic advantages of elliptic cones over circular cones. Experiments were conducted to determine the force and moment characteristics for elliptic cones at Mach numbers of 1.97 and 2.94. Elliptic cones having cross-sectional axis ratios from 1 to 6 and with lengths and base areas equal to circular cones of fineness ratios 3.67 and 5 were studied for the angle-of-attack range from 0 to about 16 degrees. The Reynolds number, based on model length, was 8×10^6 . Experimental investigations showed that bodies of elliptic cross-section exhibited higher lift to drag ratio than those of circular cross-section.

Graves ⁽¹⁵⁾ conducted an experimental investigation to compare the aerodynamic characteristics of a low drag missile concept with a body of circular cross-section, to one with a body of 3:1 elliptical cross-section, the bodies having identical cross-sectional area distributions. Tests were performed at Mach numbers from 0.5 to 4.63 and at angles of attack from about -5° to 28° . The comparison shows that at supersonic speeds the elliptical concept provides increasingly greater normal force up to Mach 2.5 to 3.0, beyond which an incremental increase of about 25% holds through the incidence range. The elliptical concept exhibited lesser longitudinal stability at all test Mach numbers. However, levels of lateral and directional stability are increased, particularly at the higher incidences.

Fournier and Spencer ⁽¹⁶⁾ investigated the aerodynamic characteristics of a series of elliptic bodies in the Mach number range from 1.5 to 4.63 and the incidence range

from -4° to 28° without any sideslip. The results indicated that increasing the power-law exponent (i.e. decreasing bluntness and increasing span), increases the lift curve slope at low angles of attack for a given value of a/b at all Mach numbers. The bodies attain a maximum value of lift to drag ratio at a value of the power-law exponent inbetween 0.5 and 0.66, especially at higher Mach number of 4.63. For all configurations tested, increasing the Mach number results in large reductions in minimum drag coefficient, as expected, so that the lift to drag ratio increases with Mach number since the lift coefficient is affected only slightly by Mach number.

2.2 Delta wing

A very large volume of data is available in the literature and reviews are available to describe the flow fields over delta wings at subsonic, supersonic, and hypersonic speeds ^(17, 18). Delta wing research has been stimulated as a result of the suitability of this planform for use on missiles, supersonic transports, supersonic fighter aircraft, aerospace planes, etc. etc. The pressure or windward side of the delta wing has an attached and orderly flow from 0° to 90° angle of incidence, however, the flow over the suction or the leeward side is found to be very complex. Delta wing research at subsonic and supersonic speeds has, therefore, been mainly concerned with the leeward side flow field. As a result this flow field is now well understood.

The interest in re-entry bodies during the '60s led to a large volume of experimental research work on the hypersonic aerodynamics of blunt leading edge delta wings at high incidence angles under NASA's X-20 Dyna-Soar program ⁽¹⁹⁻²³⁾. At hypersonic speeds the windward side of a wing or body is by far the largest contributor to the overall aerodynamic characteristics of a vehicle, the leeward side contribution being very small. In addition, the windward side experiences maximum heating so that most of the research on the aerodynamics of delta wings operating at hypersonic speeds, has been directed towards the study of flow over the windward surfaces. The experimental studies showed that windward flow field for blunt delta wings is very similar to that for the sharp delta wings with detached leading edge shocks. A few studies to investigate the flow fields over the windward as well as the leeward side of sharp delta wings with attached and detached leading edge shocks have been conducted ⁽²⁴⁻²⁸⁾.

Hefner and Whitehead ⁽²⁴⁾ made an experimental investigation of the flow field over two 60° swept delta wings of circular-arc and rhombic cross-section at a Mach number of 6.0. The windward surface pressure distributions correlate with the conical flow angle measured from the model centre line in the horizontal plane. The method of lines (a numerical technique) predicts both the magnitude and trend of the pressure data with better agreement occurring at higher angles of attack where the viscous effects are relatively small. The heat transfer distributions correlate with the normalised distance from the leading edge. The method of Spalding and Chi predicts turbulent levels of

heating over all the lower side of the wing. Oil-flow studies indicate that the windward flow is nearly two-dimensional for the incidence range from 0 to 10°. Vortices are present on the leeward side and are found to induce localised high heating in the region of the meridian. Whitehead and Keyes ⁽²⁹⁾ found that flow separation can occur at the leading edge at hypersonic speeds with a resulting flow behaviour somewhat similar to that found at the lower speeds. A coiled vortex sheet emanates from the leading edge and reattaches on the wing surface. The reduction in wing sweep from 75° to 70° moved the separation inboard so as to allow attached flow at the leading edges. At the lower speeds leading edge vortices cause suction but no such reduction in pressure below the vortex cores at the hypersonic speeds was found.

Jernell ⁽³⁰⁾ investigated the potential for optimising airfoil shape at high supersonic and hypersonic speeds using the two-dimensional shock-expansion method applied along the freestream direction. Theoretical and experimental force and moment coefficients for four delta planform wings, having 65° swept back leading edges, are compared. The wings incorporate modified diamond airfoils of maximum thickness-chord ratio of 0.06. The wings differ only in the position of maximum thickness and camber. The experimental data were obtained at Mach numbers of 3.95 and 4.63 and at a Reynolds number of 7×10^6 based on the wing root chord. The theory provides an accurate estimate of the relative effects of airfoil maximum-thickness position and camber, but overestimates the maximum lift to drag ratio by about 6-8%. A relatively simple method based upon the two-dimensional shock-expansion theory is suggested for predicting the optimum modified diamond airfoil shape at high supersonic and hypersonic speeds.

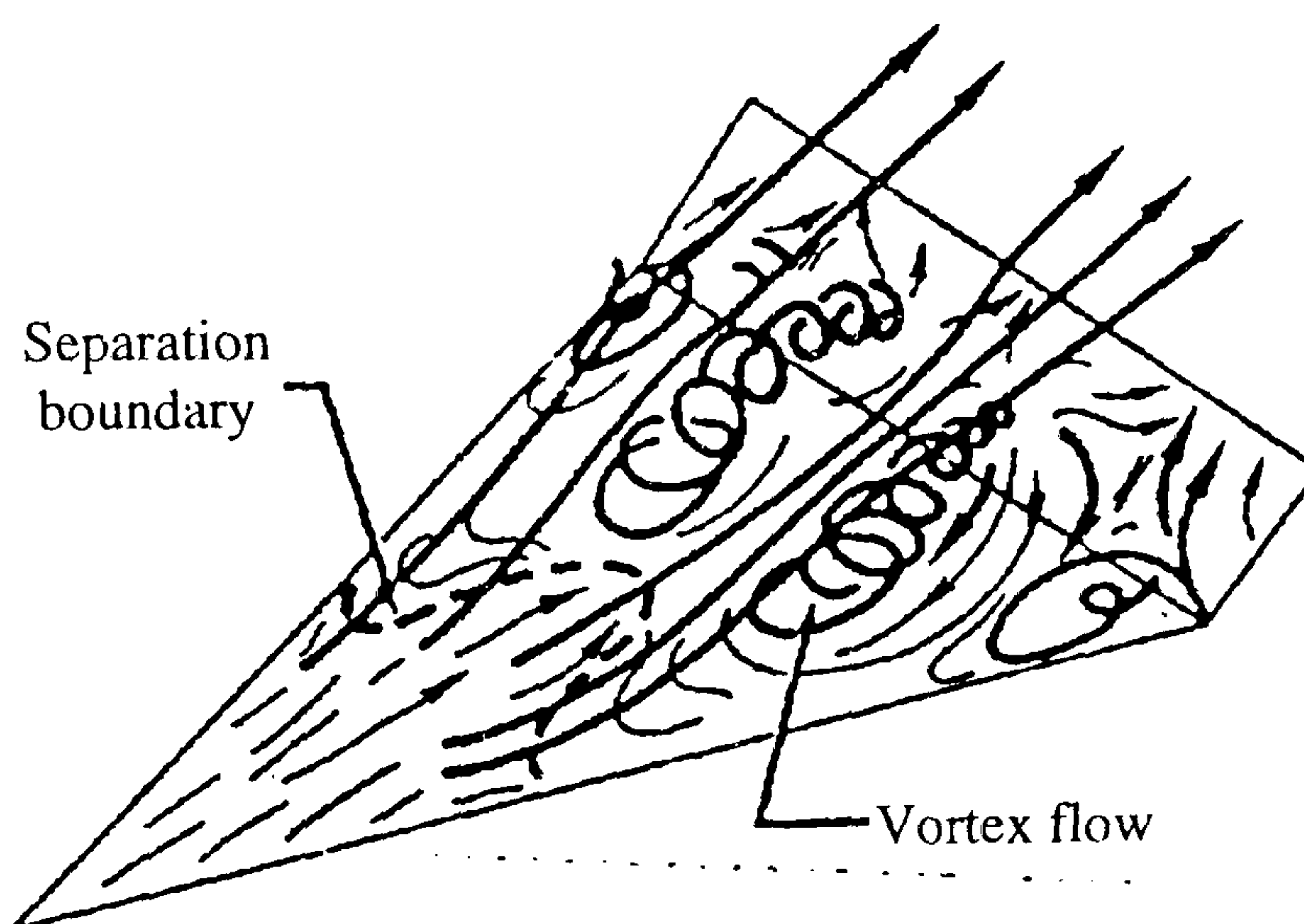
Babaev ⁽³¹⁾ obtained inviscid numerical solutions to the problem of flow over the windward surfaces of a flat delta wing with supersonic leading edges. It is concluded that at a given Mach number the normal force is practically independent of sweep angle because any pressure increase with sweep angle in the uniform region near the wing leading-edge is compensated for by a decrease in the apex region. It is therefore, possible to approximate the normal force contribution from the windward surfaces using two-dimensional theories such as the shock-expansion theory. The calculated shock

shapes fall inbetween the tangent-wedge and tangent-cone solutions in the plane of symmetry, lying closer to the tangent-cone value with increasing sweep.

Experimental force measurements by Rao ⁽³²⁾ and Opatowski ⁽³³⁾ at a freestream Mach number of 8.2 suggest that the normal force over a delta wing with the leading-edge shocks attached is well estimated by the two-dimensional shock theory. The surface pressure measurements on the windward side agree reasonably well with Squire's theory ⁽³⁴⁾ up to an incidence of 18° .

2.3 Flapped delta wing

Whitehead and Keyes ⁽²⁹⁾ investigated the Mach 6.0 flowfield over two highly swept ($\Lambda = 70^\circ$ and 75°) delta wings with trailing-edge flaps. Heat-transfer rates, surface pressure distributions, and several flow visualisation techniques were used. The 70° delta wing was tested at low incidences but the 75° delta wing was tested over a wide range from 0° to 90° . Positive flap deflection angles of 0° , 10° , 20° , 30° , and 40° were used for both wings. The freestream Reynolds number, based on the wing root chord, was varied from about 2.4×10^6 to 10.9×10^6 for the 70° delta wing and flap and from about 1.5×10^6 to 6.5×10^6 for the 75° delta wing and flap. A large sweep back angle of 70° is found to cause mixed flow over the wing, turbulent nearer the wing centre-line and laminar or transitional near the outward region of the wing. The end of transition for the delta wing with and without roughness elements near the leading edge was determined from heat transfer data. The end of transition for zero flap deflection angle at low incidences occurred along a line parallel to the wing leading edge. It was found that complex flow phenomena, like the development of vortices within the separated region which lift off the surface and reattach on the flap, occur if the type of boundary layer prior to separation differs across the span on the windward side. This was found to be the case with natural boundary layer transition and is depicted in a sketch given below.



A sketch showing the vortices on the windward side of a delta wing⁽²⁹⁾

The surface pressure in the separated flow region on the wing lower surface is found to be nearly constant across the span and is well predicted by the turbulent separation correlation ⁽³⁵⁾ for turbulent (obtained by tripping the boundary layer) as well as mixed-

flow separation. However, the surface pressure on the flap of 70° delta wing is found to increase along the span as the distance from the centre-line increases when the boundary layer is turbulent all over the span of the wing prior to flap induced separation, the behaviour is reversed for the mixed flow separation. This is true at all the instrumented locations, both upstream and downstream the flow reattachment on the flap. At low incidences when the boundary layer is turbulent over the span of the wing prior to separation, the surface pressure and heating can be estimated by a two-dimensional calculation along the centre-line of the wing and flap.

Keyes⁽³⁶⁾ reported the results of an experimental investigation to find the effect of flap deflection on the flow over a $\Lambda = 75^\circ$ delta wing for the angle of attack range of 0° to 90° and flap deflections from 0° to 40° at a freestream Mach number of 6. The freestream Reynolds number based on the wing root chord varied from 1.3×10^6 to 5.6×10^6 for 0° angle of attack and was 3.4×10^6 for angles greater than 0° . The boundary layer on the centre line in the vicinity of the hinge-line for $\alpha = 0^\circ$ is transitional at the lowest test Reynolds number and turbulent for higher Reynolds numbers, but near the wing leading edge it is laminar in both cases. The spanwise pressure variation due to the mixed-flow separation at $\alpha = 0^\circ$ (for the Reynolds number of 1.3×10^6) with transitional separation nearer the centre-line and laminar nearer the wing-edge has not been discussed. However, at low angles of attack (0° to 10°) with turbulent boundary layer near the centre line and laminar near the wing edge prior to separation, the spanwise pressures at a given chordwise station are nearly constant, except near the edges of the flap and when the flow separates. It was concluded, that even though the flow on the wing and flap is three-dimensional in nature, centre-line calculations based on existing two-dimensional methods were in good agreement with the experimental trends and in some cases predicted the maximum levels of the local pressures and heat transfer. Tangent-cone theory on the wing and oblique-shock theory on the flap were found to give good estimates of the pressure levels at moderate incidences.

Rao⁽³²⁾ carried out an extensive study of the flow and force characteristics of a 70° and a 76° delta wing with and without full-span trailing-edge flaps. Both the wings had a flat lower side, inverted-V top (with the upper ridge line inclined at 6° with respect to the lower surface) and blunt base. The wings were tested at a freestream Mach

number of 8.2. The freestream Reynolds number was 6.7×10^4 per cm for most of the tests but it was varied for a few tests to determine the unit Reynolds number effect on transition. The study shows a forward movement of transition with increasing Reynolds number and leading-edge sweep. The earlier onset of transition on the 76° delta wing produced consistently smaller separation lengths which resulted in better flap effectiveness on this wing at all incidences and flap angles. The flow visualisation studies showed evidence of mixed flow separation observed by Whitehead and Keyes⁽²⁹⁾, where the earlier onset of transition along the model centre-line delayed the separation there.

Edwards⁽³⁷⁾ conducted an experimental study of heat transfer distribution on the compression surface of a 70° delta wing. This wing was identical to the 70° delta wing used by Rao⁽³²⁾ and the tests were conducted at a Mach number of 8.2 and a freestream Reynolds number of 6.7×10^4 per cm. The main emphasis has been placed on the flow separation induced by a trailing-edge flap. The flow separation was found to be determined by the location of transition. Rao⁽³²⁾ found that the location of transition moves further upstream with an increase in the Reynolds number and the wing sweepback angle. Boundary layer development at the leading edge results in an increased pressure all along the edge so as to induce inward flow towards the centreline. The boundary layer on the wing centreline is therefore thickened to bring about an earlier onset of transition there. The flow over the lower side of a delta wing is therefore generally mixed in nature, being transitional or turbulent near the centreline and laminar further away^(29,32,36). The flap-induced separation on Edwards's delta wing is thought to be transitional although at low flap angles there was evidence of mixed flow separation. At zero incidence a two-dimensional strip theory is found to give a reasonable prediction of heat transfer distributions, except in the separated flow region. At higher incidences the theory over predicts the heat transfer rate on the flap which has strong three-dimensional effects.

Rao⁽²⁸⁾ reviewed the experimental data on the incipient separation characteristics of planar delta wings with 75° swept-back sharp leading edges and full-span trailing edge flaps deflected into the windward flow. The local Reynolds number range for these investigations covered laminar, transitional and turbulent conditions. It is shown that,

while turbulent boundary layer data correlates with two dimensional results, in the laminar and transitional cases, there is a nearly parallel shift to higher flap angles (approximately 2 to 5°) for incipient separation. Keyes⁽³⁶⁾ also found that the turbulent separation on the 75° sharp leading edges delta wing and that on flat plate at similar Mach and Reynolds number occurred at approximately the same flap angle.

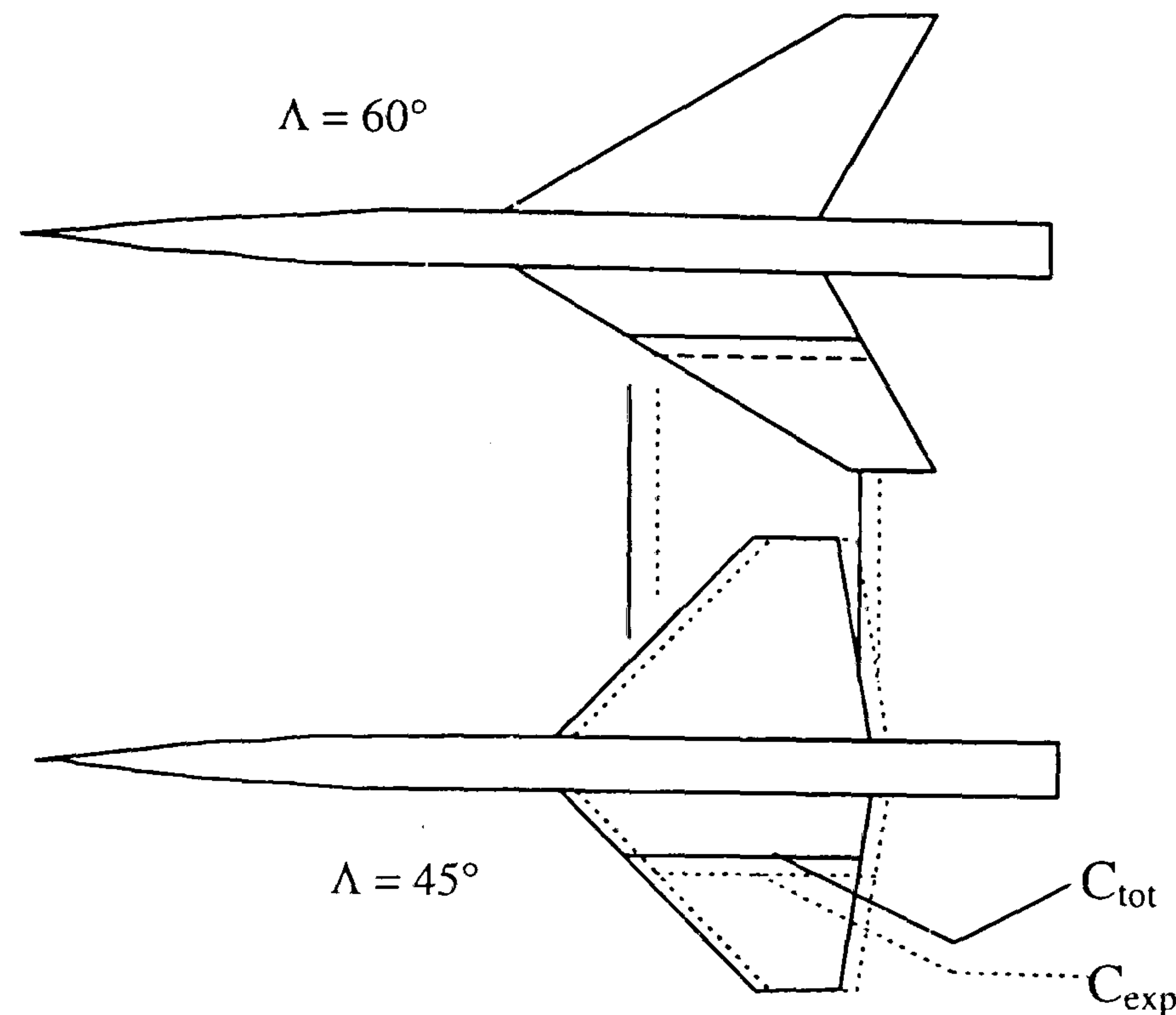
2.4 Delta wing-body

Jernell⁽³⁸⁾ reported the effectiveness of the shock-expansion method in predicting the surface pressure distribution over the wing of a wing-body configuration at high supersonic speeds. A delta wing, with 65° leading edge sweep and consisting of a 6% thick symmetrical double wedge aerofoil section, was mounted along the centreline of a body. The body consisted of a sharp contoured nose section and a cylindrical afterbody. The data show that at a Mach number of 4.63 and moderate angles of attack, the experimental pressure distributions over the wing surfaces affected by the expanded flow are practically constant and agree well with the estimates based upon two-dimensional Prandtl-Meyer expansion. However, over the lower surfaces of the wing at angle of attack considerable pressure gradients exist both in the chordwise and the spanwise direction. The average pressure coefficient can be estimated with reasonable accuracy using the two-dimensional shock-expansion method. Further, the data show a decrease in the pressure gradient and improvement in the agreement between the theory and experiment as the Mach number is increased.

Clark and Richie⁽³⁹⁾ investigated the hypersonic aerodynamic characteristics of an air-launched, delta-wing research aircraft concept at Mach 6. The effects of various components such as nose shape, wing camber, wing location, centre vertical tail, wing tip fins, forward delta wing, engine nacelle, and speed brakes were also studied. Tests were conducted with a 0.021 scale model at a Reynolds number, based on model length, of 10.5 million and over an angle of attack range from -4° to 20° .

Penland and Pittman⁽⁴⁰⁾ conducted an experimental investigation to determine the effect of wing leading edge sweep and wing translation on the aerodynamic characteristics of a wing body configuration at a freestream Mach number of about 6 and Reynolds number (based on body length) of 17.9×10^6 . Seven wings with leading edge sweep angles from -20° to 60° were tested on a common body over an angle of attack range from -12° to 10° . All wings had a common span, aspect ratio, taper ratio, planform area, and thickness ratio. Wings were translated longitudinally on the body to make tests possible with the total and exposed mean aerodynamic chords located at a fixed body station. Aerodynamic forces were found to be independent of wing sweep

and longitudinal position, and pitching moments were constant when the exposed wing mean aerodynamic chord was located at a fixed body station.



A sketch showing the two of the wing-body models⁽⁴⁰⁾

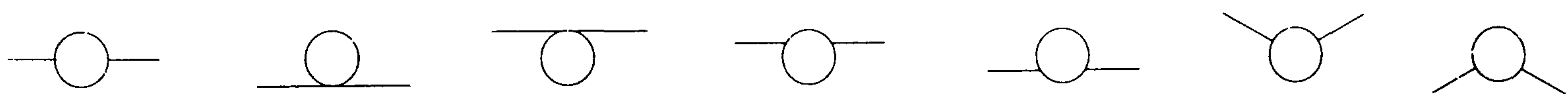
Theory applied with tangent wedge pressures on the wing and tangent cone pressures on the body provided excellent predictions of aerodynamic force coefficients but poor estimates of moment coefficients.

Dillon and Pittman⁽⁴¹⁾ investigated the static aerodynamic characteristics of a 1/30-scale model of a wing-body concept for a high speed research aeroplane in a Mach 6 wind tunnel. The model configuration was build-up from the basic body by adding a wing, centre vertical tail, three module scram jet, and six module scram jet engine. The tests were conducted at incidences from -4° to 20° with a constant sideslip angle of 0° , -2° , and -4° and the Reynolds number based on the model body length was 13.7×10^6 . The elevons were deflected from 10° to -15° for pitch control. The hypersonic arbitrary-body aerodynamic computer program⁽⁴²⁾ gave good predictions for the longitudinal but not for the lateral-directional aerodynamic characteristics.

Whitehead⁽⁴³⁾ investigated the aerodynamic characteristics of bodies and wing-body combinations with triangular, rectangular, and elliptical body cross-sectional shapes and with body width-height ratios of 2 and 3 at a freestream Mach number of 6.9 and a Reynolds number based on length of 1.4×10^6 . The two delta wings tested in

combinations with these bodies had leading-edge sweep angles of 70° and 75° . The results of the investigation show that for either bodies alone or wing-body combinations the possibility of an increase in the maximum lift-drag ratio with an increase in width-height ratio depends on the cross-sectional shape and the orientation of the configuration. For the flat-top wing-body combinations, neither the small increase in width-height ratio nor the change in cross-sectional shape from the basic conical to a triangular, rectangular, or elliptical wing-body combination produces any significant increase in the lift-drag ratio. However, for the flat-bottom wing-body combinations, body cross-sectional deviations from the conical body in some instances are shown to provide higher values of lift-drag ratio.

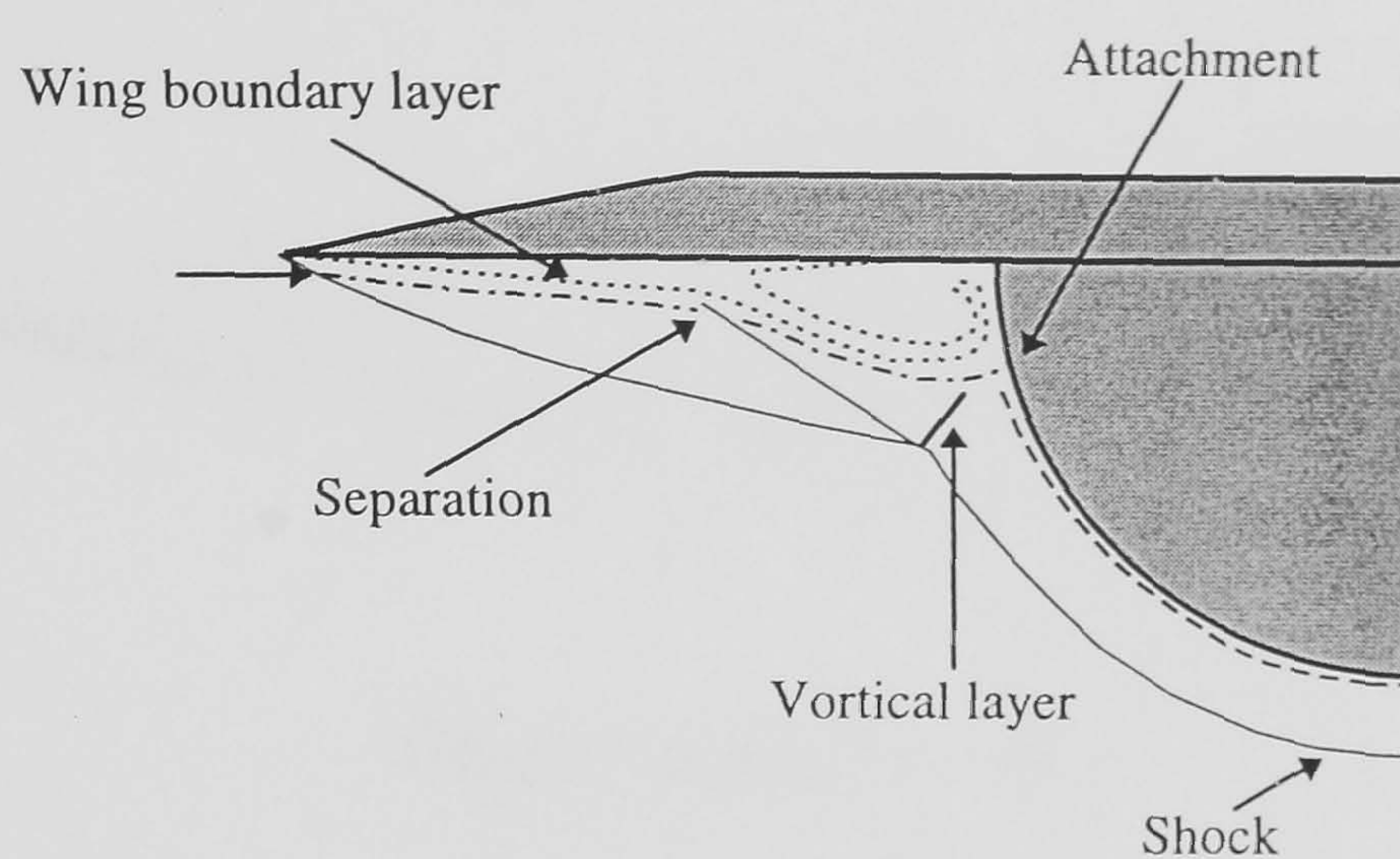
Allen and Watson ⁽⁴⁴⁾ performed an experimental study at supersonic speeds to measure wing and body spanwise pressure distributions on an axisymmetric-body delta wing model on which the vertical location of the wing on the body was systematically varied from low- to high-mounted positions. In addition, for two of these positions both horizontal and radial wing angular orientations relative to the body were tested. Roll angle effects were investigated for one of the positions. Seven different wing-body configurations and a body-alone configuration were studied. A sketch of the wing-body configurations showing the locations of wings on the body is given below.



The tests were conducted at Mach numbers from 1.7 to 2.86 at incidences from -4° to 24° . The important observation was that for a given incidence and at a roll angle of 0° , the pressures were virtually constant in the spanwise direction across the windward surfaces of the wing-body combinations. The vertical location of the wing on the body was found to have a very strong effect on the body surface pressures and the surface pressure on the lower side of the wing was affected favourably depending upon the vertical location of the wing.

Meyer and Vail ⁽⁴⁵⁾ reported experimental results for a flat-topped half-cone-and-delta-wing lifting configuration. The experiments were conducted on a model with a 60° swept back delta wing mounted on a 12° semi-cone angle flat-topped half-cone at a

Mach number of 12.6 and Reynolds number of 4.4×10^5 (based on the length of the model). A sketch of the configuration along with the important flow features is shown in the figure below.

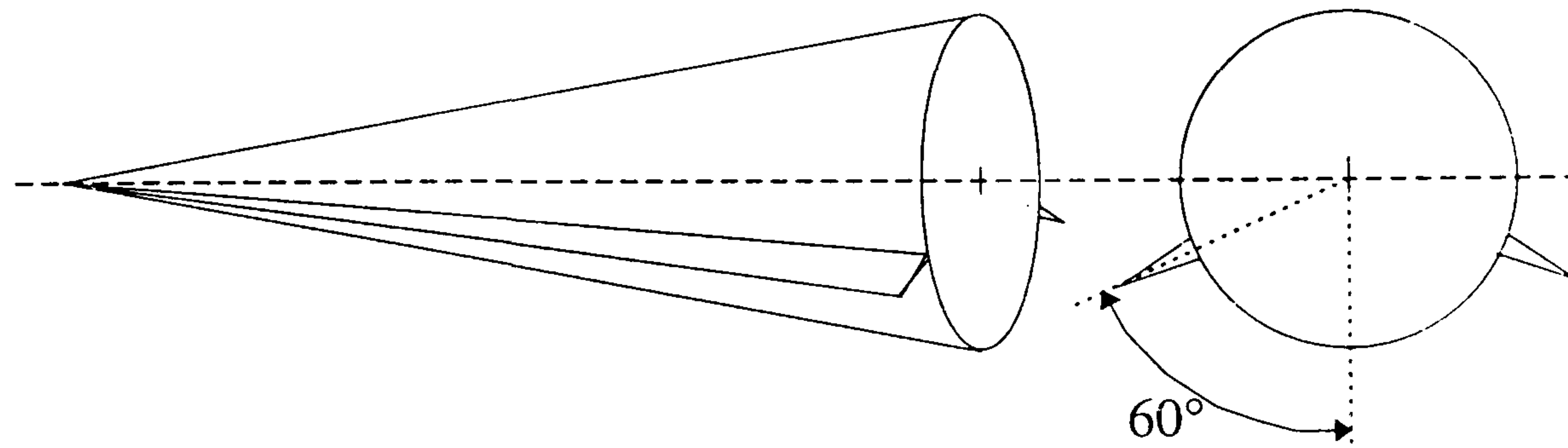


A sketch of the configuration along with the important flow features⁽⁴⁵⁾

The lifting effectiveness of the configuration is supposed to benefit from the favourable interference of the cone pressure field with the lower surface of the delta wing. It was found that flow separation is a particularly important feature of the flow leading to high rates of heat transfer at flow reattachment. At an incidence of 0° the laminar boundary layer over the lower surface of the wing separates as a result of interaction with the cone shock. However, the separated flow region collapses to a very small region near the wing-body junction as the incidence is increased to 30° because the cone shock merges with the detached wing leading edge shock.

Reggiori⁽⁴⁶⁾ conducted an experimental study to obtain surface pressure distribution and the total forces (lift and drag) on a wing-cone configuration. The experiments were performed at a freestream Mach number of 5.8 and the Reynolds number was varied from 7.7×10^5 to 15×10^6 for the pressure model and from 5×10^5 to 10×10^6 for the force model based up on the base diameter. The configuration consisted of a 20° right circular cone with 75° swept back delta wings located at 60° to the plane of symmetry as shown in the figure given below. The interference effect of the wing on the cone pressure is found to be very strong at positive angle of attack. The pressures on the windward side of the cone are found to be nearly constant. The force measurements

show that the lift-drag ratio is increased by both the negative as well as the positive dihedral wings but the negative dihedral configuration gives much larger increase in comparison to the simple cone.



Wing-cone configuration

EXPERIMENTAL SET-UP AND PROCEDURE

3.1 Gun-tunnel facility

The experimental investigation was conducted using the College of Aeronautics gun-tunnel. This tunnel was originally established at Imperial College, London. A description of the set-up is given by Stollery, Maull, and Belcher⁽⁴⁷⁾. Further description of the tunnel and the calibration of the facility is given by Needham⁽⁴⁸⁾. A sketch of the tunnel showing the main components is presented in figure 1. The mass of the piston used in this study was approximately equal to 100 grams. The test-section and the barrel end are separated by a Sellotape diaphragm on the upstream end of the nozzle assembly. 12 gauge (0.104 inch thick) commercial grade aluminium alloy (H15) diaphragms (15 cm square) were used in the double-diaphragms assembly. The tunnel is equipped with contoured axisymmetric nozzles for Mach numbers of 8.2 and 12.2. The Mach 8.2 nozzle, used in this study, was designed and constructed by Bristol Siddely Engines Limited. The test section flow calibration with Mach 8.2 contoured nozzle was done by Opatowski⁽⁴⁹⁾. Opatowski found the flow to be uniform inside the test section other than along the centre. Along the centre comparatively large variations in Mach number were found and are shown in Figure 2(a). The flow was found to be parallel to the floor to

within 0.1°. The axisymmetric contoured nozzle provided a useful jet of 15 cm diameter inside the test section.

The high pressure vessel was filled with air to a pressure of 2015 psia keeping the barrel at atmospheric pressure. The vacuum chamber and the test-section were evacuated to pressures less than one mm of mercury. The rupturing of the two aluminium diaphragms applies 2015 psia pressure on the piston placed at the reservoir end of the barrel. The piston is accelerated creating a shock ahead of it. The air entrapped ahead of the piston is compressed and heated and then expanded in the nozzle to give Mach 8.2 flow inside the test-section.

3.1.1 Mach number survey

Sometimes the model is required to be mounted in the test-section in such a way that a part of the model is inside the nozzle. It was therefore decided to calibrate the flow up-to 20 cm inside the nozzle. Limited Mach number calibration inside the nozzle and the test section showed significant variations along the axis, figure 2 (b) & (c). A pitot tube with a 15 psid strain gauge pressure transducer was used for measuring the pitot pressure for Mach number calibration. The transducer was calibrated against a vacuum gauge during the evacuation phase of the gun-tunnel run. A single measurement was done during each run by acquiring the transducer output on a transient recorder after suitable amplification and filtering through an active 2 Khz. low pass filter. The signal was then analysed with a 386 personal computer immediately after the run. A typical pressure signal trace is shown in figure 2(d).

Pitot pressures measured along, and at 2 cm from, the nozzle axis are shown in figure 2(b). These measurements had a repeatability of 2.5%. The Mach number was obtained from measured pitot pressures using the following relation

$$\frac{P_{02}}{P_{01}} = \left[\frac{0.5(\gamma + 1) M_1^2}{1 + 0.5(\gamma - 1) M_1^2} \right]^{\gamma/\gamma - 1} \left[\frac{2\gamma M_1^2 - (\gamma - 1)}{\gamma + 1} \right]^{-1/\gamma - 1}$$

Variation of Mach number along the nozzle axis and along a line 2 cm above the axis is shown in figure 2(c). There are large variations in Mach number along the axis of the nozzle. These variations are considerably reduced at other points inside the test diamond

and are thought to be because of the focusing effect of axisymmetric nozzles. The variations are reduced off the axis and the Mach number there can be considered to be uniform within $\pm 2\%$ of 8.2 inside the test diamond. To find out the lateral extent of uniform core flow at 20 cm inside the nozzle, pitot pressure measurements were made at points up to ± 5 cm from the nozzle axis. The Mach number variation along a vertical line normal to the axis at 20 cm inside the nozzle is shown in figure 2(e) and shows the flow is reasonably uniform in a core of 5 cm radius (except for along the nozzle axis) for which measurements have been done.

3.2 Models

Four different models were used for this study. A sketch of these models giving the basic dimensions is shown in figure 3.

1. Spherically blunted cone-cylinder model, figure 3(a).
2. Power-law body with and without strakes, figure 3(b).
3. Sharp leading edge flat plate with trailing-edge flap, figure 3(c)
4. Wing-body model with the above mentioned spherically blunted cone-cylinder body, figure 3(d) and 3(e).

3.2.1 Spherically blunted cone-cylinder model

The lift, drag and pitching moment coefficients for the spherically blunted cone-cylinder model were determined using a three-component strain-gauge balance. The model consisted of three separate pieces; a hemi-spherical nose, a 5° half angle cone frustum and a circular cylindrical section. The cone-cylinder portion of the body was made out of Jelutong wood. This wood is hard and light and is generally used for pattern making. However, the hemispherical nose of the body was made out of aluminium alloy to withstand the high heat transfer rate and dust erosion nearer the nose. Figure 3(a) shows the basic dimensions of the spherically blunted cone-cylinder model. The model was very light and weighed only 29 grams including the metallic adapter to rigidly mount the model on the balance sting.

Initial tests proved that the position of the model centre of gravity relative to balance moment centre (the point at which the model is attached to the balance), and the mass of the model affected the natural frequency of the balance/model combination. Increasing the mass and / or the distance between the model C. G. and the balance moment centre decreased the natural frequency, making it impossible to filter the noise without affecting the signal itself. The mass of the model was therefore, reduced to the minimum possible by hollowing it from inside and the C. G. was made to overlap the moment centre, by using dead weights at the base of the model, to get a natural frequency of around 200 Hz. The centre of gravity of the model and the balance moment centre was located at 5.2 cm from the model base.

3.2.2 Elliptic power-law body

Figure 3(b) shows the basic dimensions of the wooden models. The power-law body design co-ordinates are presented in Table 2. These models are 18.75 cm long. The strakes are approximately 1.67 mm square at the base. Dead weights were used at the base of the models to get the C. G. position at about 2.6 cm from the base. The mass of the models was 56 grams and 33 grams with and without strakes respectively. The models had an elliptical cross section and a one half-power-law contour along the length. The models were rigidly screwed on to the balance sting and fixed at zero roll angle. A photograph of the model is shown in figure 4(a).

3.2.3 Sharp flat plate

A flat plate model shown in figure 3(c) was used to obtain a few pictures to validate the behaviour of a liquid crystal layer on the model exposed to the gun tunnel flow. The model was made of steel and had a sharp leading edge. solid wedge blocks were used to simulate flap deflection angles of 5, 15, and 25°. A photograph of the model is shown in figure 4(b).

3.2.4 Wing-body model

Three different models were constructed for the experimental investigation of the hypersonic flow over the flapped wing-body model. A simple aluminium alloy model was constructed during the first phase of the experimental schedule for flow visualisation using schlieren, oil-dot, and liquid crystal techniques. During the second phase detailed surface pressure measurements were made on the lower side of the delta wing of the wing-body model. The pressure model had an integral flap surface hinged to the wing. To keep the model construction simple, the two flaps were made as a single piece running through the cylindrical portion of the body. The flap angle was changed by replacing a small section near the base of the body. This construction introduced a small difference inbetween the geometries of the wing-body models used for flow visualisation and that used for pressure measurements. The pressure model had no gap inbetween the body and the flap root because the flap was really made in a single piece. However, two separate wedge blocks were used to simulate the flap deflection on the

port and starboard sides of the wing-body model, causing a gap inbetween the body surface and the flap blocks. Finally a set of four very light-weight force models were used to measure the lift, drag and the pitching moment. These models were identical to the wing-body pressure model.

3.2.4.1 Flow visualisation model

The model was assembled from separate pieces of hemispherical nose, 5° half angle cone, cylindrical body with a groove to fix the 70° swept back delta wing. Solid wedge pieces of 5° , 10° , 15° , 20° , 25° , and 30° angles were rigidly attached to the wing (one on the port and one on the starboard side) to simulate the flap deflection angles of 0° , 5° , 10° , 15° , 20° and 25° . A photograph of the model along with the add-on wedge pieces is shown in figure 4(c). These tests involving oil-dot flow and schlieren photographs were later repeated using the wing-body pressure model to find out any significant difference in the flow in the absence of any gap inbetween the body surface and the flap root.

3.2.4.2 Pressure model

The wing-body pressure model was made of steel to withstand the necessary wear and tear of the longer test schedule. It was possible to vary the flap angle from -5° to 0° , 10° , 15° , and 25° . The small clearance inbetween the wing and the flap surface was filled and smoothed with plasticine to make it air-tight. The model was originally constructed to have 58 pressure tapings, however four tapings nearest to the wing tip were found to interfere with the flow over the lower side of the wing and were therefore removed. All the pressure measurements were therefore done at 54 pressure tapings. The pressure tapings were distributed over the two sides of the wing. There were 29 orifices on the port side and 25 over the starboard side of the wing. A sketch of the pressure model showing the location of the pressure tapings is shown in Figure 3(e). The location of the pressure tapings is tabulated in Table 1. The pressure tapings consisted of 1.672 mm internal diameter steel tubes inserted in to the wing and rigidly fixed with an adhesive. These tubes were flush with the wing surface on the windward side but protruded about 10 mm out on the leeward side of the wing. A photograph of

the model is shown in figure 4(d). No pressure measurements were made on the spherically blunted cone-cylinder and the power-law body.

3.2.4.3 Force models

The wing-body model used for the force measurements was required to be as light as possible. The cone-cylinder portion of the body was made out of Jelutong wood, as already mentioned. However, the hemispherical nose and the wing was made out of aluminium alloy to withstand the excessive aerodynamic forces and temperatures. A set of four models were made each with a fixed flap deflection angle i.e. 0° , 5° , 15° , and 25° . A photograph of these models is shown in figure 4(e). The mass of these models was about 65 grams including the mass of the adapter to rigidly mount the model on the balance.

The mass of the model was reduced to the minimum possible and the C. G. was made to overlap the moment centre to get a natural frequency of around 200 Hz. The model C. G. and the balance moment centre were located at 5.2 cm from the base of the model.

3.3 Flow visualisation

Three different methods of flow visualisation were used. The schlieren technique is well established for the visualisation of the density gradients throughout the flow field so as to enable the flow structures like shock waves, vortices, boundary layer separation and transition etc. to be determined. The second technique using liquid crystals has been used by many researchers as a surface flow visualisation technique. However, an effort was needed to establish the technique in the gun-tunnel facility at Cranfield University. Babinsky and Stollery⁽⁵⁰⁾ used a high speed video camera to acquire the liquid crystal pictures of their spaceplane model. The present work not only used a different liquid crystal formulation but also used a single shot camera to record the images. Some experimentation was required to find the optimum relative locations of the camera and the light source. It was required to know the optimum type of photographic film, camera settings, light source, thickness of the liquid crystal layer to be deposited and the time instant during the tunnel run time for acquiring the image because the technique was being used for the first time.

The surface oil flow technique quite routinely used in wind-tunnel experimentation is not suitable for gun-tunnels because the oil film can not be moved in the short run duration. Discrete oil dots were therefore tried. A large number of mixtures and proportions were tested to get a suitable mixture which does not flow during almost 45 minutes of tunnel preparation, but flows sufficiently to register the surface flow direction when the tunnel is fired.

3.3.1 Schlieren technique

The tunnel is equipped with a single pass conventional optical arrangement for taking schlieren pictures. A short duration (one hundred nanoseconds to one microsecond) spark source is used as a point light source. A Polaroid camera using ISO 3000 film is used for recording the density gradients inside the test-section.

3.3.2 Liquid-crystal technique

Liquid crystals have been available for a long time but they have only found application as a diagnostic tool in aerodynamic and aerothermal studies during the last

30 years. Many researchers have reported promising results regarding the use of the crystals as a temperature measuring tool. However, these crystals are found to be particularly useful for the preliminary study of an unknown thermal flow field enabling the accurate positioning of other well established temperature measuring transducers and also for a qualitative analysis of the thermal field over a hypersonic vehicle. In general an organic compound is optically isotropic in its liquid phase at temperatures above its melting point and nonisotropic in its crystalline solid phase. However, certain organic compounds exhibit behaviour inbetween the isotropic liquid and the nonisotropic crystalline solids forms. They exhibit optical properties characteristic of a crystalline (solid) phase but mechanical properties characteristic of a liquid phase. These compounds are commonly called liquid crystals.

Cholestric liquid crystals have a molecular structure containing a large number of cholestrol compounds. These crystals make very thin molecular microlayers with the long axis of the molecules parallel to the plane of the layers. The molecular axis in the adjacent layers traces a helical path. The Cholestric liquid crystals exhibit a peculiar characteristic called circular dichroism. The electric vector of the incident white light is split into clockwise and anticlockwise rotating components. One component is transmitted through the crystals and the other is reflected back giving rise to a colour change. The wavelength reflected back depends upon the temperature, shear stress, pressure, and orientation of the incident and the reflected light, the composition of the organic compound, the imposed electric and magnetic fields and other factors which can change the pitch of the helical microlayer molecular structure.

The use of shear sensitive liquid crystals has become an established technique for diagnostic flow visualisation. Smith⁽⁵¹⁾ presented an overview of the state of the art of the liquid-crystal technique, including the historical development and a discussion of how it is used. Sample results from several researchers were used to demonstrate the range of flow features that can be illustrated, including laminar boundary layer transition, laminar separation bubbles, shocks, and separation. The technique has been demonstrated in flight and wind-tunnel environments from subsonic to hypersonic speeds. Reda and Aeschliman⁽⁵²⁾ conducted experiments to test the surface shear-stress capabilities of shear-stress sensitive liquid crystal compounds in hypersonic flows.

Liquid crystal coatings were applied to the surface of a conical model which was then exposed to a high unit Reynolds number ($2.3 \times 10^7/\text{m}$) Mach 5 flow. The model was illuminated by white light and the response of the liquid crystal layer was recorded with standard video and high-speed movie cameras. Abrupt changes in the surface shear stress (for example due to a transition front) are made visible by an abrupt change in the colour of the liquid crystal layer. The technique was demonstrated to be a viable diagnostic tool for use in transient/compressible flow.

The pressure sensitivity of the liquid crystals is important only if the range of operation is close to the phase change. The liquid crystals are made insensitive to shear stress if they are to be used for temperature measurements. This is done by enclosing the crystals in micro capsules of 5-30 μm diameter. Encapsulated liquid crystals are believed to have a spatial resolution of a fraction of a millimetre and surface temperatures can be measured to within 2°C with a negligible response to shear stress. Smith and Baxter ⁽⁵³⁾ and Babinsky and Edwards ⁽⁵⁴⁾ demonstrated the use of encapsulated liquid crystals to measure surface heat flux in short duration hypersonic facilities. Haq, Roberts, and East ⁽⁵⁵⁾ used the technique for qualitative analysis of the surface heat transfer distribution near fin body junctions. Roberts, and East ⁽⁵⁶⁾ reviewed the use of liquid crystals for quantitative heat flux measurements in hypersonic wind tunnel facilities with run times $O(1)\text{sec}$.

In the present study encapsulated liquid crystals were used for qualitative heat transfer rate measurements. The model was made out of an aluminium alloy. The crystals reflect a certain component of the incident white light allowing the other components to pass through the crystal layer. The model was, therefore, painted matt black to make the crystals colour change more vivid. The image was recorded on a single shot camera film. The colours perceived depend both upon the orientation of the incident light as well as the viewing angle. It is therefore important to keep the set-up fixed throughout the test schedule to make a meaningful analysis. The set-up is shown in figure 5(a). The model was mounted so that the windward side of the model was facing the camera as model incidence varied from 0° to 10° . The SLR camera pointed in a direction normal to the optical window of the test-section. An electronic flash (Vivitar 3500) using extra wide manual mode was used to illuminate the model for

about half a millisecond at an incidence of about 45° to the test-section window. The flash was triggered using a delay circuit while the camera was used in 'B setting' mode. The liquid crystals mixture R23C10W supplied by Hallcrest was sprayed on the model surface. The crystals were mixed with binder in 1:3 ratio. This mixture was diluted with distilled water in 1:4 ratio to give a uniform layer of about $50\text{ }\mu\text{m}$. A single photograph was taken just before the end of each run, i.e. with a delay setting of 35 msec with respect to the time at which the reservoir end diaphragms are ruptured. The liquid crystals colour change temperature are given in Table 3.

Before the gun tunnel run the initial colour of the model is dependent upon test section ambient temperature. During these experiments the initial colour of the model was always light red so that regions of high and low temperature can be detected by eye on the photographs. The model used for these experiments had flap angles of 0, 5, 15, and 25 degrees.

3.3.3 Oil-dot flow technique

Meyer and Vail⁽⁴⁵⁾ used the oil-dot flow technique to study the surface flow over the windward side of flat-topped half-cone-and-delta-wing lifting configuration. A number of dots of high vacuum oil, coloured with lamp black were placed on the surface of the model. Depending upon the magnitude of the shear stress, these dots flow to indicate the direction of the limiting streamlines. The length of the oil streaks formed were found to give a good qualitative indication of the shear stress distribution. Rao⁽³²⁾ used the technique to study the three-dimensional flow over his flapped delta wing in the same gun tunnel facility in which the present tests have been conducted. The method was found to be useful for locating reattachment boundaries as well as reflected wave impingement over the lower side of the trailing-edge flap.

In the present study, discrete dots of a mixture consisting of linseed oil, high vacuum silicone oil and titanium dioxide powder with a few drops of oleic acid were applied on the matt-black painted surface of the model. An oil-dot pattern on the model before the run is shown in figure 5(b). The oil dots flow in a direction depending upon the surface shear stress. However, because of reduced shear stress over the separated flow regions the oil dots do not flow. Similarly at reattachment the oil dots are wiped

away because of large increase in shear stress there. The oil-dot flow pattern was recorded on a 35 mm black and white film immediately after the completion of each run.

3.4 Instrumentation and data acquisition

The surface pressures over the lower side of the wing were measured using a set of eight transducers. The lift, drag, and pitching moment coefficients for the spherically blunted cone-cylinder body, the wing-body, and the power-law body with and without strakes were measured using a three component balance.

The data acquisition system used for acquiring the data from the above mentioned instruments is shown in figure 6. The system consists of a ten channel analogue Flyde amplifier, an eight channel Datalab DL2800 series transient recorder, and a 386SX personal computer. The signal from the transducers can be amplified by a factor of up to 1000 before recording on the transient recorder. The data is stored as a 10 bit digital signal. A maximum of 4096 data points can be recorded on each channel. Four out of the eight channels have a maximum sampling frequency of 2 MHz, while the rest of the four have 0.2 MHz. However, a maximum sampling frequency of 50 KHz was used to ensure the recording of the entire tunnel run along with a few samples before the firing and a few after the end of the run. The data acquired by the transient recorder is transferred to the 386SX computer via a KERMIT link at a rate of 9600 bits/sec for further analysis.

The tunnel is equipped with a trigger and delay circuit to capture various events during the operational 25 msec run time of the tunnel. The sound generated by rupturing of the diaphragms is picked up by a microphone. There is a time lag of around 12 msec inbetween the rupturing of the diaphragms and the actual establishment of the Mach 8.2 hypersonic flow inside the test-section. The microphone output is used to trigger a 5 volt square wave generator which in turn triggers the recording mechanism of various pieces of equipment. However, as various events are recorded during a particular period of the tunnel run a certain amount of time delay is introduced inbetween the firing of the tunnel and the start of the event recording. The schlieren pictures are taken about 15 msec after the firing of the tunnel but the liquid crystal photographs are taken 25 msec after the firing of the tunnel, i.e. just before the end of the run.

3.4.1 Pressure transducers

All pressure measurements were made on the windward side of the wing. A pressure transducer housing, consisting of eight Kulite 0-15 psia miniature strain gauge transducers, was rigidly fixed at the foot of the model mounting. This was helpful in preventing leakage because the test section pressure was of the same order of magnitude as the freestream static pressure and secondly the transducers were not affected by any ambient temperature variations. The surface pressure at eight locations on the wing was measured in each run by connecting eight pressure tapings to as many transducers by approximately 15 cm long PVC tubings. The pressure tapings not connected to the transducers were effectively sealed using plasticine. The pressure transducers were calibrated against a vacuum gauge during the test-section evacuation phase of each run. The transducer outputs after suitable amplification were acquired on eight channels of the Datalab transient recorder to obtain 4096 data points from each transducer at a sampling rate of 50 samples/msec. The transient recorder stores the incoming analogue signal by dividing the voltage full scale setting of the recorder channel into 1024 steps. It is, therefore, very important to keep the VFS setting sufficiently close to the expected signal so as to have good resolution and at the same time the VFS setting should be high enough not to saturate the recorder channel. A typical pressure trace is shown in figure 7. The transient recorder output was acquired and analysed on a 386SX computer after the end of each run. 10 bit digitised signal output is processed at the end of the run to obtain the surface pressure at each of the 8 orifice locations using the following algorithm

$$p = \frac{(Tr - Tr_i)}{1024 \times G} \times VFS \times C + p_i$$

Where ' p_i ', and ' Tr_i ' are the initial static pressure and the initial transient recorder reading before the firing of the tunnel and ' C ' is the calibration constant of a particular transducer obtained during the evacuation phase of the run. ' Tr ' is recorder reading after the firing of the tunnel and ' G ' is the amplifier gain used. The raw data was converted into ratio of local surface pressure to freestream static pressure, p/p_∞ .

3.4.2 Three-component strain gauge balance

A three component strain gauge balance was used to measure axial, and normal forces and pitching moment. Miniaturised silicon strain gauges are used to measure the strain in the sensing elements for the three components of the balance. Each component uses four strain gauges to form a Wheatstone bridge powered by a regulated dc power supply. The balance uses thin tension / compression links as sensing elements to measure lift and drag but the pitching moment is sensed by a cantilever link on which the model is mounted. Further details of the balance are given by Opatowski ⁽⁴⁹⁾. A photograph of the balance is given in figure 8.

During the operation of the tunnel the test section was found to vibrate. These vibrations are picked up by the balance to give a high amplitude noise carrier to the signal. The transient recorder voltage-full-scale setting (VFS) had to be very high in comparison with the expected signal voltage in order to capture the signal riding over high amplitude noise. The transient recorder resolution depends upon the VFS selected because it records the signal by dividing the VFS into 1024 equal steps. It was found that the resulting resolution was too low to extract the actual signal from the noise so that any filtering of the recorder output to get rid of the noise was useless. It was therefore decided to build three active 170 Hz low pass filters to pre-filter the input signal to the recorder. This arrangement was found to be satisfactory for the measurement of comparatively small forces over the slender models under consideration.

The balance is generally excited with a 5V dc power supply but for these tests 7.5V was used to enhance the sensitivity of the balance because the forces, particularly drag, for the slender models under investigation were very low at small incidences. The balance output was low pass filtered through a 170 Hz active filter after suitable amplification. The filtered signal was then acquired on three channels of Datalab transient recorder at a sampling rate of 50 samples per msec. 10 bit digitised signal output is processed at the end of the run to obtain forces and pitching moment using the following algorithm

$$F = \frac{(Tr - Tr_i)}{1024 \times G} \times VFS \times C$$

Where C is the calibration constant obtained by static calibration of the balance.

3.4.2.1 Balance calibration

The balance was calibrated using a locally made test rig. The rig provides a suitable stable platform for the balance as well as allowing application of pure axial, and normal forces and pure torque loads using a series of weights and pulleys. A pure load (Normal, axial, or pitching moment) is applied to the balance and the output from the three channels of the balance is noted. Readings are taken for both increasing and decreasing loads. The calibration curves are shown in Appendix-1. A method proposed by Rae and Pope⁽⁵⁷⁾ was used to obtain the balance calibration matrix. A typical filtered signal trace from the balance for a hemi-spherically blunted cone-cylinder is shown in figure 9.

The model incidence was changed by bolting the balance to the existing linkage described by Opatowski⁽⁴⁹⁾. The incidence settings were measured using an inclinometer with an accuracy of ± 1 minute. Output from each component of the balance is amplified and low pass filtered before acquiring it on the transient recorder. An error analysis by Opatowski⁽⁴⁹⁾ shows that the maximum possible error for the three balance channels is as given below:

Normal force: 7.5%

Axial force: 10%

Pitching moment: 7.5%

However, the above analysis had an error of 3% due to trace reading and 1% due to the oscilloscope. The present study used computerised data acquisition and analysis instead of the oscilloscope used by Opatowski so that maximum possible errors involved should be 4% less than those given above.

The measurements by Rao⁽³²⁾ were found to have the overall accuracy given below:

Normal force: 6%

Axial force: 7.5%

Pitching moment: 4.5%

In the present study the repeatability of the results was found to be very good. The calibration of the balance was repeated many times and showed insignificant difference. In fact the calibration constants with balance excitation voltage increased

from 3 to 7.5 Volts showed a simple increase by 2.5 times. The present results are expected to have maximum overall errors given below:

Normal force: 3.5%

Axial force: 6%

Pitching moment: 3.5%

3.5 Test Conditions

The tests were conducted in the College of Aeronautics gun-tunnel at a freestream Mach number, $M_\infty = 8.2$ and a unit Reynolds number, $Re_\infty = 9.35 \times 10^4 / \text{cm}$. Other freestream test conditions are summarised in a table given below. The model was at room temperature so the tests can be considered as cold wall tests.

$M_\infty = 8.2$	$Re_\infty = 9.35 \times 10^4 / \text{cm}$
$P_{0\infty} = 1580 \text{ Psia} \cong 109 \times 10^5 \text{ N/m}^2$	$P_\infty = 0.138 \text{ Psia} \cong 951.5 \text{ N/m}^2$
$T_{0\infty} = 1290 \text{ }^\circ\text{K}$	$T_\infty = 89.3 \text{ }^\circ\text{K}$
$a_\infty = 189.4 \text{ m/sec}$	$V_\infty = 1553 \text{ m/sec}$
$\rho_\infty = 0.0371 \text{ kg/m}^3$	$\mu_\infty = 6.161 \text{ kg/m-sec}$

Freestream conditions

RESULTS AND DISCUSSION

4.1 Spherically blunted cone-cylinder

A spherically blunted cone-cylinder model shown in figure 3(a) was used for this investigation. The experimental programme consisted of measuring the lift, drag and pitching moment characteristics of the model in the incidence range from 0° to 10° using the three component balance. In addition, schlieren pictures were taken to find out about the flow field around the body. The schlieren pictures showed flow separation on the leeward side as found by Stetson⁽⁵⁸⁾.

4.1.1 Theoretical estimates of the forces and moments for a spherically blunted body

The equations obtained by Trimmer⁽⁵⁹⁾ were used to obtain theoretical estimates for the lift, drag, pitching moment and centre of pressure of the spherically blunted cone portion of the body. The contribution of the cylindrical portion of the body was then added to obtain theoretical estimates of the aerodynamic characteristics of the complete body. The equations are separated into two incidence ranges to take care of the shadow region. The pressure coefficient is assumed to be zero in the shadow region. A comparison of experimental aerodynamic coefficients, for a sharp 9° half angle cone at $M_\infty = 6.77$ over an angle of attack range from 0 to 180° by Neal⁽⁶⁰⁾, shows better

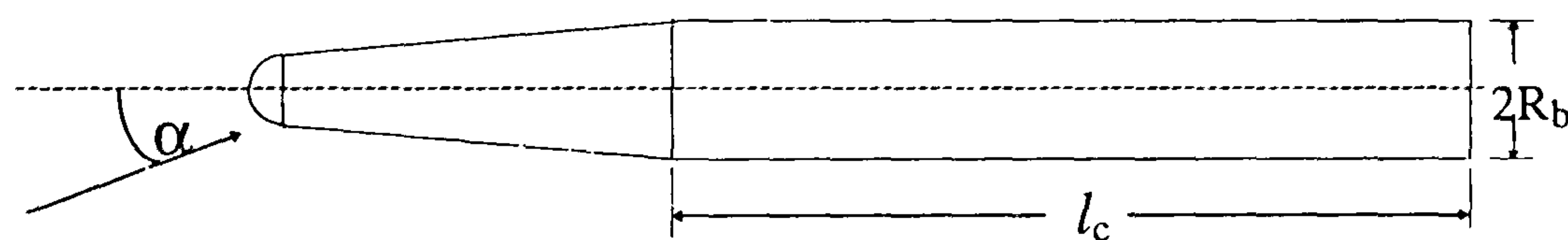
agreement with the coefficients calculated using the Newtonian flow model than that with the modified Newtonian flow model⁽⁶¹⁾. However, Trimmer suggests that for blunt bodies the modified Newtonian flow model may give better results. It was, therefore, decided to use $K = 1.828$ instead of 2. The equations obtained by Trimmer are given in Appendix 2.

For the cylindrical portion of the body the axial force was considered zero for the inviscid flow and the centre of pressure at half length. The normal force coefficient is given by

$$C_{N_{Cylinder}} = \frac{8}{3\pi} \frac{l_c}{R_b} \sin^2 \alpha, \text{ from the Newtonian theory and if the centrifugal forces}$$

are also considered (Newton-Busemann theory) then,

$$C_{N_{Cylinder}} = \frac{2.4}{\pi} \frac{l_c}{R_b} \sin^2 \alpha$$



Effect of nose bluntness on the characteristics

The stagnation point convective heat transfer is roughly proportional to $(\rho_\infty)^{0.5} (U_\infty)^3 (R_N)^{-0.5}$. Thus, a blunter body will have a lower convective heat transfer rate at the stagnation point although for velocities greater than orbital velocities, radiative heating may become significant as a result of dissociation and ionisation of the gas inside the shock layer. The stagnation point radiative heat transfer to the body is roughly proportional to $(\rho_\infty)^{1.6} (U_\infty)^{8.5} R_N$, indicating an increase in radiative heat transfer rate with increase in nose radius. So that a blunter body will experience higher stagnation-point radiative heat-transfer rate. The nose radius becomes a conflicting requirement for re-entry bodies entering the atmosphere at velocities above the orbital speeds.

Nose bluntness creates a low density, high temperature, variable entropy, and variable stagnation pressure layer next to the surface of the body persisting for hundreds of nose radii downstream. The thickness of this layer depends upon the amount of

bluntness, freestream Mach number and the incidence of the body. Near the nose the boundary layer grows inside the entropy layer which affects transition due to the changing boundary layer edge conditions. Eventually the whole of the entropy layer flow is swallowed by the boundary layer but not in the tests reported here.

Trimmer's equations were used to assess the influence of nose bluntness on the aerodynamic characteristics of the body alone and the results are shown in figure 10. Nose bluntness, R_N/R_b , was increased from 0 to 0.5 keeping the cone angle and the base radius constant. Figure 10(a) shows the non-linear nature of the C_L - α curve. This is expected because the Newtonian pressure coefficient varies with $\sin^2\alpha$. The effect is small but the slope of the curve decreases with an increase in bluntness. The normal force due to the pressure acting on the conical surface area is reduced because of the reduction in the conical area for a constant semi-vertex angle. The reduction in the lift coefficient with increasing bluntness is found to be small because of the increased contribution of the hemispherical nose. The drag coefficient increases significantly with increasing bluntness, although the increase is more at lower, and less at higher incidences, figure 10(b). The pitching moment coefficient about the moment centre (5.2 cm from base) decreases with increasing bluntness partly because the lift coefficient decreases and partly due to the shift in the location of the centre of pressure towards the moment centre, figure 10(c). Finally, the lift to drag ratio reduces drastically with increasing nose radius because of reduction in lift and increase in drag coefficient. The incidence at which the maximum value of L/D is achieved increases with bluntness, figure 10(d).

4.1.2 Navier-Stokes computations

A three-dimensional Navier-Stokes solver developed by Qin and Richards⁽⁶²⁾ has been used to calculate the flow field around the blunted cone-cylinder body. Flow field results for two cases of $\alpha = 0^\circ$ and $\alpha = 10^\circ$ are obtained assuming a perfect gas and a laminar boundary layer over the full length of the body. The freestream conditions used are $M_\infty = 8.2$ and $Re_\infty = 9.35 \times 10^4 / \text{cm}$. A $65 \times 65 \times 65$ grid is used for discretization of the flow field. The Navier-Stokes solution converged to the experimental shock position after about 6000 iterations on a Silicon Graphics workstation Indigo-2. The flow field

results plotted in the form of contour plots using the 'Fieldview' software package are shown in figure 11, 12 and 13.

Figure 11 shows density, velocity, entropy and total stagnation pressure contours inside the shock layer for the $\alpha = 0^\circ$ case. The bow shock due to the hemispherical nose is clearly visible in figure 11(a), showing the density contours. There is a large increase in density shown by the red coloured contours as air is compressed there through the normal shock wave. However, the density is reduced as the flow expands around the nose. The density is further reduced as the flow is expanded around the cone-cylinder junction. The relative position of the boundary layer can be seen in the velocity contour plot shown in figure 11(c). The laminar boundary layer is very thin over the cone. The thickness increases over the cylinder but still remains very thin in comparison to the low density region around the body.

Figure 11(b) shows the entropy contours inside the shock layer. The main feature of this figure is the existence of an almost constant thickness entropy layer over the cylindrical portion of the body. This feature is important for the present study because the wing as well as the trailing-edge flaps of the wing-body model will be immersed in this thick entropy layer. The figure shows that the thickness of the entropy layer at the body base is more than the base radius. The effect of this entropy layer on the aerodynamic characteristics of delta wing fitted with trailing-edge flaps will be considered later while discussing the wing-body results. It was not possible to obtain the N-S solutions for the wing-body during the limited duration of present study. Figure 11(d) shows the reservoir pressure contours inside the shock layer. The bow shock causes a thick region of low reservoir pressure around the body. The thickness of this region at the base of the body is nearly equal to the diameter of the cylindrical section of the body i.e. at $X/R_N = 35.4$ the thickness of the low stagnation pressure region is about 4 times the nose radius.

Figure 12 shows density, velocity, and entropy contours inside the shock layer for the $\alpha = 10^\circ$ case. The effect of increase in incidence is to decrease the thickness of the shock layer as it gets compressed on the windward side and a large increase on the leeward side of the body due to the expansion of the flow, figure 12(a). The thickness of

the entropy layer is likewise reduced on the windward and increased on the leeward side of the body.

Figure 13 compares the entropy and Mach number contours for $\alpha = 0^\circ$ and 10° . The delta wing on the wing-body under consideration was mounted along the centre line of the body in the horizontal plane. The variations in the shock layer, in the horizontal plane, with incidence are shown in figure 13(a), (b), (c), and (d). In these figures, flowfield on the port side of the body is shown because in the horizontal plane the flow remains symmetrical at zero yaw angle. Figure 13(a), (b) show a thick entropy layer of constant thickness over the cylindrical portion of the body as has been described earlier. However, figure 13(c), and (d) show a significant reduction in the thickness of the entropy layer as the incidence increases to 10° . The layer is found to taper towards the base of the body. Figure 13(e) compares the Mach number contours at the base across the length of the body for $\alpha = 0$ and 10° . The contours are symmetrical about the body axis and show a thick region of low Mach number as a result of loss through the stronger portion of the bow shock at $\alpha = 0^\circ$. The picture is much more complicated for $\alpha = 10^\circ$, being no more symmetrical about the body axis. The flow is separated on the leeward side forming a strong vortical flow.

4.1.3 Experimental characteristics of the body

4.1.3.1 Flow visualisation

Schlieren pictures for the body alone model are shown in figure 14. The spherically blunted nose creates a strong bow shock. At zero incidence the bow shock is concentric with the body. An expansion fan is generated at the cone-cylinder junction. It can be seen as a black line representing the leading expansion wave over the top side and as a white line over the lower side, at approximately half-way inbetween the body surface and the bow shock. Nothing can be said about the boundary layer as it is obscured by the effects of entropy layer created by the blunt nose. It is believed that the laminar boundary layer is attached everywhere over the surface of the body. As the incidence is increased the bow shock no longer remains concentric. It moves further away from the surface on the top side but closer to the surface on the lower side. The

shock radius of curvature increases over the top side making the entropy layer thicker than that on the lower side. The pictures do not show any cross flow separation on the leeward side up to $\alpha = 7^\circ$ but for $\alpha = 8^\circ$, figure 14(h), there is clear indication of separation. It is thought that the cross flow separates to form a pair of counter rotating vortices on the leeward side. A black line starting from about the middle of the cylinder length shows the vortex core. Further increase in incidence to 10° moves the separation point further up to about the cone-cylinder junction. Stetson⁽⁵⁸⁾ found that the leeward flow over a cone separates at about $3/4$ the cone half angle for sharp as well as a spherically blunted 5.6 degree half angle cone. He proposed a flow model containing symmetrical supersonic helical vortices with an attachment line on the most leeward ray. The vortices are in contact with the surface (at least up to $\alpha = 18^\circ$) and there is no subsonic reverse flow or singular points associated with the vortex pattern.

4.1.3.2 Aerodynamic characteristics of the body

The lift, drag and pitching moment of the body alone were measured using the three component strain gauge balance in the incidence range from 0° to 10° . The measured values are compared with theoretical calculations. The Newtonian flow model was used for theoretical analysis because the method is very simple and is expected to give reasonable estimates for the type of geometric model and freestream conditions under consideration. Figure 15 compares estimated and experimental variations of lift, drag, pitching moment, ratio of lift to drag, and the centre of pressure with change in incidence of the spherically blunted cone-cylinder body.

Lift Coefficient

Figure 15(a) shows the variation of lift coefficient with incidence. The estimated values using modified Newtonian theory compare reasonably well with the experimental values. The experimental values are slightly above Newtonian values. The gap further widens above $\alpha = 5^\circ$, the angle around which the leeward flow is expected to separate forming counter rotating leeward vortices. At hypersonic speeds the pressure distribution on the leeward side generally does not contribute much towards the aerodynamic forces, however, over the present model leeward suction created by flow separation increases C_L values further beyond that predicted by Newtonian flow model.

Newtonian flow model assumes zero pressure coefficient in the shadow region of the body. The experimental results agree very well with the N-S results (only two cases of $\alpha = 0$, and 10° were computed).

Drag Coefficient

Figure 15(b) compares the drag values. The experimental values are lower than the theoretical values even though Newtonian values neglect viscous and base drag. Zoby and Thompson⁽⁶³⁾ obtained flow field solutions for a blunted 5° half angle cone in a Mach 15 flow at an altitude of 150,000 feet using a three dimensional viscous shock layer code. Detailed computations assuming both transitional as well as fully laminar flow showed that increasing the nose radius from 0.125 to 0.75 ft., for a fixed cone half angle and base radius, reduces both the drag coefficient and the convective stagnation point heat transfer rate. The overall heating rates to the vehicle surface are significantly reduced indicating a similar reduction in skin friction drag. However, the experimental measurements by Campbell⁽⁶⁴⁾ and Cleary⁽³⁾ showed that the drag coefficient of a blunted cone is always more than that of a sharp cone. The skin friction and base drag for the spherically blunted cone-cylinder model should be very small in comparison to the pressure drag. Glover and Hagan⁽⁶⁵⁾ note that the pressure drag for blunted bodies may account for nearly 100% of the total drag. The Newtonian pressure over the surface of the blunt cone-cylinder model depends only upon the local surface inclination relative to the freestream. The pressure is a maximum at the stagnation point, reducing over the spherical cap before becoming constant over the conical generator. However, for actual flow the surface pressure may be under-expanded or over-expanded depending upon the distance from the vertex, the cone half angle and the freestream Mach number. Analysis by Zoby and Thompson showed that the surface pressure over the smaller nose radius body is constant and equal to the sharp cone pressure beyond 200 nose radii from the vertex with under-expansion before it. However, the surface pressure was found to be lower than the sharp cone value over most of the cone with the larger nose radius. Detailed surface pressure measurements by Stetson⁽⁵⁸⁾ also showed that the flow over a blunted cone over-expands near the nose before recompression to return to the approximate sharp cone value. It is therefore believed that the surface pressure over most of the conical portion of the blunted cone-cylinder body will have under-expansion

resulting in a reduction of drag coefficient in comparison to the modified Newtonian values. The experimental drag coefficient value for $\alpha = 10^\circ$ is in good agreement with the N-S value but the agreement for $\alpha = 0^\circ$ is poor for unknown reasons.

Pitching moment Coefficient

The variation in pitching moment coefficient, figure 15(c), closely follows the theoretical estimate indicating the body becoming increasingly unstable with incidence. This is to be expected because the centre of pressure lies well ahead of the moment centre (point about which the moments are taken). At a non-zero incidence the cylindrical section of the body starts producing normal force so as to shift the centre of pressure of the body downstream towards the base, figure 15(e). However, in the incidence range for which the tests have been conducted the effect of increased body normal force is more dominant than the rearward shift in the centre of pressure. Again, the N-S results agree very well with the experimental result at $\alpha = 10^\circ$.

Lift-Drag ratio

The experimental L/D ratio shows improvement over the estimate above 5° incidence due to increased suction over the leeward side, figure 14(d).

4.2 Half-power-law elliptic body

The surface co-ordinates of a power law body are given by $Y/R_b = (X/l)^n$, where
Y - is the local radius of the body
 R_b - body base radius
X - axial distance from the nose
l - body length, and
n - the power exponent

The $n = 1$ value represents a sharp cone and a reduction in the value of n from 1 to 0 continuously increases the bluntness of the body. In general nose bluntness is useful to reduce the stagnation heating as well as to increase the volume for a given fineness ratio (l/d) body. One way of introducing the nose bluntness is to use a hemi-spherically blunted nose which is preferable for vehicles using terminal homing guidance systems. However, power-law bodies with elliptical cross-section are attractive to obtain favourable aerodynamic characteristics and internal volume for a given body fineness ratio.

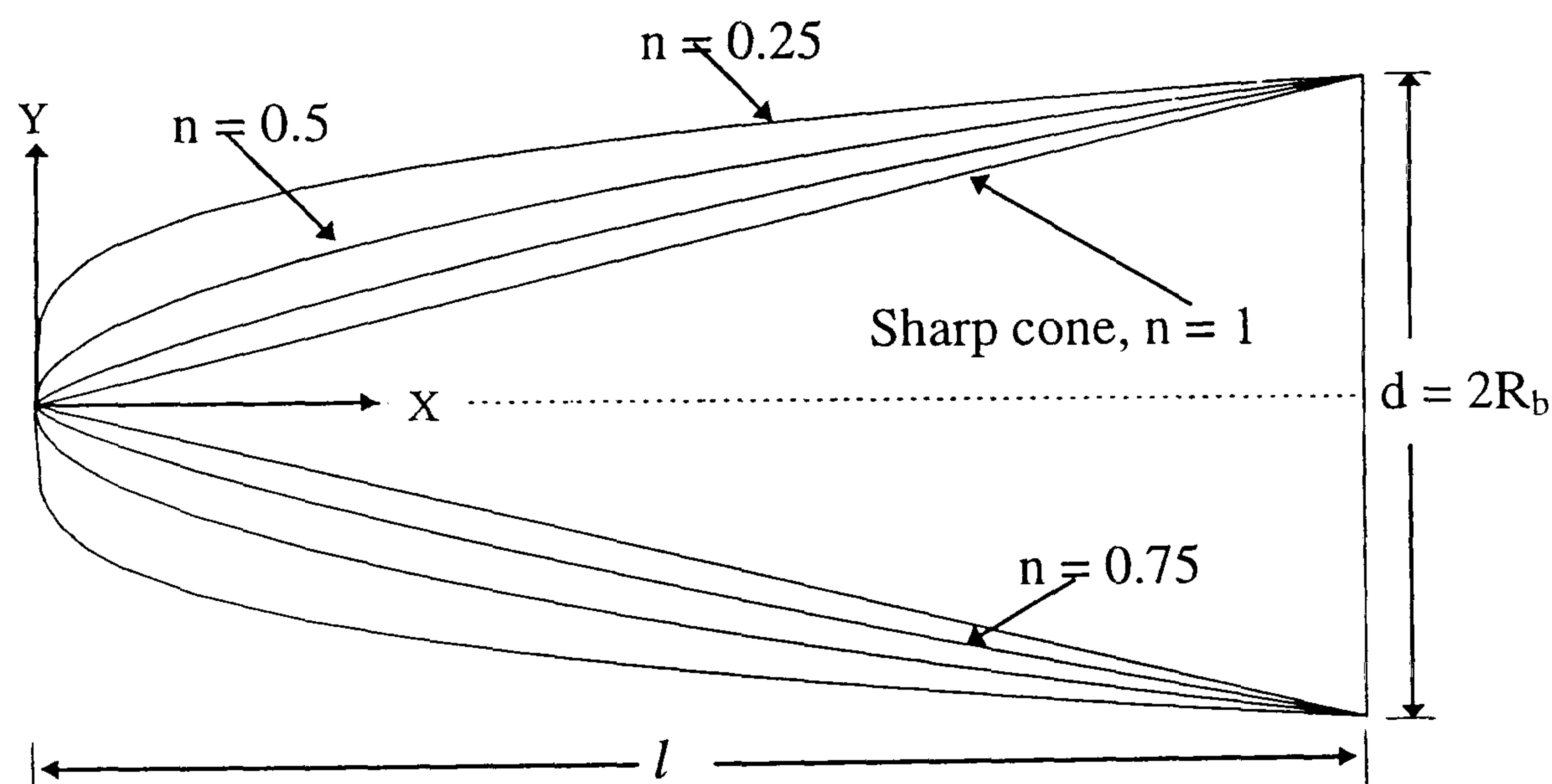


Figure showing the surface contour of power-law bodies

In the present study experiments were conducted to obtain lift, drag and pitching-moment coefficients for a one-half power-law body having an elliptical cross-section of $a/b = 1.6$. The force measurements were made with and without strakes attached to the power-law body. All the tests were conducted keeping the major axis of the body in the horizontal plane. It was believed that the addition of very small span

strakes will result in a favourable interference inbetween the body and the strakes so as to improve the lift to drag ratio of the straked configuration.

The experiments were repeated by Kontis⁽⁶⁶⁾ to obtain more detailed measurements. A comparison of the present measurements with those of Kontis is shown in figure 16(a) and 16(b). In general a very good repeatability is achieved for the measurements for the power-law body configuration without the strakes. However, there is a scatter in the data for the centre of pressure location and Kontis's data shows a higher value of the maximum L/D ratio because it occurs at an incidence for which the tests were not conducted in the present study.

4.2.1 Force measurements

Schlieren pictures for the elliptical cone model are shown in figure 17. No significant difference in schlieren pictures was noticed due to the inclusion of strakes on the cone model. This is because any changes brought about by the inclusion of the strakes will not be visible in the schlieren pictures because of the body obstruction. The leading edge shock and attached laminar boundary layer for $\alpha = 0^\circ$, figure 17(a), can be clearly seen. However, at $\alpha = 3^\circ$ the lee side flow separates close to the base of the model, probably to form a pair of counter rotating vortices. Stetson⁽⁵⁸⁾ found that the leeward flow over a cone separates at about 3/4 the cone half angle for sharp as well as a spherically blunted 5.6 degree half angle cones. The incidence at which the cross flow over a power law body separates will probably depend upon the power-law exponent. The white line nearer the surface on the leeward side indicates the vortex core. As the incidence increases the separation point moves forward and by $\alpha = 9^\circ$ the separation region extends over the whole upper surface. At $\alpha = 15^\circ$ the vortices are no longer visible. This is probably because the large expansion of the flow over the leeward side of the body reduces the local density beyond the sensitivity of the schlieren system.

The two strakes on the windward side of the body have negative dihedral and the other two on the leeward side have positive dihedral. Any interference between the flow over the body and that over the windward side of the anhedral strakes or over the leeward side of the dihedral strakes will result in changes in the aerodynamic characteristics of the Straked configuration in comparison to the body alone

configuration. Any interference effects on the leeward side of the anhedral strakes and that on the windward side of the dihedral strakes will be acting on a very small area of the body surface and will therefore not cause any significant changes in the aerodynamic characteristics. A comparison of C_L , C_D , C_m , L/D , and X_{cp}/L versus incidence curves for the elliptical cone with and without strakes is presented at figure 18. It was not possible to compare the present measurements for an elliptical power-law body-with-strakes with other data because no other results are available. However, the lift and drag measurements for elliptical power-law body without strakes are compared with the experimental results obtained by Fournier and Spencer⁽¹⁶⁾ for a one-half-power-law elliptic body at $M_\infty = 4.63$ and by Fox and Spencer⁽⁶⁷⁾ for a theoretical minimum-wave-drag body (volume and length constraints) at $M_\infty = 10.03$, figure 19.

4.2.1.1 Lift coefficient

The effect of the addition of strakes is to shift the C_L - α curve upwards increasing the C_L value at all positive incidences at which the tests have been done, figure 18(a). This increase is due to increased pressure over the lower side of the cone caused by increased amount of air entrapped by the strakes. The two strakes having negative dihedral create strong interference with the windward side of the body and increase the surface pressure on the body. The cross flow component on the windward side remains very small due to the blockage created by the anhedral wing and should therefore result in a more or less constant pressure on the windward side of the body and the anhedral strakes. Similarly, expansion of the flow over the leeward surfaces of the other two strakes having positive dihedral reduces the pressure on the leeward side of the body in comparison to the pressure there for a body alone configuration. The favourable interference created by the strakes is thus increasing the lift coefficient at all the incidences tested. The surface pressure measurements were not conducted in this study, however surface pressure measurements by Reggiori⁽⁴⁶⁾ on a wing-cone configuration show these interference effects.

Figure 19(a) compares the power-law body alone lift coefficient measurements obtained during the present study with those reported by Fournier and Spencer⁽¹⁶⁾. The elliptical cone models used in the present study and that used by Fournier and Spencer

are identical so far as the power-law exponent is concerned ($n = 0.5$) but the tests have been conducted at different Mach numbers and also the ellipticity ratio, a/b , is different in the two studies. The present results obtained for $a/b = 1.6$ fall inbetween those for $a/b = 1$ and 2 obtained by Fournier and Spencer, following the trend of increasing C_L with increasing a/b . This increase in C_L is primarily due to the increased aspect ratio of the body. The effect of Mach number variation on the C_L - α curve, for the range under consideration, seems to be negligible as concluded by Fournier and Spencer for the Mach number variation below 4.63. The lift curve is non-linear with its slope increasing with α and following the well known α^2 variation. The separation of lee side flow further increases the slope due to increased suction over the leeward side of the cone.

Figure 19(b) compares the present lift measurements with that of Fox and Spencer⁽⁶⁷⁾ for elliptical cone alone configuration. The measurements by Fox and Spencer were obtained on a theoretical minimum-wave-drag body (volume and length constraints) of elliptical cross-section with $a/b = 2.0$, however this body did not have a power-law longitudinal contour and the tests were conducted at $M_\infty=10.03$. The minimum-wave-drag body consistently produces less lift than the half power law body of lesser a/b ratio.

4.2.1.2 Drag coefficient

Figure 18(b) shows that the addition of the strakes increases the drag coefficient of the configuration, as expected. However, the increase in C_D is small particularly at the lower incidences. This is probably because a one-half-power law body is a comparatively blunt body so that the major source of the drag is the wave drag due to the detached nose shock. The addition of strakes on the present configuration does not change the nose shape and hence the nose shock, any increase in the drag coefficient due to comparatively small strakes is therefore expected to be small.

The variation of C_D with α at $M = 4.63^{(16)}$ and at $M = 8.2$ for the power-law body alone configuration is compared in figure 19(a). The drag measurements by Fournier and Spencer show that the drag coefficient does not vary significantly with the variation in a/b from 1 to 2, however comparison of their results with the present measurements show a large reduction in C_D values close to $\alpha = 0^\circ$ as the Mach number

varies from 4.63 to 8.2. This reduction in the minimum drag coefficient is believed to be due to the reduction in the base drag as the Mach number is increased from 4.63 to 8.2.

Figure 19(b) compares the present drag measurements with that of Fox and Spencer⁽⁶⁷⁾ for elliptical cone alone configuration. The two bodies have approximately same zero-lift drag but at higher incidences the half power-law body has significantly higher drag than the minimum wave-drag-body.

4.2.1.3 Pitching moment coefficient

Figure 18(c) shows that at lower incidences the addition of the strakes does not cause any significant change in the pitching moment coefficient about the balance moment-centre. The scatter in the data is of the order of the change in the coefficient shown in the figure. At lower incidences, the effect of increased lift is being balanced by a downstream shift in the centre of pressure. However, at higher incidence there is an increase in the pitching moment coefficient because the increase in the lift coefficient for the cone with strakes over that without the strakes continues to increase with α . Any shift in the centre of pressure as a consequence of the addition of the strakes is constant with α .

4.2.1.4 Lift-Drag ratio

The L/D ratio determines the overall aerodynamic efficiency of a lifting configuration. Both the lift and the drag coefficients increase with the addition of the strakes. However, there is an overall increase in L/D ratio by approximately 17% with the addition of strakes, figure 18(e).

Figure 19(a) compares the power-law body alone L/D ratio obtained during the present study with that reported by Fournier and Spencer⁽¹⁶⁾. The L/D ratio increases with an increase in a/b ratio because increase in a/b ratio increases the lift coefficient without any significant change in the drag coefficient so that L/D values for the a/b=1.6 configuration should lie inbetween the values for the a/b=1.0 and a/b=2.0 configurations. However, as a consequence of Mach number effect which reduces the drag coefficient at the low incidences, the L/D ratio for the a/b=1.6 configuration tested at $M_\infty = 8.2$ is in fact more than that of the a/b = 2.0 configuration tested at $M_\infty = 4.63$.

Figure 19(b) compares the present lift-drag measurements with that of Fox and Spencer⁽⁶⁷⁾ for the elliptical cone alone configuration. On the whole the minimum-wave-drag body does perform better than the power-law body by displaying higher L/D ratios at most of the incidences under consideration.

4.2.1.5 Centre of pressure position

The measurements show a scatter in the location of the centre of pressure with the average value of 0.64 for the cone with the strakes and of 0.6 without the strakes, figure 18(d). The difference is well covered by the scatter so that there is no noticeable shift in the centre of pressure due to the addition of the strakes or due to the change in the incidence of the two configurations. According to slender body theory⁽⁶⁸⁾ the location of the centre of pressure of power-law bodies is given by $\frac{x_{cp}}{l} = \frac{2n}{n+1}$, for $n \neq 1$.

For $n = 0.5$, this relationship gives $x_{cp}/l = 0.67$ which is very near to the experimental value obtained.

4.2.2 conclusions

1. Forces measured on a half power law slender elliptical cone compare reasonably well with the existing results, figure 19.
2. The flow separates on the leeward side of the body at a very low incidence. The separation spreads further forward with incidence, figure 17.
3. The addition of strakes on the configuration tested increases the lift as well as the drag but an overall increase in L/D is realised, figure 18.

4.3 Sharp flat-plate with flap

A few experiments were conducted to gain experience in taking schlieren and the liquid crystals pictures. The well known flow field around a sharp leading edge flat plate with a trailing edge flap was used for this purpose. It was hoped that these experiments would be helpful in interpreting the pictures for the flow field around the wing-body model.

4.3.1 Schlieren photography

A schlieren photograph showing the flow field on a sharp leading edge flat plate at zero incidence with a 25° deflected flap is shown in figure 20. The photograph shows the leading edge shock as a result of viscous interaction there. Next to the surface of the flat plate a white line indicates the density gradient near the edge of the thermal boundary layer. At hypersonic speeds the edge of the thermal boundary layer and the laminar velocity boundary layer are fairly close. The flow separates well ahead of the hinge line as is indicated by sudden movement of the white line away from the plate surface. In addition a separation shock can be seen to be emanating from just above the separation point. The separated shear layer attaches to the flap, becomes very thin and turns parallel to the flap surface. The reattachment shock can be seen in the photograph. Thus flow separation and reattachment on this model can easily be detected from the schlieren pictures.

4.3.2 Liquid crystals flow visualisation

Vehicles operating at hypersonic speeds experience aerodynamic heating due to the temperature gradient inside the boundary layer. The temperature gradient is produced as the high speed flow is retarded by the viscous effects converting the kinetic energy into thermal energy. Heat transfer rate to the skin is given by Fourier's law of heat conduction,

$$\dot{q} = -k \left[\frac{\partial T}{\partial Y} \right]_{Y=0}$$

where the temperature gradient in the boundary layer is evaluated at the wall. $\dot{q}$ depends upon flow field stagnation conditions, wall temperature, vehicle incidence, type

of boundary layer, and viscous effects. Heat transfer data are generally expressed in terms of heat transfer coefficients defined as

$$C_h = q / \rho_\infty U_\infty C_p (T_r - T_w)$$

where T_r is recovery temperature and can be approximated by the following equation

$$T_r = T_e \left[1 + \left(\frac{\gamma - 1}{2} \right) \sqrt{\text{Pr}} M_e^2 \right]$$

Various flow phenomena such as the state of the boundary layer, separation, vortex development, transition etc. can be detected by surface heat transfer. For example the general trend of the heat transfer rate distribution over a sharp flat plate model⁽³⁷⁾ with natural transition from laminar to turbulent flow is shown in figure 21. These flow phenomena can therefore be detected if the surface temperature distribution over the body under consideration could be determined. The experimental techniques used to determine surface temperature distribution can be broadly grouped into two categories :-

- (a) Electrical sensors such as thermocouples and thin film gauges etc. Platinum thin film gauges deposited on an insulating substrate are generally used to measure the surface temperature.
- (b) Temperature sensitive surface coatings such as encapsulated liquid crystals and thermographic surface coatings. Temperature sensitive encapsulated liquid crystals were used to get a qualitative picture of the heat transfer distribution over the model to detect the surface flow phenomena. This technique can be used to obtain a quantitative heat transfer distribution by a suitable colour change versus temperature calibration and using computerised image analysis to detect the colour change. Direct colour change detection by the naked eye can be used for a qualitative heat transfer distribution only.

The technique was initially applied to a flapped sharp leading edge flat plate because the flow field is comparatively well understood and the heat transfer ahead of the hinge line for attached flow can be evaluated using simple engineering methods (e.g. the reference temperature method) with reasonable accuracy. A more accurate distribution determined experimentally using thin film gauges was available at a later stage so that a comparison could be made. The liquid crystal pictures showing the change in surface temperature distribution with a change in flap angle from 0° to 25° are presented in figure 22. A comparison of heat transfer values ahead of the hinge line

calculated using the reference temperature method and thin film gauge measurements is shown in figure 23.

The Reference Temperature Method

The reference temperature concept makes use of the simplified expressions available for the prediction of skin friction and heat transfer coefficients for a flat plate in incompressible flow. However in the extension to compressible flow, the thermodynamic and transport properties in these expressions are calculated at a reference temperature, T^* , indicative of the temperature somewhere inside the boundary layer. The reference temperature varies with Mach no. and wall temperature. Many formulations are available for T^* , but that due to Eckert⁽⁶⁹⁾ is generally used. Eckert's formulation is based on numerical solutions evaluated by Young and Janssen⁽⁷⁰⁾ and is given by

$$T^* = T_e + 0.5(T_w - T_e) + 0.22(T_r - T_e)$$

For laminar plate flow, the skin friction coefficient can be written as

$$C_f \approx \frac{0.664\sqrt{C^*}}{\sqrt{\text{Re}_x}}$$

Where C^* is Chapman-Rubesin constant and can be calculated from the following approximation

$$C^* = \frac{\rho^* \mu^*}{\rho_e \mu_e} \approx \left(\frac{T^*}{T_e} \right)^{-1/3} \text{ assuming that } \mu \propto T^{(2/3)}$$

The heat transfer coefficient C_h can then be calculated using Reynolds analogy

$$C_h \approx \frac{C_f}{2\text{Pr}^{*2/3}} \quad \text{where } \text{Pr}^* = \frac{\mu^* C_p}{k^*}$$

4.3.3 Flat plate attached flow

The liquid crystal picture, figure 22(a), for the sharp leading edge flat plate showed a continuous decrease in the surface temperature from the leading edge towards the trailing edge. The liquid crystals turn colourless near the leading edge indicating a surface temperature above 43 °C due to excessive heating caused by the high viscous shear. The viscous shear and the consequent rapid growth of the boundary layer leads to strong viscous interaction and development of a leading edge shock there. Further

downstream, the surface temperature falls continuously and the crystals change colour to blue, green and finally red near the trailing edge. This pattern suggests that the flow is laminar everywhere over the plate as there is no indication of surface temperature increase associated with transition. For the flap deflection angle of 5° the photograph ahead of the hinge line was very similar to the one shown in figure 22(a). However, there is a larger reduction in the temperature just ahead of the 5° flap leading edge indicated by the red colour. This reduction in temperature ahead of the hinge line is caused by the thickening of the boundary layer as a result of the adverse pressure gradient due to the flap shock. The surface temperature increases over the flap surface, indicated by change in colour from red to green, as a result of thinning of the boundary layer caused by the increased pressure there.

4.3.4 Flat plate separated flow

The compressive turning of the flow over a trailing edge flap generates a shock wave whose strength depends upon the upstream Mach no. and the flow deflection angle due to the flap. For two dimensional inviscid and attached flow the shock is generated at the flap hinge-line and the shock angle can be evaluated using oblique shock tables. However, for real flow the boundary layer converts the sudden pressure increase at the corner into a gradual increase. The pressure increase due to the corner is felt upstream through the subsonic part of the boundary layer. The adverse pressure gradient decelerates the inner part of the boundary layer so as to increase its thickness. In turn, the outer boundary layer and the inviscid flow is deflected outwards away from the surface generating compressive waves inside the supersonic part of the boundary layer. These waves coalesce to form an oblique shock. The thickening of the boundary layer flow ahead of the flap reduces the velocity and temperature gradients near the surface so reducing the skin friction and heat transfer rate coefficients. However, further increases in flap angle soon reduce the skin friction coefficient to zero at the hinge line. This flap angle is known as the incipient separation angle. The following correlation by Needham and Stollery⁽⁷²⁾ has been found to give a good prediction of incipient angle for a flapped flat plate⁽⁷¹⁾ at zero incidence.

$$M_{\infty} \beta_i = 80 \sqrt{\chi_{h.l.}}$$

For the present test conditions the correlation predicts the flap deflection angle for the incipient separation condition equal to 6.6°. Hence the flow should be attached for $\beta = 5^\circ$ but should be separated for 15° and 25°.

The liquid crystal picture corresponding to $\beta = 15^\circ$, figure 22(c), has a dark red coloured region extending over a third of the region ahead of the hinge line, indicating the flow to be separated there. The separated shear layer is reattaching somewhere over the flap. However, the liquid crystals pictures can not indicate the reattachment location because the crystals turn colourless along a line about one third the length of the flap downstream of the hinge line. Further increasing the flap angle to 25°, figure 22(d), extends the red coloured separated region further upstream. Further it can be seen that the liquid crystals change from red to green just ahead of the hinge line of 15° flap

suggesting that transition is occurring near the hinge line. The flow is, therefore, thought to be transitional. Similarly it can be seen that the red to green colour change is occurring well ahead of the hinge line over the 25° flap. This suggests that transition is moving upstream by increase of the flap angle.

Comparison with quantitative measurements

The qualitative findings from the liquid crystal pictures are in reasonably good agreement with the quantitative heat transfer measurements using thin film gauges. Kumar and Stollery⁽⁷¹⁾ have reported an extensive experimental study of the flow over the flapped sharp flat plate used here for liquid crystal flow visualisation. The flow was noted to be laminar everywhere over the plain flat plate at zero incidence. The flow remained attached at $\beta = 0$ and 5° but separation occurred as β increased to 10° . The separation of the boundary layer corresponding to $\beta = 10^\circ$ promoted shear layer instability ahead of the reattachment on the flap, so that the flow was thought to be transitional. The flap boundary layer was found to be laminar at $\beta = 0$ and 5° , transitional at $\beta = 15^\circ$ and turbulent at $\beta = 25^\circ$.

The experimental results by Kumar and Stollery have been superimposed on the liquid crystal pictures. The blue coloured dots joined by a blue line are the heat transfer measurement results. The measurements are plotted using the length of the plate along the X-axis and the width along the Y-axis. Along the Y-axis local values of the heat transfer coefficient are plotted on the log scale. These measurements are reproduced in figure 23 and are compared with the theoretical results obtained using the reference temperature method. The reference temperature method is found to give reasonable results for the attached flow conditions over the plate. However, the experimental values fall below the theoretical estimate as soon as the laminar boundary layer gets separated ahead of the flap. The reduction in the heat transfer is due to the reduction in the skin friction. The liquid crystals indicate the separated flow region by changing their colour to red in the region. The separated shear layer is found to turn transitional just ahead of the hinge line for $\beta = 15^\circ$ and well ahead of the hinge line for $\beta = 25^\circ$ as indicated by a change from the decreasing heat transfer to an increasing one ahead of the hinge-line. The liquid crystals indicate it by turning green from red in colour. The heat transfer continues to increase over the flap as a result of the thinning of the separated shear layer

as it turns towards the flap surface to reattach. The maximum heating occurs nearer the reattachment where the boundary layer is thinnest. The liquid crystal photographs can not pin-point reattachment because they turn colourless well before the reattachment due to very high heating rate over the flap. The liquid crystals used in the present study turn colourless as the temperature exceeds about 43°C. A different formulation of crystals can of course be used for fixing the location of reattachment. Now it is interesting to note that the surface temperature over the flap increases from about 23°C to temperatures beyond 43°C in a short span of about 5-10 msec after the hypersonic flow is established in the tunnel.

Repeatability of pictures

A few tests were conducted to ascertain the repeatability of the pictures from one run to the other as well from one time to the other during the run-time of the tunnel. However the wing-body model with $\alpha = 5^\circ$ and $\beta = 25^\circ$ was used instead of the flat-plate model. Pictures were taken at about 5, 10, 15, 20, 25, and 30 msec after the establishment of flow in the tunnel. Four of these pictures are shown in figure 24. Figure 24(a), taken about 5 msec after the flow establishment in the tunnel, shows that the liquid crystals colour pattern is hazy and the colour contrast between the attached and the separated flow regions has yet to emerge in the photograph. It could be because the crystals do not respond within 5 msec time duration or because the separation pattern over the model has not fully established. Needham and Stollery⁽⁷²⁾ reported flow establishment times of 600 μ sec for separated flows. The response time for the liquid crystals is also reported to be a few msec, Roberts and East⁽⁵⁶⁾. It is therefore believed that these tests need more than 5 msec run duration most likely because the crystals response time is more than 5 msec. The photographs taken at 10, 15 and 25 msec look very similar to each other.

4.4 Wing-body model

The wing-body configuration used for this investigation consisted of a spherically blunted cone-cylinder body and a 70° swept back delta wing fitted with trailing-edge flaps. The characteristics for the cone-cylinder body have already been described. The importance of the flow field around the body is that the wing operates within it. The flow field around the cone-cylinder body obtained using the Navier-Stokes solver was found to be very useful for describing the aerodynamic behaviour of the wing-body configuration, particularly the behaviour of the flow over the wing and the flaps. A brief description of the body alone flowfield is therefore given first. Figure 25 shows the variation of the body surface pressure with axial length for $\alpha = 0^\circ$. The surface pressure is found to decrease from the nose to the base of the body. The initial reduction from the stagnation pressure at the nose-tip occurs as the flow expands around the hemi-spherically blunted nose and continues over the cone frustum. Expansion around the cone-cylinder junction causes a further reduction in the surface pressure resulting in a pressure less than the freestream static pressure over the rear portion of the cylinder.

The spherically blunted nose causes a curved bow shock. The entropy increase (or the total pressure decrease) across the shock is proportional to the local inclination of the bow shock wave. A streamline passing through the nearly normal portion of the shock suffers a larger entropy increase than does the streamline passing through the more oblique portion of the shock. Since the entropy remains constant along a streamline in an inviscid, adiabatic and steady flow, the entropy will vary continuously from a very high value next to the surface to a comparatively low value near the edge of the shock layer. The wing of the wing-body configuration will be operating in this variable entropy flow. An estimate of the thickness of the entropy layer has been made from the Navier-Stokes solution for the flow-field around the spherically-blunted cone-cylinder body. Figures 26 (a) and (b) show the variation of Mach number inside the shock layer with the body at an incidence of 0° and 10° at the cone-cylinder junction and at the base of the body. The bow shock is seen to cause a small drop in Mach number in the outer part of the shock layer. There is a small increase in Mach number as one moves from the shock towards the body surface, particularly for the case near the body

base. This increase is thought to be due to the expansion fan emanating from the cone-cylinder junction. As one moves further inward the Mach number is reduced because of the entropy layer. The outer edge of the entropy layer has been chosen to be point 'A' on these Mach number variation curves in figure 26. It can be seen that for $\alpha = 0^\circ$, most of the wing is immersed in the entropy layer, figure 26(c). However, $\alpha = 10^\circ$, most of the wing is outside the entropy layer, figure 26(d).

4.4.1 Flow visualisation

Three different flow visualisation techniques were used. Schlieren photography was used for the flow field and liquid crystals and oil-dot flow for the surface flow visualisation.

4.4.1.1 Schlieren photography

The schlieren pictures were not as useful as in the case of flat plate for the analysis of the wing-body flow for the following reasons :-

1. The flow field details near the surface of the wing and the flap are not visible in the pictures because of the body obstruction.
2. The density gradients produced by the entropy layer generated by the blunt nosed body obscure the density gradients due to the boundary layer.

The schlieren pictures showing the top and side view of the flow field around the wing-body model at zero incidence and 25° flap angle are shown in figure 27(a). The model was designed to avoid the bow shock striking the wing. A bow shock is produced because of the spherically blunted body nose. The shock detachment distance, δ , is so small that it is difficult to measure accurately from the schlieren photograph. However, an estimation can be made using the following relation by Hayes in the book by Cox and Crabtree⁽⁷³⁾

$$\delta \cong \frac{\varepsilon R_N}{1 + \sqrt{2\varepsilon}}$$

Where $\varepsilon = (\gamma - 1) / (\gamma + 1)$ and R_N is body nose radius. The shock stand off distance is dependent upon the flow near the sonic point on the body because the high pressure air at the stagnation point has to be expanded around the nose and the mass flow will

depend upon the distance inbetween the sonic point on the body and on the bow shock. The stagnation pressure can be calculated using the Rayleigh pitot tube formula ⁽⁷⁴⁾

$$\frac{p_{02}}{p_{\infty}} = \left[\frac{(\gamma+1)^2 M_{\infty}^2}{4\gamma M_{\infty}^2 - 2(\gamma-1)} \right]^{\gamma/(\gamma-1)} \left[\frac{1-\gamma+2\gamma M_{\infty}^2}{\gamma+1} \right]$$

This high pressure of $p_{02}/p_{\infty} \approx 87$ must fall around the spherical nose to a value a few times the freestream pressure aft of the sphere-cone junction through an expansion. The strong interaction between the bow shock and the expansion causes a high shock curvature some distance down stream of the nose. The shock curvature produces a gradient in the flow properties inside the shock layer. The layer of fluid next to the body surface having a large entropy gradient is called the entropy layer. It is a region of low density and high temperature. The flow is further expanded around the cone-cylinder junction through an expansion fan there. The bow shock becomes more or less straight and approaches the 5° -sharp-cone shock-angle of 9° approximately near the cone-cylinder junction . There is no indication of any interaction between the expansion fan from the cone-cylinder junction and the bow shock because the leading wave of the expansion fan should be inclined at an angle of approximately 12° to the freestream.

A comparison of the bow shock co-ordinates from the schlieren picture with that obtained from the blast wave theory and a correlation by Billig ⁽⁷⁵⁾ for a spherically blunted circular cylinder is presented in figure 28. For blast wave theory the second order relation obtained by Lukasiewicz ⁽⁷⁶⁾ is used for comparison.

$$\frac{R/d}{M_{\infty} C_D^{1/2}} = 0.795 \sqrt{\frac{(x/d)}{M_{\infty}^2 C_D^{1/2}} \left[1 + 3.15 \frac{(x/d)}{M_{\infty}^2 C_D^{1/2}} \right]}$$

Where x = distance measured from the nose, in the flow direction

C_D = wave drag coefficient of the nose

d = nose diameter

R = local distance inbetween the body axis and the shock in the lateral direction.

The blast wave theory slightly underestimates the shock shape. The relations used are applicable to spherically blunted cylinders but are found to give reasonable values for the spherically blunted cone-cylinder provided the cone is slender. The cone under investigation is a slender cone of 5° half angle. The power-law shock shape in the blast

wave limit has been shown ⁽⁷⁷⁾ to be actually supported by a blunt body slowly expanding normal to its axis. Assuming a hyperbolic shock shape Billig obtained the following experimental correlation for sphere-cone bodies

$$x = r + \delta - R_c \cot^2 \beta \left[\left(1 + \frac{y^2 \tan^2 \beta}{R_c^2} \right)^{1/2} - 1 \right]$$

Where r = radius of the nose of the body

R_c = radius of curvature of the shock wave at the vertex of the hyperbola

δ = shock stand off distance

β = shock wave angle for a sharp cone

$$\frac{\delta}{r} = 0.143 \exp[3.24 / M_\infty^2] \text{ and}$$

$$\frac{R_c}{r} = 1.143 \exp[0.54 / (M_\infty - 1)^{1.2}]$$

Billig's correlation overestimates the shock slightly. The experimental shock is found to lie outside the shock shape obtained by the blast wave theory but inside of that obtained from the correlation. The bow shock envelops the body and does not intersect the wing. The shock stand off distance of 0.5 mm obtained by Hayes and 0.75 mm by the correlation are too small to be measured from a schlieren photograph.

An increase in incidence to 5 and 10° moves the bow shock nearer to the windward surfaces because of higher compression of the flow. The expansion fan at the cone-cylinder junction can be seen and interacts with the bow shock to make it bend round downstream of the cone-cylinder junction, figure 27(b). Again the bow shock does not intersect the wing anywhere. The bow shock due to the blunt nose envelopes the model so that the air wetting the model is first being processed by the bow shock. The local conditions ahead of the wing leading edge will therefore depend upon the distance inbetween the bow shock and the leading edge. In particular the static pressure ratio, p/p_∞ , over the outward spanwise locations on the wing at zero incidence is likely to be greater than that at more inboard locations. These effects will be discussed later when considering the pressure measurements.

The development of the flow field over a delta wing is dependent upon the position of the wing leading edge shock and whether it is attached or detached. The

leading edge shock detachment depends upon the incidence, leading edge sharpness and chamfer, freestream Mach number, ratio of specific heats and the sweepback angle of the wing. The leading edge shock becomes detached if the normal component of the freestream Mach number becomes subsonic or if the flow deflection angle required at the leading edge is more than the maximum permissible corresponding to the normal component of freestream Mach number. A method based on the equivalent wedge technique due to Stetson and Scaggs⁽⁷⁸⁾ was used to estimate the shock detachment angle. The method predicted shock detachment to occur at $\alpha = 9^\circ$. The schlieren picture corresponding to $\alpha = 10^\circ$ shown in figure 27(b) indicates that the shock is just detached, although the oil-dot flow technique discussed later failed to show it at this early stage of detachment.

4.4.1.2 Liquid crystals flow visualisation

The liquid crystal pictures indicating the variation of surface temperature distribution over the wing-body model with a change in the flap deflection as well as the model incidence are presented in figures 29, 30, and 31. In general, the heat transfer to the hemispherical nose is very high turning the crystals colourless. This is to be expected because of the presence of the strong bow shock ahead of it. Similarly, the sharp delta wing leading edges have a high heat transfer rate not only because of the leading edge shock but also because of the leading edge sharpness. The stagnation point heating being inversely proportional to the square root of the nose radius.

The liquid crystal pictures corresponding to the sharp flat plate with 5° flap deflection, figure 22(b), did not indicate the occurrence of transition. It can therefore be assumed that the transition Reynolds number for a sharp flat plate at zero incidence in CoA gun-tunnel Mach 8.2 flow is above 1.8×10^6 . Johnson⁽⁷⁹⁾ has reported a transition Reynolds number of 1.99×10^6 for a sharp flat plate in Mach 8 and unit Reynolds number $8.69 \times 10^6/\text{m}$ flow. Jillie and Hopkins⁽⁸⁰⁾ have shown that increasing the sweep back of a sharp leading-edge flat plate beyond 45° moves transition forward. Deem & Murphy⁽⁸¹⁾ suggested a simple empirical multiplication factor of $\sqrt{\cos \Lambda}$ to estimate the transition Reynolds number for swept back delta wings. This correlation suggests that the transition Reynolds number for the plain delta wing at zero incidence in the

present test facility will be above 1.05×10^6 . The variation of unit Reynolds number and Mach number just outside the boundary layer on the windward side of the delta wing is shown in figure 32 as the incidence increases from 0° to 25° . The local flow properties ahead of the hinge-line were obtained by passing the inviscid freestream through the leading edge shock wave because of incidence and the leading edge chamfer before expanding through the expansion fan as shown in figure 33. Sutherland's law for viscosity- temperature relationship was used.

$$\frac{Re_{\infty}/m}{Re_{\infty}/m} = \frac{p}{p_{\infty}} \frac{M}{M_{\infty}} \left(\frac{T_{\infty}}{T} \right)^2 \left(\frac{T + 110}{T_{\infty} + 110} \right)$$

It can be noted that the inviscid unit Reynolds number ahead of the hinge line initially increases, attaining a peak value of about 1.05×10^7 corresponding to $\alpha \approx 6^\circ$, before continuously decreasing with further increase in the incidence. On the other hand, the local Mach number decreases continuously as the incidence is increased from 0° to 25° . Figure 34 shows that the inviscid Reynolds number based on the local chord length along various spanwise stations is nowhere exceeding 1.1×10^6 (based upon the wing root chord) for the incidence range of 0° to 10° , so that the flow is expected to be laminar with zero flap deflection at all incidences tested. Further, the flow will probably remain laminar with a flap deflection of 5° even though an adverse pressure gradient is known to promote transition. However, as β is raised to 15 and 25° , separation of the laminar boundary layer will probably cause transition to occur prematurely. Particularly, the separated shear layer is likely to turn transitional along the inward spanwise locations although outward locations may still remain laminar. Then, there are other factors like body nose bluntness and spanwise-flow which are likely to affect both separation and transition.

Wing-body attached flow

For $\beta = 0$ and 5° , the liquid crystal pictures indicated attached flow over the lower side of the delta wing at all incidences tested (i.e. $\alpha = 0, 5$ and 10°). This is expected because the correlation for two dimensional laminar flows by Needham and Stollery⁽⁷²⁾ predicts the flap deflection angle for the incipient separation condition equal

to 6.6° . Figure 29(b), (c), and (d) show the thermograph for $\alpha = 0, 5$ and 10° with a flap deflection of 5° and figure 30(a) for $\alpha = 0^\circ$ with a flap deflection of 0° . These liquid crystal pictures indicate higher temperatures along the outer and lower temperatures along the inner spanwise locations. The difference is probably due to the variation in the boundary layer thickness, suppressing comparatively higher temperature entropy layer next to the body surface. The surface temperatures increase as the incidence is increased from 0 to 5 and 10° .

Another important feature indicated by these pictures is glancing shock interaction inbetween the wing leading edge shock and the boundary layer over the body. A large amount of research work on glancing interaction has been reviewed by Stollery⁽⁸²⁾. Most of the high speed work reported is regarding glancing interaction for a turbulent boundary layer. A simple criterion for incipient separation of turbulent boundary layers by Korkegi⁽⁸³⁾, $M_\infty \alpha_i = 17^\circ$, suggests that the turbulent boundary layer on the side wall will separate when the shock generating surface is inclined just above $\alpha_i = 2^\circ$, to the Mach 8.2 flow. The flow structure of the interaction involving a laminar boundary layer is similar to that with a turbulent boundary layer. However, the laminar boundary layer will separate more easily resulting in the formation of very complicated corner flow. In a glancing interaction the pressure rise across the shock is fed forward on the side wall through the boundary layer to deflect the surface stream lines well before they reach the shock wave. For sufficiently strong shocks, $\alpha > \alpha_i$, the surface flow separates from the side wall to roll into a vortex as shown in figure 35 by Kubota and Stollery⁽⁸⁴⁾. In these studies the flow over the side wall is affected by the interaction only so that an increase in α increases the separated flow region over the side wall. However, in the present study the flow over the body is affected by both the body-wing interaction and changes in α . The liquid crystal pictures (figure 36) show red coloured separation and blue coloured reattachment lines on the body indicating the existence of a separated shear layer turning into a free vortex whose thickness is typically of the order of the boundary layer thickness⁽⁸⁴⁾. It should however be noted that the crystals reflect a particular colour for a bandwidth of temperatures so that only those flow features causing a significant change in temperature are properly resolved. This requires different

formulations of crystals to be tried for resolving the flow features under different model orientations and flow conditions.

At $\alpha = 0^\circ$ the flow is indicated to be separated over most of the cylindrical portion of the body, figure 36(a), which clearly could not be true. The actual situation is that the crystals are not able to resolve the regions having attached flow from those having separation because the surface temperature in both regions happen to fall in the bandwidth for which the crystals reflect red colour. However, as the incidence is increased the temperature in the attached flow regions increases so that separation can be resolved. Figures 36(b) and (c) clearly show separation lines with the model at an incidence of 5° and 10° respectively. Reattachment results in a very large increase in the surface temperature so that it can be seen in all three cases.

Wing-body separated flow

The intended use of flap deflection is to produce a change in the pressure distribution over the flap to pitch the vehicle but any further deflection beyond the incipient separation angle causes separation of the flow ahead of the hinge line. The separated shear layer will reattach somewhere along the flap. The increase in pressure due to separation on the wing ahead of the hinge line and reduction in pressure over the flap in comparison to that which would have been attained with attached flow, reduces the effectiveness of the flap to produce the desired pitching moment. The flap deflection also promotes transition which in turn affects separation, the pressure and surface heating rate distribution in the separated flow region and over the flap. It is, therefore, very important for space vehicle designers to know the incipient separation angle which can be predicted with reasonable accuracy for laminar flow over sharp flat plates by the following correlation⁽⁷²⁾

$$M_\infty \beta_i = 80 \sqrt{\chi_{h.l.}}$$

However, for delta wings Rao⁽³²⁾ found that flap induced incipient separation is postponed to larger flap angles (almost by a factor of two) for given local flow Mach and Reynolds numbers. The incipient separation angle is affected by the spanwise flow. The spanwise outflow of low momentum air near the hinge-line thins the boundary layer

making it more resistant to separation and increasing the incipient separation angle. On the other hand the turbulent incipient separation behaviour is very similar to the two dimensional case because exchange of momentum between different layers of fluid restricts any spanwise outflow. However, the turbulent incipient separation itself for delta wings occurs at lower Reynolds numbers because of the lowered transition Reynolds number on delta wings in comparison with the 2-D flat plates.

For a meaningful analysis of the separated flow it has been found very useful to categorise the flow into laminar, turbulent and transitional based on the position of transition relative to the separation and reattachment points. The flow is termed laminar if transition does not occur or occurs downstream of reattachment. The flow is described as turbulent if transition occurs upstream of separation. The third category called transitional flow is associated with transition occurring inbetween the separation and reattachment points. The location of transition can be determined experimentally from the surface pressure or heating rate distribution or from schlieren or shadograph pictures. In the present work the categorisation was based upon the variation in the separation region length with change in the hinge line Reynolds and Mach numbers. The hinge line Reynolds number and Mach number were changed by variation of the model incidence. The length of the separated flow region depends upon many factors such as Reynolds and Mach number, flap deflection angle, flap length (if its too small), surface temperature, amount of outflow, and the type of separated flow i.e. laminar, turbulent or transitional. Needham⁽⁸⁵⁾ showed that both the laminar and turbulent separation region lengths increase with increase in Reynolds number but the separated flow length decreases for transitional flow. The separated region length increases with increase in Mach number. However, the laminar incipient separation angle also increases with increase in Mach number so that it is possible for laminar separation length to decrease with increase in Mach number for a small range of flap deflection angles just above the incipient angle. Kumar and Stollery⁽⁷¹⁾ found that the separation length on their flat plate with 10° flap ($\beta_i = 6.6^\circ$) decreased with increasing incidence from 0 to 5° even though separation was found to be laminar for both incidences.

The separation length as well as the incipient separation angle is affected by the spanwise flow. The spanwise outflow near the hinge line occurs particularly on low

aspect ratio flat plates. The spanwise outflow thins the boundary layer making it more resistant to separation and increasing the incipient separation angle. The local Reynolds number, based upon the local chord to the hinge-line, reduces gradually from the wing root to the tip for a delta planform so that the boundary layer remains thin along the outer spanwise locations. The local heating rates, particularly along the outer spanwise locations are thus increased due to the thinner local boundary layer.

Separated flow at zero incidence

As discussed earlier the liquid crystal pictures showed attached flow over the windward side of the wing for $\beta = 0$ and 5° indicated by the expected behaviour of the crystals, figure 29(b), (c), (d), 30(a), and 31(a). However, an increase in β to 15° resulted in a drastic change in the colour pattern, figure 30(b). A red coloured low temperature separated flow region develops well ahead of the hinge line. The separated shear layer is seen to be reattaching on the flap causing very high heating rates turning the crystals colourless. In fact, the crystals indicate increasing temperature downstream of the hinge line which is probably due to transition occurring in the separated shear layer very near the hinge line before reattachment.

An increase in β to 25° increases the length of the separated flow region, moving separation further forward, figure 31(b). The separation line becomes more or less parallel to the wing leading edge. On the other hand, reattachment on the flap moves downstream away from the hinge line. It is interesting to note that the crystals indicate increased surface temperature over the region just ahead of the hinge line all across the span. It is believed that the increased temperature near the tips is due to thinning of the boundary layer resulting from the spanwise flow and not due to the transition.

Separated flow at incidence

The increase in incidence from 0 to 5° with 15° flap deflection results in a drastic reduction in the length of the separated flow region, being a maximum at the root and continuously decreasing to almost zero at the tip as indicated by the red coloured area ahead of the hinge line, figure 30(c). The reattachment on the flap can not be determined from these pictures because most of the flap turns colourless under the

intense heating associated with reattachment and for the same reason transition is also concealed which is probably occurring downstream of the flap hinge line. As the incidence is increased to 10° the separation region reduces further and the separation line becomes more or less parallel to the hinge line, figure 30(d). The flap turns completely colourless right from the hinge line to the trailing edge which does indicate further forward movement of reattachment on the flap. The region inbetween the hinge line and the separation line on the wing does show an increase in the heating rate indicating the movement of transition upstream of hinge line. A careful comparison of these pictures for $\alpha = 0, 5$ and 10° corresponding to $\beta = 15^\circ$, figure 30(b), (c), and (d), shows that the separation region length ahead of the hinge line decreases along the inward spanwise stations, however, it increases slightly along the outward spanwise stations as the incidence is increased. The increase in incidence causes an increase in the hinge line Reynolds number and a decrease in the hinge line Mach number as shown in figure 32 calculated assuming inviscid attached flow. It follows from the above discussion that the separated flow along the inward spanwise stations is transitional which may be becoming turbulent over the flap upstream or downstream of the reattachment. However, the flow is still remaining laminar along the outward spanwise locations. This difference in the type of separated shear flow over the wing will further complicate the shape of separated flow region.

The observations made above are supported by the behaviour of the flow with a flap angle of 25° , figure 31. The separated flow region length ahead of the hinge line drastically reduces along the inward spanwise locations but increases slightly along the outward spanwise locations as the incidence increases from 0 to 5 and 10° . Reattachment moves forward towards the hinge line and the flap becomes colourless due to excessive heating. The crystals show increasing temperature from immediately downstream of the separation line on the wing making clear the forward movement of transition with increase in flap angle as well as increase in incidence. The glancing shock interaction due to separation and reattachment shock waves further complicates the boundary layer flow over the cylindrical portion of the body.

The following conclusions can be made from the above discussion :

Encapsulated liquid crystals are an excellent aid in the qualitative analysis of hypersonic flow involving complicated three dimensional separated flow regions. Photographs taken with an ordinary single-shot camera using commercially available ISO 200 film with a normal flash light are adequate for qualitative analysis of heat transfer distributions over comparatively complex vehicle geometries in short duration gun-tunnel flow. The pictures taken at different instants during the gun-tunnel run showed that there is practically no difference between the pictures taken inbetween 20 and 35 msec after firing of the tunnel. However, pictures taken at 15 msec after firing of the tunnel did not clearly show the separation line over the wing-body model at $\alpha = 5^\circ$ and $\beta = 25^\circ$. As there is a delay of around 10 msec inbetween the firing of the tunnel (i.e. rupturing of the diaphragms) and the actual start of the hypersonic flow inside the test-section a minimum of 5 to 10 msec of tunnel run duration is required for suitable use of the technique. The gun-tunnel used for the present work (run duration about 25 msec) is therefore more than adequate for this type of work using liquid-crystals. The pictures taken do show model shadow because only one flash light at 45° to the plane of the delta wing was used. Better pictures could be obtained by the use of two synchronised flash lights.

The glancing shock interaction (due to wing leading edge shock, separation and reattachment shocks on the windward surface of the wing) results in separated flow over the cylindrical portion of the windward side of the body.

The flow over the delta wing is attached for small flap angles at all incidences tested. Increasing regions of separated flow result as the flap is deflected to 15° moving the separation line forward becoming more or less parallel to the leading edge when $\alpha = 0^\circ$ but becoming parallel to the hinge line when $\alpha = 10^\circ$ at $\beta = 25^\circ$. This feature of reducing separation length with increasing hinge-line Reynolds number shows the transitional nature of the separated flows. The location of the reattachment line can not be determined from the pictures but indications are that it moves downstream away from the hinge line as β increases.

Transition moves upstream with increase in β as well as increase in incidence. The separated flow is transitional along the inward and laminar along the outward spanwise stations.

4.4.1.3 Oil-dot flow visualisation

Surface flow information regarding separation, transition and reattachment is generally obtained from the schlieren and shadograph pictures although for three dimensional bodies like delta wings the information is limited to the wing centre line. However, in the present study even this limited information could not be extracted mainly because of the body obstruction and partly because the entropy layer envelops the density gradients due to the boundary layer. The liquid crystal technique could only be used for qualitative analysis because colour change detection by the naked eye is highly subjective. In the absence of expensive equipment for the quantitative analysis of liquid crystal images^(53, 54, 56), recourse was made to the inexpensive oil-dot technique. The technique needed a lot of effort and patience to make it work in the short duration gun-tunnel. The experiments were conducted at an incidence of 0, 5 and 10° for a flap deflection angle of 0, 5, 10, 15, 20 and 25°.

Attached flow

A typical oil dot pattern before the run is shown in figure 5(b) and the patterns after the run for a flap deflection angle of 0° at $\alpha = 0, 5$, and 10° are presented in figure 37. During the run the oil dots flow in a direction depending upon the surface shear stress. Because of reduced shear stress in the separated flow regions the oil dots do not move there. Conversely, at reattachment the oil dots are wiped away because of the large shear stress there. The quality of these pictures has left something to be desired though some useful information can be obtained. The length by which the oil dots move depends on the magnitude of the skin friction and the direction of movement depends on the surface flow direction. These pictures clearly show attached flow over the lower side of the delta wing. There is a small inward spanwise flow over the wing at $\alpha = 0^\circ$, however, over the flap the flow is more or less in the direction of the freestream. The inward flow is a consequence of higher static pressure along the outward spanwise locations, in turn caused by the shape of the bow shock. The skin friction reduces along the wing chord as well as in the spanwise direction. The reduction in skin friction from tip towards root clearly suggests a thickening of the boundary layer towards the root. As the incidence is increased to 5 and 10° the length by which the oil dots move increases

indicating increased skin friction (and hence increased surface heating rates as shown by the liquid-crystal pictures). However, it should be noted that the change in surface temperature will change the oil viscosity to affect the length moved. The viscosity of the mixture was not adjusted to eliminate the problem. The surface flow still has an inward component at $\alpha = 5^\circ$ but at $\alpha = 10^\circ$ the spanwise flow gets divided. There is an inward component for inward spanwise locations due to reduced pressure near the body surface caused by the entropy layer effects, and an outward component over the outer locations due to the tip effect. The oil-dot pattern over the wing corresponding to $\beta = 5^\circ$ at $\alpha = 0, 5$, and 10° presented at figure 38 is very similar to the one with $\beta = 0^\circ$, figure 37, but on the flap there is an inward component as the flow tries to escape to the lower pressure region at the body base, and an outward flow near the flap tip.

Separated flow

A further increase in β to 15 and 25° results in complex three dimensional separated flow regions on the windward side of the delta wing. The oil-dot patterns corresponding to $\beta = 15$ and 25° at $\alpha = 0, 5$, and 10° are presented in figures 39 and 40 respectively. At $\alpha = 0^\circ$ and $\beta = 25^\circ$, figure 40(a), the flow separates along a line approximately parallel to the leading edge, supporting the description of laminar separation on the lower side of a delta wing. The separated shear layer reattaches on the flap. The distance inbetween the reattachment line and the hinge line is a maximum along the root and minimum along the wing tip chord. The reverse flow inside the separation bubble on the flap is clearly visible. Figure 39(a) shows a similar pattern at $\alpha = 0^\circ$ and $\beta = 15^\circ$ except that the separated flow region is smaller, as expected. The separation as well as the reattachment line, both, move towards the hinge line. The increase in incidence from 0 to 5 and 10° results in significant changes in the oil-dot patterns, figures 39 and 40. The separation length is drastically reduced in the inward spanwise stations but increased a little in the outward stations. The glancing interaction inbetween the boundary layer over the body and the separation and/or reattachment shock further complicates the flow over the body. In fact figure 40(c) for $\alpha = 10^\circ$ and $\beta = 25^\circ$ shows a small separated flow region on the body near the base. The separated reverse flow inside the bubble has an inward component (figure 39(c) and 40(c)).

A comparison of the liquid-crystal pictures portraying surface temperature distribution with the oil-dot pictures indicating the shear stress distribution is made in figure 41(a) and (b). The oil-dot pictures for $\alpha = 0, 5$ and 10° with $\beta = 15^\circ$ and 25° have been superimposed upon the corresponding liquid-crystal pictures. The oil-dot pictures indicate a reduction in the shear stress over the model wherever the liquid-crystals indicate a reduction in the surface temperature. This is expected as the Reynolds analogy gives a direct relationship between the heat transfer coefficient and the skin friction coefficient. The separation region indicated by the two methods matches quite well with each other and is shown in a sketch at figure 41(c). The position of separation and reattachment lines indicated by oil-dot on the body surface, due to the wing shocks (leading-edge, separation, and reattachment shocks) glancing interaction, can be seen to be very close to the positions indicated by liquid-crystals. The position of separation and reattachment lines can therefore be quantified by combining the images from the two techniques. A sketch showing the variation in separation and reattachment over the lower side of the delta wing with a change in flap deflection angle from 10 to 25° at an incidence of $0, 5$ and 10° is presented in figure 42. The same information has been replotted in figure 43 to show the changes in separation pattern as the incidence is increased for a given flap deflection angle.

For delta wings Rao⁽²⁸⁾ found that flap induced incipient separation is postponed to larger flap angles (almost by a factor of two) for given local flow Mach and Reynolds number. However, the oil-dot pictures for $\beta = 10^\circ$ indicated separated flow just ahead of the hinge-line as shown in figure 42 although flow should have separated around $\beta = 13^\circ$ to follow Rao's findings. An estimate of the incipient separation angle can be made by extrapolating the separation length to zero (figure 44). The estimation gives values ranging from 5° to 10° depending upon the model incidence and the spanwise location. For the present model configuration the following factors are believed to cause the discrepancy in the incipient angle value:

- (a) Body nose bluntness effects
- (b) Type of boundary layer
- (c) Three dimensional effects

The inner spanwise locations are affected by the body nose bluntness effects and due to three-dimensional effects have a comparatively thicker boundary layer. These locations are found to have transitional separation. However the outer spanwise locations are least affected by the bluntness effects and have laminar separation. The three-dimensional effects causing spanwise flow thin the boundary layer there.

At $\alpha = 0^\circ$ the incipient separation angle, β_i , is found to increase from the wing-root towards the tip because the nose effects reduce towards the wing-tip. As the incidence is increased to 5 and 10° , the innermost spanwise location is still affected by strong nose effects so that β_i remains constant at about 6.5° . However, the behaviour at the outer most spanwise location is no longer affected by the bluntness effects so that β_i is determined by the effect of Reynolds number and of spanwise flow. β_i decreases with an increase in Reynolds number (due to incidence increase) for a laminar boundary layer and it increases with an increase in the spanwise flow (due to incidence increase) causing thinning of the boundary layer and hence more resistant to separation. β_i decreases from 10° to 5° as the incidence is increased from 0 to 5° . However, the spanwise flow becomes the dominant factor at $\alpha = 10^\circ$ causing a small increase in β_i from 5 to 6.5° . The behaviour at the inbetween spanwise stations is similarly determined by the relative dominance of the controlling factors.

The following conclusions can be made from the above discussion of the oil-dot results:

- (1) The oil-dot patterns follow the trend shown by the liquid crystal images. The position of separation and reattachment lines can therefore be determined by combining the images from the two techniques. The two techniques complement each other and help to determine the flow on a fairly complicated wing-body model.
- (2) The flow is laminar all over the windward side of the model at all the incidences tested with flap deflection angles of 0° and 5° . The flow is attached all over the windward side of the model except for small regions over the body surface due to glancing interaction.
- (3) However, further increase in β to 10, 15 20 and 25° results in an increasing extent of separated flow ahead of the hinge line at all incidences. The flow is therefore believed to be laminar before separation takes place at all the incidences tested because

turbulent separation length should be comparatively small and occur at higher flap deflection angles.

(4) Along the inner spanwise locations separation is transitional because the separation length is found to decrease and laminar along the outer spanwise locations because the separation length is found to increase with increasing hinge-line Reynolds number.

4.4.2 Pressure measurements

The wing-body model used for the surface pressure measurements is shown in figure 4(d). A test programme involving detailed surface pressure measurements on the windward side of the wing with and without the wing-trailing-edge flap deflected was conducted. The surface pressures over the upper surface of the wing were not measured due to the lack of suitable pressure transducers and also because the leeward side of the wing does not contribute much to the overall aerodynamic behaviour of a vehicle at hypersonic speeds. No pressure measurements were made on the cone-cylinder body to complete the investigation within the stipulated time schedule. The experimental surface pressures on the windward side of the delta wing are compared with a simple two dimensional method of prediction. Attached flow over the wing and the flap is compared with the inviscid values calculated using the shock-expansion theory, applied stream-wise, with a single shock at the flap hinge-line. As the flow separates comparison is made with a two shock model. In the two shock method the flow passes through a separation shock on the wing and a reattachment shock on the flap. The separation shock strength is calculated corresponding to the plateau pressure obtained from the correlation ⁽³²⁾ given below. The reattachment shock strength is calculated to make the flow downstream of the reattachment parallel to the flap surface.

$$C_{p_{Pl.}} (M^2 - 1)^{0.25} = 1.74 \text{ Re}^{-0.25}$$

4.4.2.1 Pressure measurements without flap deflection

The pressure model had 54 pressure tapings, 29 on the port side and 25 on the starboard side of the wing as shown in figure 3(e). Figure 45 shows the surface pressure, p , non-dimensionalised with the freestream static pressure, p_∞ , plotted along conical rays from the root of the wing leading-edge for zero degrees flap deflection angle but the incidence varying from 0° to 10° . Figures 45(a) and (b) show the pressure distribution on the port and starboard sides of the wing respectively and figure 45(c) shows measurements at all the 54 pressure tapping plotted together. It can be seen that the measurements on the two sides of the wing at same spanwise locations are very similar to each other as expected with the model tested at zero yaw angle. It was therefore

decided to plot the measurements from the two sides of the wing as if they were obtained on one side.

Jernell ⁽³⁸⁾ found that Prandtl-Meyer expansion from the tangent-wedge (or oblique shock) value of the forward surface, of a symmetrical double-wedge cross-section, provides a good estimate of the average pressure over the lower rearward surface of the symmetrical double-wedge X-section delta wing of his wing-body model at $M_\infty = 4.63$. Figure 45(c) compares the experimental values with the theoretical estimates obtained by using the shock-expansion theory along the stream-wise strips. At $\alpha = 0^\circ$ the experimental surface pressure values are found to be very close to the freestream static pressure. However, the experimental values show a trend of increasing pressure beyond $\phi/\phi_{L.E.} \cong 0.55$ which can be attributed partly to the leading-edge chamfer and the viscous interaction effect and partly due to the bow shock generated by the hemispherical nose of the body. At higher incidences of $\alpha = 5^\circ$, and 10° Prandtl-Meyer expansion from the oblique shock value over the chamfered leading-edge portion of the wing provides a good estimate of the average pressure ratio. The method is commonly known as the shock-expansion method and consists of processing the freestream flow through the leading-edge shock corresponding to the flow deflection angle equal to the sum of the leading-edge chamfer and the angle of incidence. The surface pressure value over the flat portion of the wing is then obtained by processing the flow obtained downstream of the leading edge shock through the Prandtl-Meyer expansion. The experimental measurements at $\alpha = 5^\circ$ follow the same trend as at $\alpha = 0^\circ$, of more or less uniform pressure at the inner spanwise locations but a trend of increasing pressure beyond $\phi/\phi_{L.E.} \cong 0.2$. The theoretical value still provides a reasonable estimate of the average static pressure over the lower side of the delta wing although it overestimates at the inner and underestimates at the outer spanwise locations. The picture changes with a further increase in incidence to 10° . The pressure distribution appears reasonably uniform over most of the instrumented portion of the wing span and is well estimated by the shock-expansion theory, however there is a considerable dip in the pressure values at the spanwise locations next to the body and extending up to $\phi/\phi_{L.E.} \cong 0.2$. This dip in the static pressure is believed to be due to the nose bluntness effect. A comparison of the experimental chordwise pressure distribution over the surface of the

wing with the static pressure distribution inside the shock-layer of the body alone configuration at $\alpha = 0^\circ$ is given in figure 46(a). A good agreement is obtained between the experimental values and the results obtained from the Navier-Stokes solver. This supports the argument that the variation of surface pressure over the wing is due to the body nose bluntness effects. However, a similar comparison at $\alpha = 10^\circ$, figure 46(b), shows that the experimental values are considerably less than the 'N-S estimations'. The values for 'N-S estimations' over the lower side of the delta wing were obtained by adding the shock-expansion pressure values (calculated assuming $M = 8.2$ ahead of the wing leading-edge) to the local static pressure in the shock layer (obtained from the N-S solution for the body alone). The comparison is not suitable for a quantitative analysis, however it does show that considerably higher pressure can be achieved over the lower surface of the wing by reducing the spanwise outflow, for example by using wing-lets.

4.4.2.2 Pressure measurements with the flap deflected

The intended use of the trailing edge flap deflection is to change the pressure distribution over the flap so as to generate a pitching moment to manoeuvre the vehicle. The pressure measurements were made with the flap deflected through 5° , 15° and 25° to find the suitability of simple two dimensional techniques to estimate the average pressure ahead of the hinge line and the peak pressure achieved over the flap. The body nose bluntness is known to affect the entire flow field around a space vehicle. For example the body nose bluntness can change the effectiveness of a flare on a cone-cylinder-flare body ⁽⁶⁾. It was therefore interesting to find the effect of the body nose bluntness on the wing-trailing-edge-flap effectiveness.

Figures 47, 48, and 49 show chordwise pressure distributions at different spanwise locations on the lower side of the delta wing at $\alpha = 0^\circ$, 5° , and 10° respectively. To show the measurements at different Y/b clearly, these measurements are plotted in the form of two charts, upper and lower chart on each page. The upper chart uses a single log scale y-axis for all the spanwise locations, however the lower chart uses a displaced scale along the y-axis for each spanwise location. The measured static pressure, p , has been non-dimensionalised by the freestream static pressure, p_∞ and plotted on a log scale. Stream-wise distance from the leading edge of the wing root

chord to the pressure tapings, X , is non-dimensionalised by the wing-root-chord up to the hinge-line, C_r , and is plotted along the X -axis. $X/C_r = 1.0$ is the hinge-line, H. L.

Flap deflection angle, $\beta = 5^\circ$

The pressure distributions for $\beta = 5^\circ$ at $\alpha = 0^\circ$, 5° , and 10° (figure 47(a), 48(a), and 49(a)) show attached flow over the wing and the flap. The pressures over the wing and flap are compared with the estimates made using shock-expansion theory applied strip wise across the wing (i.e. in the flow direction). The pressure over the flap shows a spanwise gradient due to the body nose bluntness. The peak in the chordwise pressure distribution at various spanwise locations increases from the root towards the tip indicating a higher pressure recovery at the spanwise stations less affected by body nose entropy layer effects. The maximum pressure over the flap is found to be less than the single shock value. The difference between the peak pressure, p/p_∞ , at a given spanwise station and the maximum pressure achieved over the flap, $(p/p_\infty)_{\max}$, is shown in figure 50 as a percentage of $(p/p_\infty)_{\max}$. Figure 50(c) for $\beta = 5^\circ$ shows that the spanwise pressure gradient decreases with increase in α .

Flap deflection angle, $\beta = 15^\circ$

Further increase in β to 15° at $\alpha = 0^\circ$, 5° , and 10° results in separated flow over the wing, figures 47(b), 48(b), and 49(b). The surface pressure starts increasing because of the separation shock, well upstream of the hinge-line, forming a small plateau region along the innermost chordwise locations. There is a further increase in pressure over the flap as the separated shear layer turns towards the flap surface, thinning in the process. The reattachment of the separated shear layer causes the reattachment shock to increase the pressure still further. The maximum pressure attained further downstream over the flap is found to be less than the single shock value at $\alpha = 0^\circ$ but as the incidence is increased to 5° the pressure recovery improves as the flap surface starts moving out of the entropy layer resulting in peak pressures higher than the single-shock theoretical values at the most outward spanwise locations. Further increase in the incidence to 10° further improves the pressure recovery over the flap, being well above the two-shock

value at all except the innermost spanwise instrumented location. The chordwise peak pressure attained on the flap increases from the root towards the tip as in the $\beta = 5^\circ$ case. The variation in the flap recovery pressure along the span is greatest at $\alpha = 0^\circ$ because the flap is immersed in the entropy layer flow, however, figure 50(b) for $\beta = 15^\circ$ shows that the spanwise pressure gradient decreases with increase in α to 5° and becomes more or less constant at $\alpha = 10^\circ$ except for the innermost instrumented spanwise location because the flap has moved out of the entropy layer flow.

Flap deflection angle, $\beta = 25^\circ$

The chordwise pressure distribution for $\beta = 25^\circ$ at $\alpha = 0^\circ$, 5° , and 10° is shown in figure 47(c), 48(c), and 49(c). The surface pressure starts increasing even further upstream of the hinge-line forming a small plateau region all along the instrumented spanwise locations. The maximum pressure over the flap is found to be less than the single shock value at $\alpha = 0^\circ$ at all except the two outermost instrumented spanwise locations but as the incidence is increased to 5° the pressure recovery improves resulting in peak pressures higher than the single-shock theoretical values at the most outward spanwise locations. Further increase in the incidence to 10° further improves the pressure recovery over the flap, being well above the two-shock value at all except the innermost spanwise instrumented location. In fact, at $\alpha = 10^\circ$ the peak pressure attained on the flap at various spanwise locations is more or less uniform at all except the innermost instrumented spanwise location, figure 50(a).

The variation in peak pressure over the flap, figure 50, is because of the entropy layer effects of the blunt nosed body. Increasing α and β moves the flap out of the nose entropy layer resulting in comparatively uniform loading over the flap. At $\alpha = 0^\circ$ all flap configurations of 5° , 15° , and 25° are affected by the entropy layer causing a large reduction in the peak pressures obtainable on the flap, hence reducing the flap effectiveness. Increasing incidence to 5° reduces these effects but at an incidence of 10° most of the 15° and 25° flap is out of the entropy layer and the peak pressures are more or less constant at all except the innermost instrumented spanwise location.

4.4.2.3 Separated flow

The effect of the incidence angle, α , and the flap deflection angle, β , on the pressure distribution over the lower surface of the wing and flap can be more clearly seen by plotting the chordwise pressure distribution at a particular spanwise location on the wing for different flap angles. Figures 51 shows the change in the chordwise pressure distribution at different spanwise locations, $Y/b = 0.31, 0.44, 0.56, 0.69$, and 0.82 , as the flap deflection angle increases from 0° to 25° when the wing-body is at 0° incidence. Similarly figures 52, and 53 are for $\alpha = 5^\circ$ and 10° respectively. The surface pressure distribution ahead of the hinge-line is compared with the shock-expansion theory in the attached flow region and with a correlation ⁽³²⁾ in the plateau pressure region. This correlation though strictly applicable to two-dimensional laminar separated flows has been found to also correlate three-dimensional separations, such as those induced by part-span flaps ⁽⁸⁶⁾. In these figures the location of the separation point on a particular spanwise location as obtained from the oil-dot flow pictures is marked as ‘|’ and the reattachment by ‘+’.

Angle of attack, $\alpha = 0^\circ$

The chordwise pressure distribution at $\alpha = 0^\circ$ along different spanwise locations with β ranging from 0° to 25° is shown in figure 51(a) to 51(e). For $\beta = 0^\circ$, and 5° the average surface pressure ahead of the hinge-line is well estimated by the freestream static pressure value. The flow is attached over the wing and the flap as indicated by the oil-dot and liquid-crystals pictures also. The pressure starts increasing as a result of the flap deflection only just ahead of the hinge-line and remains well below the separation plateau pressure correlation value ahead of the hinge-line.

The upstream influence (i.e. the point where the pressure distribution is first increased due to the flap deflection) moves further up as the flap angle is increased from 5° to 25° at all the instrumented spanwise locations. The flow is attached with $\beta = 5^\circ$ so that the pressure interaction starts from just ahead of the hinge-line and it takes the most length of the flap to reach the maximum pressure value over the flap. However, at $\beta = 15^\circ$ the upstream influence moves well ahead of the hinge-line, indicating separated

flow as found from the oil-dot and the liquid-crystals techniques. The chordwise location at which peak pressure is attained shifts towards the hinge-line as one moves from the wing-root towards the tip. The experimental plateau pressure values are found to be less than the correlation values at all the spanwise locations. However, the experimental values do follow the trend shown by the correlation. The plateau pressure increases from the root towards the tip as the Reynolds number decreases from the root to the tip. Further increase in β to 25° further increases the separation length moving the upstream pressure interaction further up although there is no significant change in the peak pressure locations on the flap. The plateau pressure still remains less than the correlation value indicating the effect of nose bluntness on the plateau as well as the flap peak pressure at all the instrumented spanwise locations.

Angle of attack, $\alpha = 5^\circ$

The chordwise pressure distribution at $\alpha = 5^\circ$ along different spanwise locations with β ranging from 0° to 25° is shown in figure 52(a) to 52(e). The experimental surface pressure for the attached flow conditions is found to be less than the shock-expansion theory estimate for the inner spanwise locations but becomes higher than the estimate at the outer spanwise locations. So that the experimental surface pressure for the attached flow conditions (i.e. for $\beta = 0$ and 5°) increases from the root towards the tip. The flow is separated for $\beta = 15^\circ$. The upstream influence moves further up for $\beta = 25^\circ$. The inward spanwise locations $Y/b = 0.31$ and $Y/b = 0.44$ still have the plateau pressures less than the correlation value, however beyond these spanwise locations the experimental pressure values exceed the correlation values indicating the reduction in the wing area affected by the body nose bluntness effects.

Angle of attack, $\alpha = 10^\circ$

The chordwise pressure distribution at $\alpha = 10^\circ$ along different spanwise locations with β ranging from 0° to 25° is shown in figure 53(a) to 53(e). The experimental surface pressure values for the attached flow conditions vary very little from the shock-expansion theory estimate except for the most inward spanwise location. Similarly, the spanwise pressure gradient for the attached flow conditions is very small.

The plateau pressure values are above the estimate obtained from the correlation, the most inward spanwise location being the exception again. The upstream pressure interaction starts at more or less the same chordwise location at all values of Y/b . Similarly the locus of maximum pressure locations on the flap for $Y/b = 0.44$ to 0.82 is a straight line parallel to the hinge-line. The pressure measurements show negligible effect of the body nose bluntness effects as the flap deflection is further increased to 25° .

4.4.2.4 Effect of incidence on separation

The pressure measurements discussed above showed that the flow over the lower surface of the delta wing is separated for $\beta = 15^\circ$ and 25° . It is interesting to know how the separation is affected by change in incidence from 0° to 10° . A change in the wing-body incidence causes a change in the local Reynolds number and the Mach number over the wing as shown in figure 32.

Figure 54 shows the variation of the chordwise pressure distribution at different spanwise locations with the wing-body incidence, α , for $\beta = 15^\circ$. The pressure measurements ahead of the hinge-line for $\beta = 0^\circ$ are also included to find the upstream effect of flap deflection. The upstream influence is clearly reduced for the two innermost spanwise instrumented locations as the incidence goes up from 0° to 5° and 10° . At $Y/b = 0.56$, the upstream influence reduces as α is increased from 0° to 5° , however, with further increase to 10° the upstream influence does not change. Now, as we further go outward along the span, at $Y/b = 0.69$ and 0.82 , the upstream influence is clearly moving further upstream with increase in the incidence. The liquid-crystals and the oil-dot flow pictures showed a similar behaviour of the separation over the wing. The separation length and the upstream influence depend upon both the hinge-line Reynolds number and the Mach number, the flap angle, and the nature of the boundary layer. For a fully laminar or a fully turbulent boundary layer the separation length and the upstream influence decreases with an increase in Reynolds number and a decrease in Mach number. However, for a transitional boundary layer the upstream influence as well as the separation length increases with an increase in Reynolds number or a decrease in Mach number. Figure 32 shows that the hinge-line Reynolds number increases with

increase in incidence but reaches a maximum value at $\alpha \cong 6^\circ$ so as to decrease slightly at $\alpha = 10^\circ$, but the Mach number decreases with increasing α throughout.

Figure 55 shows the variation of the chordwise pressure distribution at different spanwise locations with the wing-body incidence, α , for $\beta = 25^\circ$. The results are very similar to those corresponding to $\beta = 15^\circ$. It is believed that the inward spanwise stations where the upstream influence decreases with an increase in α have transitional flow and the outward spanwise stations where the upstream influence increases exhibit laminar flow.

At $\alpha = 0^\circ$ and 5° the distance from the hinge-line to the chordwise location where the peak pressure is located is a maximum at the root and a minimum at the tip for $\beta = 15^\circ$ and 25° . Similarly the length of separated flow is a maximum at the root and a minimum at the tip. In the absence of any effects of nose bluntness the hinge-line Reynolds number should decrease linearly from the wing-root to the tip resulting in a separation line parallel to the wing leading-edge. However, body nose bluntness reduces the Reynolds number so that the separation line tends to bend towards the hinge-line for the affected spanwise locations. The existence of different types of boundary layer separation also causes change in the shape of the separation line. At $\alpha = 10^\circ$ the distance from the hinge-line to the chordwise location with peak pressure, as well as the length of the separated flow is approximately constant all along the span of the wing. The main reason for this change in the shape of separated region is the difference in the type of separation across the wing span i.e. transitional along the inward and laminar along the outward spanwise locations. The interaction of the entropy layer, due to the body nose, with the wing boundary layer makes the situation very complex.

4.4.3 Force measurements

The wing body models used for the force measurements are shown in figure 4(e). The measurements were done on the wing-body combination at incidences ranging from 0° to 10° keeping the flap deflection angle, β , constant inbetween 0° to 25° . At $\beta = 0^\circ$ the leeward surface of the flap caused an expansion of 9° with respect to the wing surface as can be seen from the wing x-section given below. The results on the model without flap deflection will be discussed first.



Wing x-section for $\beta = 0^\circ$ configuration

4.4.3.1 Zero degrees flap deflection angle

Rao ⁽³²⁾ and Opatowski ⁽³³⁾ reported lift, drag, and pitching moment characteristics of thin 70° and 76° swept-back caret delta wings. Figure 56 shows a comparison of their measurements with theoretical estimates obtained by using two dimensional oblique shock theory results along stream wise strips. The pressure coefficient on the windward side of the symmetrical wedge airfoil is calculated using the exact oblique shock relation for pressure ratio across an oblique shock. The shock angle, θ , is calculated from the relation given below in an iterative procedure on Microsoft Excel spreadsheet computer program.

$$\frac{p}{p_\infty} = 1 + \frac{2\gamma}{\gamma + 1} (M_\infty^2 \sin^2 \theta - 1)$$

$$\tan \delta = 2 \cot \theta \left[\frac{M_\infty^2 \sin^2 \theta - 1}{M_\infty^2 (\gamma + \cos 2\theta) + 2} \right]$$

The pressure coefficient on the expansion side is calculated using the exact isentropic pressure relation. The Mach no. downstream of the expansion fan was calculated from the Prandtl-Meyer expansion fan function, v , in an iterative procedure.

$$\frac{p_2}{p_1} = \left(\frac{1 + \frac{(\gamma-1)}{2} M_1^2}{1 + \frac{(\gamma-1)}{2} M_2^2} \right)^{\gamma/\gamma-1}$$

$$\delta(M) = v(M_2) - v(M_1)$$

$$v(M) = \sqrt{\frac{\gamma+1}{\gamma-1}} \left[\tan^{-1} \sqrt{\frac{\gamma+1}{\gamma-1} (M^2 - 1)} \right] - \tan^{-1} \sqrt{(M^2 - 1)}$$

Figure 56(a) shows that the experimental normal force coefficient agrees rather well with the theoretical estimate, as expected. The axial force coefficient is underestimated if skin friction and base drag are not included in the estimate, figure 56(b). However, an excellent agreement is obtained by including the skin friction drag to the theoretical estimate. The base drag is expected to be very small for the thin wing. The skin friction drag was estimated using the relation obtained by Catherall⁽⁸⁷⁾ based upon the reference temperature method. The skin friction drag coefficient, C_f , is given by

$$C_f = 1.77 Q \sqrt{\frac{C^*}{\text{Re}_l}} \quad \text{where,}$$

$$Q = \frac{\rho}{\rho_\infty} \frac{U^2}{U_\infty^2}$$

$$\frac{\rho}{\rho_\infty} = \frac{(\gamma+1) M_\infty^2 \sin^2 \theta}{(\gamma-1) M_\infty^2 \sin^2 \theta + 2}$$

$$C^* = \frac{\mu^*}{\mu} \frac{T}{T^*} = \frac{T + 110}{T^* + 110} \frac{T^*}{T}, \text{ from Sutherland's viscosity law.}$$

$$\frac{T^*}{T} = 0.28 + 0.5 \frac{T_w}{T} + 0.22 \frac{T_r}{T}$$

$$\text{Re}_l = \frac{\rho}{\mu} U l$$

ρ , μ , U , and T are local density, coefficient of viscosity, velocity, and temperature respectively downstream of the leading edge shock and l is reference length.

Figure 56(b) shows a very good comparison inbetween the experimental results of Rao and Opatowski and the theoretical axial force coefficient obtained by the above

mentioned method. It was therefore decided to compare the experimental results on the wing-body with the theoretical results obtained by algebraically adding the modified Newtonian results on the body alone to the tangent-wedge theory results on the wing alone. The pressure coefficient was assumed to be constant over the wing surfaces. The average pressure coefficient over the lower surface was approximated by the following relation:

$$C_{p,lower} = 2 \delta^2 \left[\frac{\gamma + 1}{4} + \sqrt{\left(\frac{\gamma + 1}{4} \right)^2 + \frac{1}{K^2}} \right]$$

The average pressure coefficient over the upper surface was approximated by the following relation:

$$C_{p,upper} = \frac{2 \delta^2}{\gamma K^2} \left[\left(1 - \frac{\gamma - 1}{2} K \right)^{2\gamma/\gamma-1} - 1 \right]$$

Where δ = local flow deflection angle with respect to the freestream, and $K = M_\infty \delta$. In these calculations a minimum value of the upper surface pressure coefficient was limited to 70% of vacuum condition⁽⁸⁸⁾, i.e. $= -1 / M_\infty^2$.

It is to be noted here that the experimental results are expected to be influenced by the following effects:

- (a) The flow over the wing is affected by the body nose shock because the wing is completely enclosed by the shock through out the incidence range under test.
- (b) Wing-body interference
- (c) Viscous effects

The experimental determination of wing-body interference and the effect of body nose shock on the wing aerodynamic characteristics needs at least three sets of experimental data, the wing-body combination tests, and the wing and the body tested separately. In the present study wing-body and body alone tests have been conducted but no wing alone tests were carried out. The wing alone data were generated by the tangent-wedge theory and are expected to be of sufficient accuracy to delineate wing-body interference and the body nose shock effect on the wing characteristics.

The wing-body normal, and axial force, and pitching moment coefficients along with the centre of pressure location non-dimensionalised with the body length are shown

in figure 57. Figure 57(a) shows that the experimental value of the normal force coefficient is over-estimated by the theoretical method. The difference increases with the increase in incidence and is about 7% at $\alpha = 10^\circ$. This reduction in C_N is probably due to the entropy layer affecting part of the wing.

The axial force is compared in figure 57(b). The estimate falls short of the experimental values if the skin friction and the base drag are not included in the estimate. The axial force is under-estimated by approximately 20%, $\Delta C_A = 0.0752$, which is very near to the estimated skin friction drag ($\Delta C_A = 0.079$) for the slender vehicle at small incidences under consideration. The skin friction drag was estimated by adding the separate wing and body contributions. The wing contribution was calculated assuming it to be equal to that of a flat plate having surface area equal to that of the wing and length equal to the wing root chord. The skin friction drag of the equivalent flat plate was calculated applying the reference temperature method and assuming a laminar boundary layer. The contribution of the cone-cylinder body was calculated in a similar way as explained in article 4.1. A good agreement is obtained by including the skin friction drag, figure 57(b).

Figure 57(c) shows that at zero incidence the centre of pressure, X_{cp}/l , is located downstream of the balance moment centre, because of the expansion at the flap hinge-line as a result of the trailing-edge shape, resulting in a small pitch down moment, figure 57(d). However, X_{cp}/l moves ahead of the moment centre as the incidence increases, (figure 57(c)) causing pitch-up moments, figure 57(d). The variation for lift and drag coefficients is shown in figure 57(e) & 57(f). The theoretical method slightly over-estimates the lift and the lift to drag ratio, figure 57(g), but does follow the experimental trend. The ratio of experimental lift to drag coefficients increases smoothly with incidence and the configuration is probably achieving a maximum value of 2.7 at $\alpha = 10^\circ$. The small loss of L/D in comparison with the theoretical estimate is believed to be as a consequence of wing-body interference effects.

4.4.3.2 Effect of flap deflection

The theoretical method based upon algebraic summation of wing and body alone contributions has been found to give a reasonable estimate for C_N , C_A , C_m , and X_{cp}/l of

the wing-body combination. A comparison of the theoretical estimates with the experimental (balance) measurements for different flap deflection angles is made in this section. The difference between the experimental and the theoretical values is mainly due to the following effects, in the order of their importance

- (a) Flap induced separation effect
- (b) Transition effect
- (c) Reattachment of separated shear layer on the flap
- (d) Entropy layer effect
- (e) Wing-body interference effect

Normal force coefficient

Figure 58(a) shows a comparison of the experimental normal force coefficient with the theoretical method for different flap deflection angles. The liquid crystals, oil-dots, and surface pressure measurements showed attached flow over the wing for $\beta = 5^\circ$. A reasonable comparison inbetween the experimental and the theoretical estimates is obtained because for the attached laminar flow conditions all over the model only the last two effects listed above are present for the $\beta = 5^\circ$ case. The small difference between theoretical and the experimental results increases with increasing incidence and is found to be 8% at $\alpha = 10^\circ$. The normal force contribution from the cylindrical body section should be decreasing with increasing α as the separated flow region over the body due to the wing shock glancing interaction increases. The entropy layer effects are expected to reduce with increasing α but the small deflection angle will keep the most of at least the inward spanwise area of the flap surface within the entropy layer flow.

The flap induced separation increases the C_N contribution of the wing ahead of the hinge line, but reduces that of the flap. In general, there is an overall reduction in C_N due to the flap induced separation in the absence of any other affecting parameters. The entropy layer also reduces the pressure recovery over the flap resulting in a further reduction in C_N . The reattachment of the separated shear layer on the flap results in peak pressures well above the two-dimensional oblique shock value and, so that the effect should decrease the gap between the experimental and the estimated values of C_N . The contribution of the flap towards C_N will also depend upon the nature of any shock-shock

interaction. Rao⁽³²⁾ found a reduction in the effectiveness of trailing-edge flap on his delta wing model due to interaction between the wing shock and the separation shock. The transition of the separated shear layer reduces the separated region by moving the separation and the reattachment lines towards the hinge-line so as to improve C_N . Finally, the wing-body interference effects are likely to cause a change in the aerodynamic characteristics. A good comparison between the theoretical and the experimental values of C_N is obtained for the attached flow conditions corresponding to $\beta = 0$, and 5° , figure 58(a).

An increase in β to 15° results in separation of the flow ahead of the hinge-line so as to reduce the experimental normal force coefficient at $\alpha = 0^\circ$. Another reason for lower experimental values is a reduction in the flap load due to the entropy layer effects. The increase in incidence causes transition over the wing and moves the flap surfaces out of the entropy layer. Both of these effects improve the normal force coefficient. The proceedings can be seen more clearly for $\beta = 25^\circ$, where the experimental value of C_N in fact improves upon the theoretical estimate because of the transition, very small separated flow region, and the movement of the flaps out of the entropy layer at $\alpha = 10^\circ$.

Axial force coefficient

Figure 58(b) shows the variation of axial force coefficient corresponding to different flap deflection angles as the incidence is increased from 0° to 10° . A very good comparison is obtained for the attached flow conditions corresponding to $\beta = 0^\circ$, and 5° . However, at higher flap angles the axial force coefficient, C_A , is reduced in comparison to the theoretical estimate as a result of reduced recovery pressure over the flap due to separation and the entropy layer effect. The effect of separation is reduced as a result of transition at higher incidences moving the separation and the reattachment lines towards the hinge-line. Similarly, the effect of entropy layer is reduced on the flap load as the flap moves out of it at higher incidences so as to increase the axial force. These effects are magnified at $\beta = 25^\circ$ so that the experimental C_A becomes more than the theoretical estimate as a result of very high recovery pressures obtained on the flap at $\alpha = 10^\circ$.

Centre of pressure

Figure 58(c) shows the variation of the centre of pressure location, X_{cp}/l , corresponding to different flap deflection angles as the incidence is increased from 0° to 10° . The centre of pressure lies down-stream of the balance moment centre ($X_{mc}/l = 0.7$) and moves towards it (forward movement) with increasing incidence to give very small pitch-up moments for $\beta = 0^\circ$. An increase in the flap angle to $\beta = 5^\circ$ moves the centre of pressure further downstream so that the vehicle experiences a small pitch-down moment at all the incidences tested. Excellent agreement between the theoretical and the experimental position of the centre of pressure, figure 58(c), and the pitching-moment coefficient, figure 58(d), is obtained for these attached flow cases. However, the theoretical position of the centre of pressure is found to be slightly ahead of the corresponding experimental locations for the $\beta = 15^\circ$, and 25° cases for which the flow is separated over part of the vehicle.

Pitching-moment coefficient

Figure 58(d) shows the variation of the pitching-moment coefficient, C_m , about the balance moment centre for a constant flap deflection angles as the incidence is increased from 0° to 10° .

Figure 58(e), and (f) show the variation of C_L and C_D corresponding to different flap deflection angles as the incidence is increased from 0° to 10° . A plot of C_m versus C_L is shown in figure 58(g). The experimental results for the body alone as well as the wing-body combination at different flap deflection angles as the incidence is increased from 0° to 10° are plotted in figure 59.

CONCLUSIONS

The hypersonic flow over a comparatively simple wing-body configuration is found to be rather complex. However, the surface flow visualisation techniques and detailed pressure measurements over the lower surface of the wing produced a consistent picture of the flow behaviour.

The state of boundary layer is very important and determines the effect of flap deflection on the wing and the flap. The boundary layer is found to be laminar on the windward surfaces of the wing without any flap deflection. The flow is found to be attached for $\beta = 0^\circ$ and 5° at all the incidences considered. However, complex three dimensional separation patterns are formed as the flap deflection angle is increased to 10° . Further increase in the flap angle is found to produce an increasing extent of separated flow regions over the windward surfaces of the wing. The boundary layer remains laminar along the outer spanwise locations but turns transitional along the inward spanwise locations as the flap deflection angle is increased to 15° and 25° . Flap deflection not only causes separation but also promotes transition.

Nose bluntness effects are significant and remain so over the length of the wing-body configuration. Blunted nose produces a thick entropy layer around the body and is found to affect the lifting capacity of the wing as well as the flaps. The entropy layer

reduces the peak pressures obtainable over the flap and hence reduces the flap effectiveness. The thickness of the entropy layer decreases on the windward side but increases on the leeward side as the incidence is increased. The effects of the entropy layer over the flap are found to depend upon flap angle as well as the body incidence. These effects reduce as the incidence and or flap angle is increased.

Three dimensional flow over the lower surface of the delta wing of the wing-body configuration was analysed using simple two-dimensional shock wave theory. Simple strip theory is found to give reasonable estimates of the attached flow surface pressures. Separated flow plateau pressure could also be estimated using two dimensional correlation to obtain reasonable results. The peak pressure over the flap not affected by entropy layer is found to be well above single shock value but below the isentropic value.

The Navier-Stokes solutions for the blunted cone-cylinder configuration were obtained at $\alpha = 0^\circ$ and 10° . These flow-field calculations are found to be very helpful to explain the flow characteristics over the wing-body configuration.

The force measurements over the wing-body configuration were found to be in reasonable agreement with estimates based upon the shock-expansion theory. The separation of flow and nose effects caused poor agreement at higher flap deflection angles. However, the theory does help to explain the effects of various conflicting factors.

The aerodynamic characteristics for the hemi-spherically blunted cone-cylinder agree reasonably well with the modified Newtonian theory and estimations from a Navier-stokes solver.

The force measurements for a one-half-power-law body are found to be in good agreement with the other existing results. The lift, drag and pitching moment coefficients are found to increase smoothly with incidence. Leeward flow gets separated at very low incidence and moves further upstream with increasing incidence. The strakes are found to improve the L/D ratio of the configuration with out any significant change in the pitching moment characteristics.

ORIFICE NO.	S / C _r	Y / b	ORIFICE NO.	S / C _r	Y / b	ORIFICE NO.	S / C _r	Y / b
1	-0.6261	-0.3077	19	-0.2348	-0.4359	37	0.1043	0.5641
2	-0.5391	-0.3077	20	-0.1478	-0.4359	38	0.1478	0.5641
3	-0.4522	-0.3077	21	-0.1043	-0.4359	39	-0.2348	-0.6923
4	-0.3652	-0.3077	22	-0.0609	-0.4359	40	-0.1478	-0.6923
5	-0.2783	-0.3077	23	-0.0174	-0.4359	41	-0.1043	-0.6923
6	-0.2348	-0.3077	24	0.0174	0.4359	42	-0.0609	-0.6923
7	-0.1913	-0.3077	25	0.0609	0.4359	43	-0.0174	-0.6923
8	-0.1478	-0.3077	26	0.1043	0.4359	44	0.0174	0.6923
9	-0.1043	-0.3077	27	0.1478	0.4359	45	0.0609	0.6923
10	-0.0609	-0.3077	28	-0.3652	-0.5641	46	0.1043	0.6923
11	-0.0174	-0.3077	29	-0.2783	-0.5641	47	0.1478	0.6923
12	0.0174	0.3077	30	-0.1913	-0.5641	48	-0.1043	-0.8205
13	0.0609	0.3077	31	-0.1478	-0.5641	49	-0.0609	-0.8205
14	0.1043	0.3077	32	-0.1043	-0.5641	50	-0.0174	-0.8205
15	0.1478	0.3077	33	-0.0609	-0.5641	51	0.0174	0.8205
16	-0.4956	-0.4359	34	-0.0174	-0.5641	52	0.0609	0.8205
17	-0.4087	-0.4359	35	0.0174	0.5641	53	0.1043	0.8205
18	-0.3217	-0.4359	36	0.0609	0.5641	54	0.1478	0.8205

Table 1. The location of the pressure orifices on the wing-body

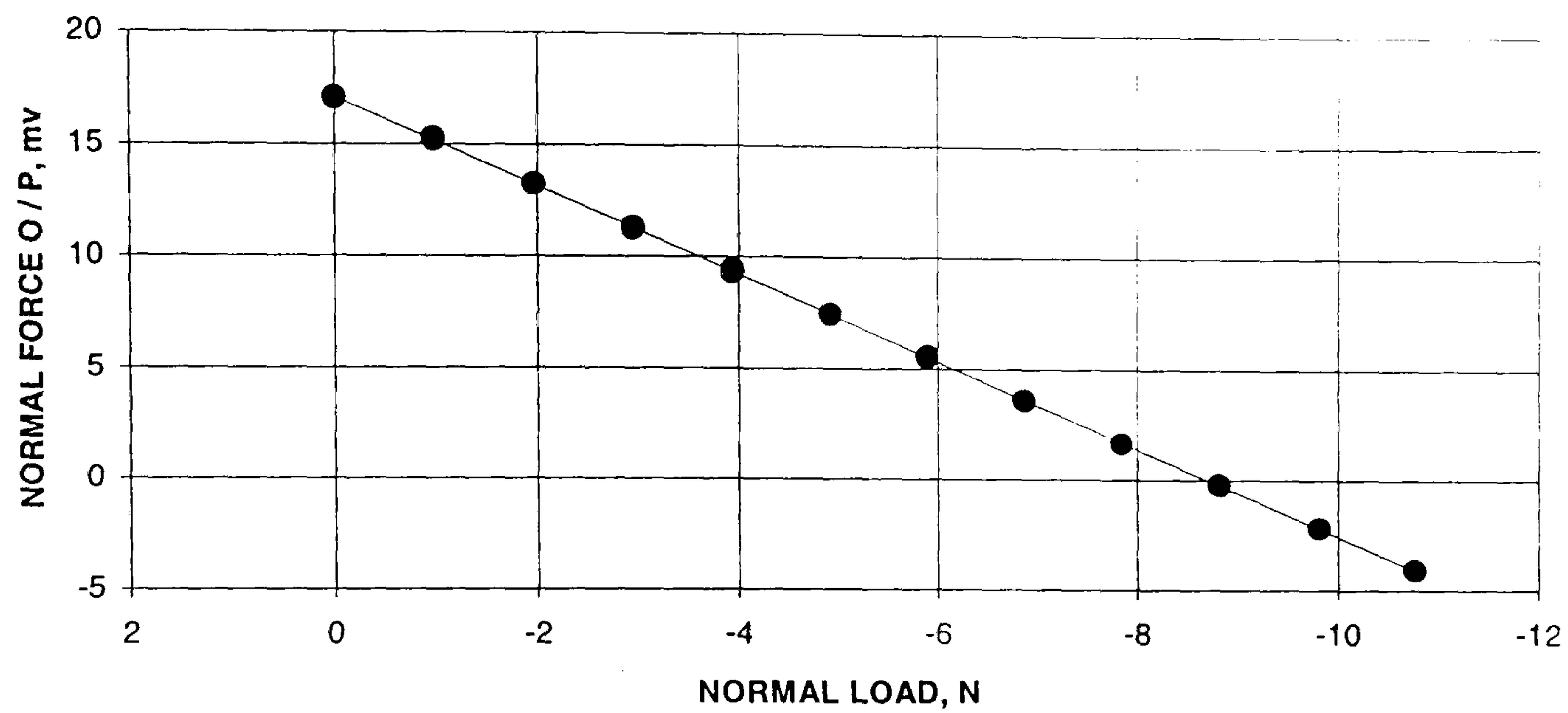
x/l	a/l	b/l	x/l	a/l	b/l
0.0000	0.0000	0.0000	0.3600	0.0429	0.0268
0.0050	0.0051	0.0032	0.4400	0.0475	0.0297
0.0100	0.0072	0.0045	0.5200	0.0516	0.0322
0.0200	0.0101	0.0063	0.6000	0.0554	0.0346
0.0400	0.0143	0.0089	0.6800	0.0590	0.0369
0.0600	0.0175	0.0110	0.7600	0.0624	0.0390
0.1200	0.0248	0.0155	0.8400	0.0656	0.0410
0.2000	0.0320	0.0200	0.9200	0.0686	0.0429
0.2800	0.0379	0.0237	1.0000	0.0716	0.0447

Table 2. Elliptic cone design co-ordinates

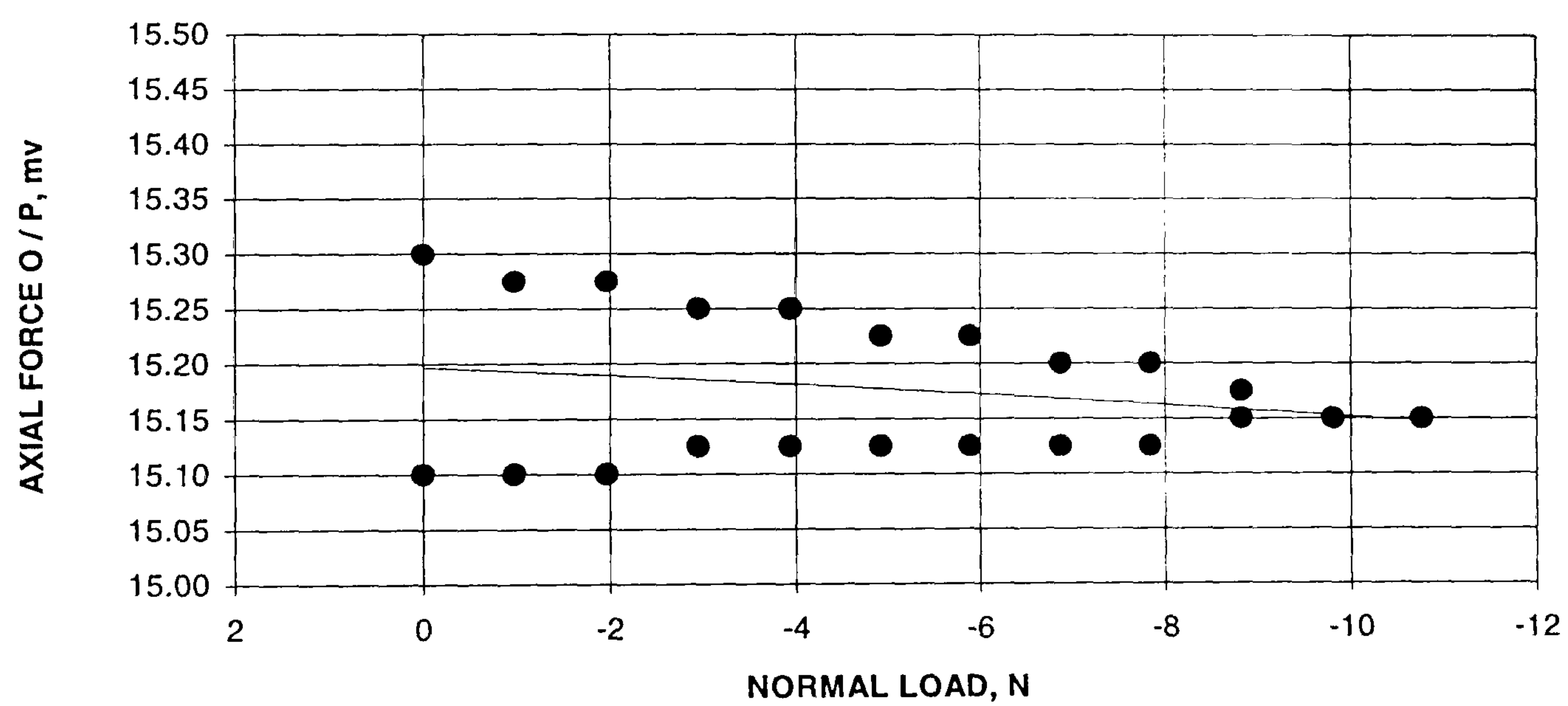
Temperature / °C	Colour emitted
22.2	Visible start
23.0	Start red
25.2	Start green
33.0	Start blue
42.9	Colourless

Table 3. Variation of colour with temperature of the liquid crystals.

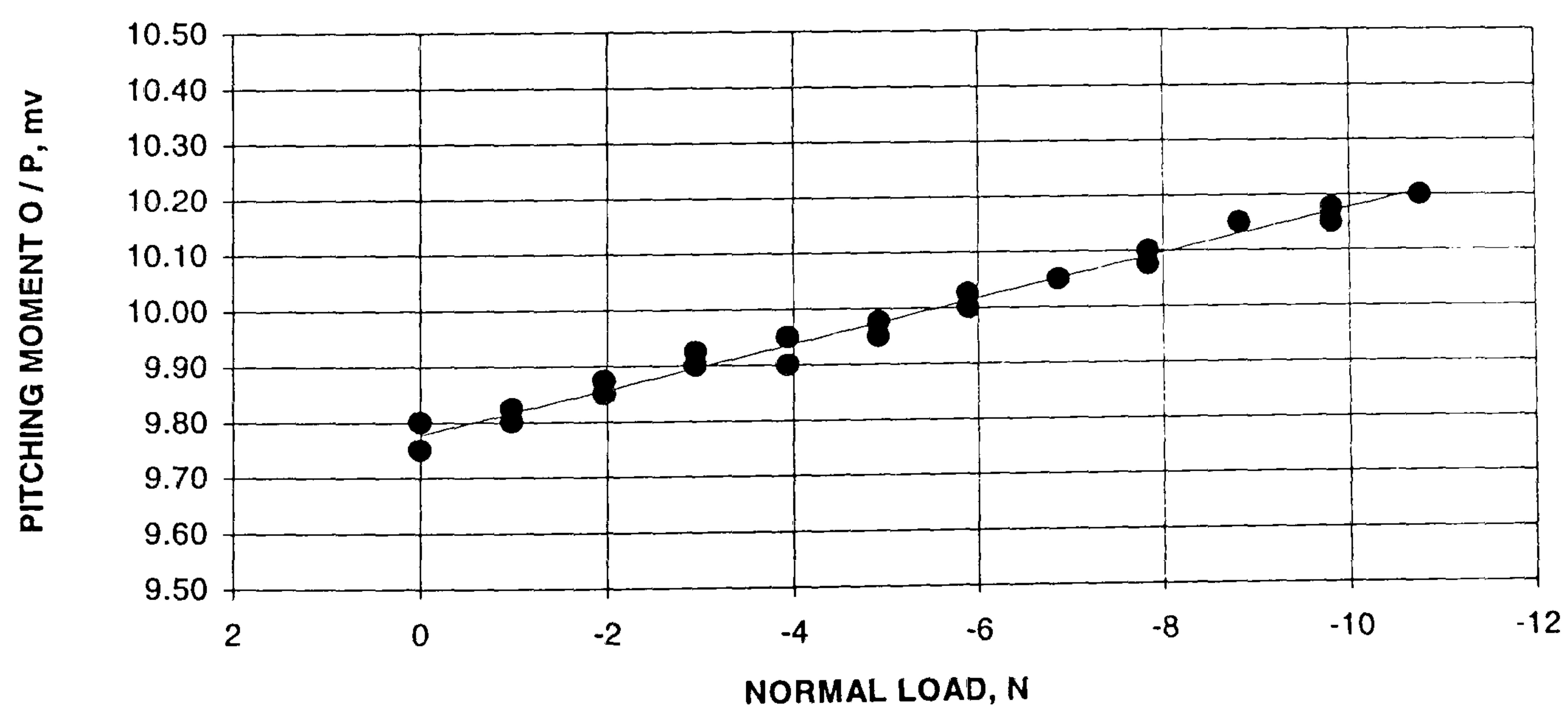
Appendix 1



(a) Effect of normal load on normal channel.

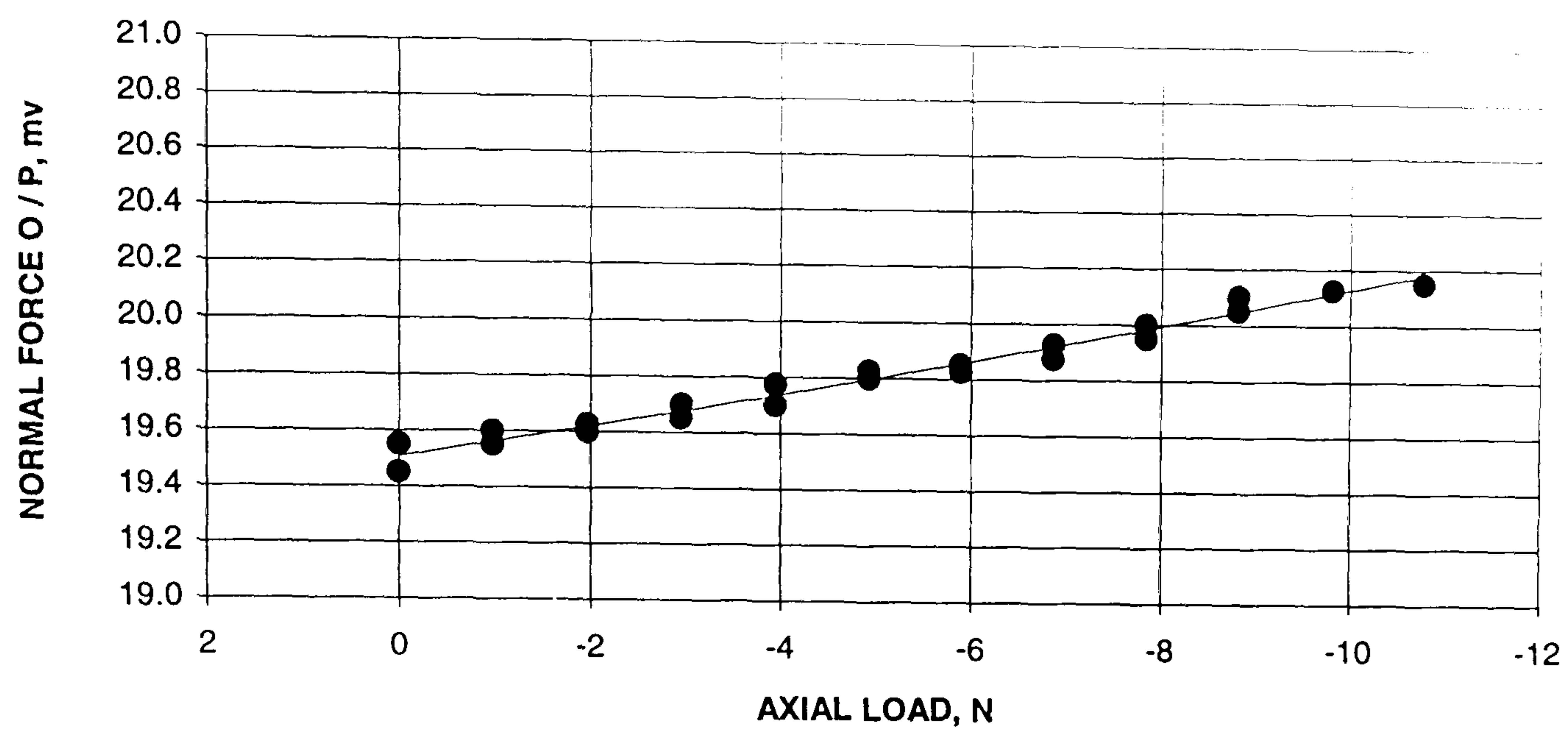


(b) Effect of normal load on axial channel.

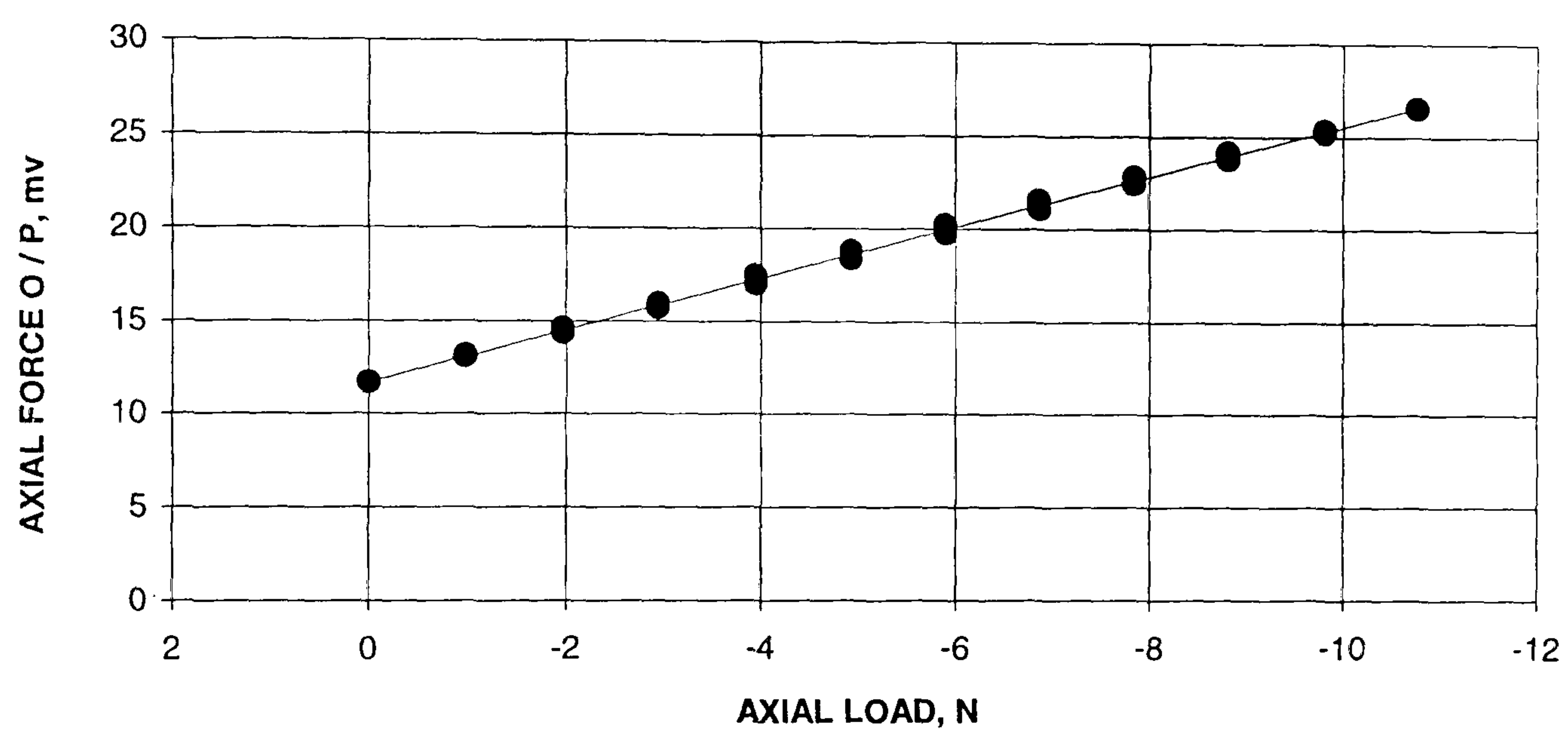


(c) Effect of normal load on pitching moment channel.

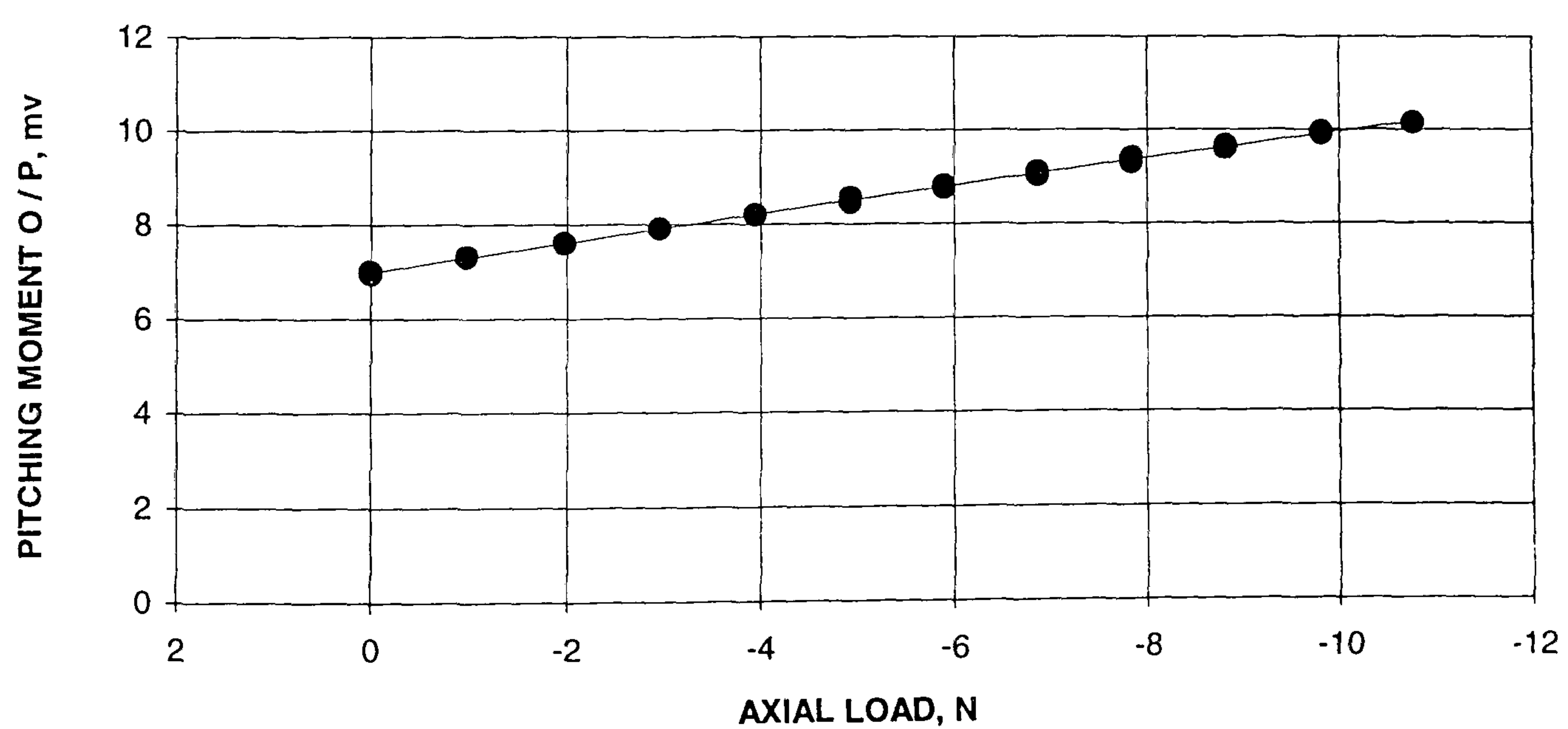
Figure 1. Calibration for the normal force loading.



(a) Effect of axial load on normal channel.

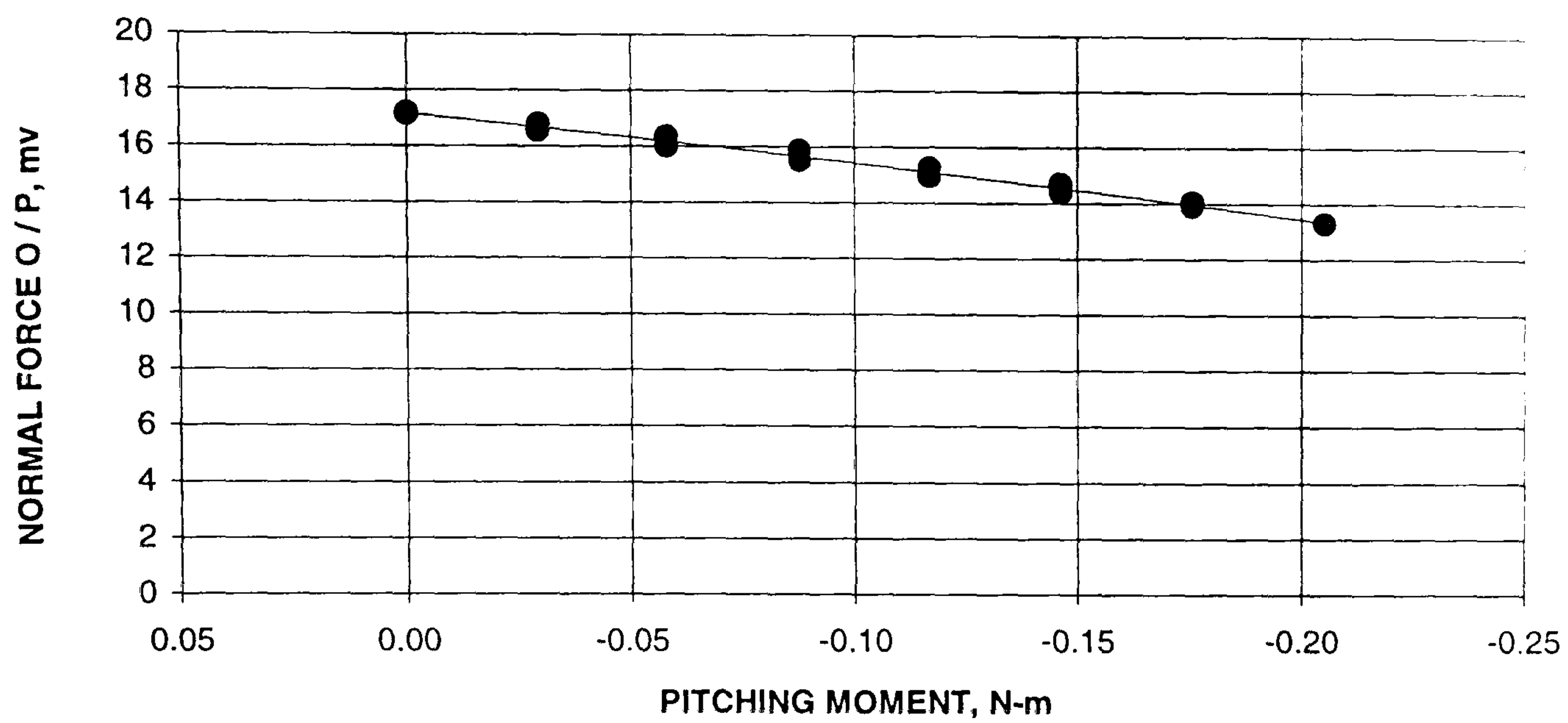


(b) Effect of axial load on axial channel.

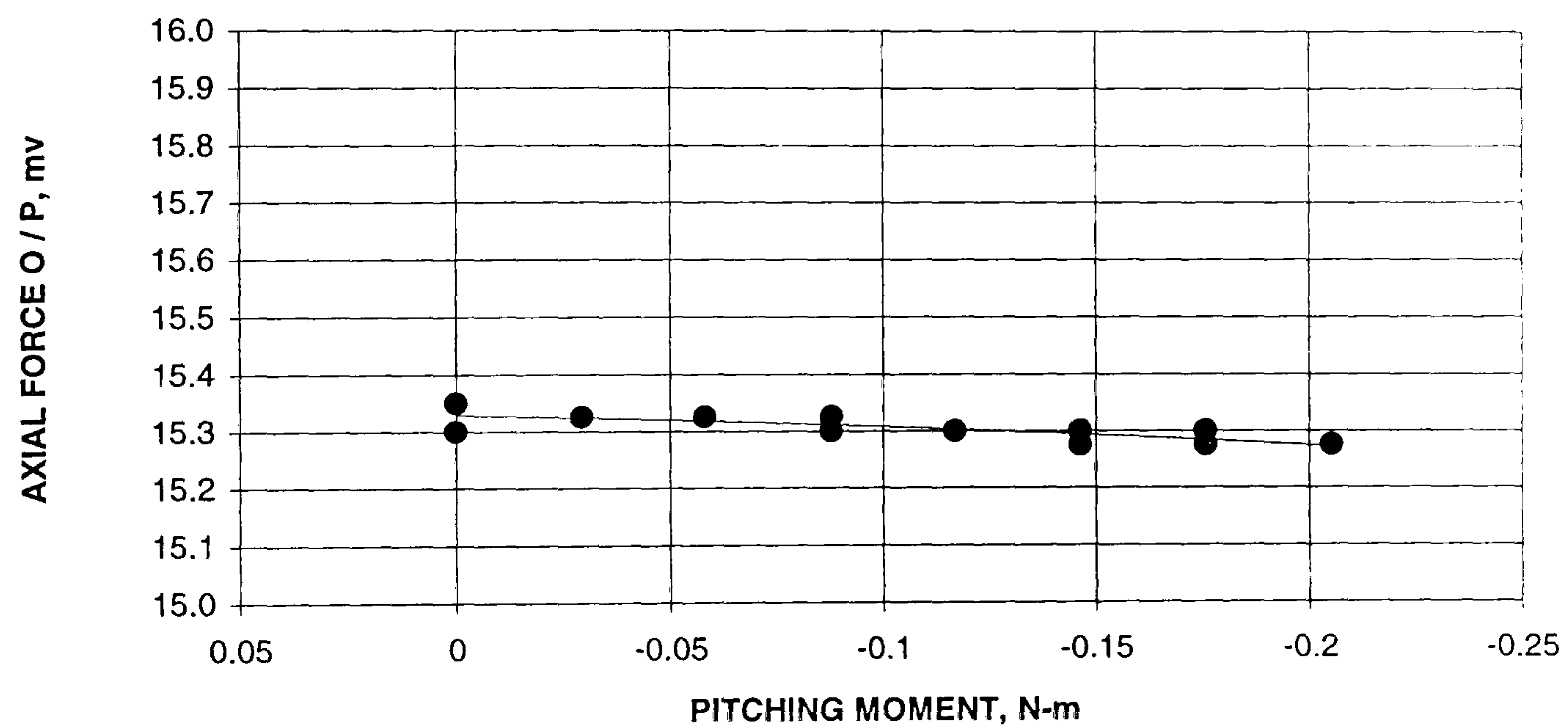


(c) Effect of axial load on pitching moment channel.

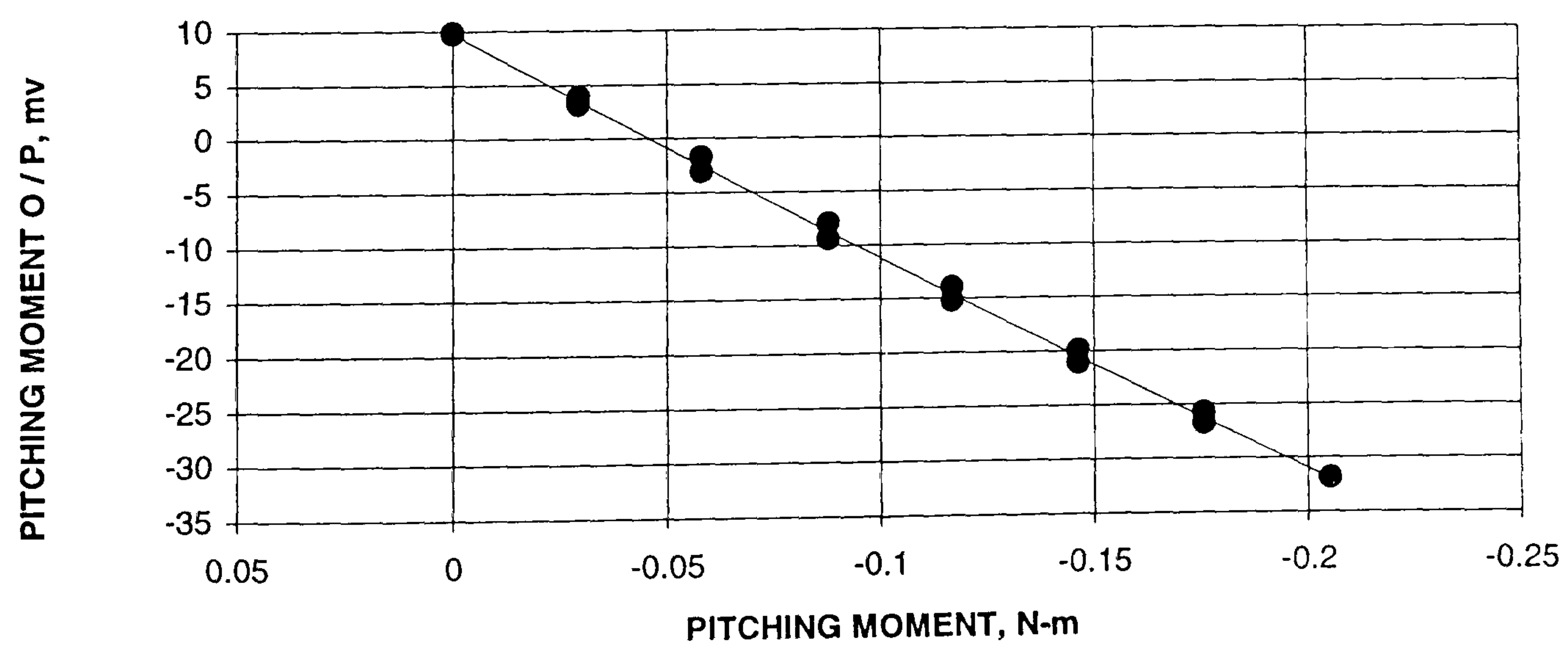
Figure 2. Calibration for the axial force loading.



(a) Effect of pitching moment on normal channel.



(a) Effect of pitching moment on axial channel.



(c) Effect of pitching moment on pitching moment channel.

Figure 3. Calibration for the pitching moment loading.

Appendix 2

The equations obtained by Trimmer are given below :-

Normal force

For $0 \leq \alpha \leq \delta$

$$\frac{C_N S}{K R_b^2} = \pi \sin \alpha \cos \alpha \cos^2 \delta \left(1 - \frac{\delta^2}{2} \cos^2 \delta \right) \text{ For } \delta \leq \alpha \leq (\pi - \delta)$$

$$\begin{aligned} \frac{C_N S}{K R_b^2} = & \sin \alpha \cos \alpha \cos^2 \delta \left(1 - \frac{\delta^2}{2} \cos^2 \delta \right) \left[\frac{\pi}{2} + \sin^{-1} \left(\frac{\tan \delta}{\tan \alpha} \right) \right] \\ & + \frac{\xi^2}{6} \sin \alpha \left[3 \cos^{-1} \left(\frac{\sin \delta}{\sin \alpha} \right) + \sin \delta \left(5 - \frac{2}{\sin^2 \alpha} + \frac{\sin^2 \delta}{\sin^2 \alpha} - 3 \sin^2 \delta - \frac{4}{\sin^2 \delta} \right) \right] \\ & + \left(\frac{2 \sin \alpha \cos^2 \delta}{3 \sin \delta} + \frac{\cos^2 \alpha \sin \delta}{3 \sin \alpha} \right) (\sin^2 \alpha - \sin^2 \delta)^{1/2} \end{aligned}$$

Axial force

For $0 \leq \alpha \leq \delta$

$$\frac{C_A S}{K R_b^2} = \frac{\pi}{2} \left[\left(1 - \frac{\delta^2}{2} \cos^2 \delta \right) (2 \cos^2 \alpha \sin^2 \delta + \sin^2 \alpha \cos^2 \delta) + \xi^2 \cos^2 \alpha \cos^2 \delta \right]$$

For $\delta \leq \alpha \leq (\pi - \delta)$

$$\begin{aligned} \frac{C_A S}{K R_b^2} = & \left[\frac{\pi}{2} + \sin^{-1} \left(\frac{\tan \delta}{\tan \alpha} \right) \right] \left\{ \cos^2 \alpha \left[\frac{\xi^2}{2} + \sin^2 \delta \left(1 - \xi^2 \cos^2 \delta \right) - \frac{\xi^2}{2} \sin^4 \delta \right] \right. \\ & \left. + \frac{\sin^2 \alpha \cos^2 \delta}{2} \left(1 - \frac{\xi^2}{2} \cos^2 \delta \right) \right\} \\ & + \frac{\cos \alpha}{2} \left\{ (\sin^2 \alpha - \sin^2 \delta)^{1/2} \left[\frac{3\xi^2}{2} \sin^3 \delta + \sin \delta \left(3 - \frac{5\xi^2}{2} \right) \right] + \xi^2 \cos^{-1} \left(\frac{\sin \delta}{\sin \alpha} \right) \right\} \end{aligned}$$

Centre of pressure

For $0 \leq \alpha \leq \delta$

$$\frac{X_{cp}}{R_b} = \frac{\cos \delta - \frac{2}{3 \cos \delta} + \frac{1}{6} \xi^2 \cos^2 \delta (\xi - 3 \cos \delta)}{\sin \delta \left(1 - \frac{\xi^2}{2} \cos^2 \delta \right)}$$

For $\delta \leq \alpha \leq (\pi - \delta)$

$$\frac{X_{cp}}{R_b} = \frac{K R_b^2}{C_N S} + \left(\sin^2 \alpha - \sin^2 \delta \right)^{1/2} \left\{ \begin{aligned} & \left[\frac{\sin \alpha \cos \alpha}{\tan \delta} \left[\frac{\pi}{2} + \sin^{-1} \left(\frac{\tan \delta}{\tan \alpha} \right) \right] \left[\frac{1}{6} \xi^3 \cos^3 \delta - \frac{1}{2} \xi^2 \cos^4 \delta + \right. \right. \\ & \left. \left. \cos^2 \delta - \frac{2}{3} \right] \right. \\ & \left. \left[\frac{\xi^2}{6} \sin \alpha \sin \delta \left[\frac{1 - \xi \cos \delta}{\tan \delta} - \xi \sin \delta \right] \right. \right. \\ & \left. \left[3 \sin^2 \delta - \frac{\sin^2 \delta}{\sin^2 \alpha} - 5 \right] \right. \\ & \left. \left. + \left[\cos^2 \delta (1 - \xi^2 \cos^2 \delta) - \frac{2}{3} (1 - \xi^3 \cos^3 \delta) \right] \right] \right. \\ & \left. \left[\frac{2 \sin^2 \alpha \cos^2 \delta + \cos^2 \alpha \sin^2 \delta}{3 \sin \alpha \sin^2 \delta \cos \delta} \right] \right. \\ & \left. + \cos^{-1} \left(\frac{\sin \delta}{\sin \alpha} \right) \left[\frac{\xi^2}{2} \sin \alpha \left(\frac{1 - \xi \cos \alpha}{\tan \delta} - \xi \sin \delta \right) \right] \right\} \end{aligned} \right.$$

Pitching moment

$$C_m = C_N \left(\frac{X_{cp} - X_{ref.}}{l} \right)$$

Where $X_{ref.}$ is the distance from the base to the desired reference point.

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1. Amarjit Singh, "Flow over a delta wing and blunt body combination at hypersonic speeds", Proceedings of the 19th congress of the international council of aeronautical sciences (ICAS), Anaheim, California, USA, 18-23 September 1994.
2. Amarjit Singh and J. L. Stollery, "Experimental investigation of hypersonic flow over a wing-body combination", AIAA-95-6083, Presented at AIAA sixth international aerospace planes and hypersonics technologies conference, Chattanooga, TN, USA, 3-7 April 1995.
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4. Amarjit Singh and J. L. Stollery, "Force measurements on a delta wing-body combination at hypersonic speeds", Submitted for AIAA 7th Annual International Spaceplanes and Hypersonic Systems and Technologies Conference, to be held at Norfolk, VA, USA, 18-22 Nov, 1996.

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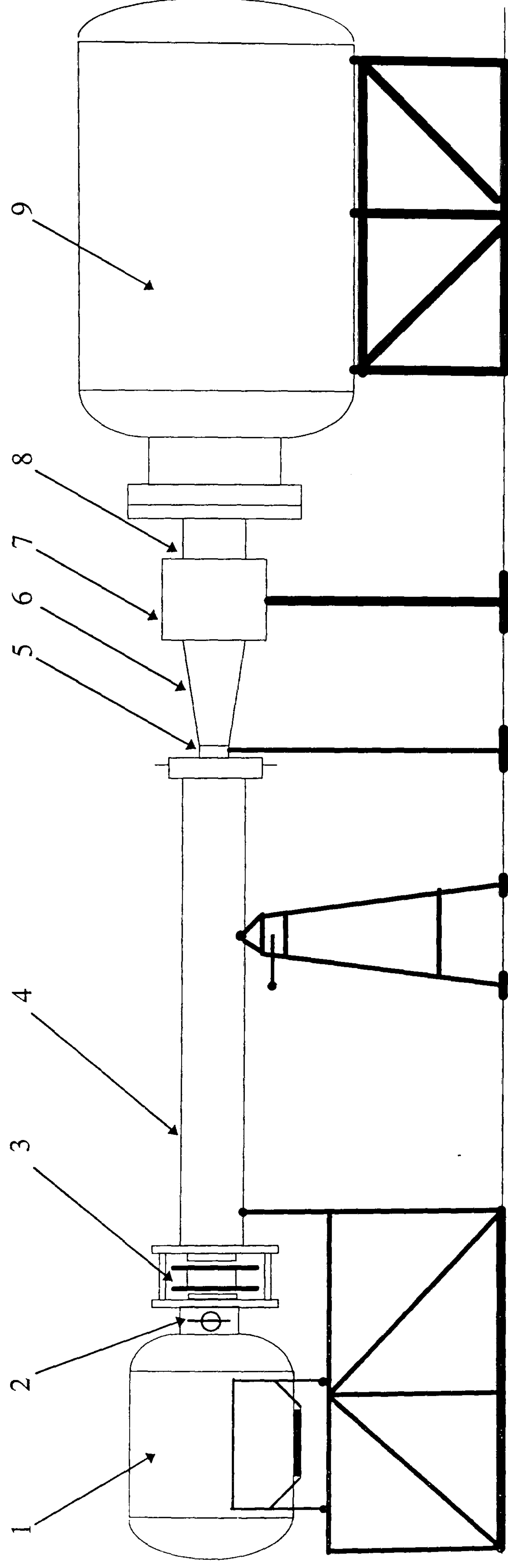
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80. Jillie, R. E. and Hopkins, E. J., "Effects of Mach number, leading-edge bluntness, and sweep on boundary layer transition on a flat plate", NASA TN D-1071, 1961.

81. Deem, R. E. and Murphy, J. S., "Flat plate boundary layer transition at hypersonic speeds", AIAA Paper 65-128, 1965.
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83. Korkegi, R. H., "A simple correlation for incipient turbulent boundary layer separation due to a skewed shock wave", AIAA J., vol. 11, no. 11, November 1973, pp. 1578-1579.
84. Kubota, H. and Stollery, J. L., "An experimental study of the interaction between a glancing shock wave and a turbulent boundary layer", J. Fluid Mech., Vol. 116, 1982.
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86. Putnam, L. E., "Investigation of effects of ramp span and deflection angle on laminar boundary layer separation at Mach 10.03", NASA TN D-2833, 1965.
87. Catherall, D., "Boundary layer effects on caret wings", RAE technical note RAE/TN/AERO 2972, 1964.
88. Maye, J. P., "A limit pressure coefficient and an estimation of limit forces on airfoils at supersonic speeds", NACA RM L8F23, 1948.



1 - Driver vessel, Volume = 0.113m^3

2 - Ball valve

3 - Double diaphragms

4 - Barrel, Length = 6.1m, Internal diameter = 0.0814m

5 - Stainless steel throat insert, Throat diameter = 0.0128m

6 - Axisymmetric contoured fibreglass nozzle, Exit diameter = 0.203m

7 - Open jet test-section

8 - Diffuser

9 - Dump tank

Figure 1. A sketch of the gun-tunnel.

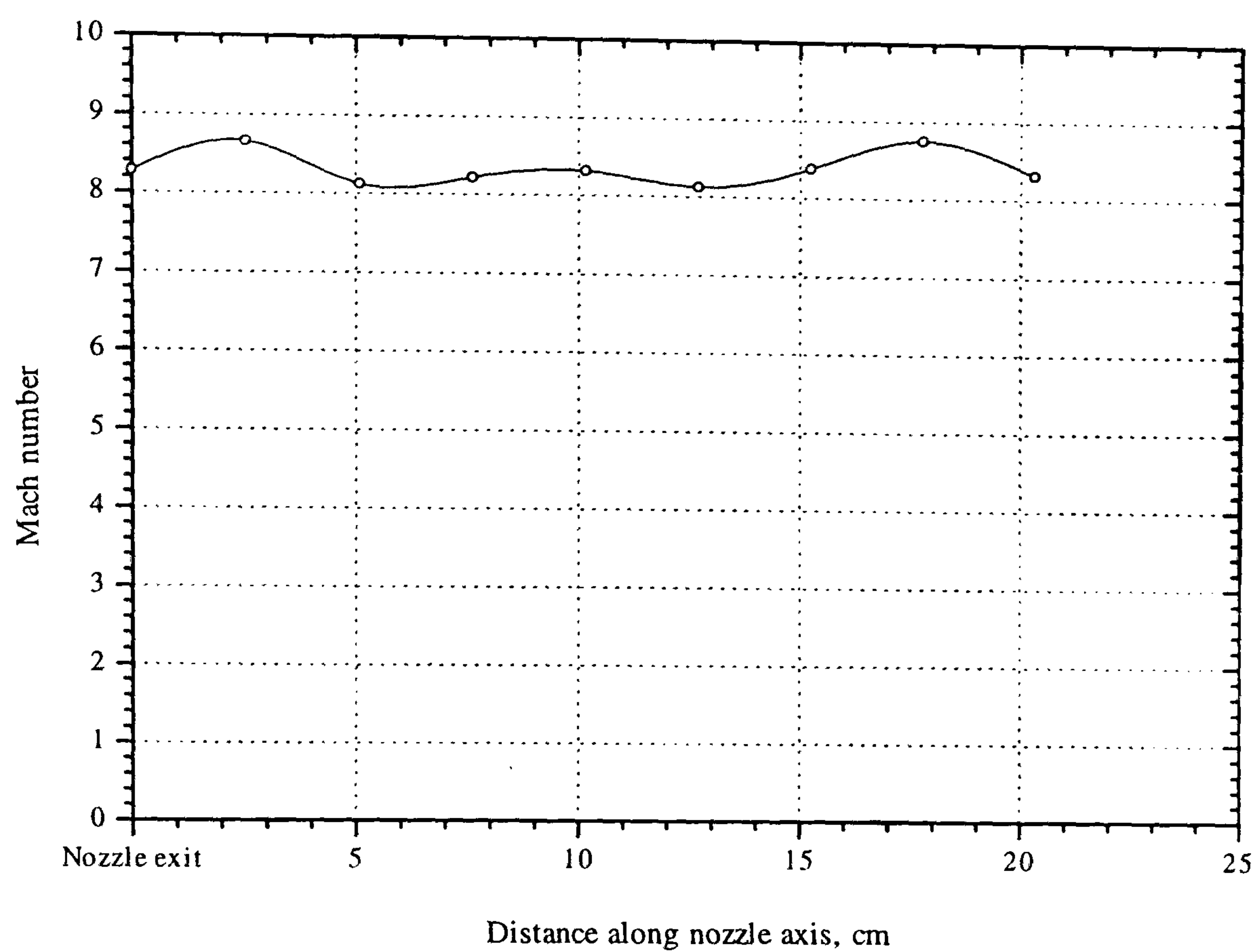


Figure 2(a). Variation of Mach number along the axis of the nozzle ⁽⁴⁹⁾.

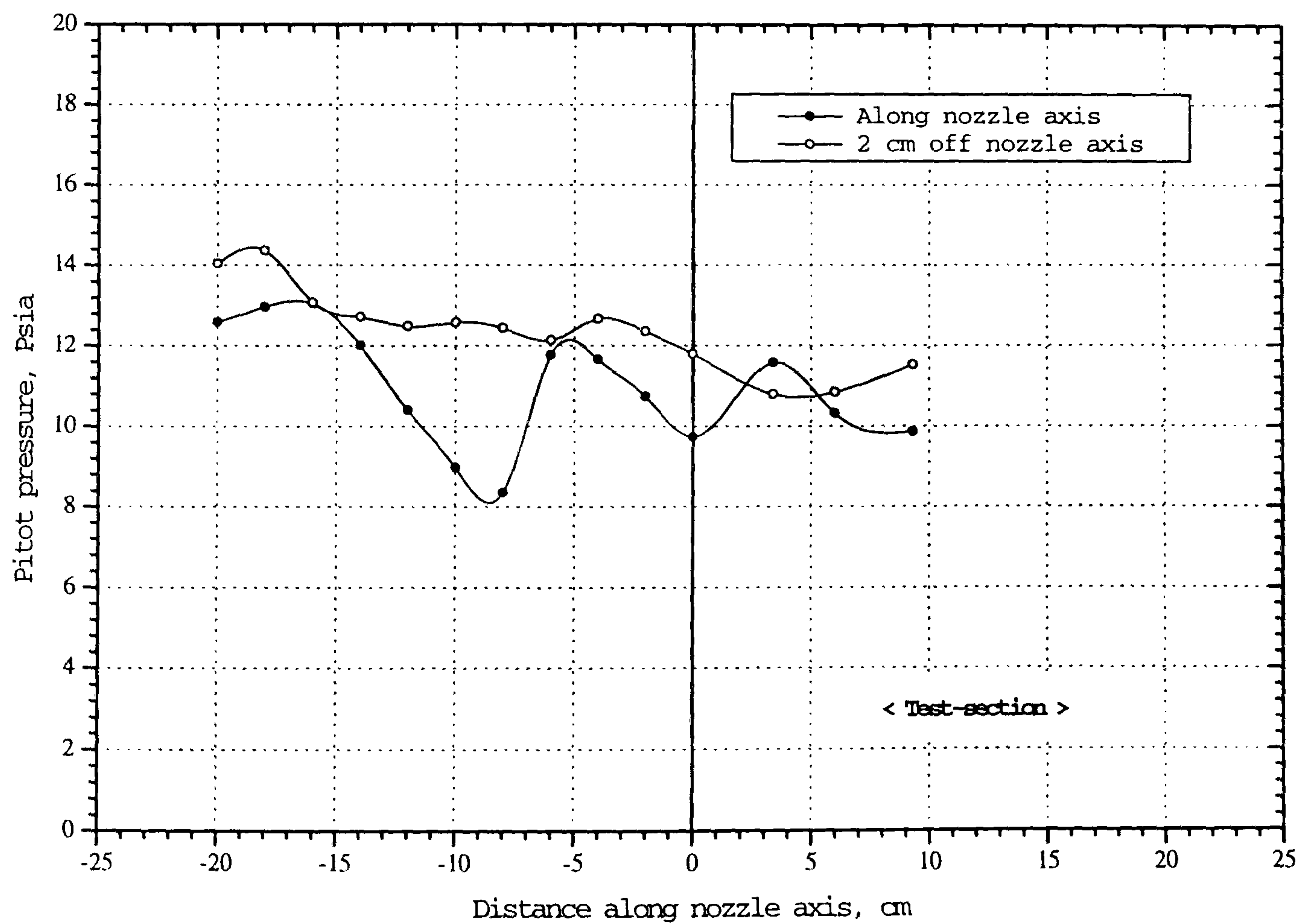


Figure 2(b). Pitot pressure measurements along and off the axis of Mach 8.2 nozzle.

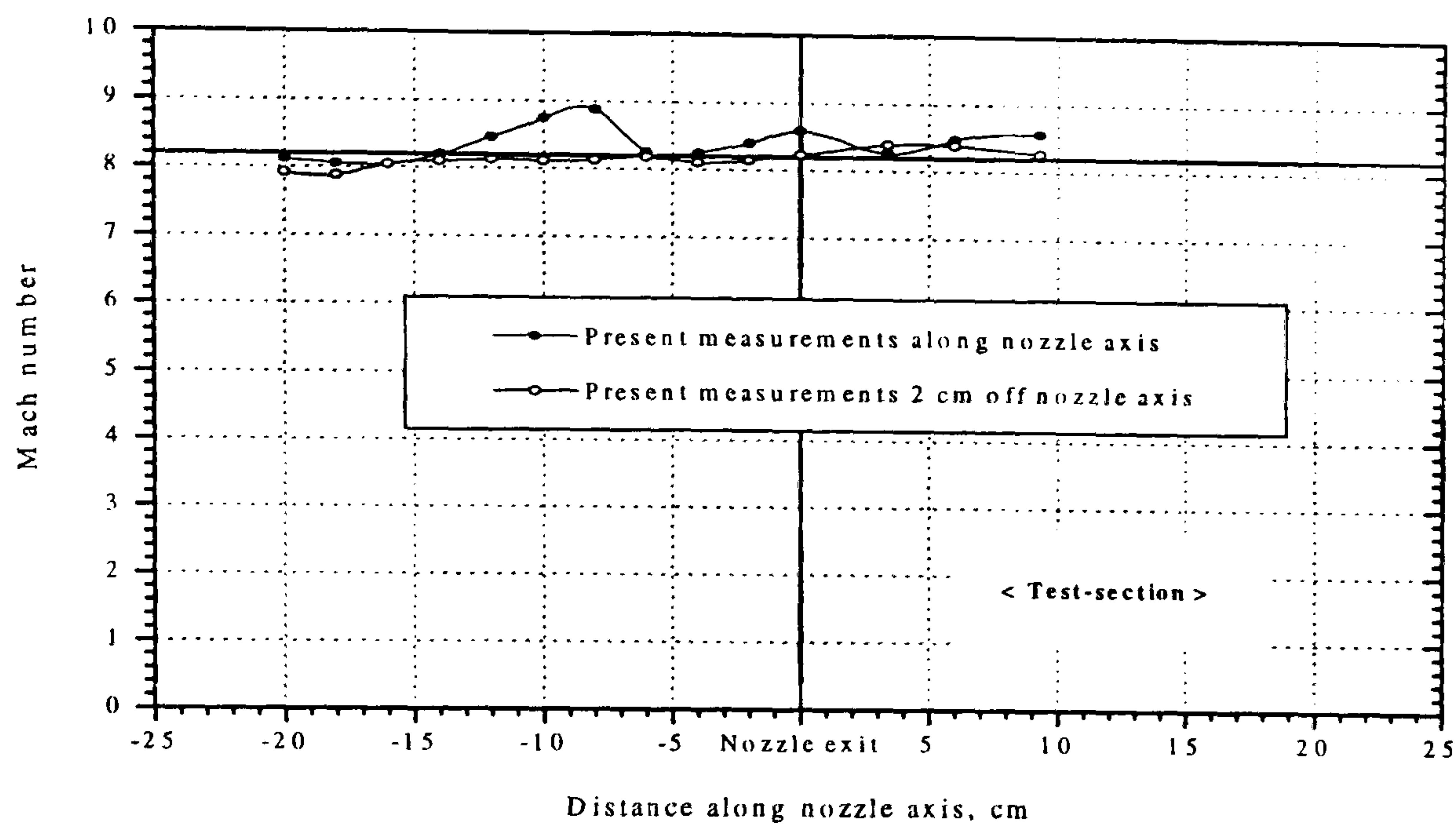


Figure 2(c). Mach number survey along and off the axis of Mach 8.2 nozzle.

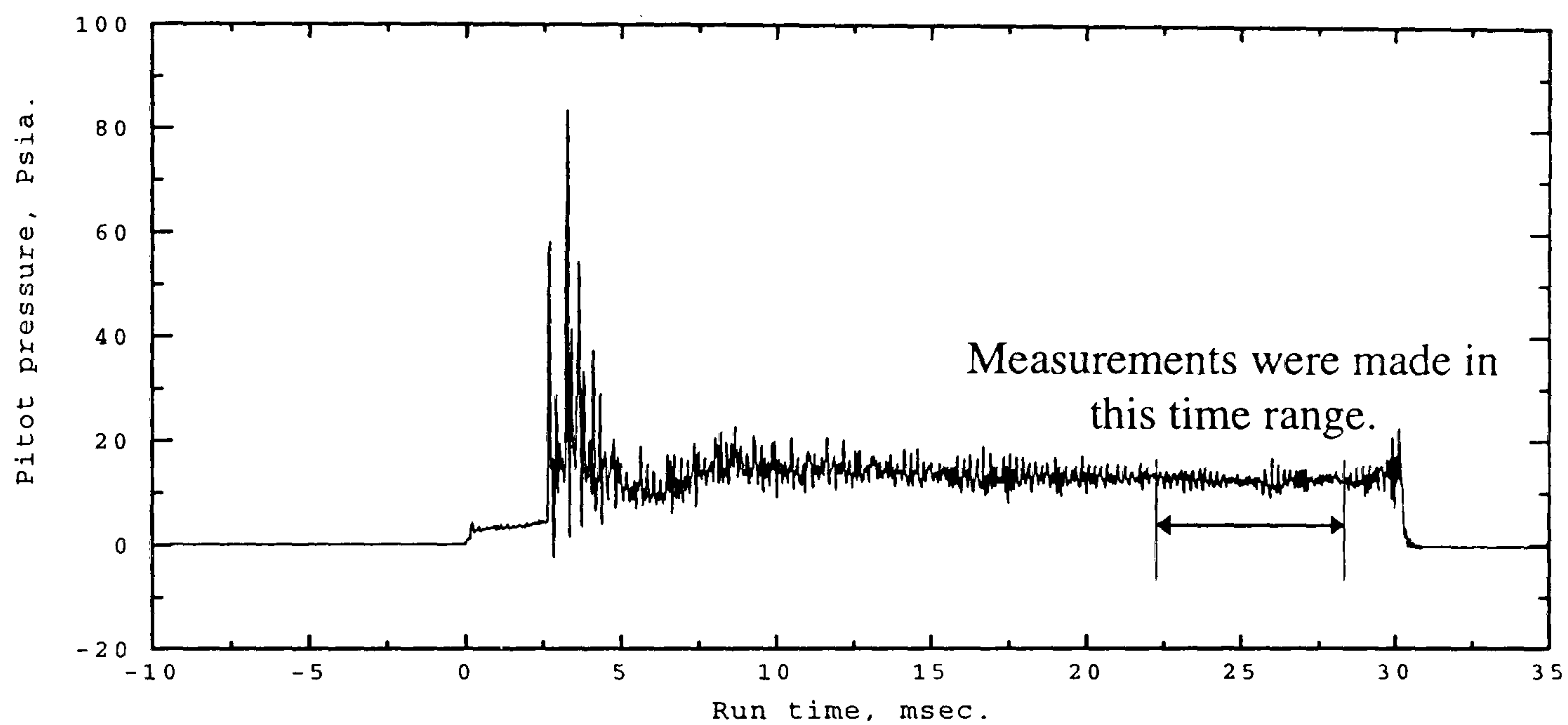


Figure 2(d). A typical pitot pressure trace.

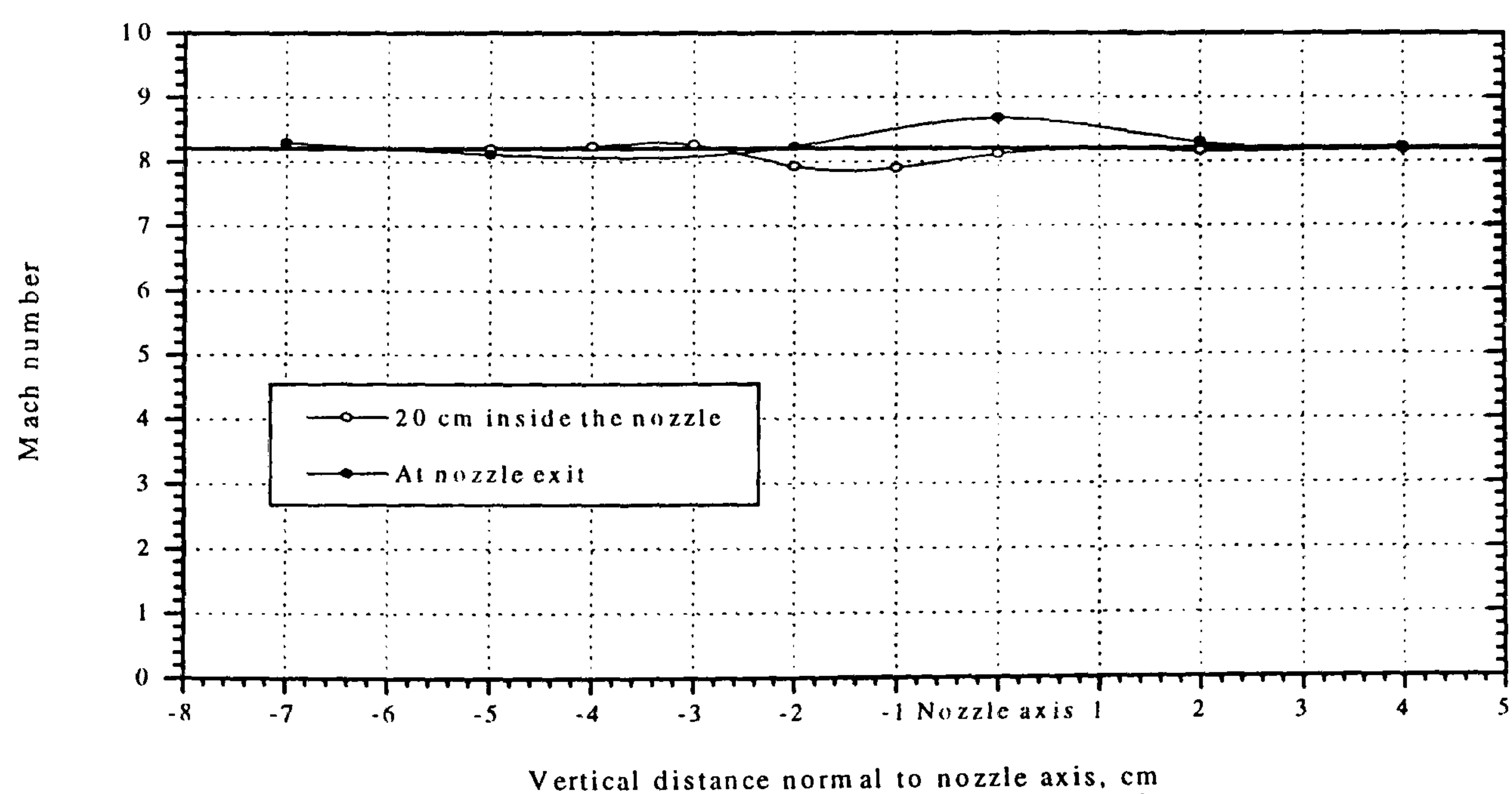
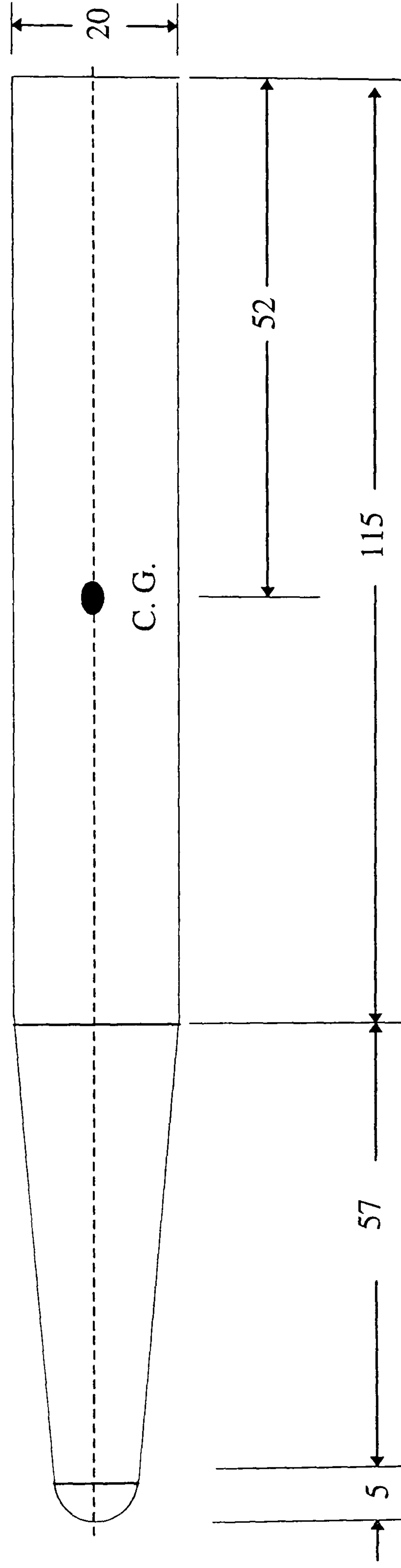
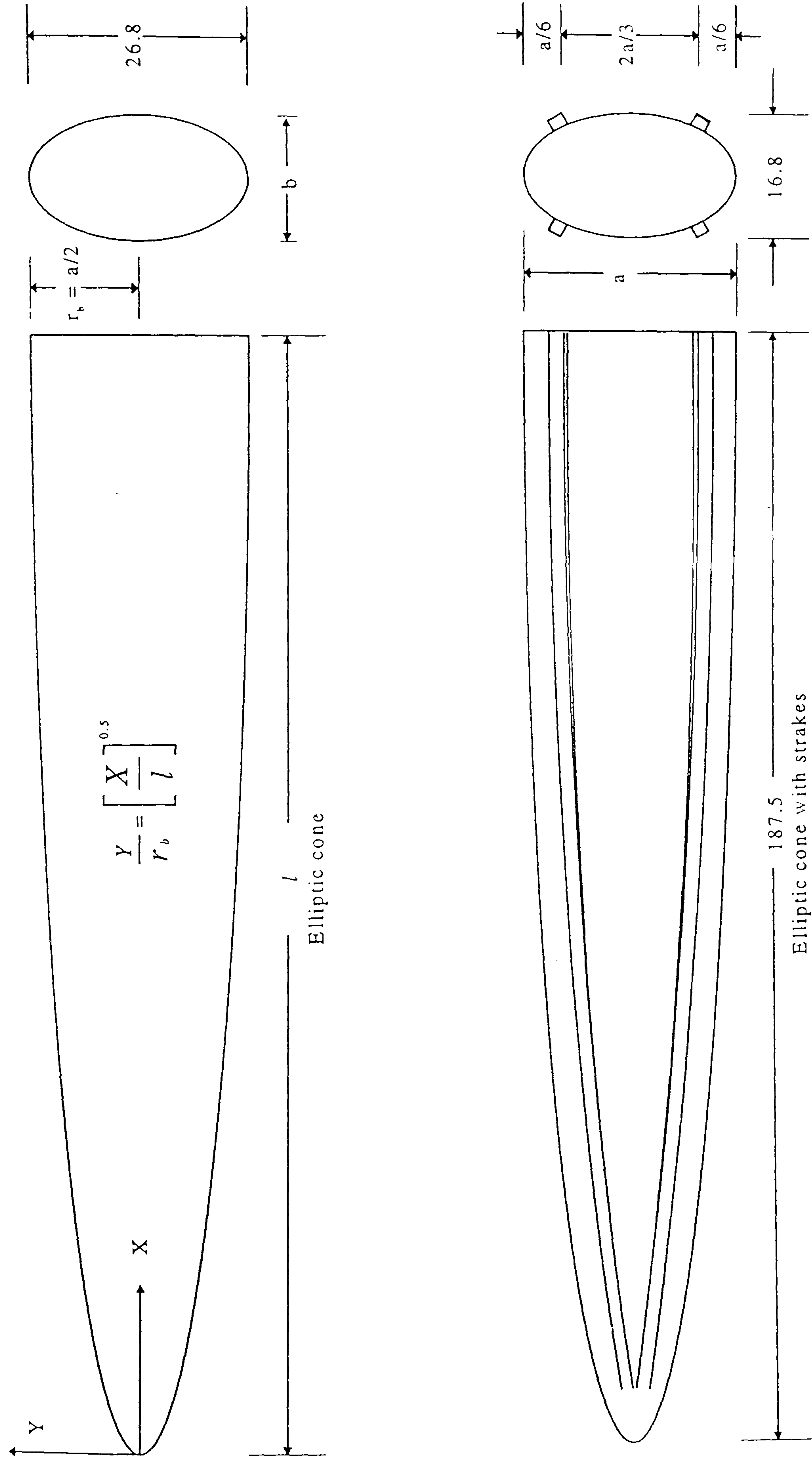


Figure 2(e). Mach number survey along a vertical line normal to the axis of the Mach 8.2 nozzle.



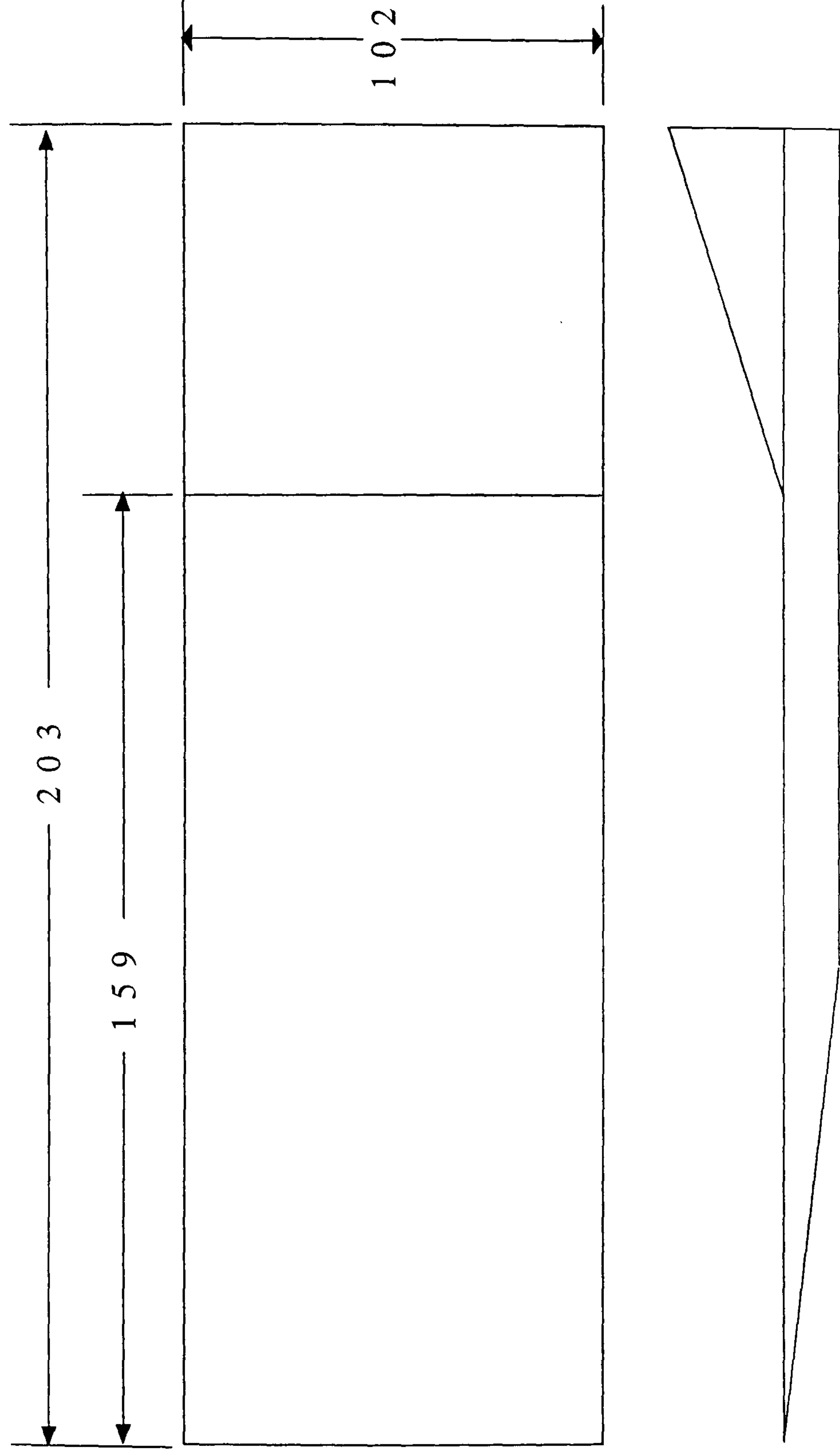
All dimensions are in mm.

Figure 3(a). A sketch of the hemi-spherically blunted cone-cylinder model.



Not to scale. All dimensions are in mm.

Figure 3(b). A sketch of elliptic cone models with and without strakes.



Leading edge diameter < 0.025

Simulated flap blocks of 5, 15 and 25°

All dimensions are in mm.

Figure 3(c) A sketch of sharp leading edge flat plate with trailing edge flap.

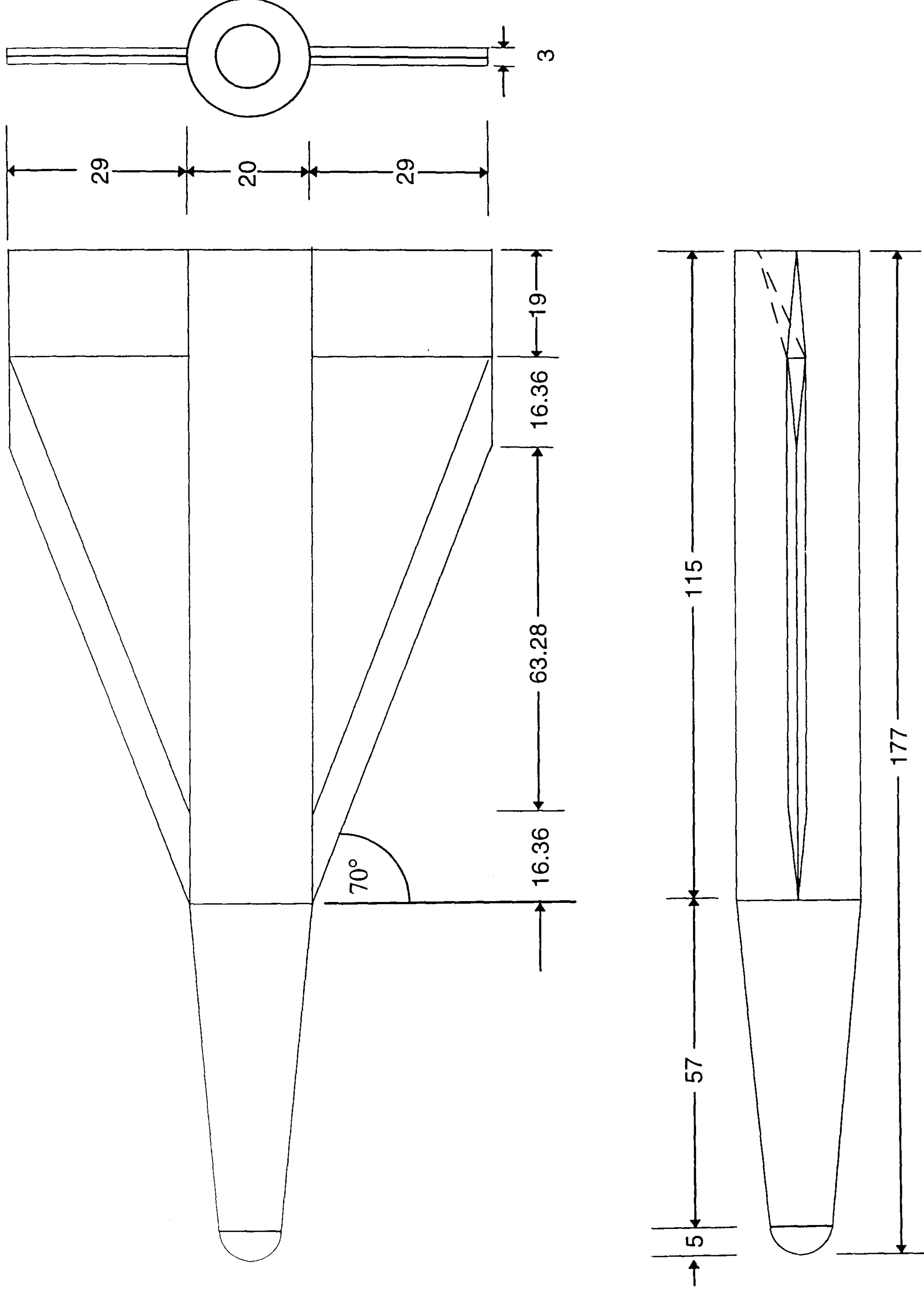


Figure 3(d). A sketch showing the dimensions of the wing-body configuration.

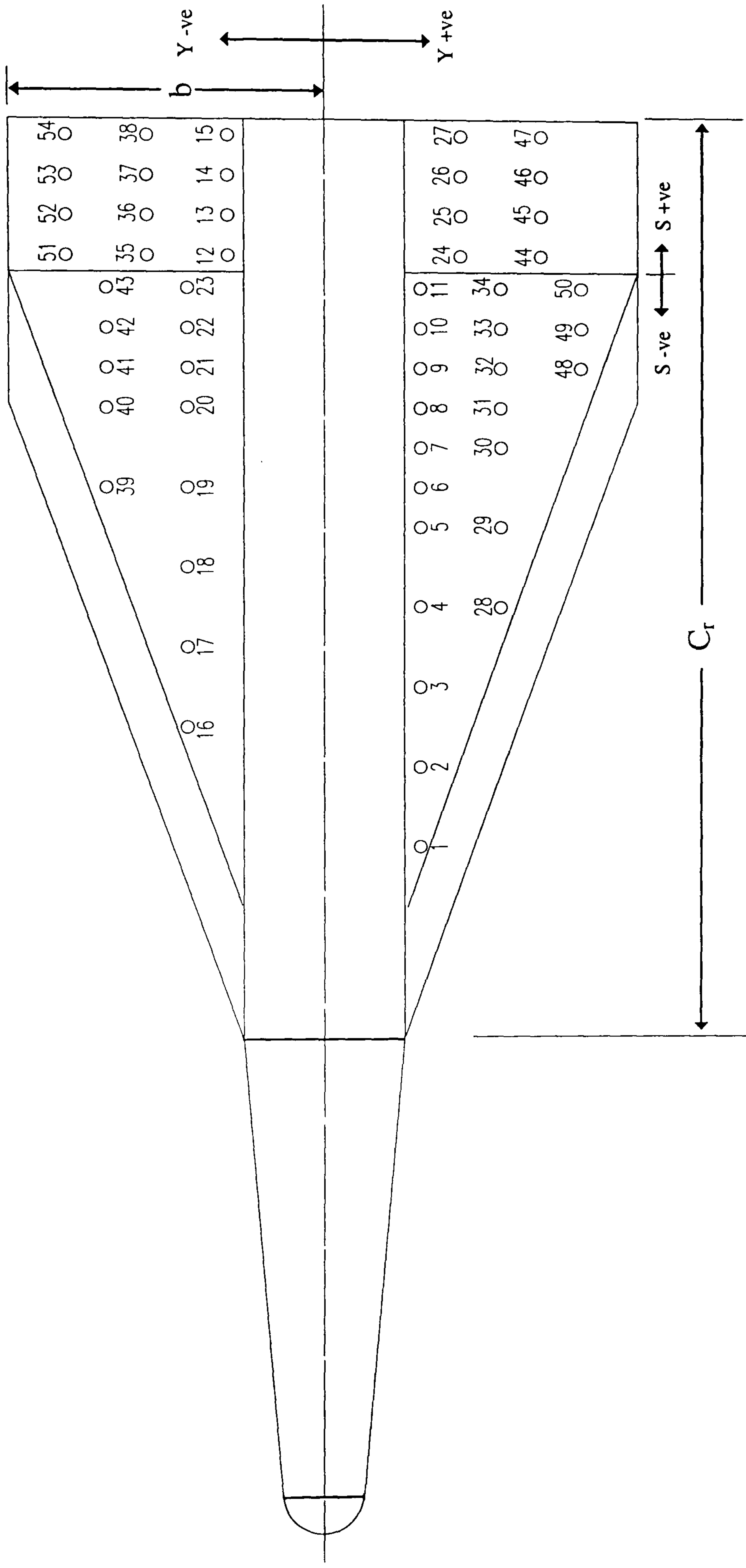
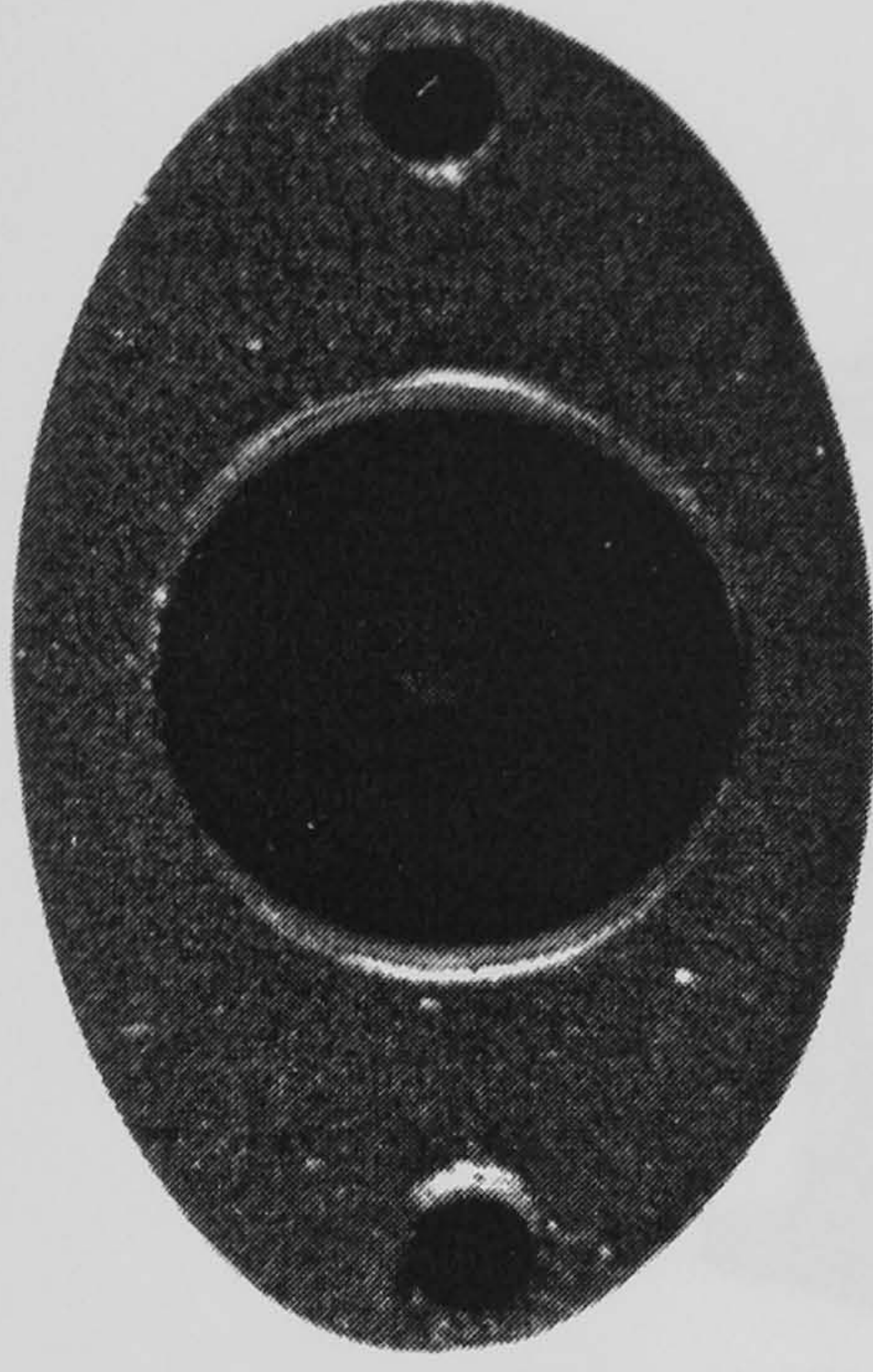


Figure 3(e). A sketch showing the location of pressure orifices on the wing-body.



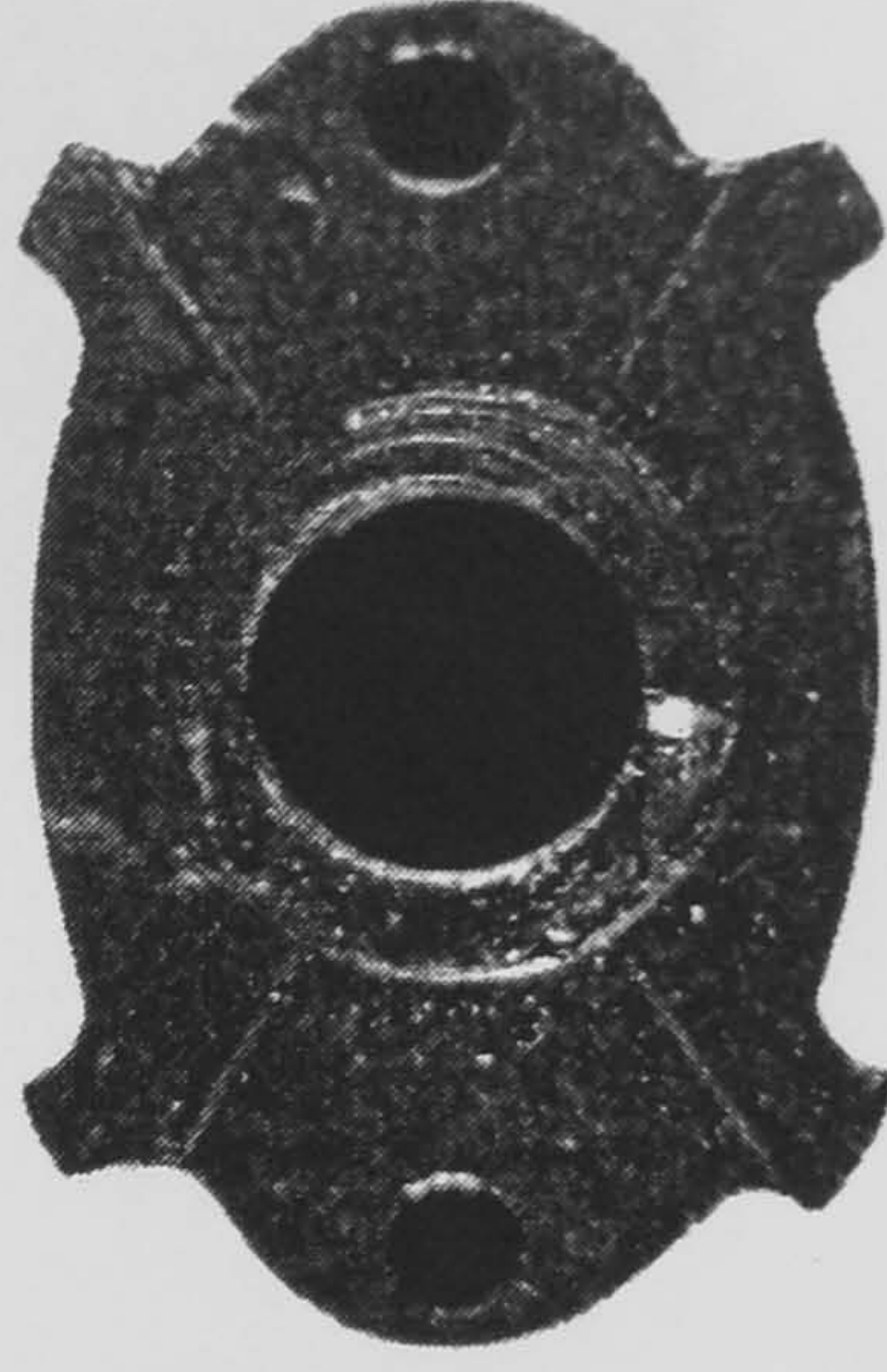
Top and side view of the power-law body without strakes



View from the base



Top and side view of the power-law body with strakes



View from the base

Figure 4(a). A photograph showing power-law body with and without strakes.

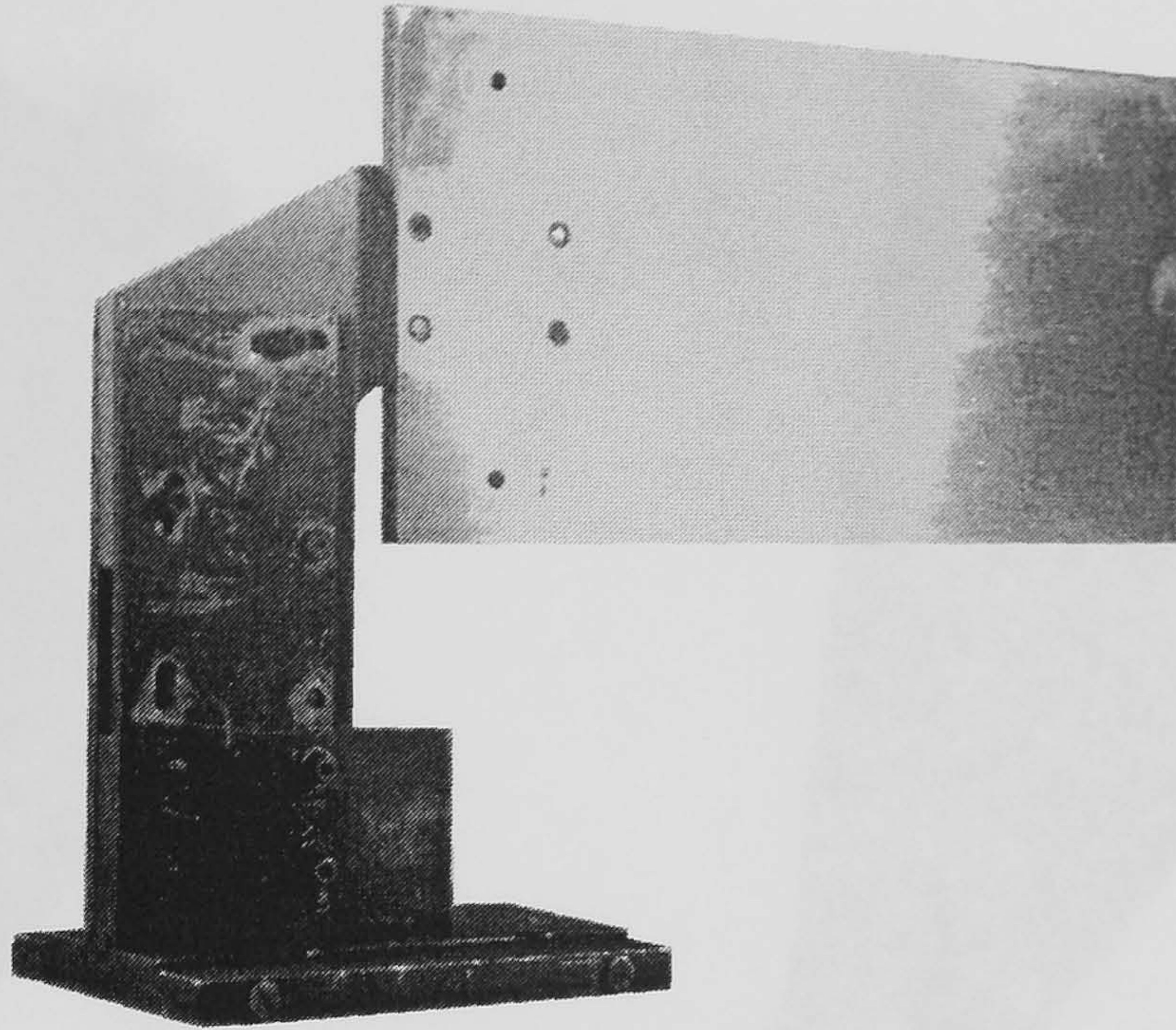


Figure 4(b). A photograph showing the flat plate model.

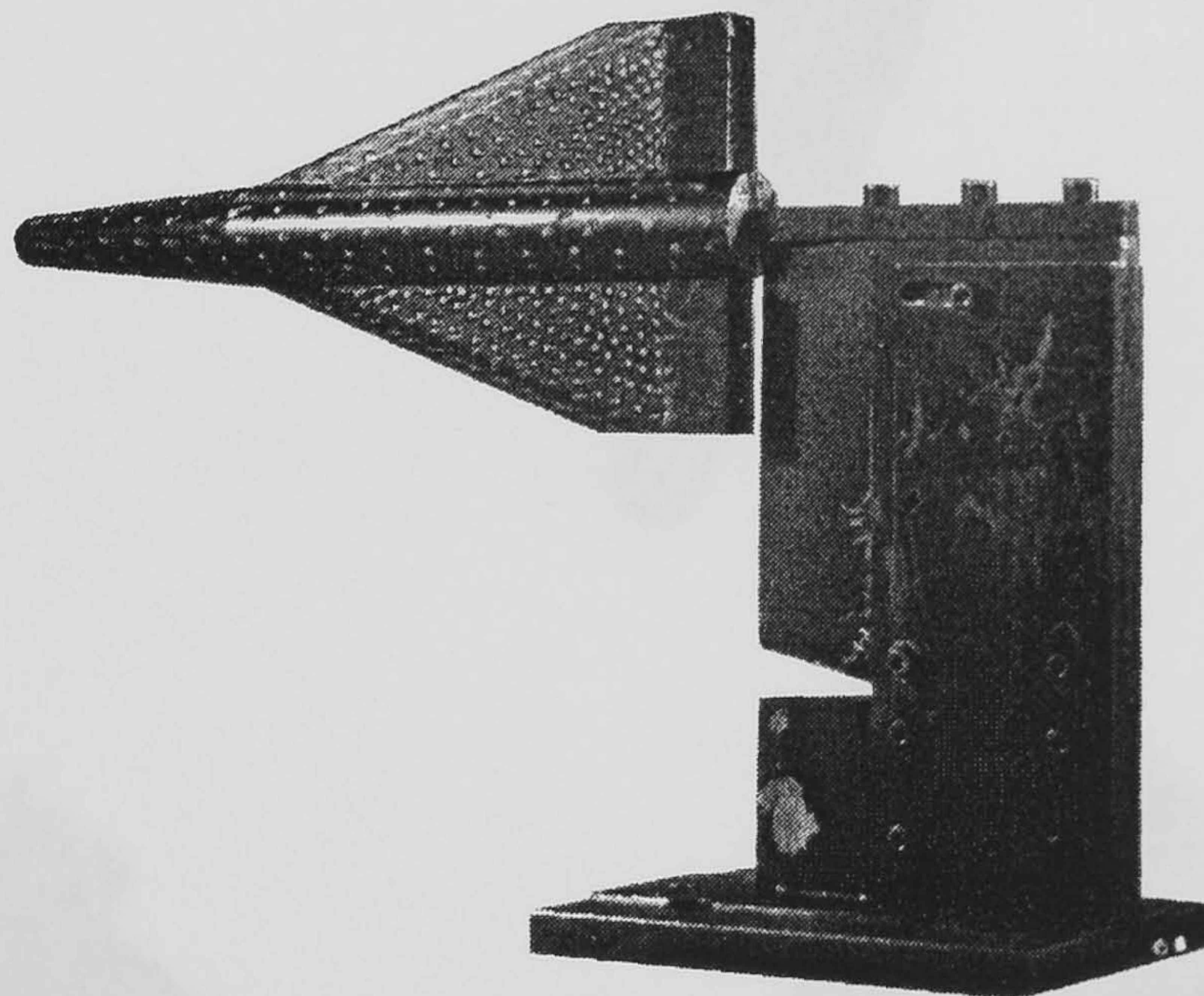


Figure 4 (c). A photograph showing the wing-body model.

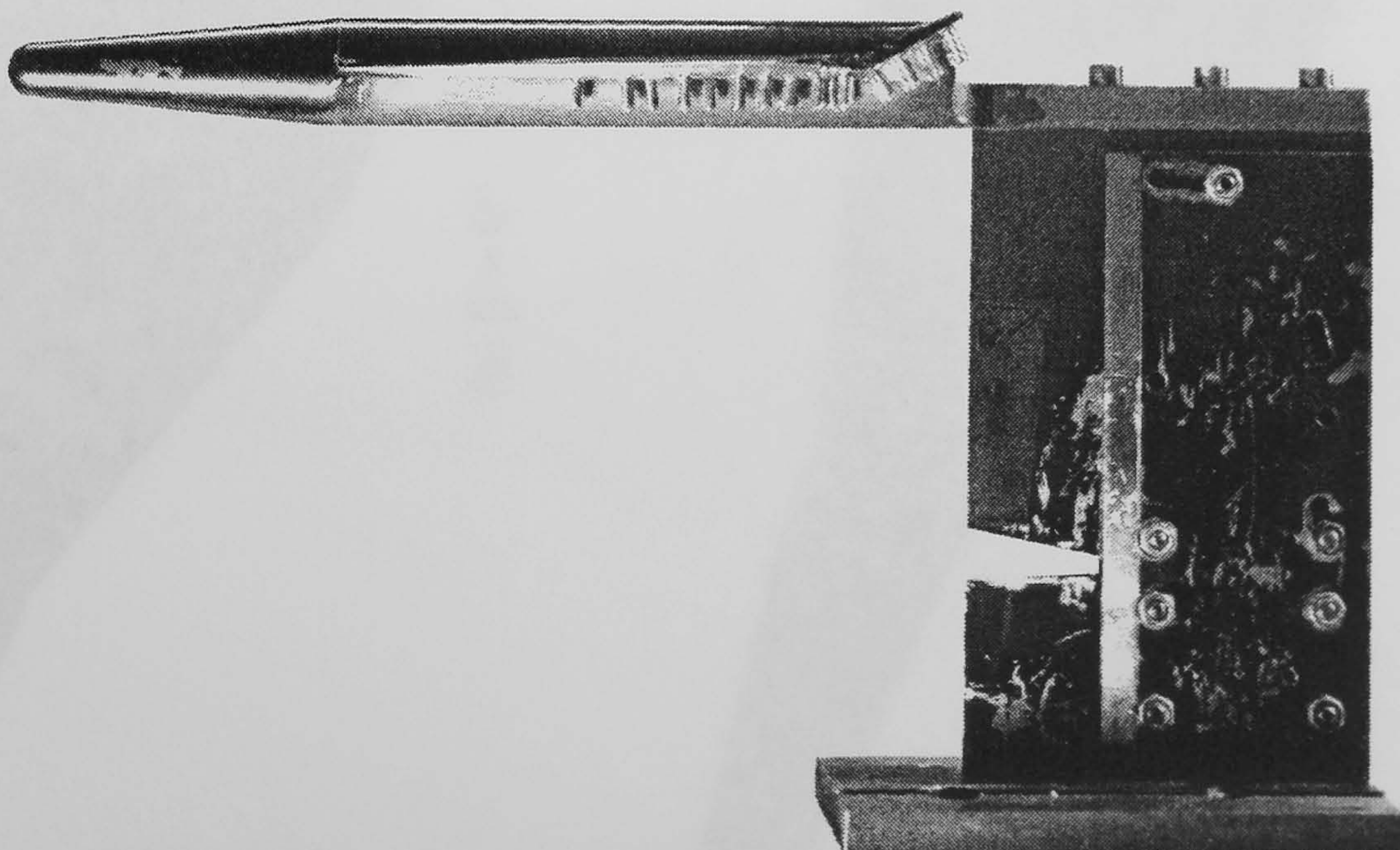


Figure 4(d). A photograph showing the wing-body model used for surface pressure measurements.



(a) $\beta = 0^\circ$



(b) $\beta = 5^\circ$



(c) $\beta = 15^\circ$



(d) $\beta = 25^\circ$

Figure 4(e). Photographs showing the wing-body models used for the force measurements.

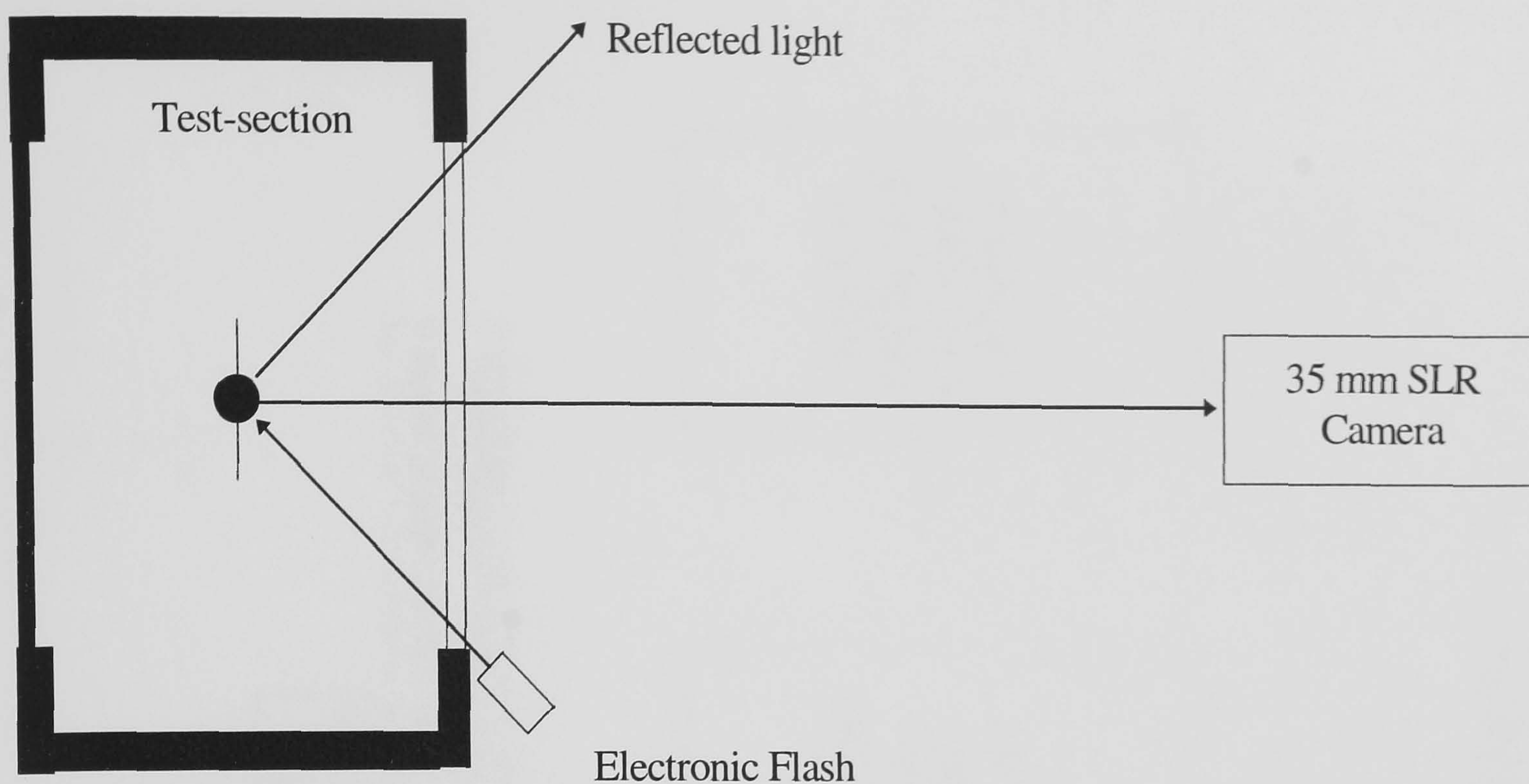


Figure 5(a). A sketch showing the set-up for the liquid crystal technique as viewed along the direction of the flow.

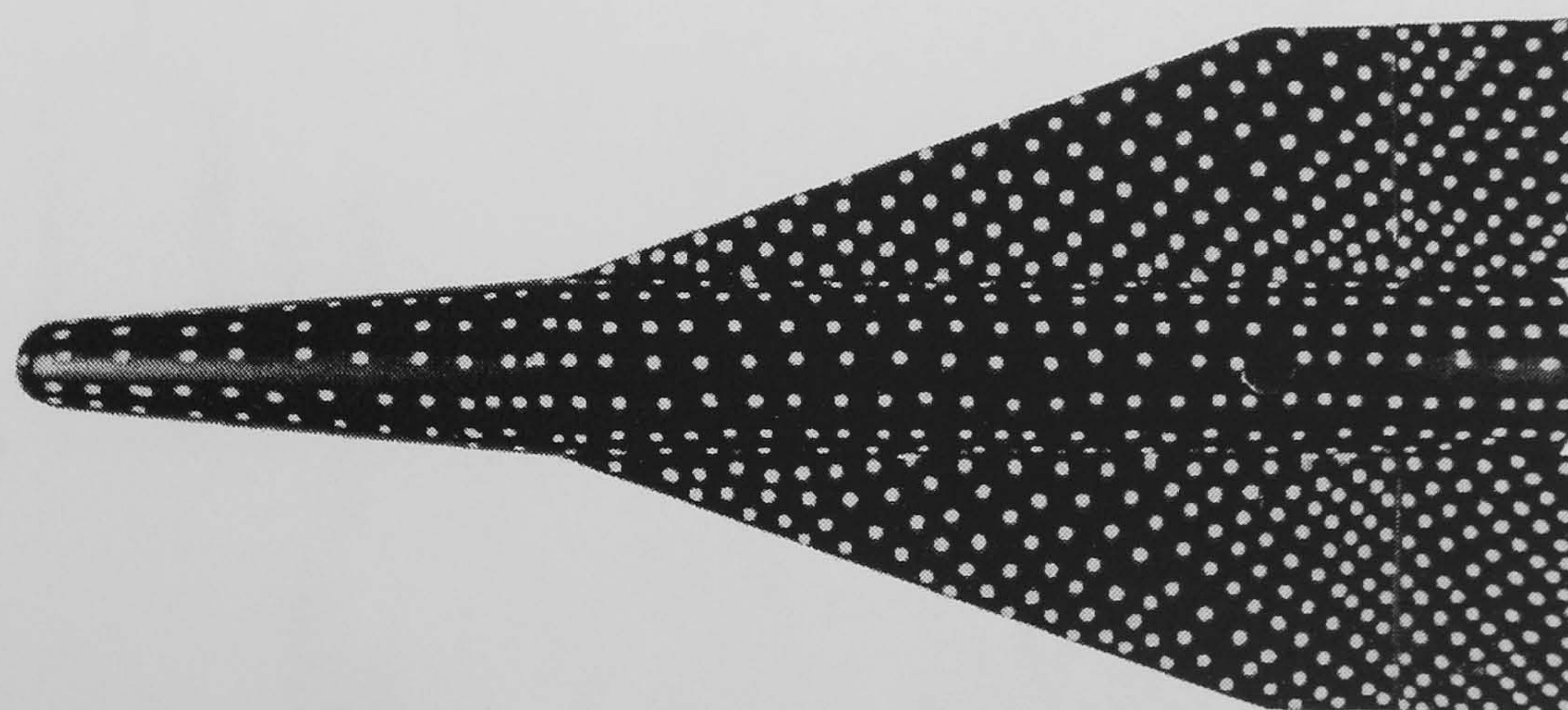


Figure 5(b). A typical oil-dot pattern before the firing of the tunnel.

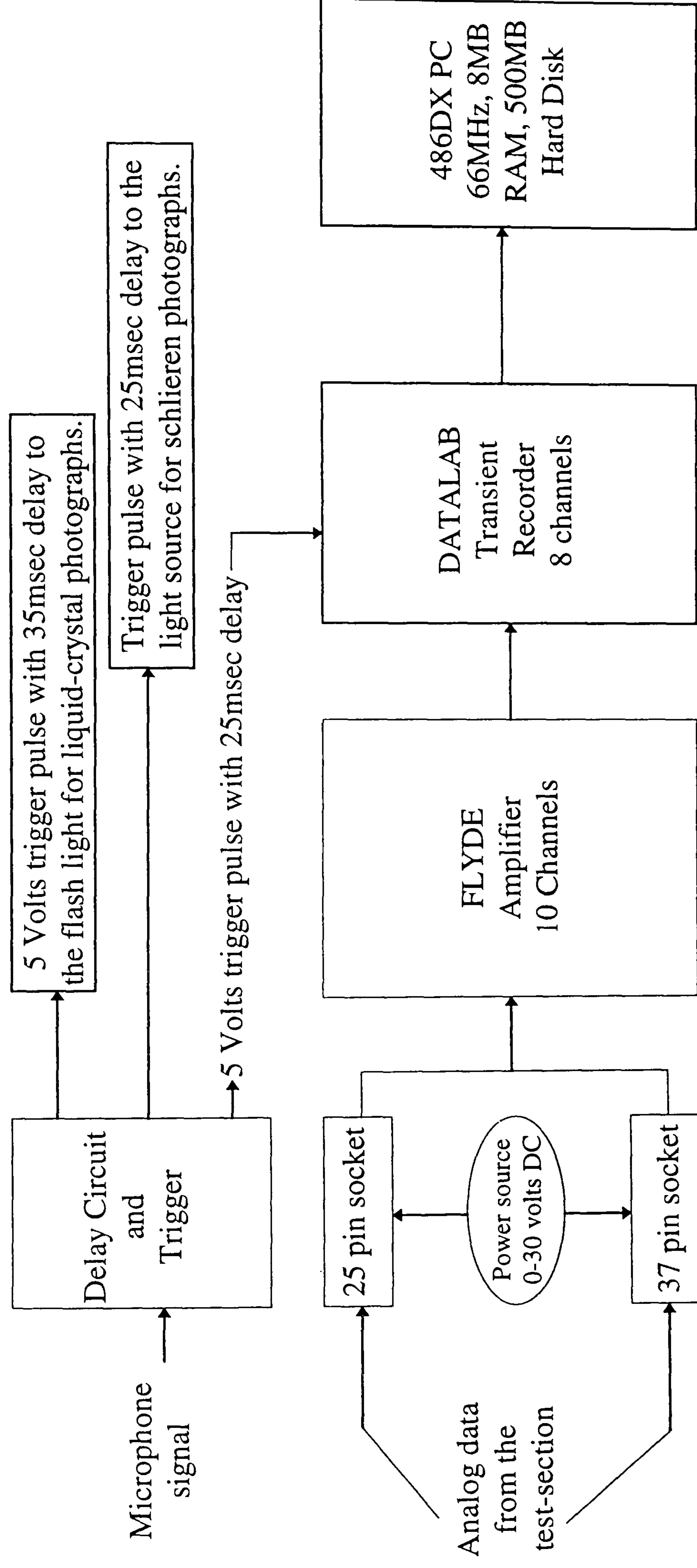


Figure 6. A schematic of the gun-tunnel data acquisition system.

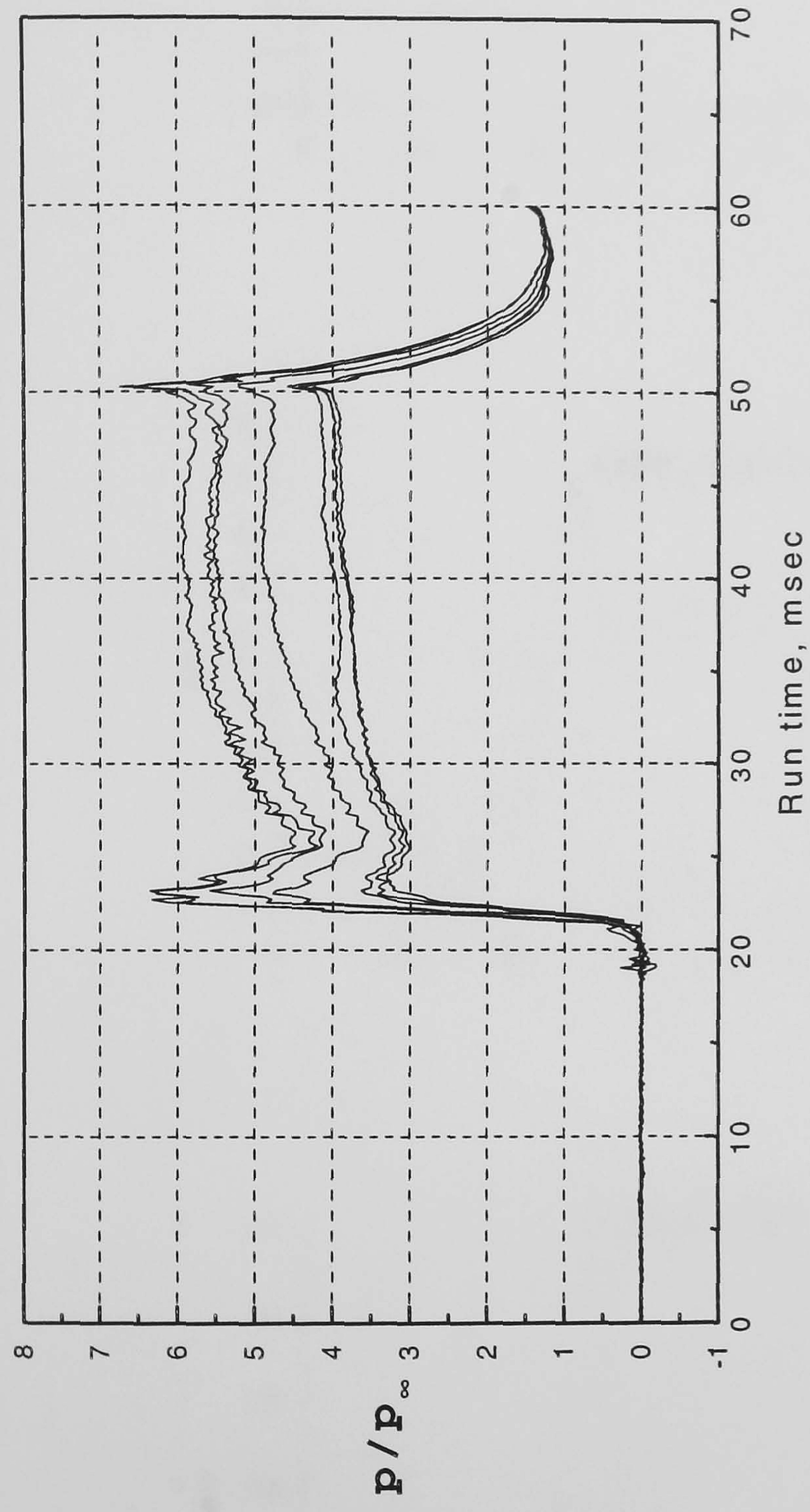


Figure 7. A typical pressure trace from seven pressure tappings.

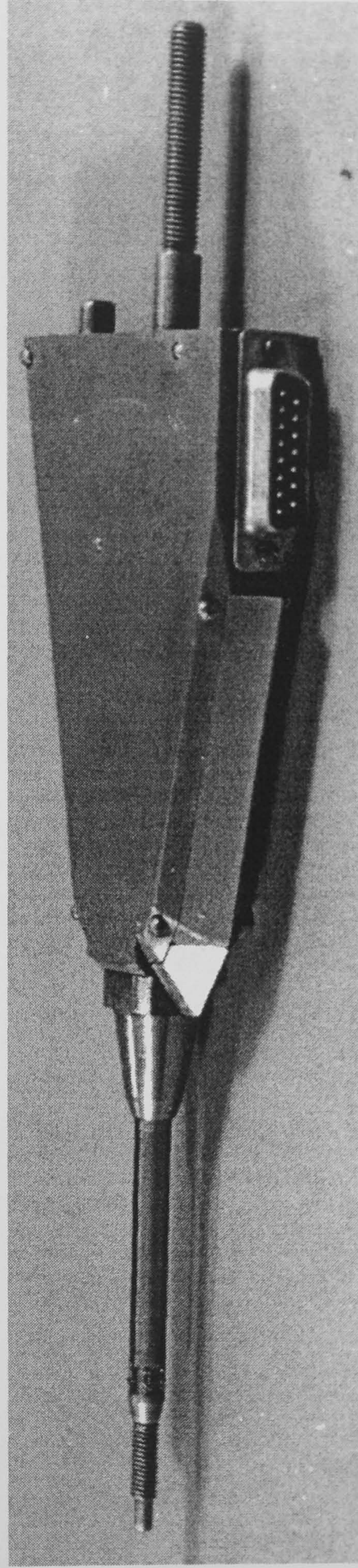


Figure 8. A photograph showing the three component strain gauge balance.

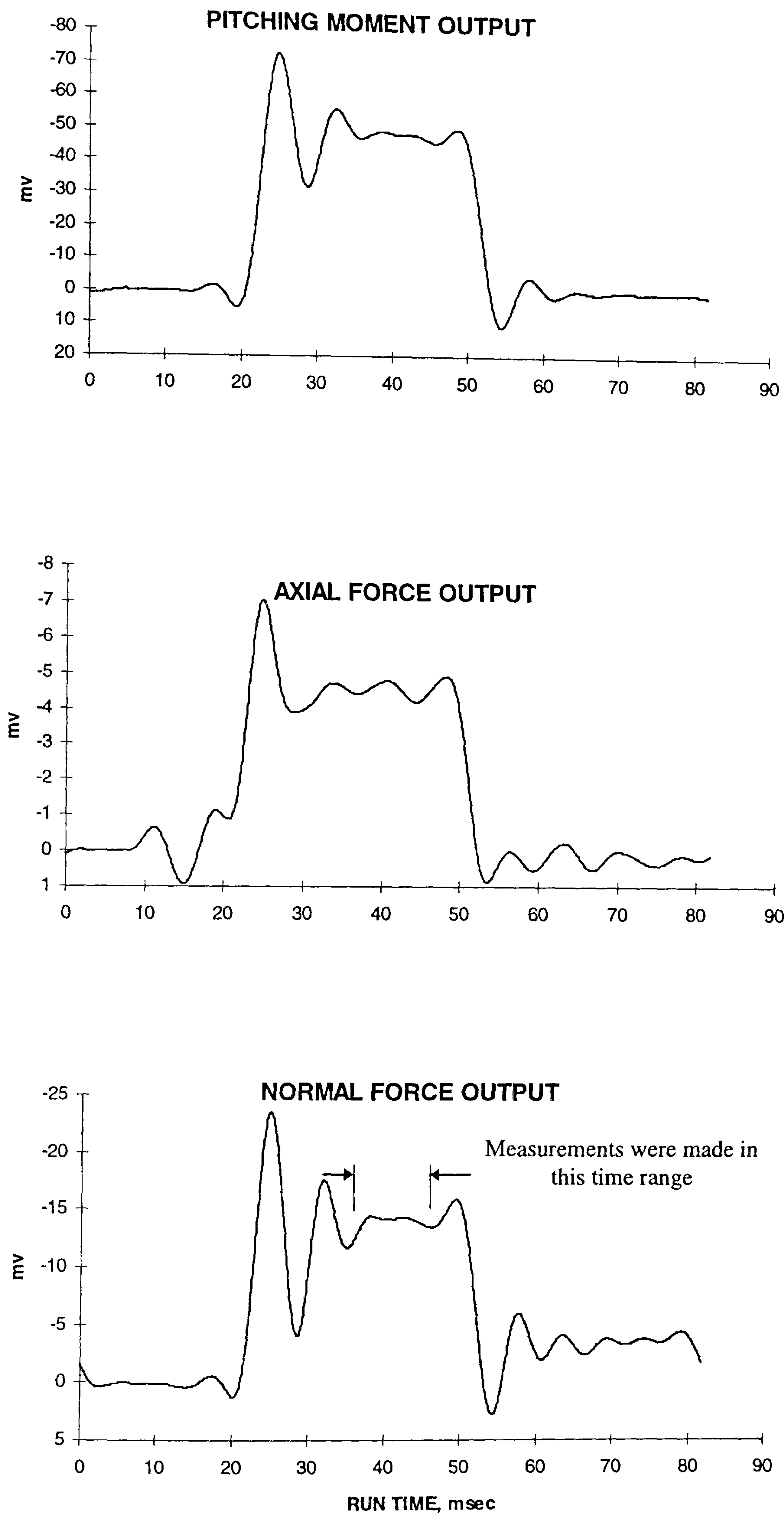


Figure 9. A typical signal trace from three channels of the strain gauge balance.

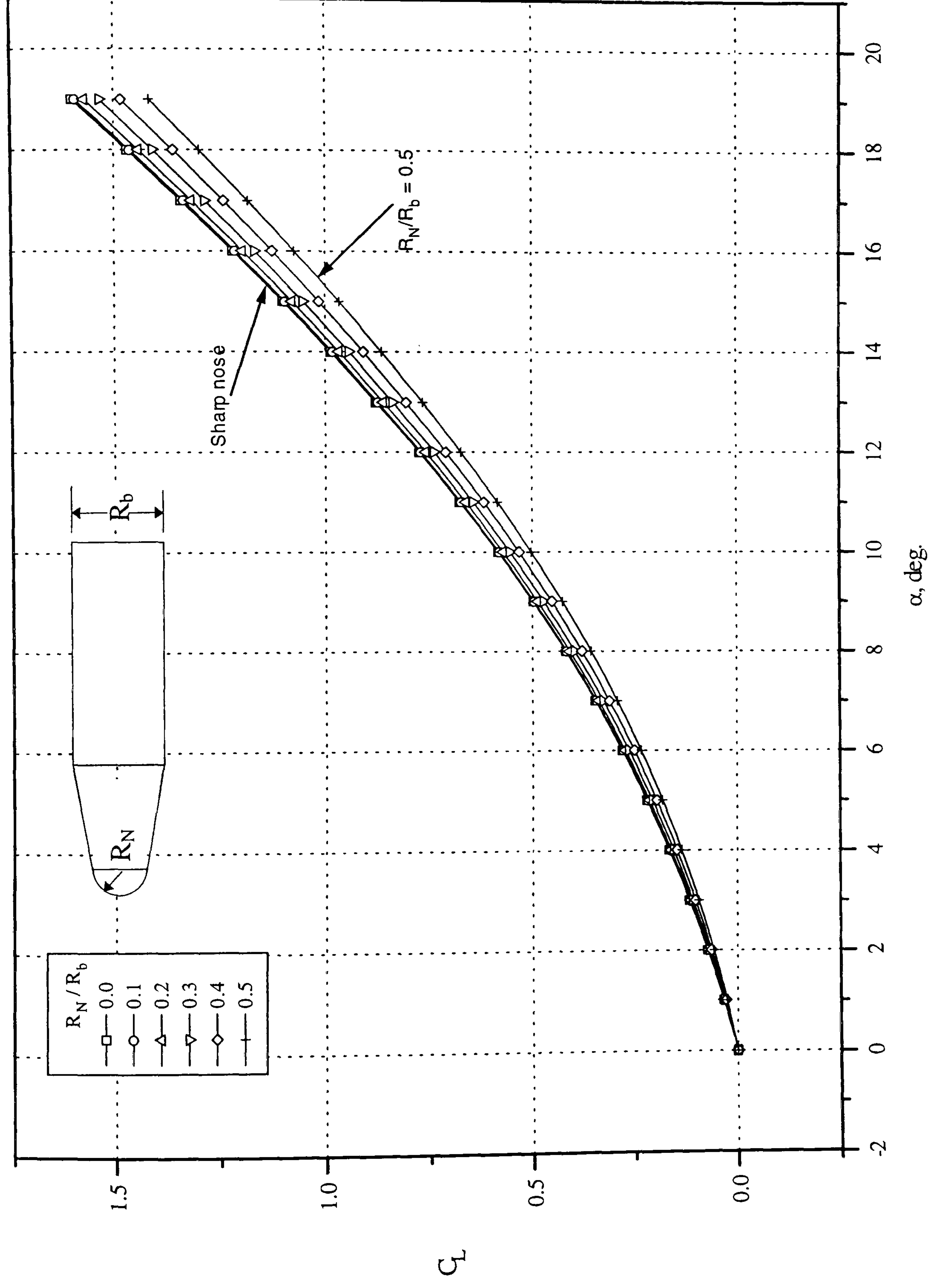


Figure 10(a). Effect of nose bluntness on Newtonian lift coefficient of spherically blunted cone-cylinder body.

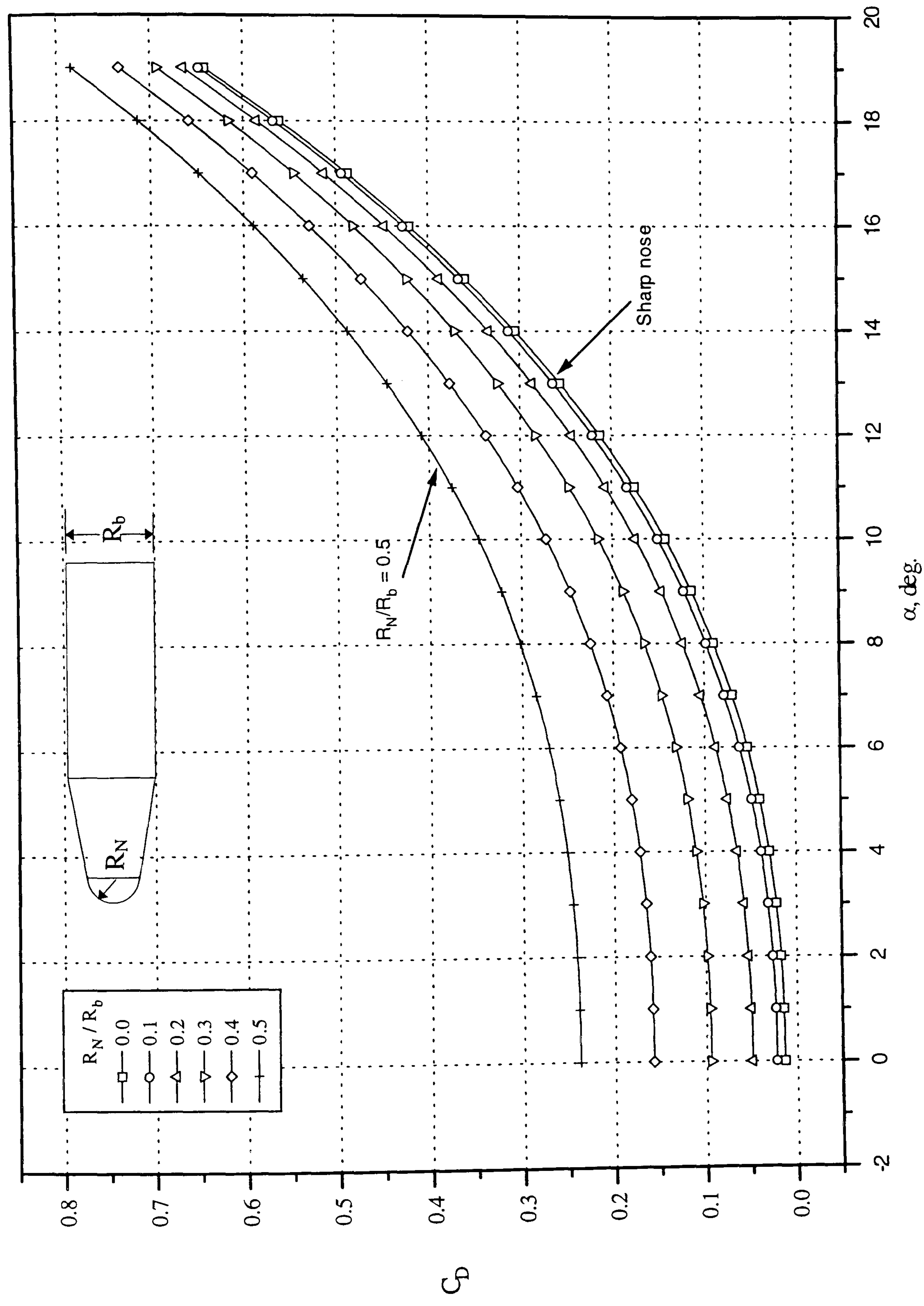


Figure 10(b). Effect of nose bluntness on Newtonian drag coefficient of spherically blunted cone-cylinder body.

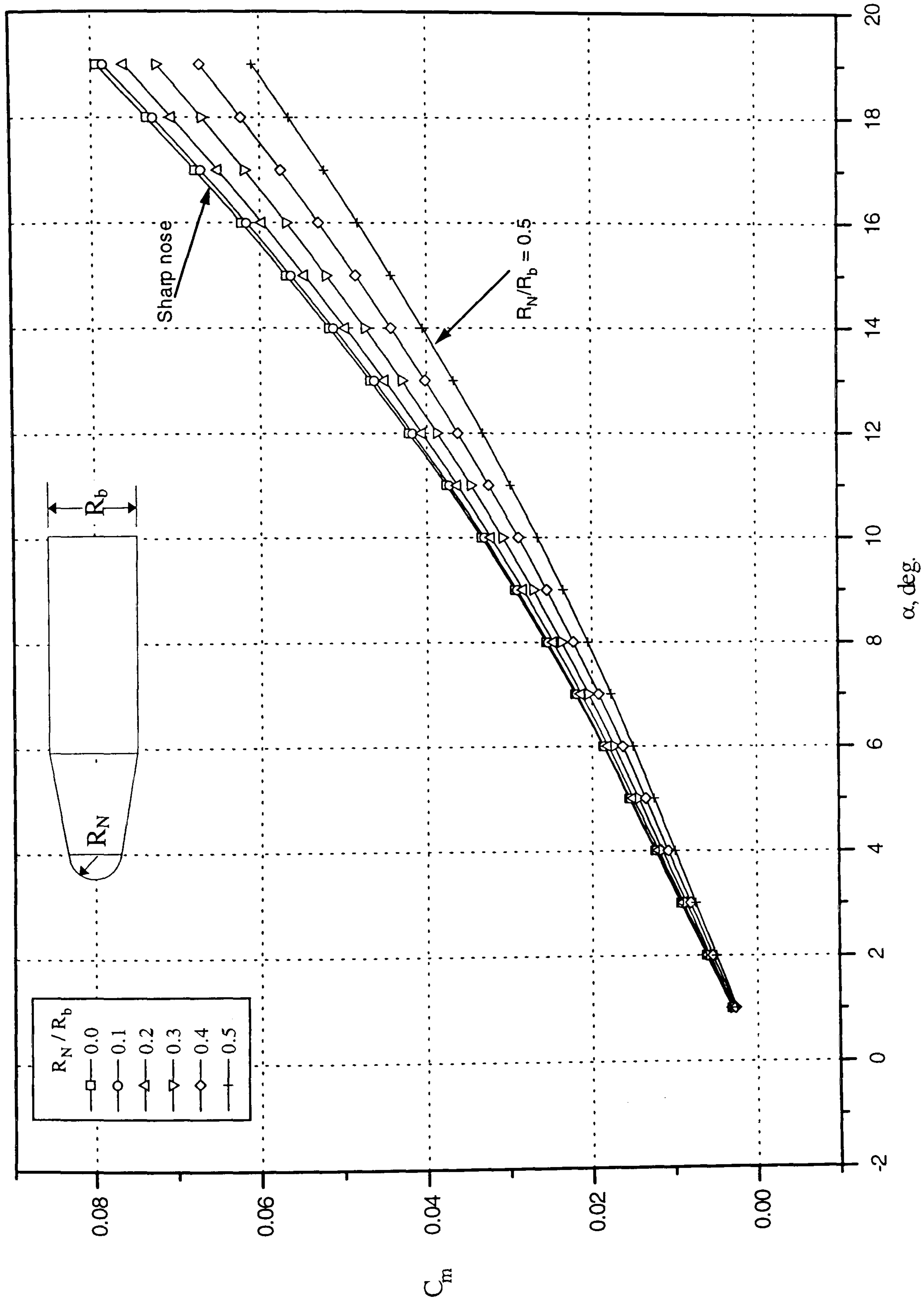


Figure 10(c). Effect of nose bluntness on Newtonian pitching moment coefficient of spherically blunted cone-cylinder body.

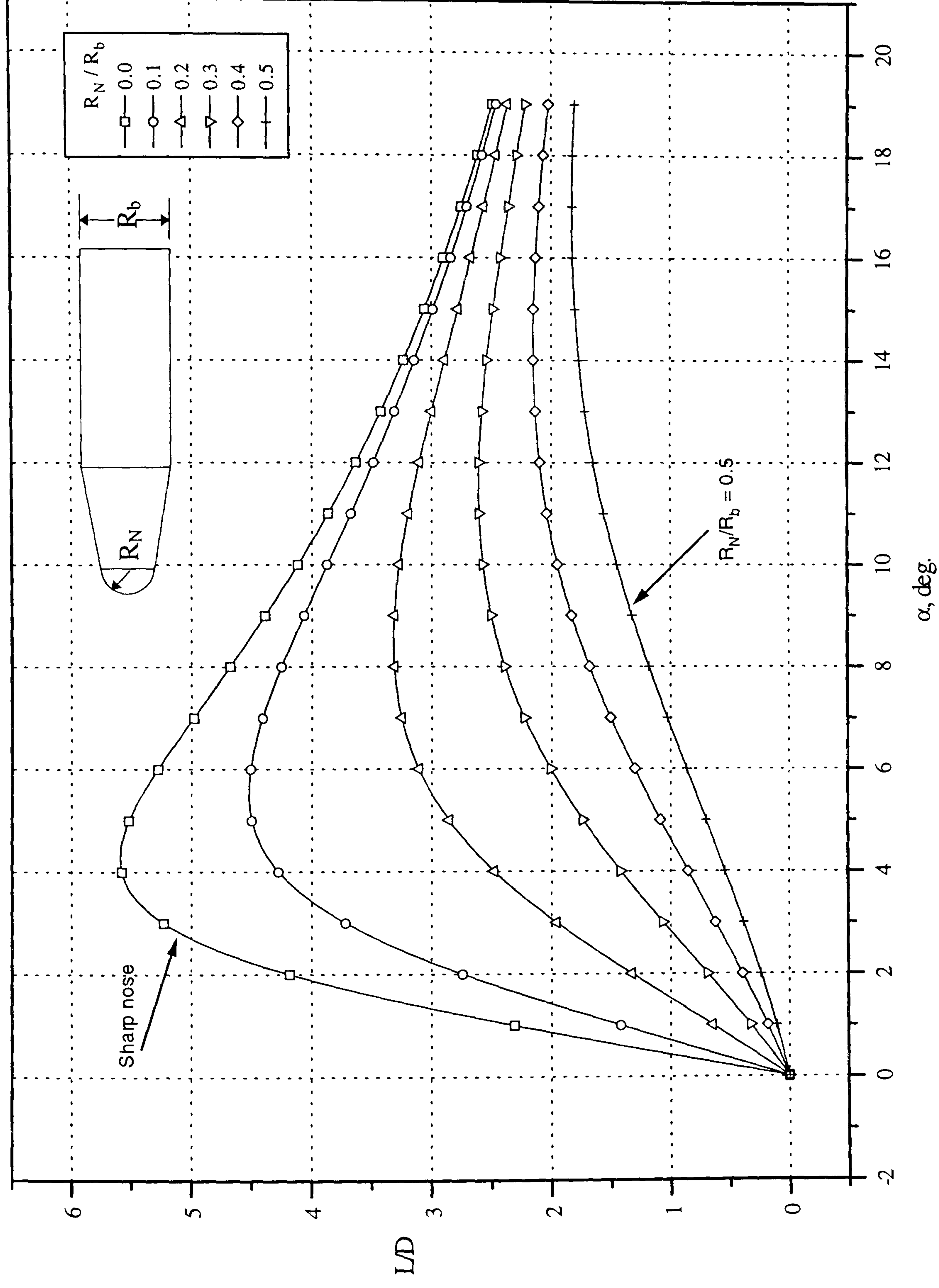


Figure 10(d). Effect of nose bluntness on Newtonian lift to drag ratio of spherically blunted cone-cylinder body.

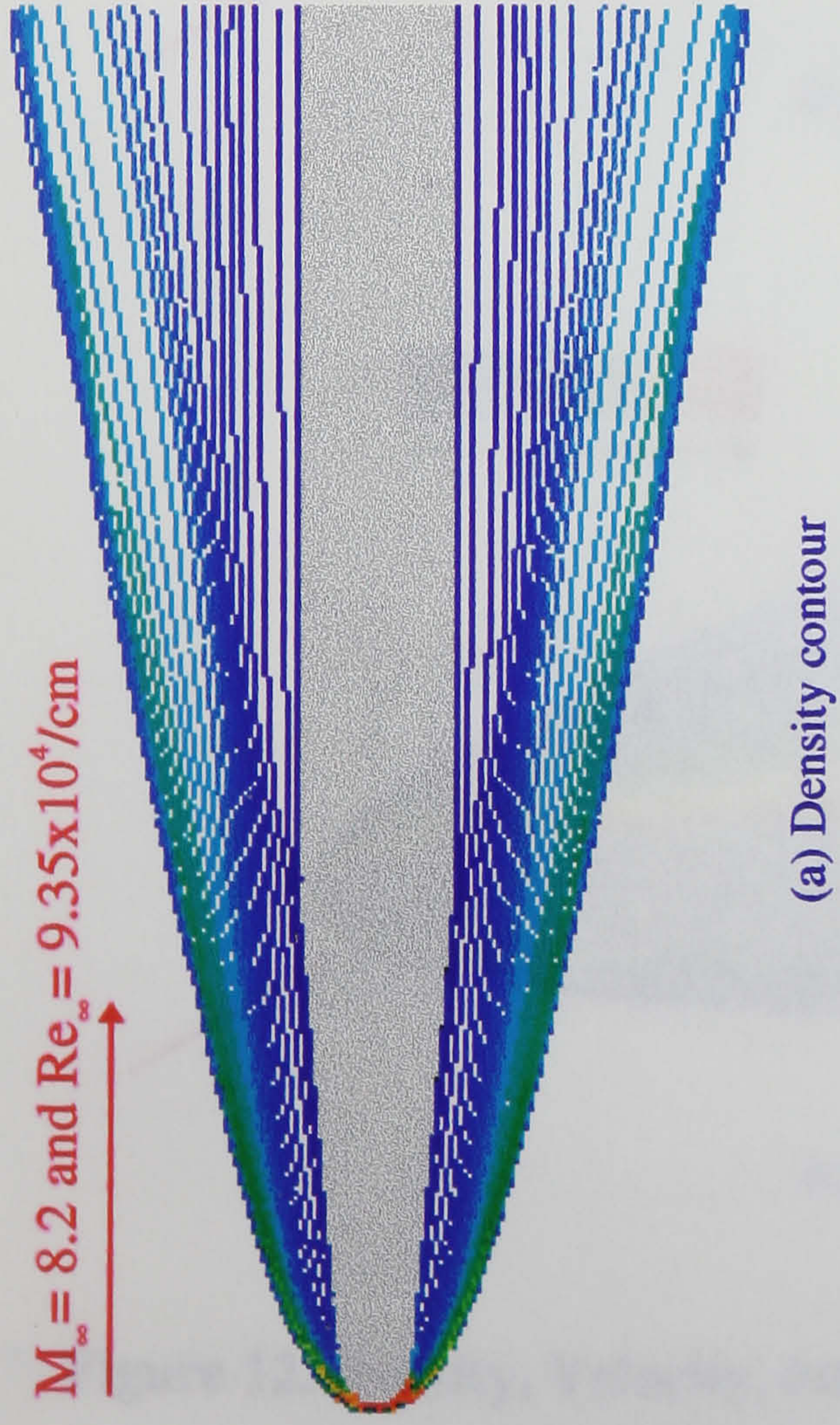
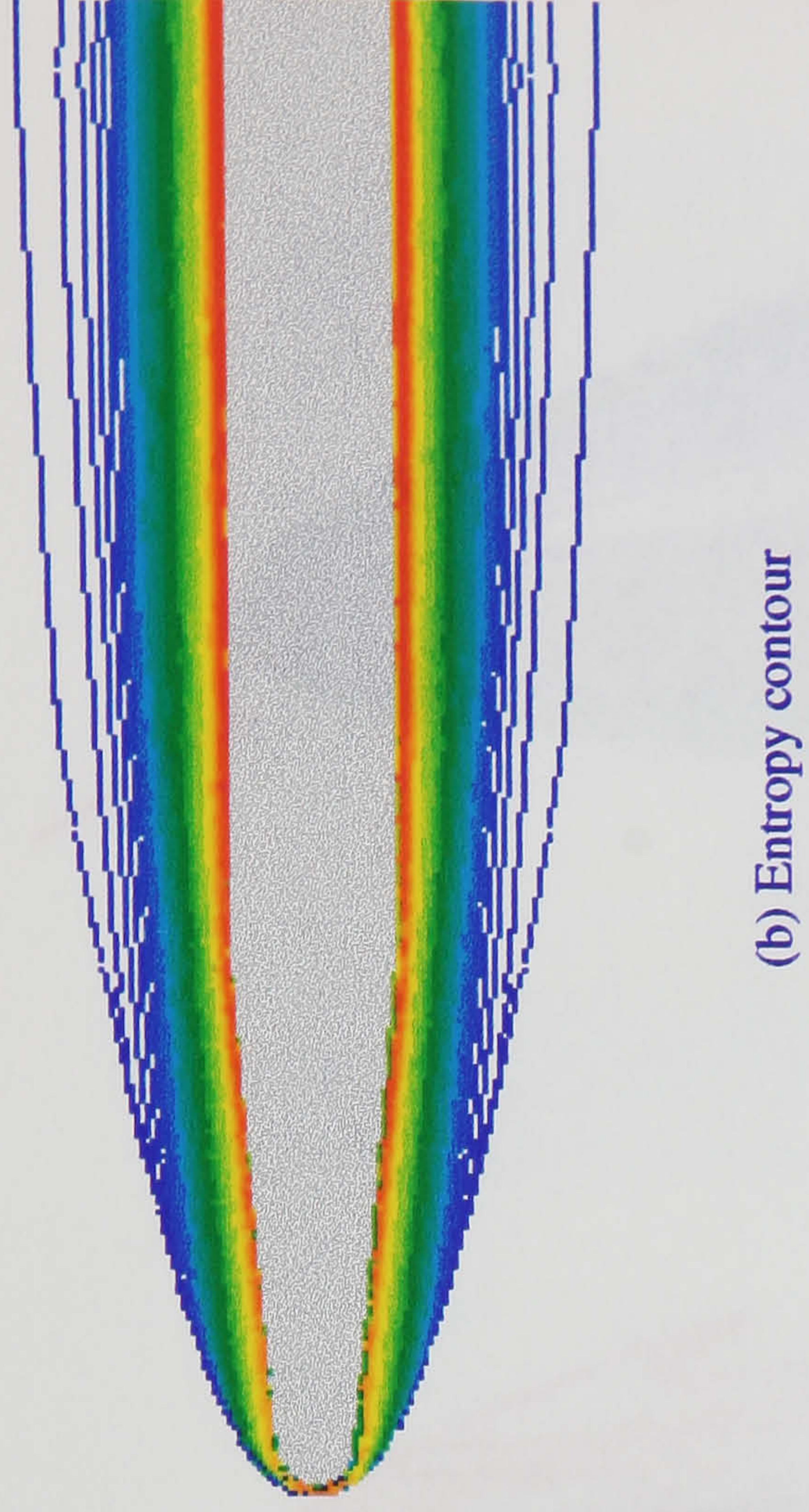
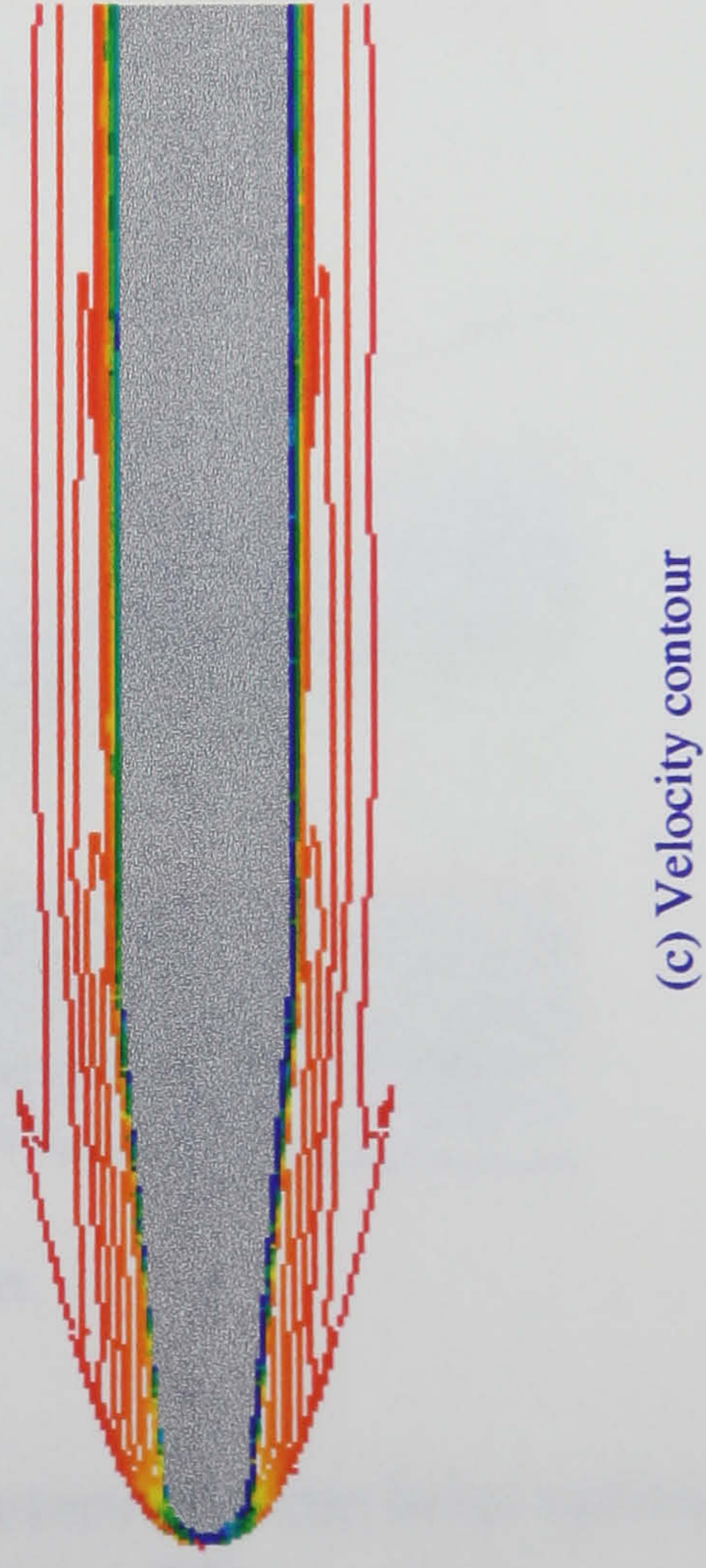
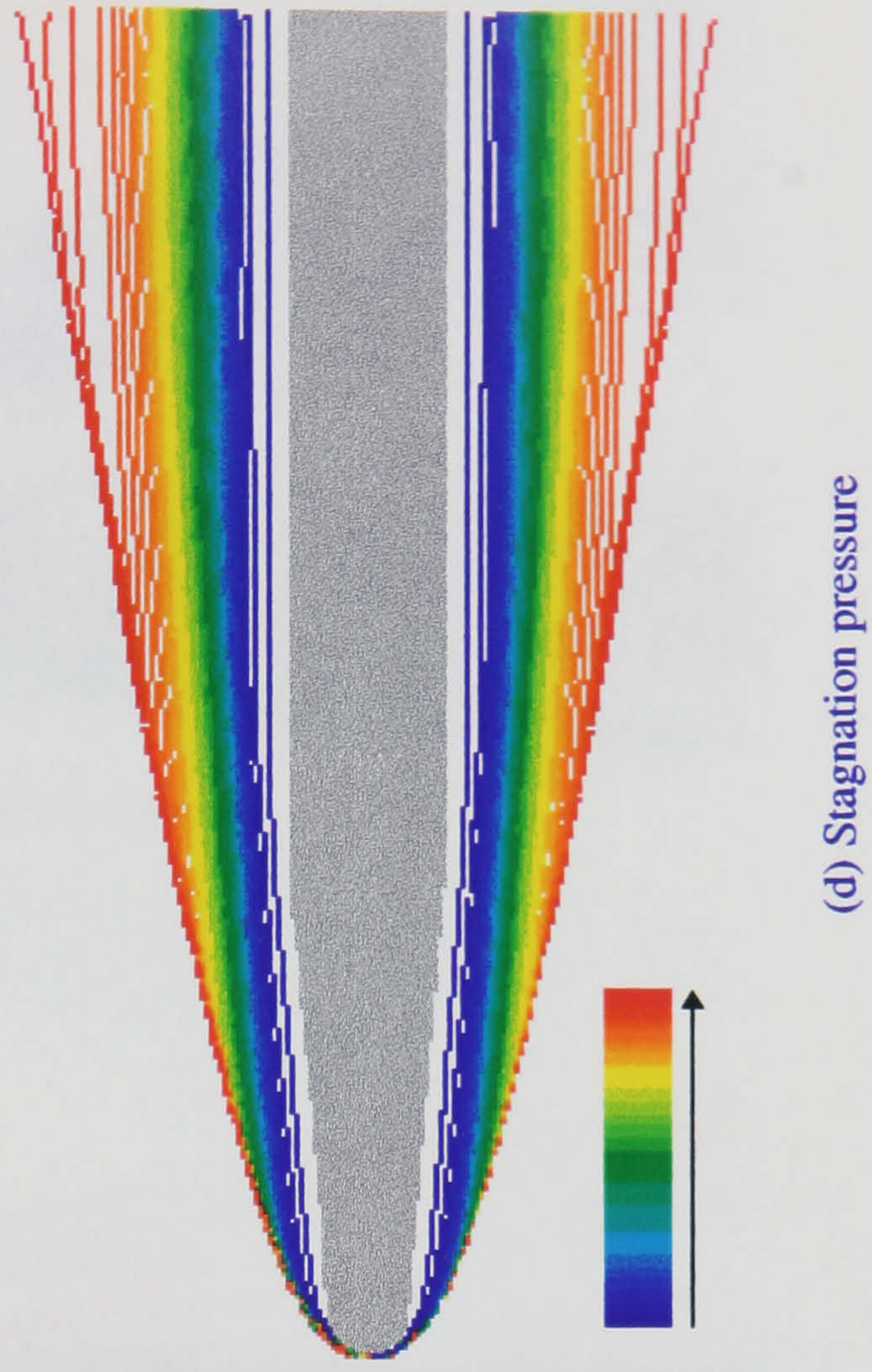
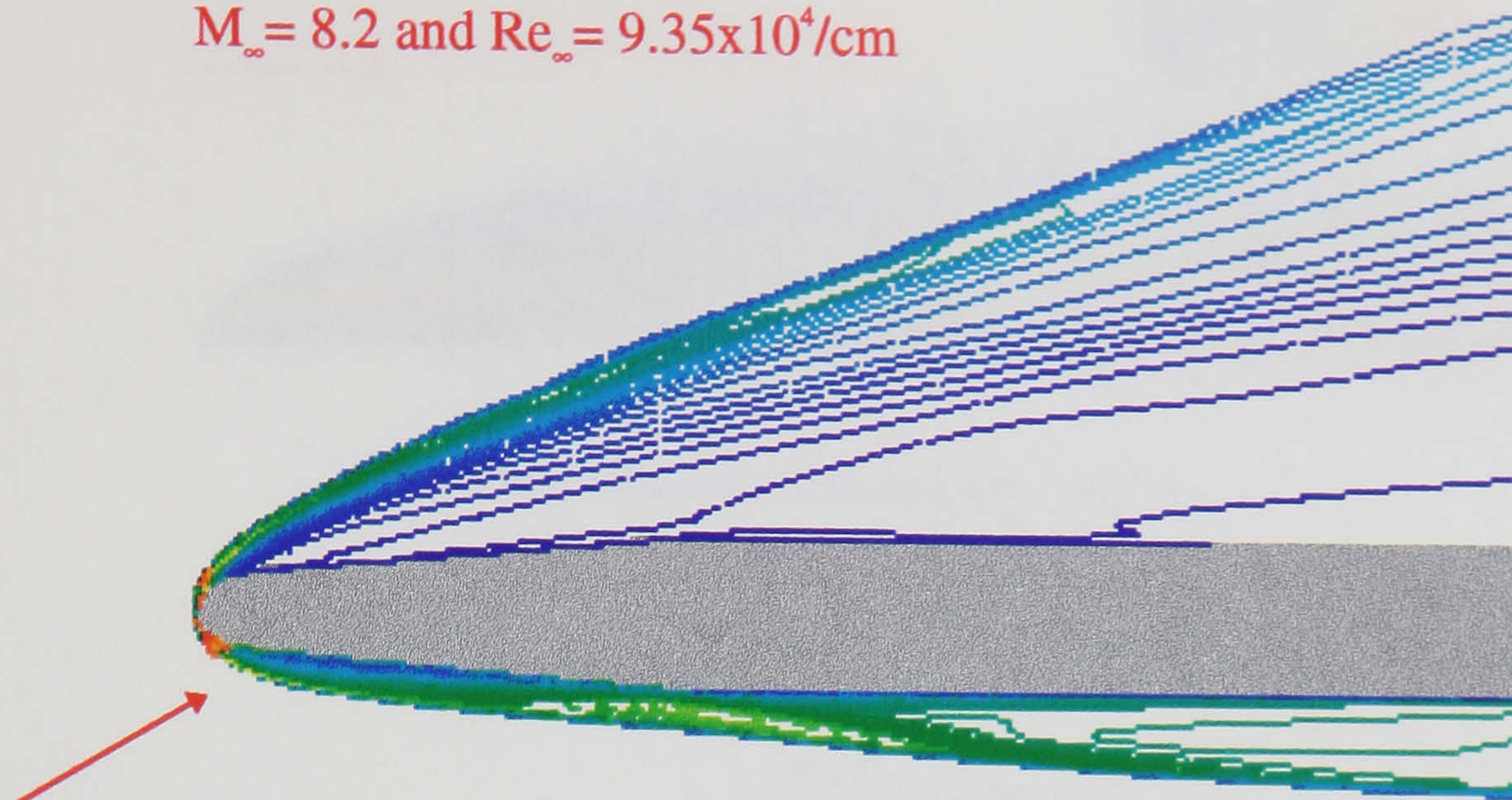
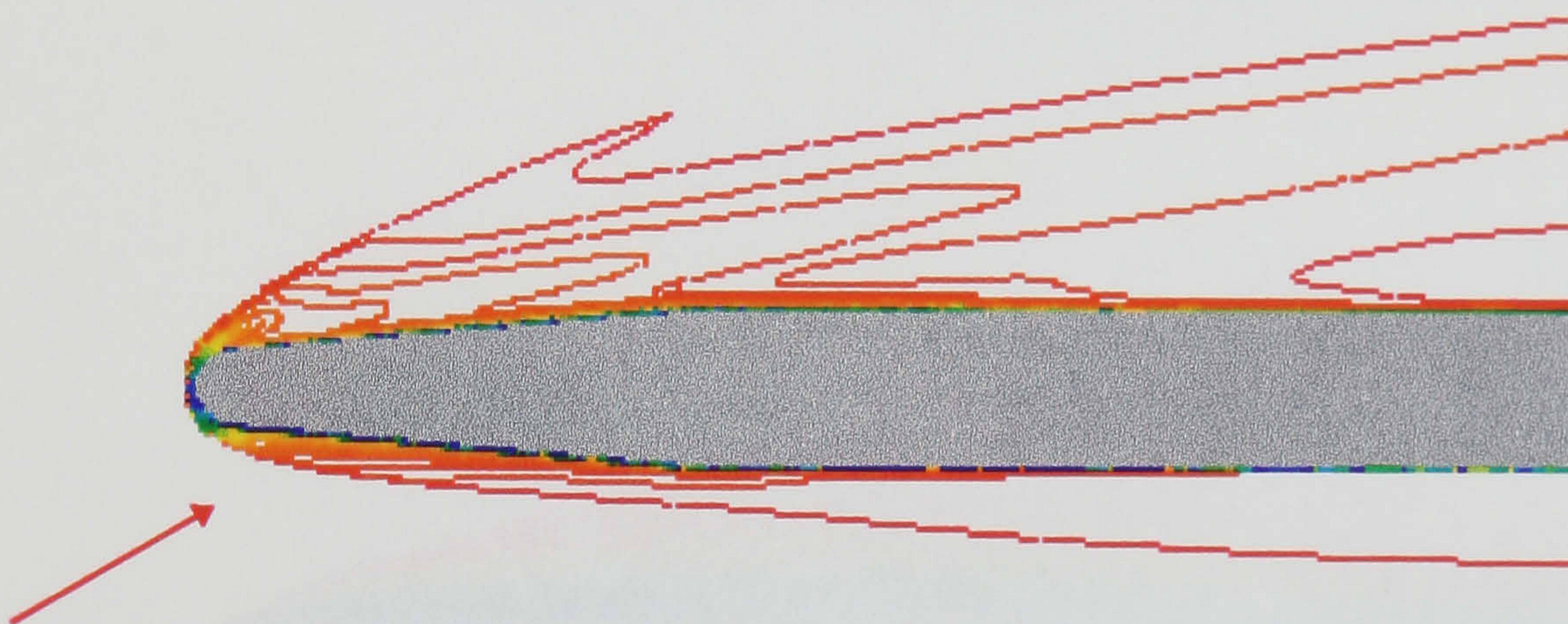


Figure 11. Density, Velocity, Entropy and Stagnation pressure contours for the hemi-spherically blunted cone-cylinder at $\alpha = 0^\circ$.

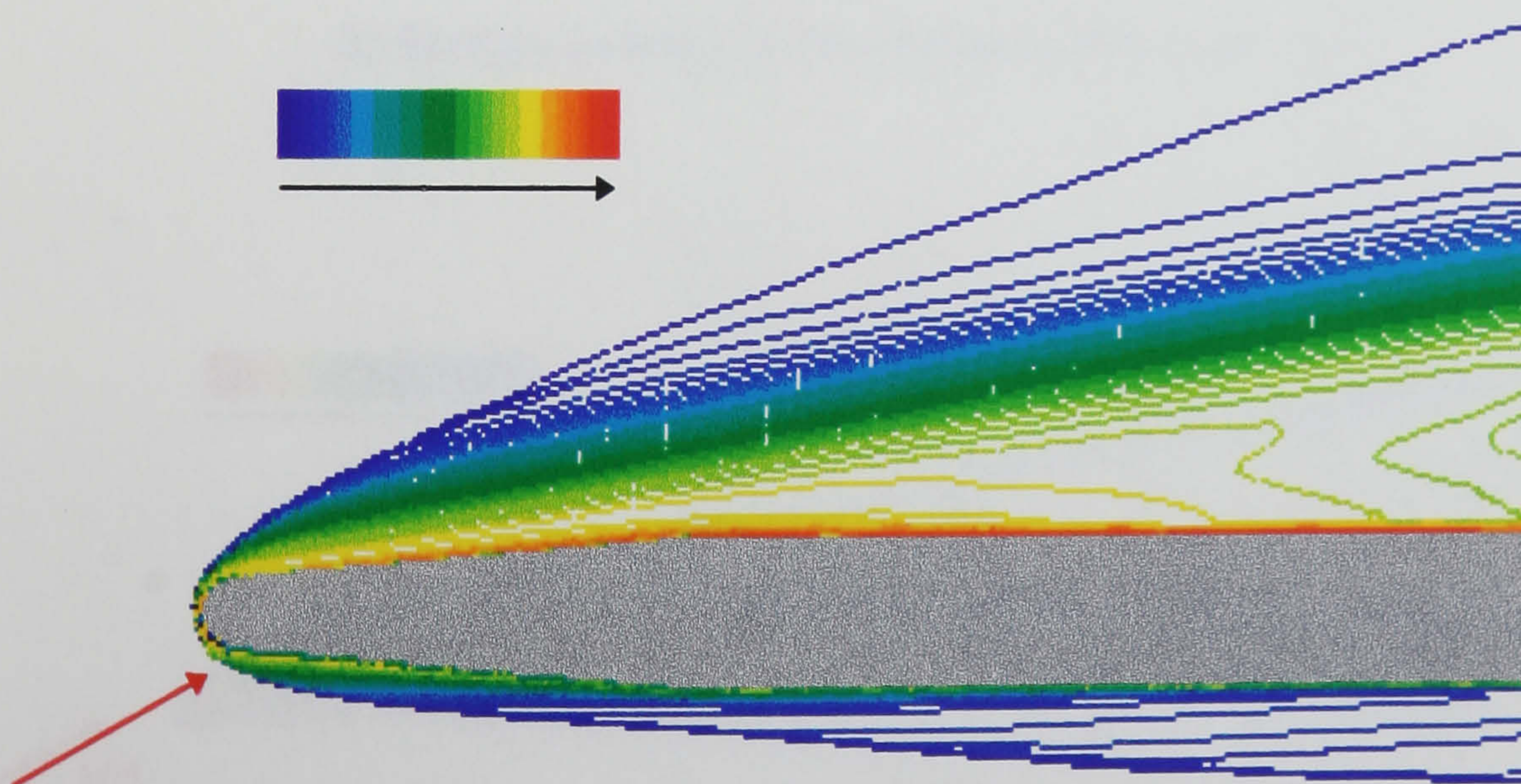
$M_\infty = 8.2$ and $Re_\infty = 9.35 \times 10^4/cm$



(a) Density contours

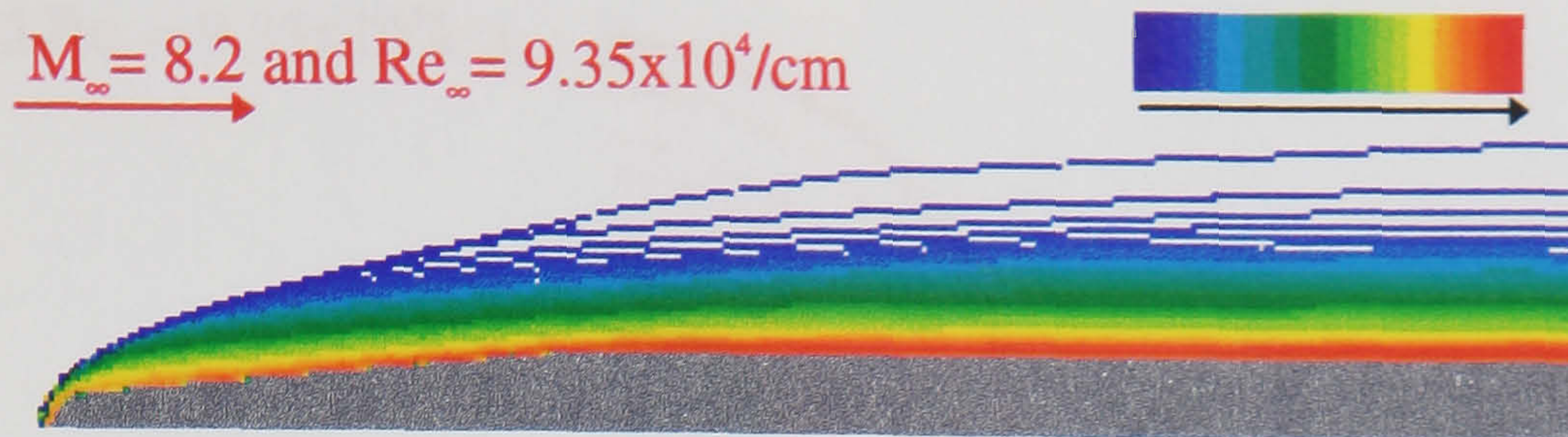


(b) Velocity contours

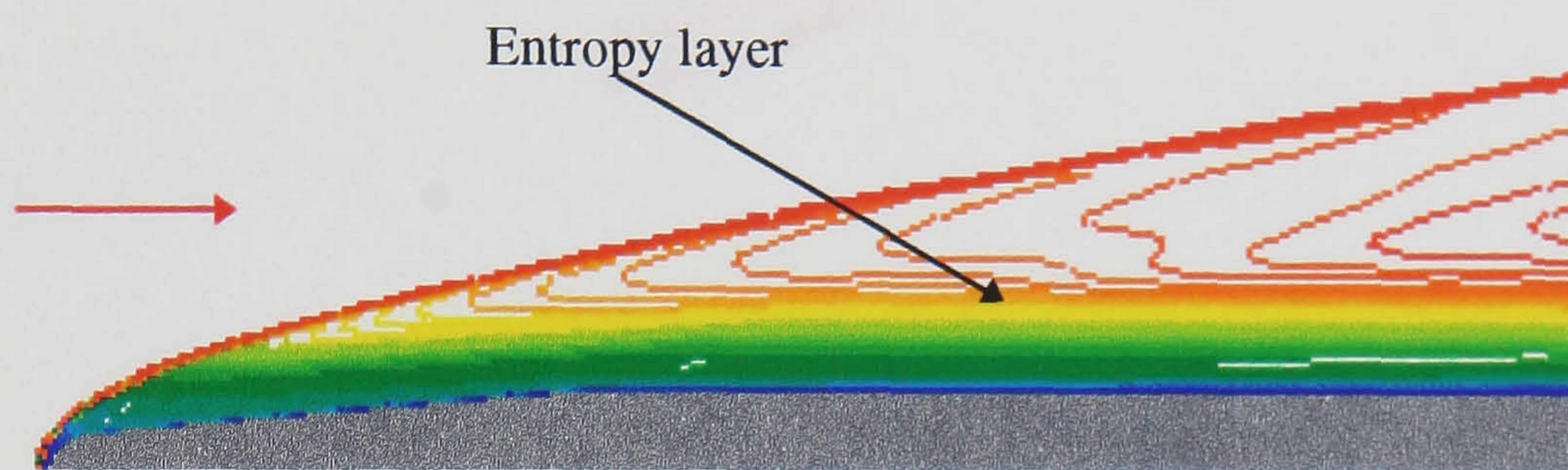


(c) Entropy contours

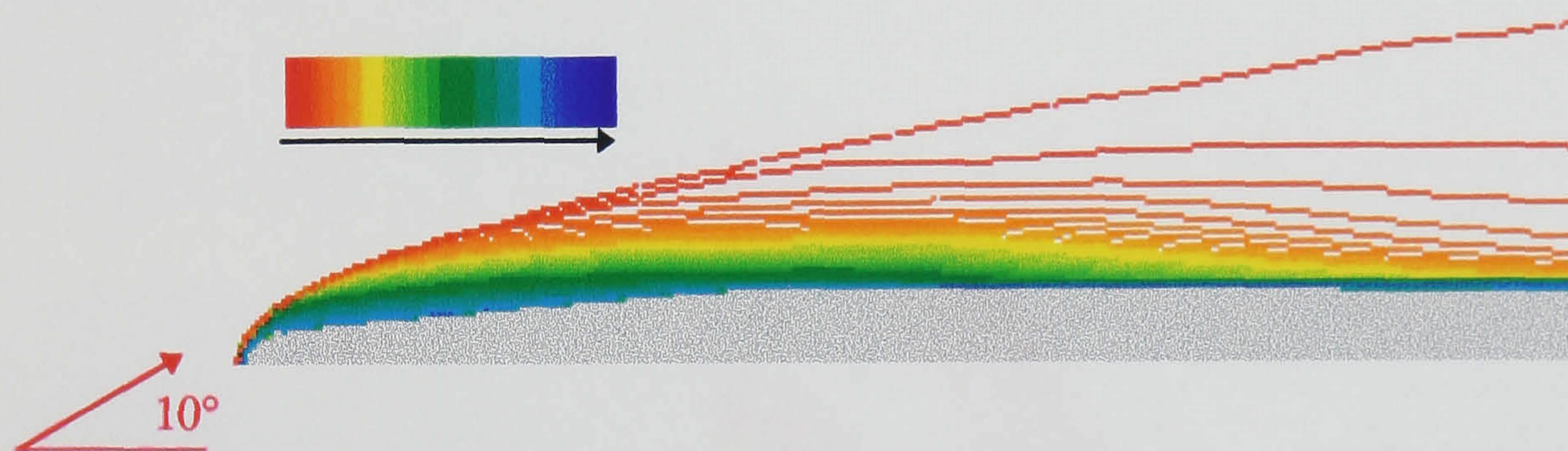
Figure 12. Density, Velocity, and Entropy contours for the hemi-spherically blunted cone-cylinder at $\alpha = 10^\circ$.



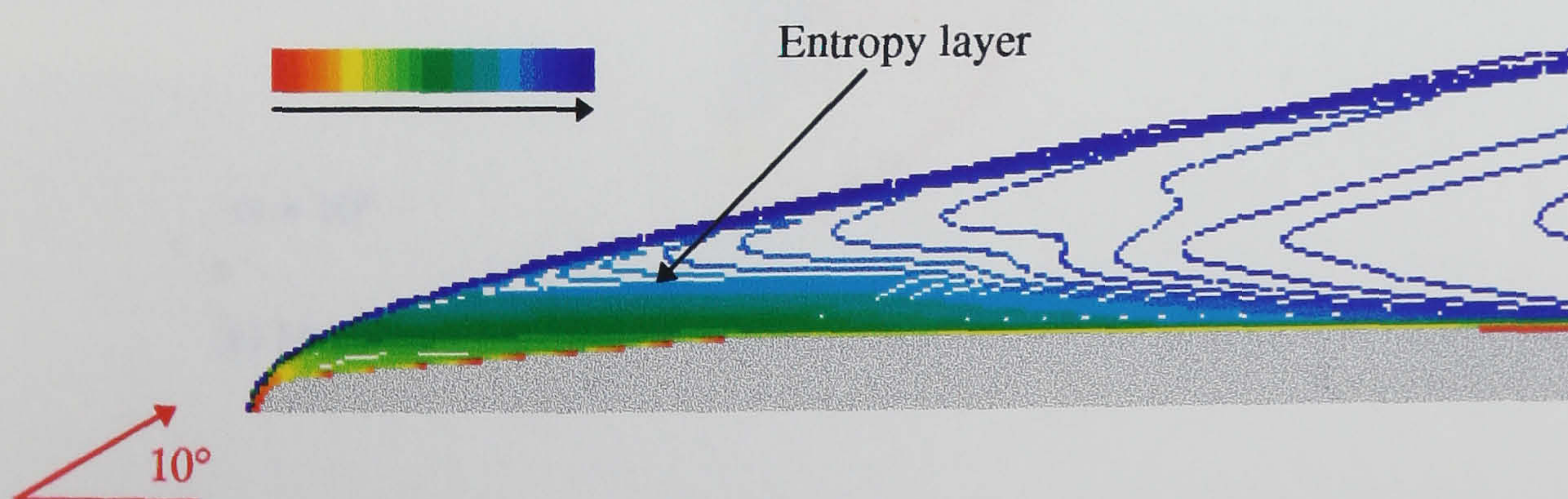
(a) Entropy contours in the horizontal plane at $\alpha = 0^\circ$.



(b) Mach number contours in the horizontal plane at $\alpha = 0^\circ$.



(c) Entropy contours in the horizontal plane at $\alpha = 10^\circ$.

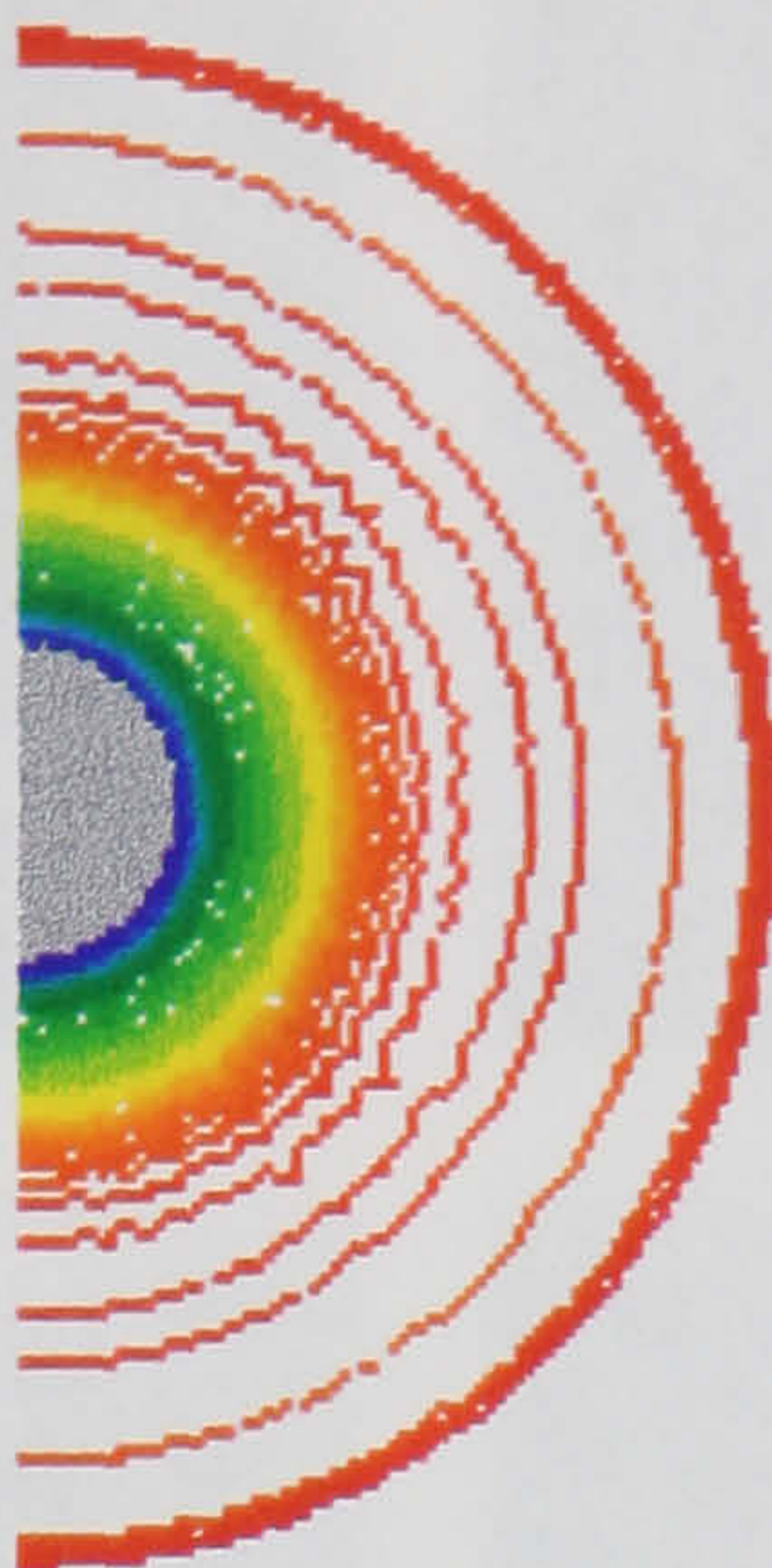


(d) Mach number contours in the horizontal plane at $\alpha = 10^\circ$.

Figure 13(a-d). A comparison of the Entropy and Mach number contours for the hemi-spherically blunted cone-cylinder at $\alpha = 0$ and 10° .

$M_\infty = 8.2$ and $Re_\infty = 9.35 \times 10^4/cm$

$\alpha = 0^\circ$

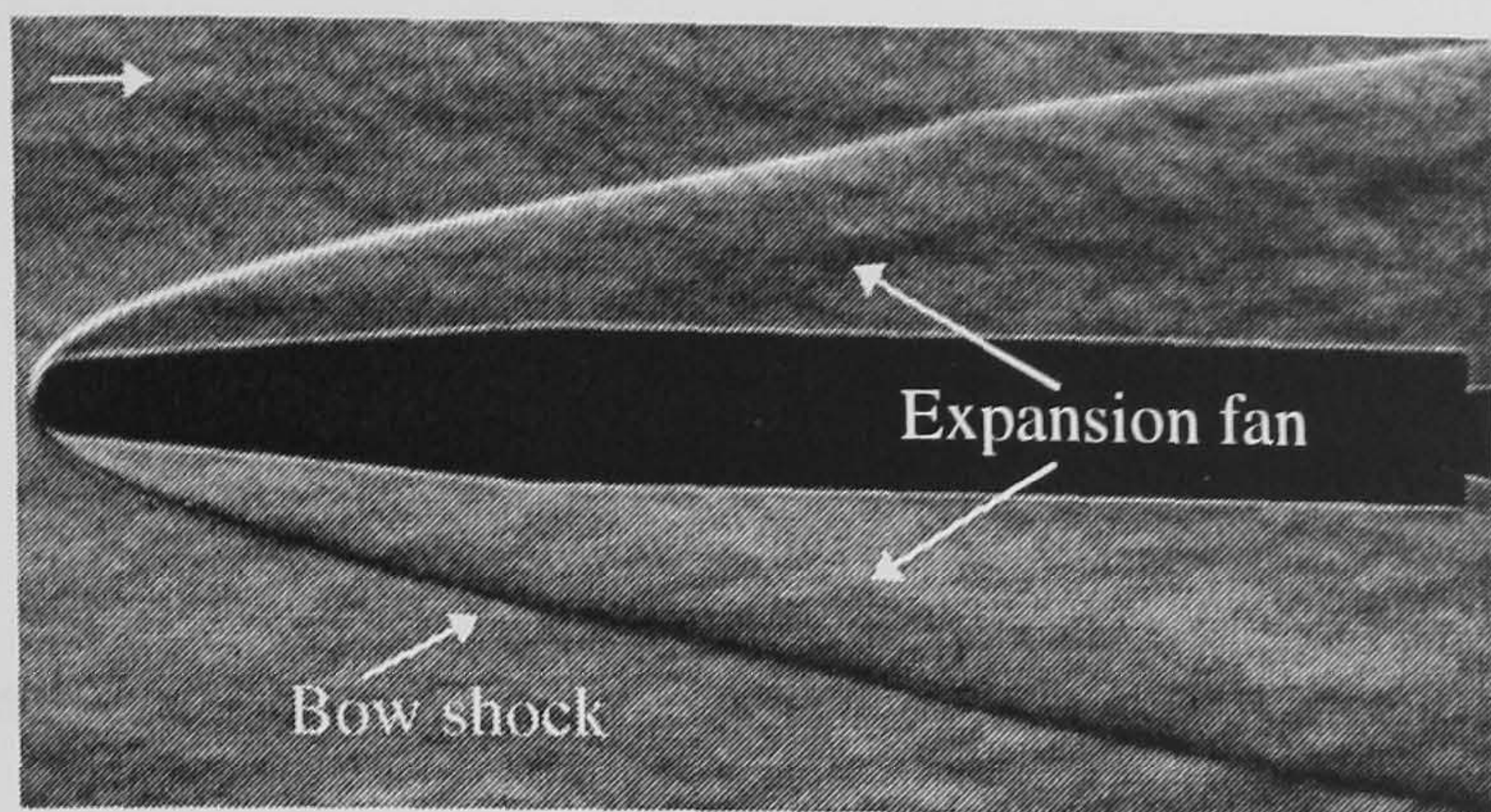


$\alpha = 10^\circ$



(e) Mach number contours in the cross-flow plane at the cone-cylinder base

Figure 13(e). A comparison of the Entropy and Mach number contours for the hemi-spherically blunted cone-cylinder at $\alpha = 0$ and 10° .



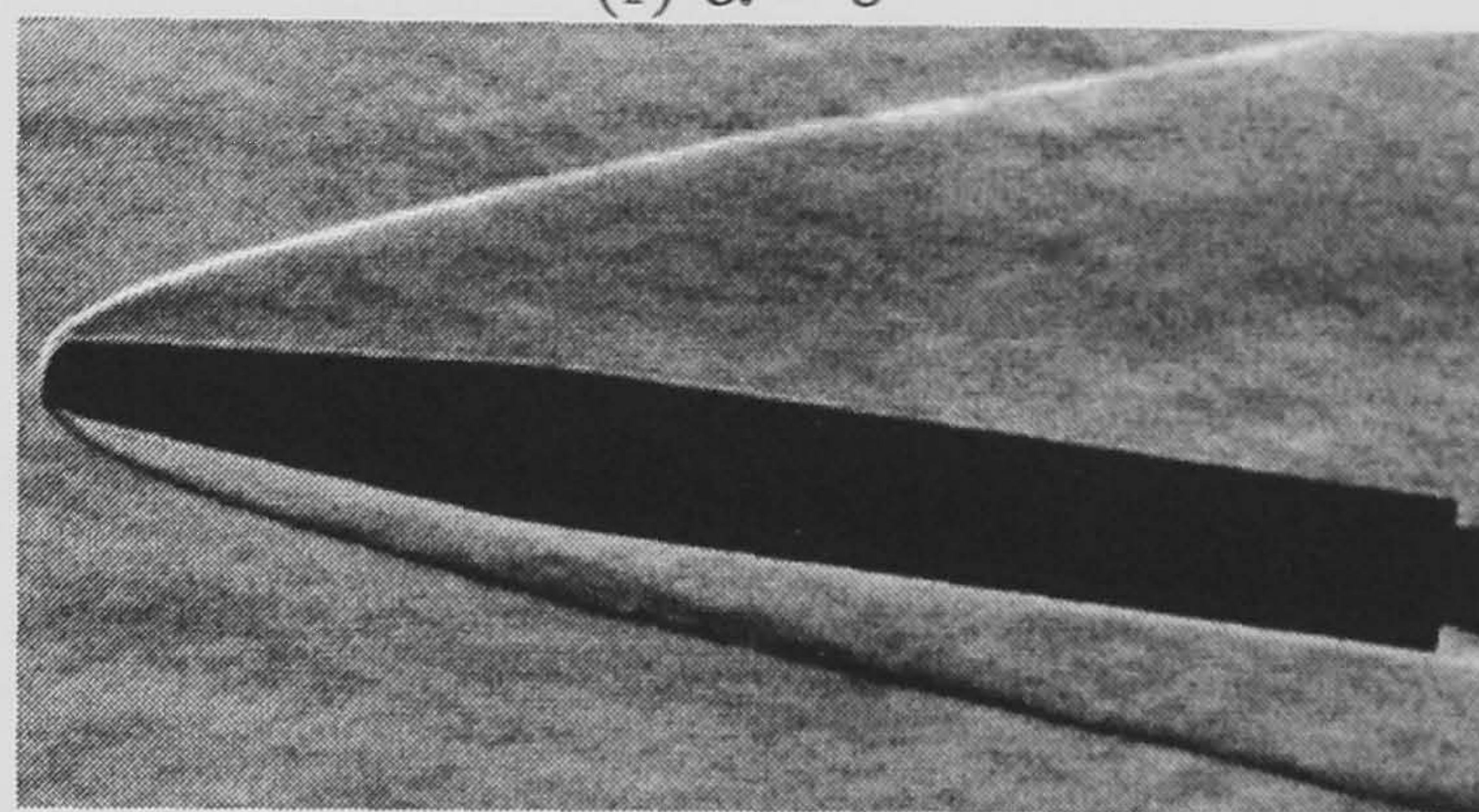
(a) $\alpha = 0^\circ$



(f) $\alpha = 6^\circ$



(b) $\alpha = 2^\circ$



(g) $\alpha = 7^\circ$



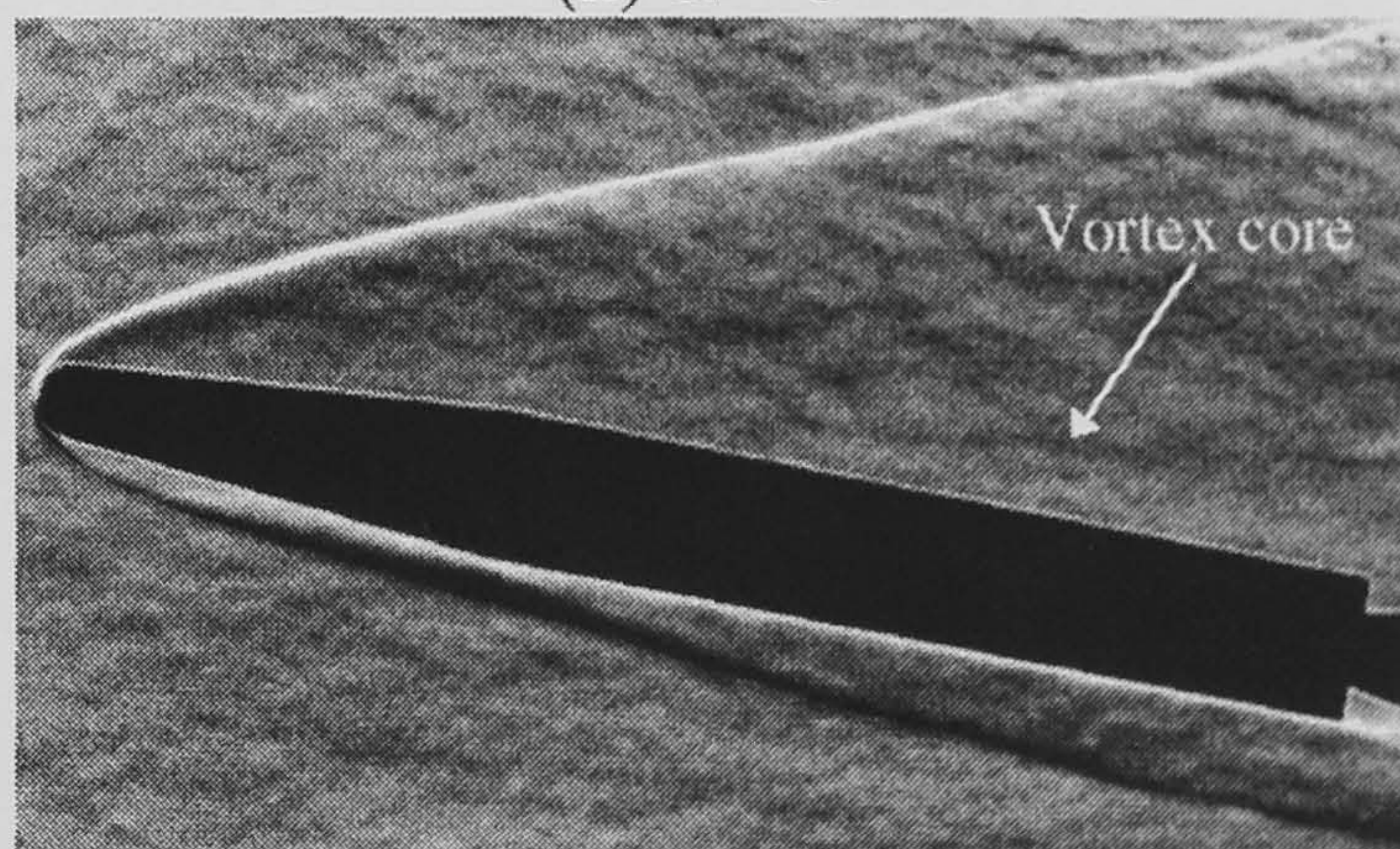
(c) $\alpha = 3^\circ$



(h) $\alpha = 8^\circ$



(d) $\alpha = 4^\circ$

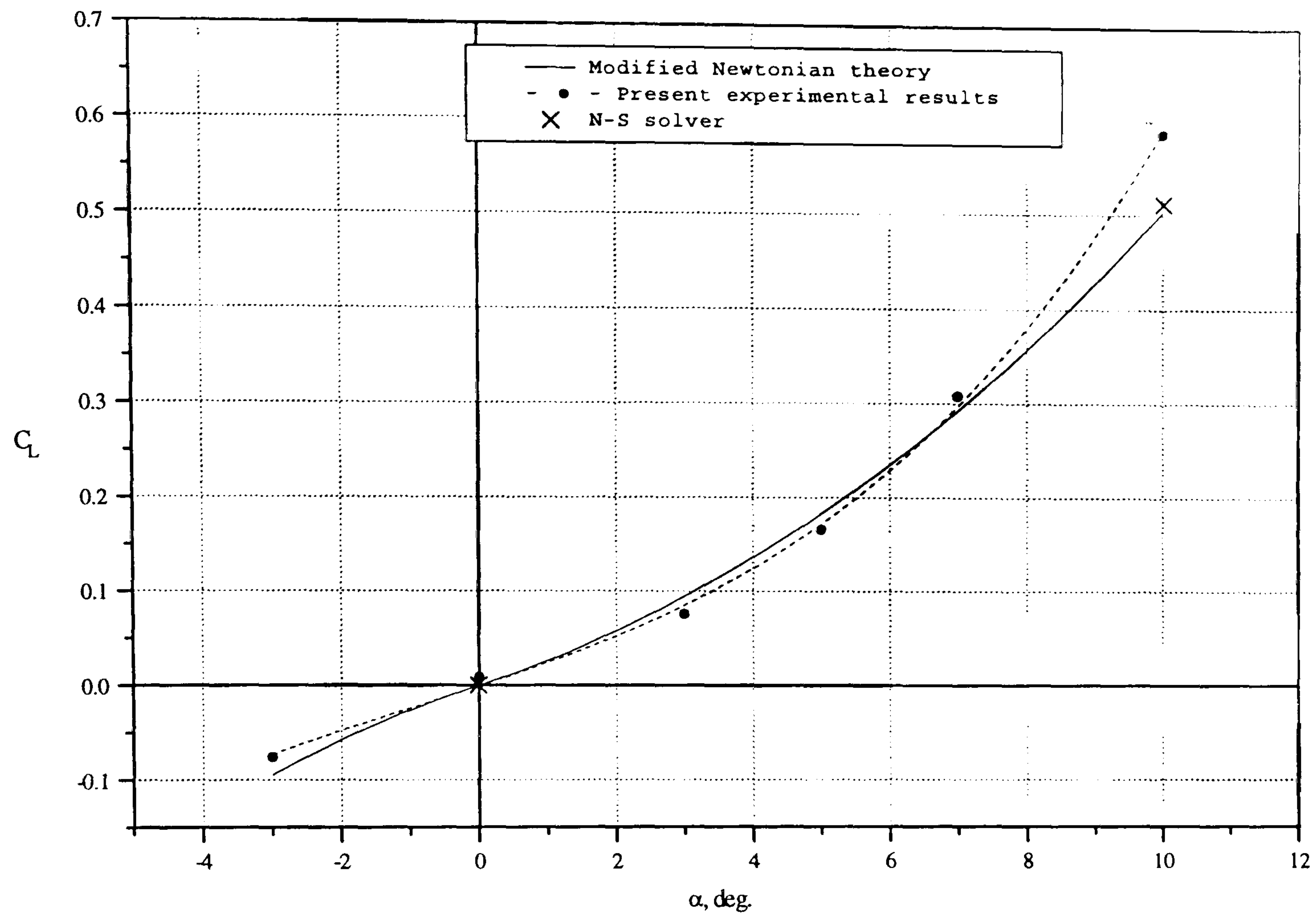


(i) $\alpha = 10^\circ$

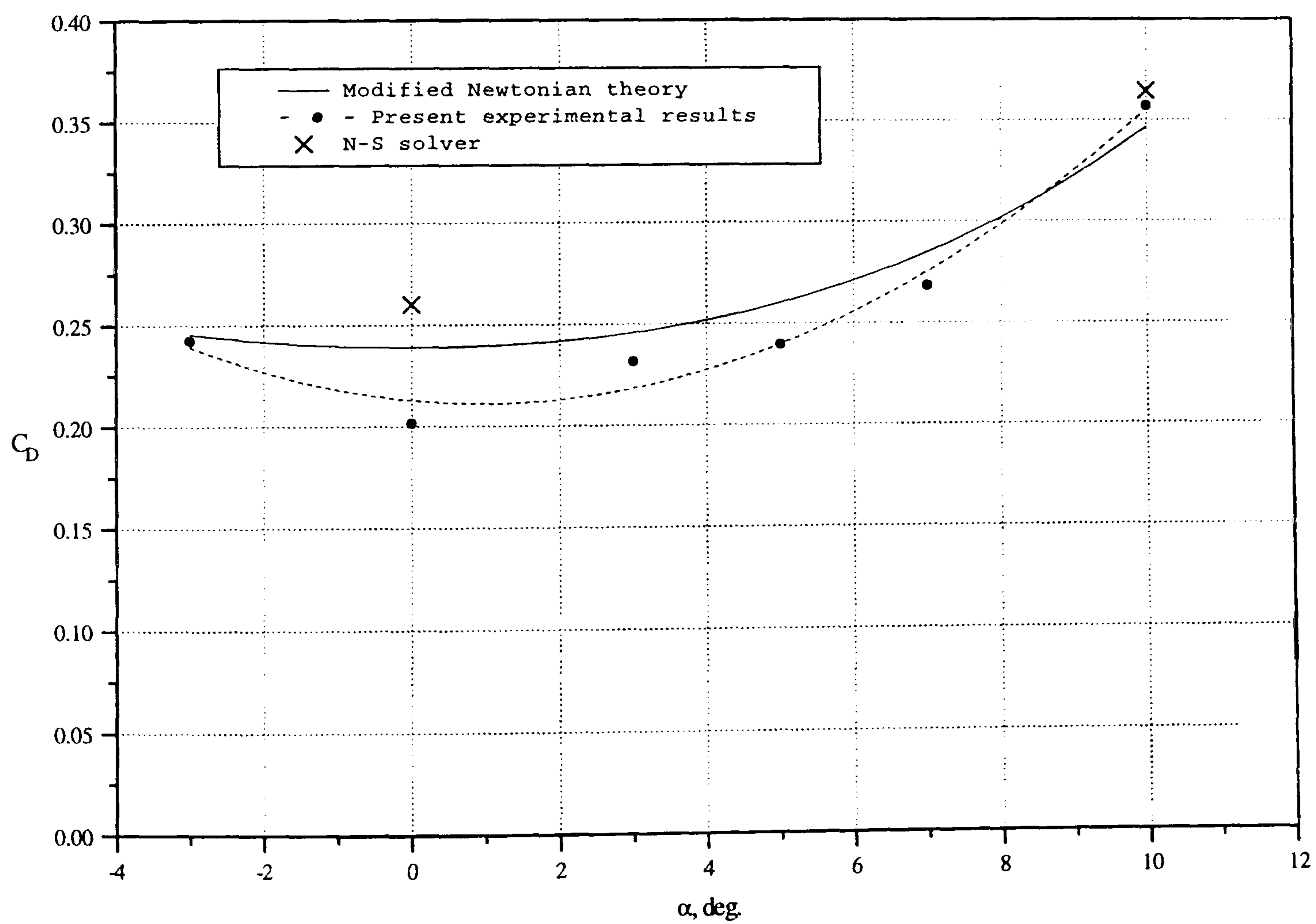


(e) $\alpha = 5^\circ$

Figure 14. Schlieren pictures showing change in shape of the bow shock generated by hemi-spherically blunted cone-cylinder body with increase in α from 0° to 10° .

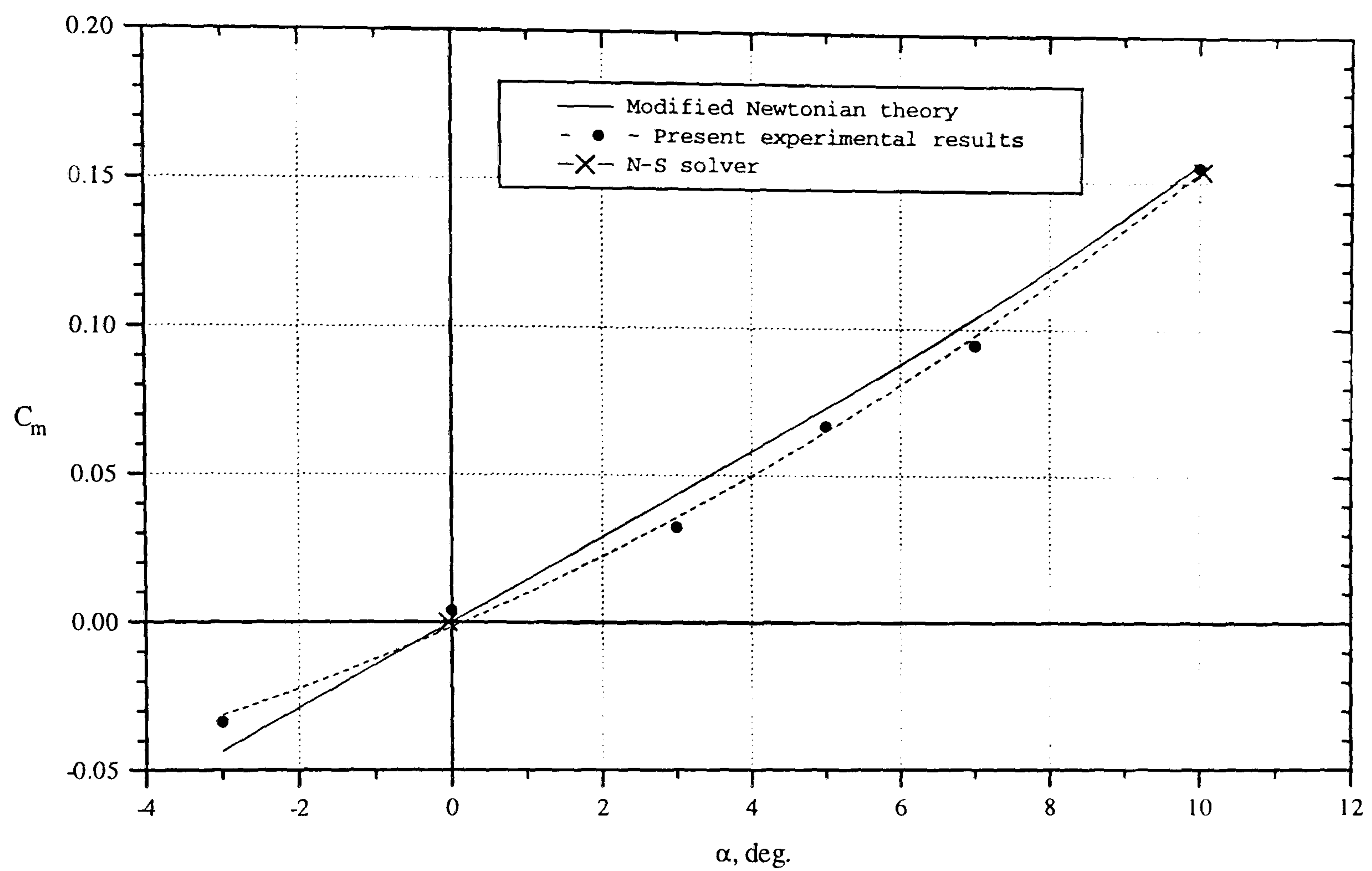


(a) Lift coefficient

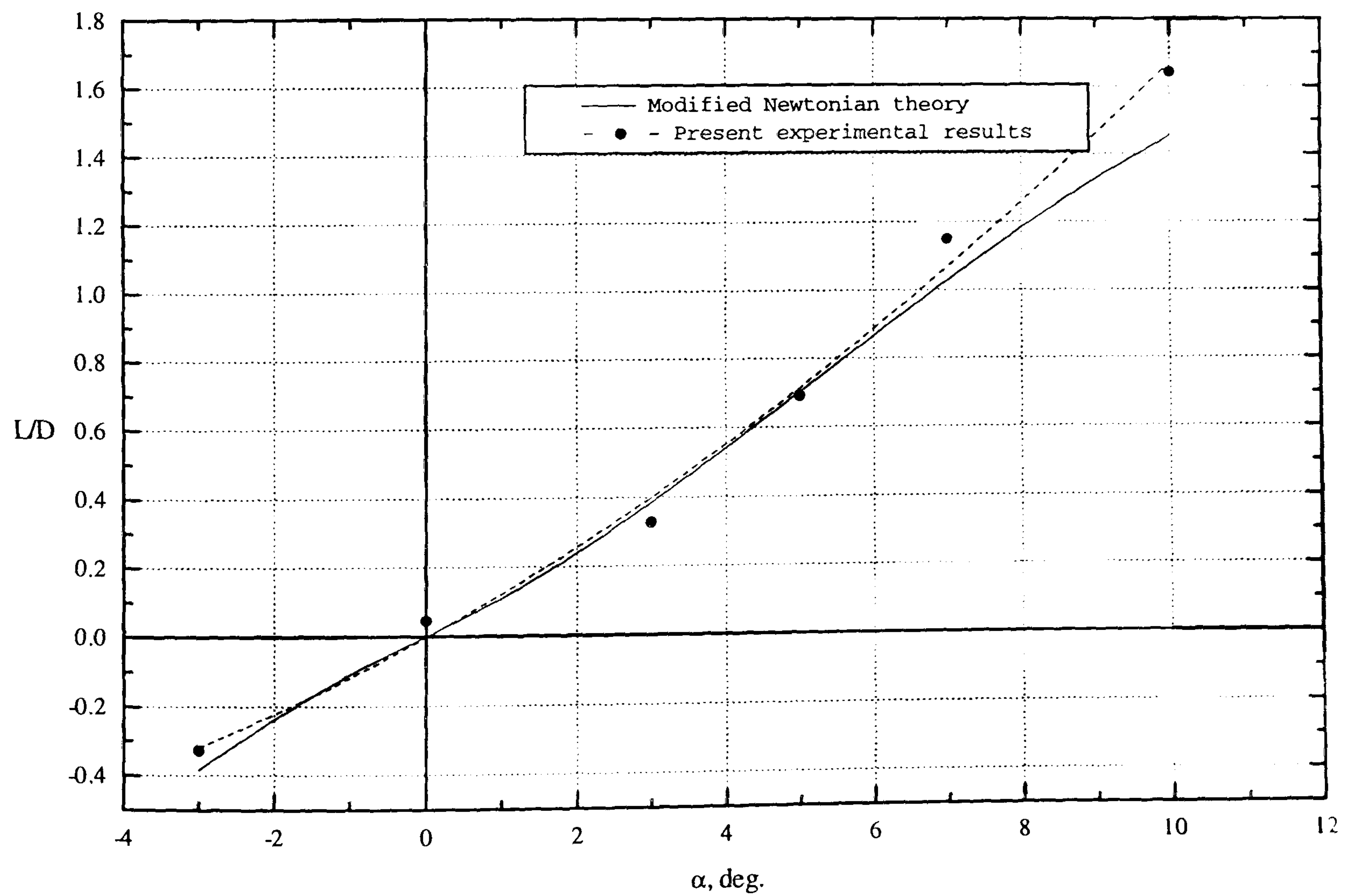


(b) Drag coefficient

Figure 15. A comparison of experimental and theoretical aerodynamic characteristics of hemi-spherically blunted cone-cylinder body.



(c) Pitching moment coefficient



(d) Lift to drag ratio
Figure 15. Continued

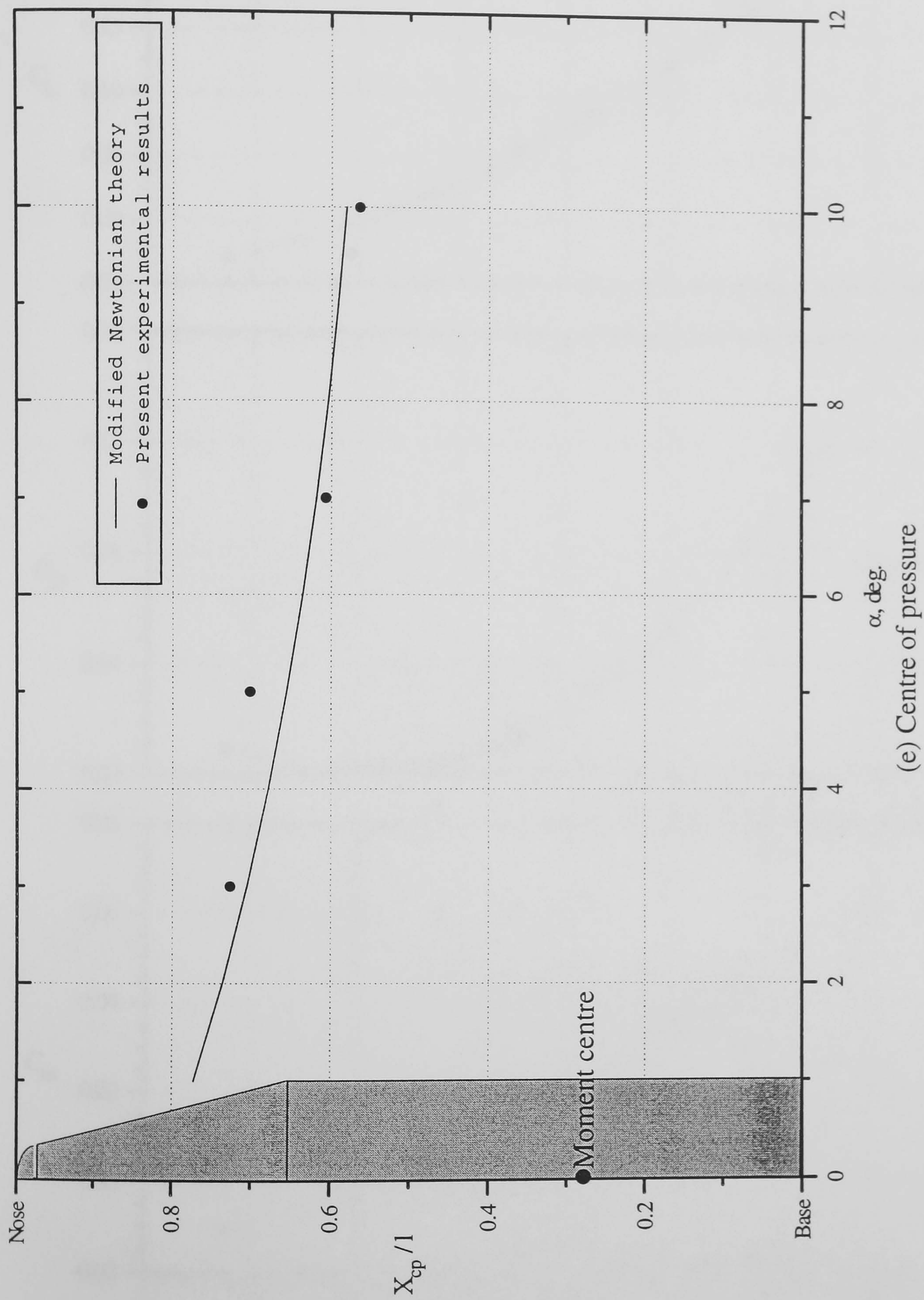


Figure 15. Continued

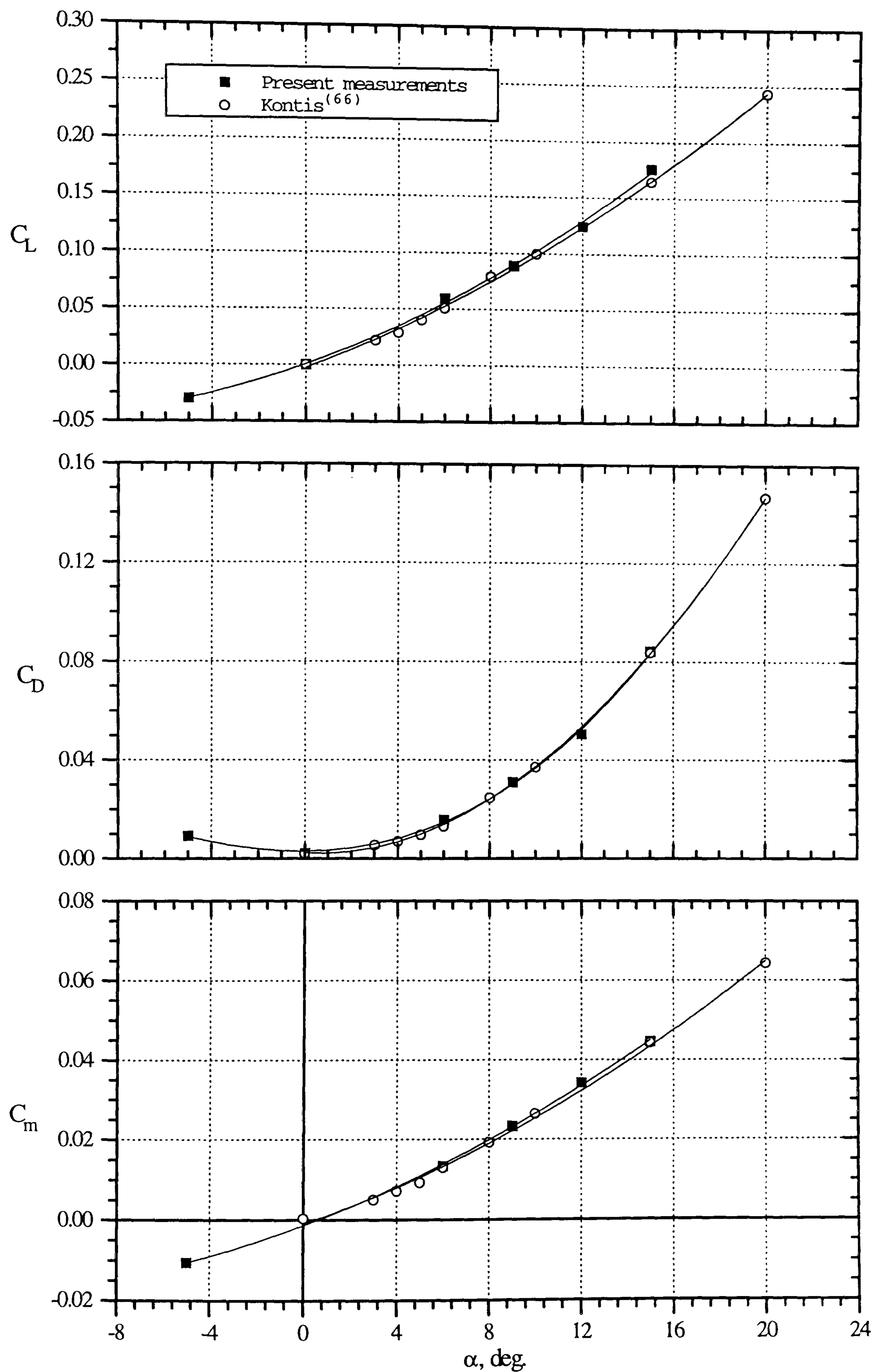


Figure 16(a). Aerodynamic characteristics of the one-half power-law body without strakes.

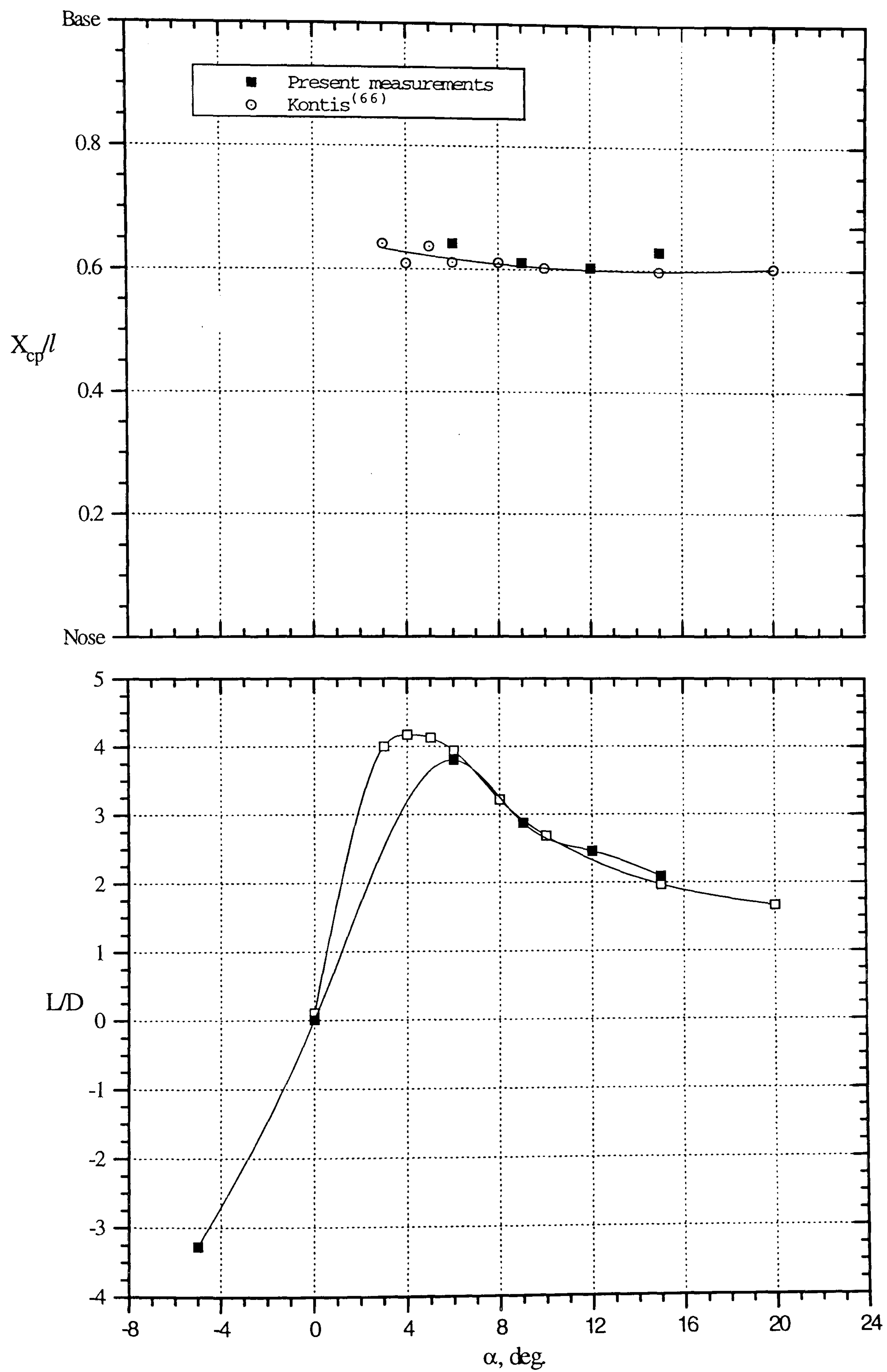


Figure 16(a). Continued

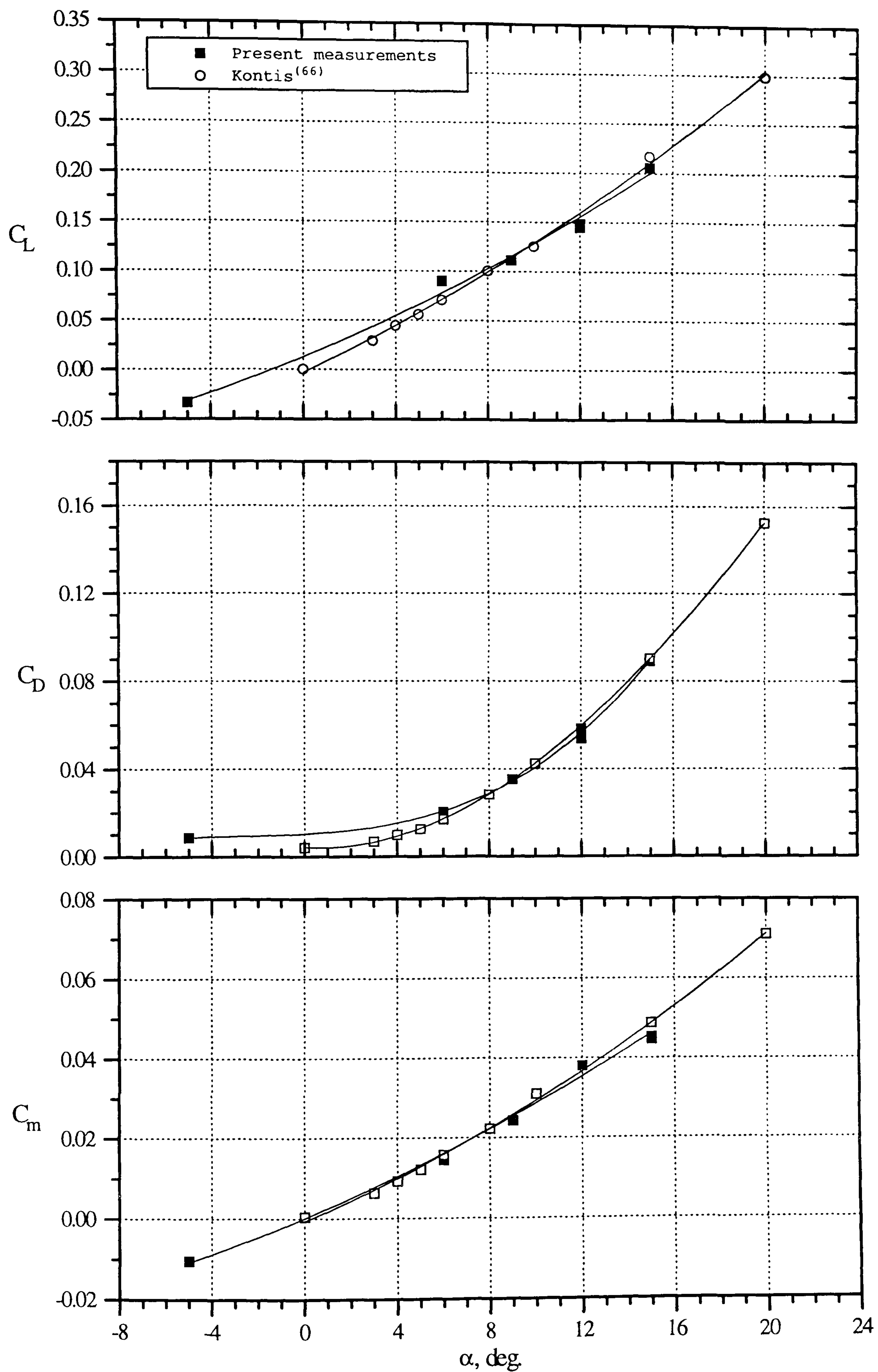


Figure 16(b). Aerodynamic characteristics of the one-half power-law body with strakes.

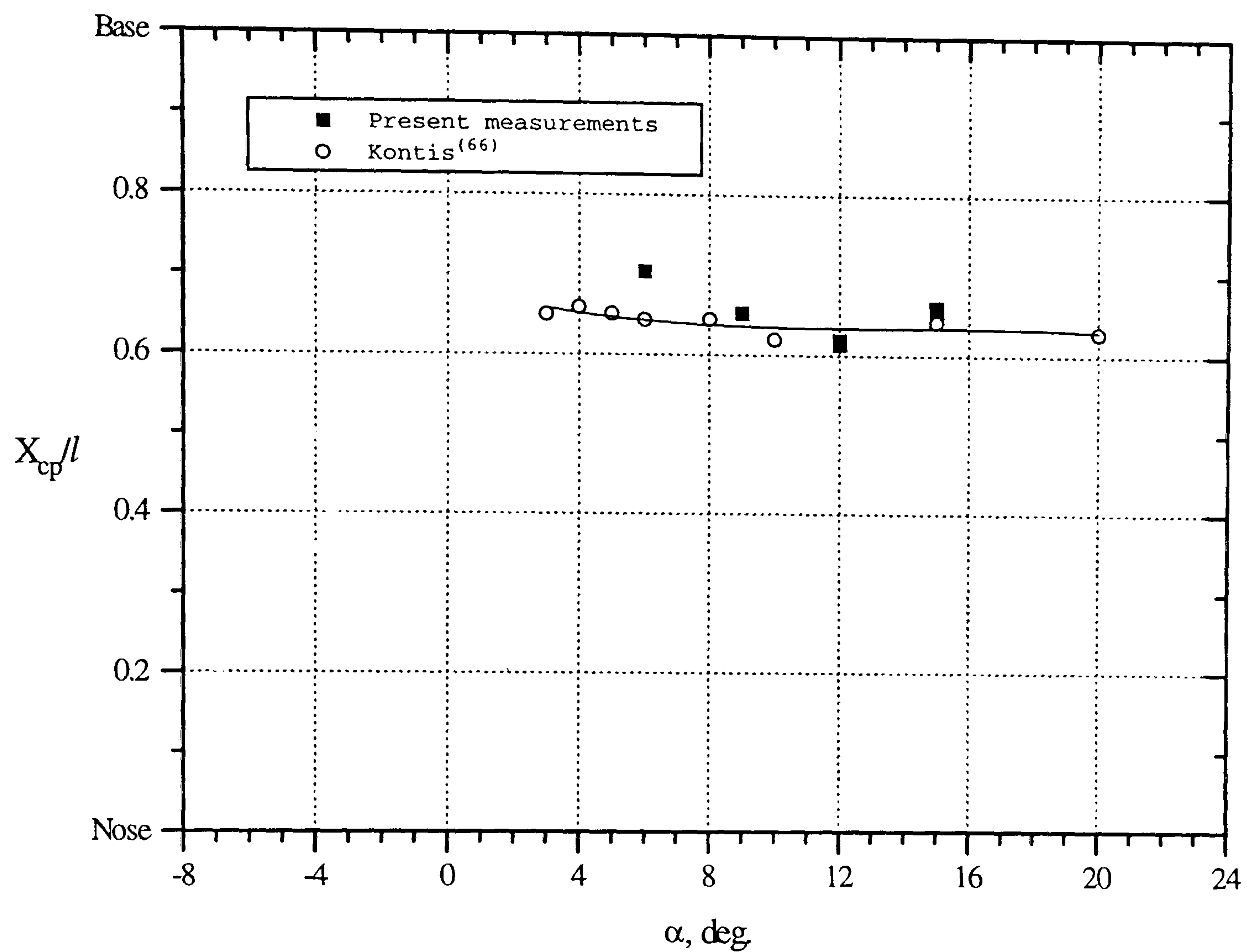
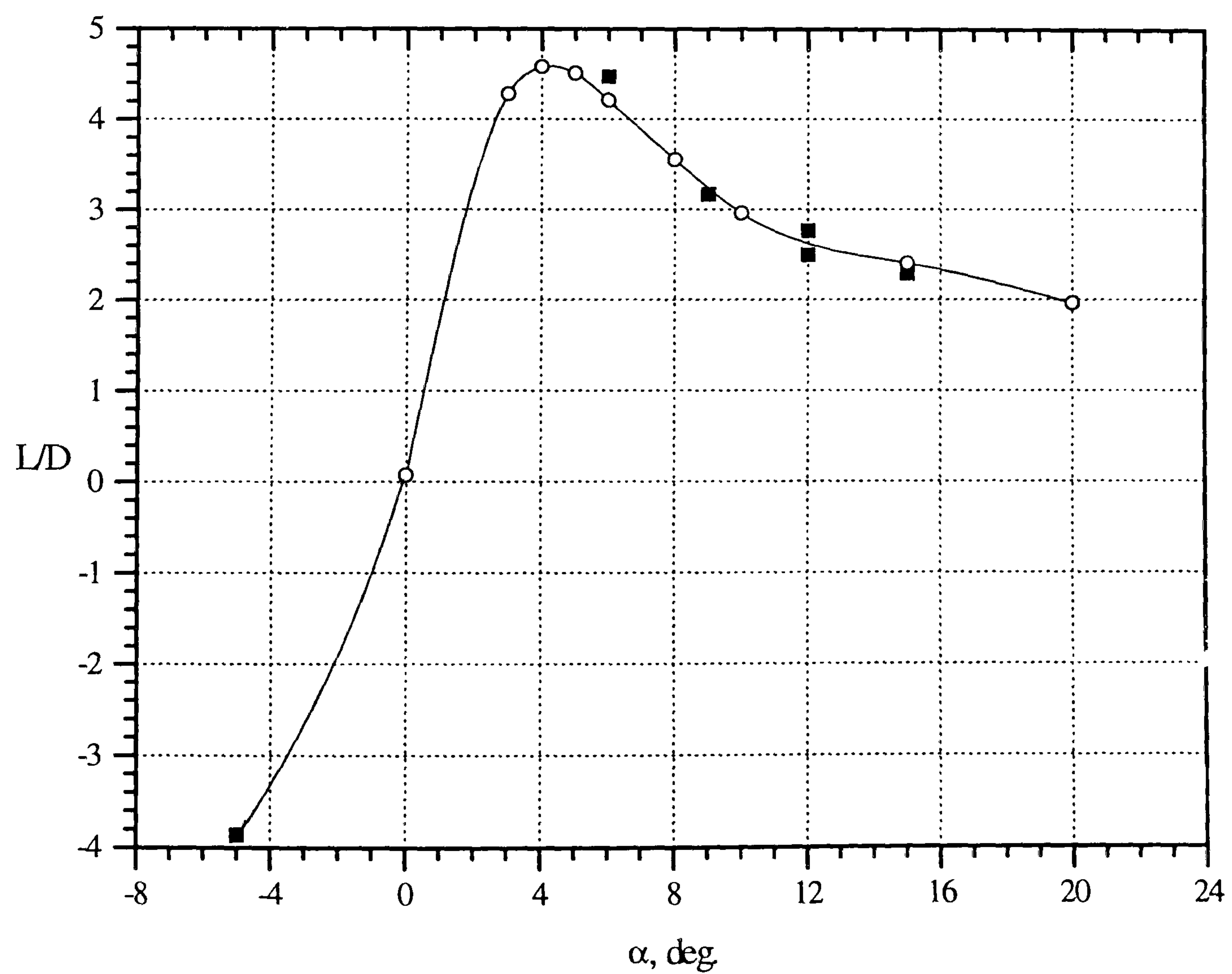
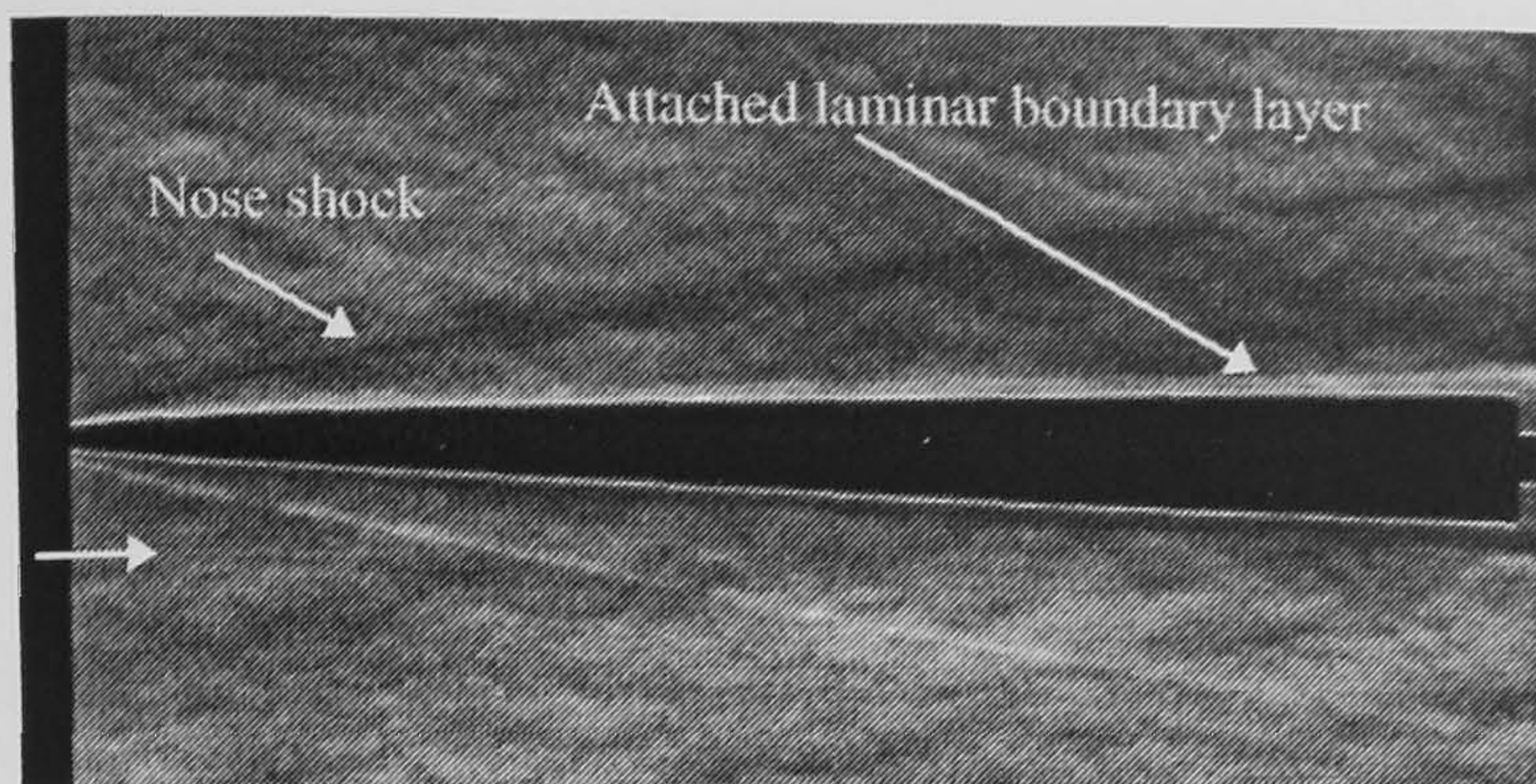
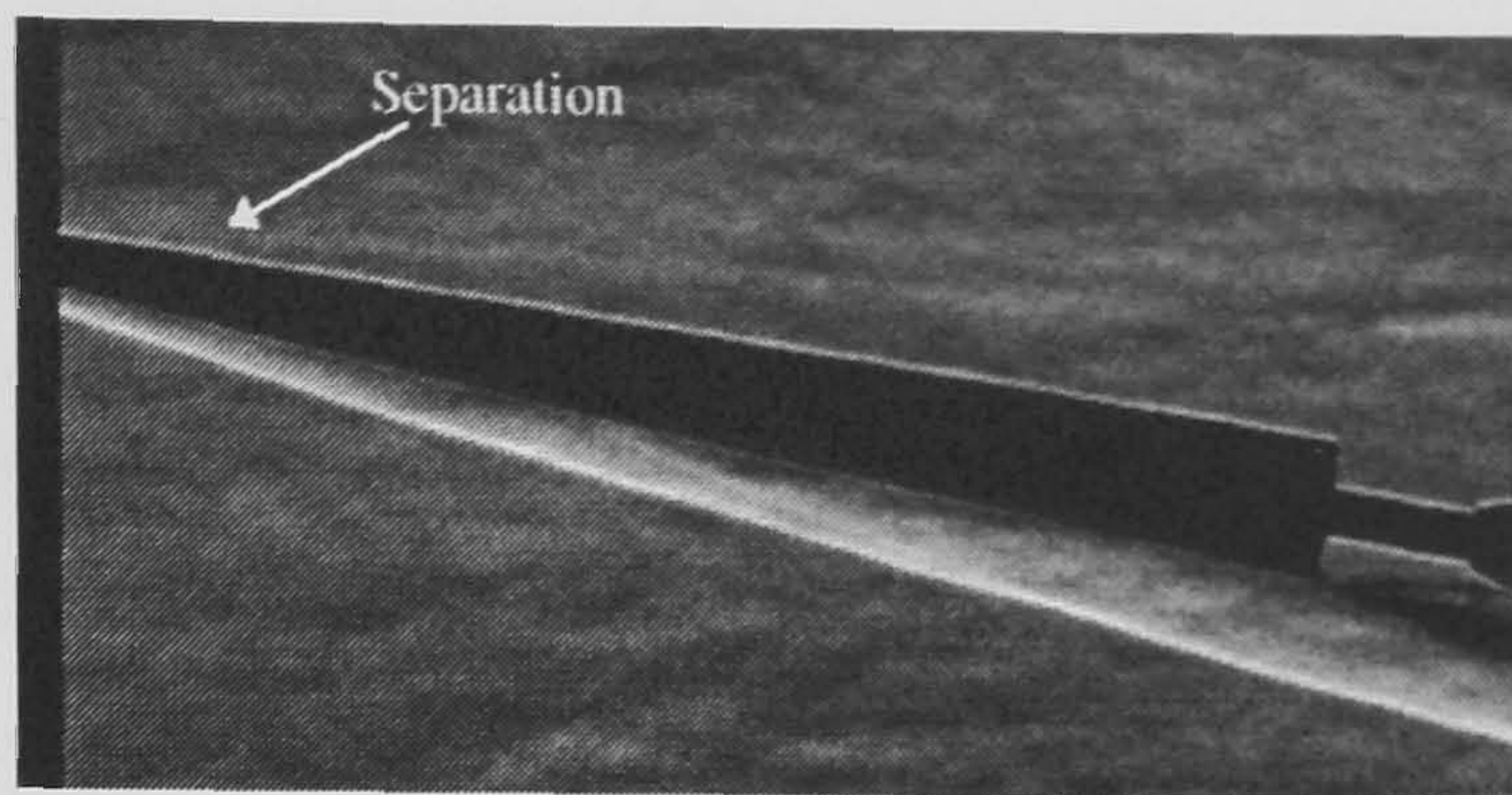


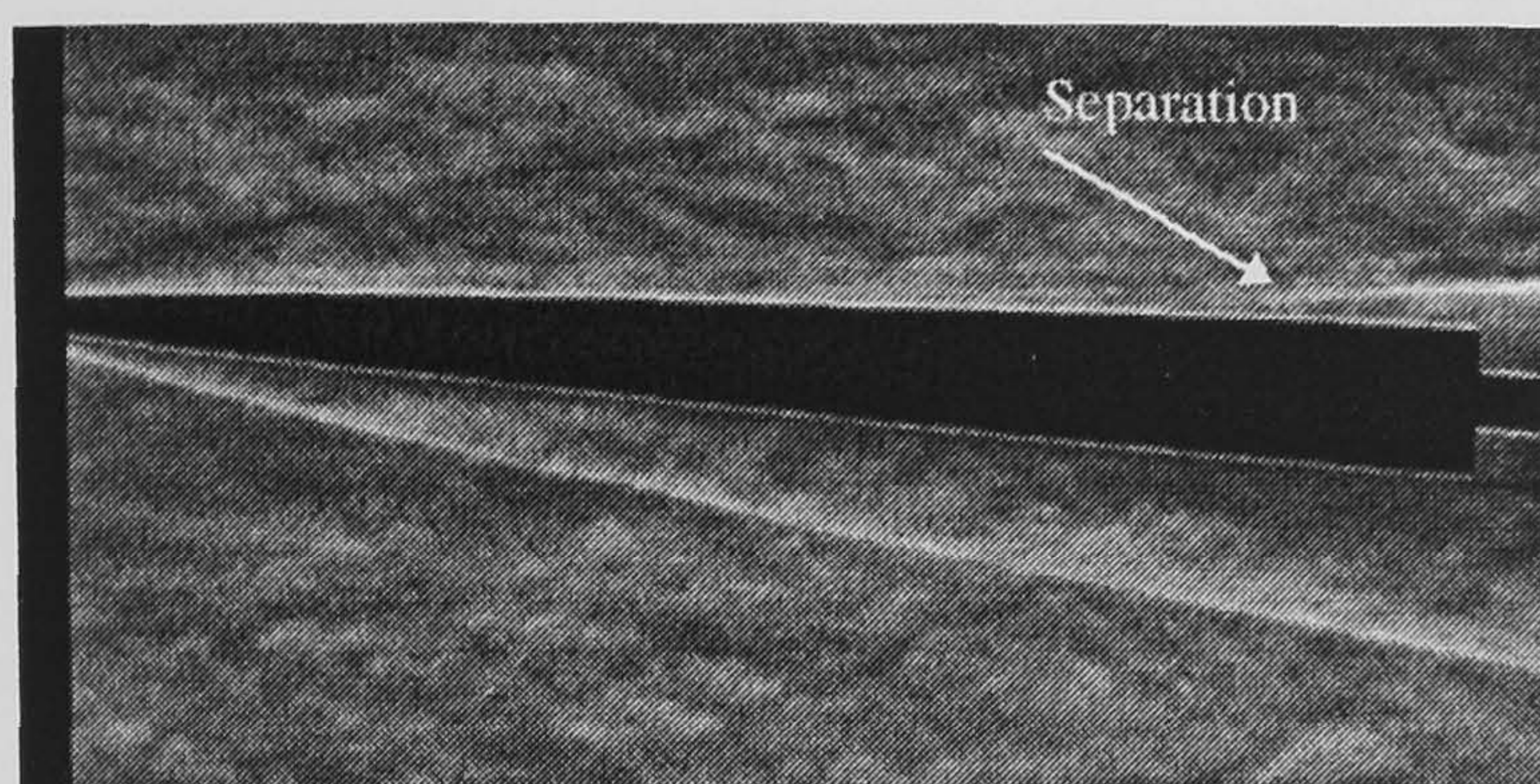
Figure 16(b). Continued



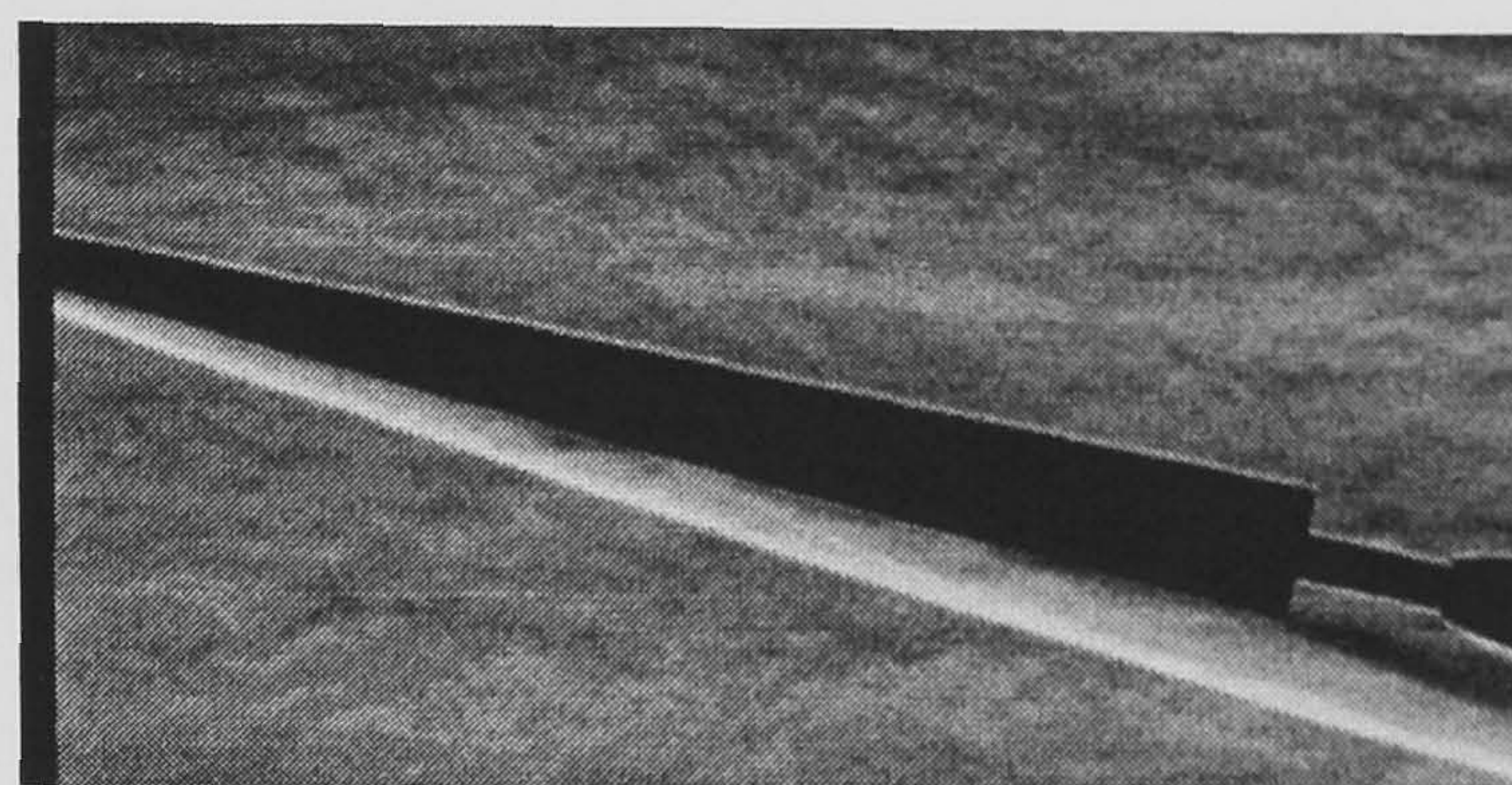
(a) $\alpha = 0^\circ$



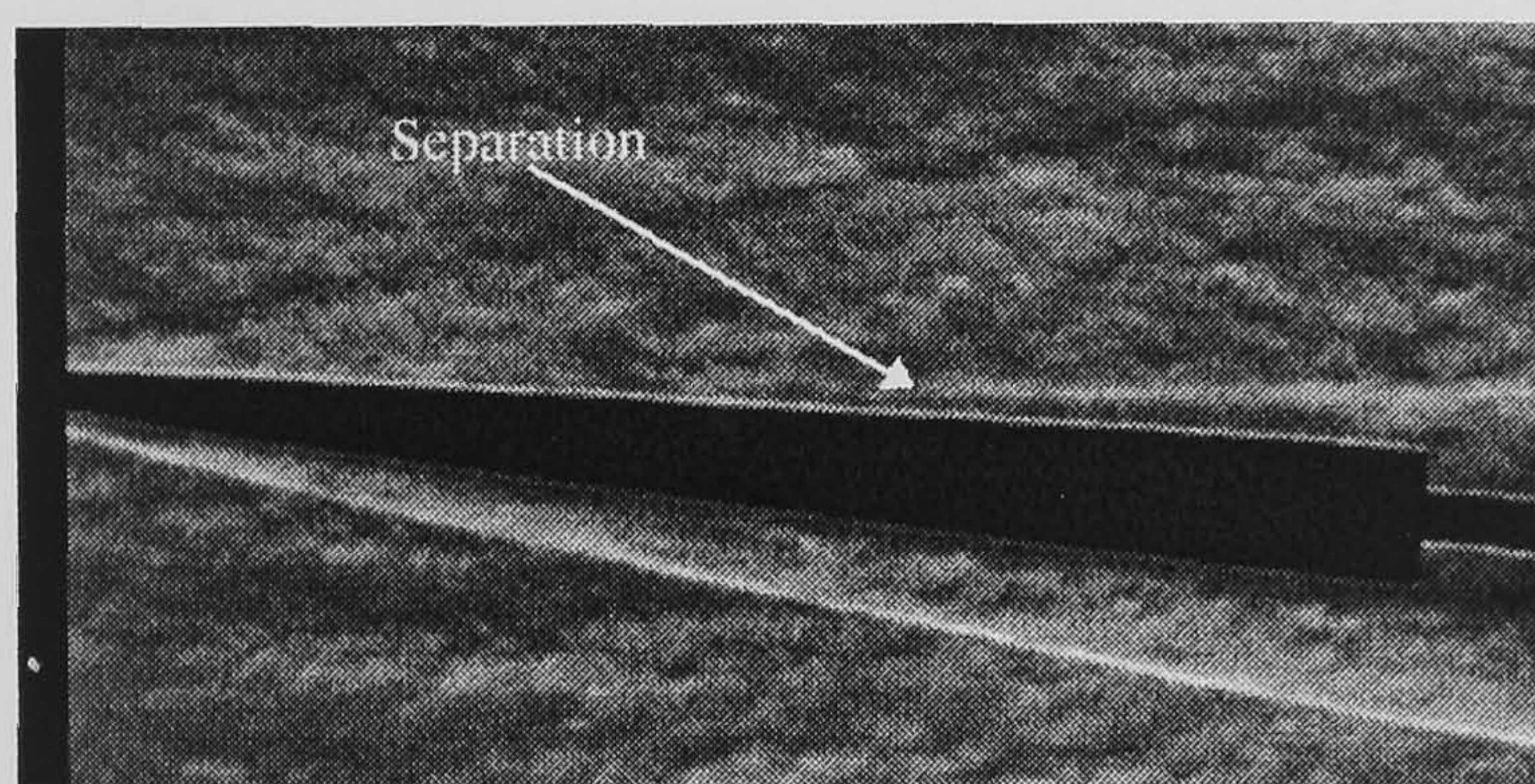
(e) $\alpha = 9^\circ$



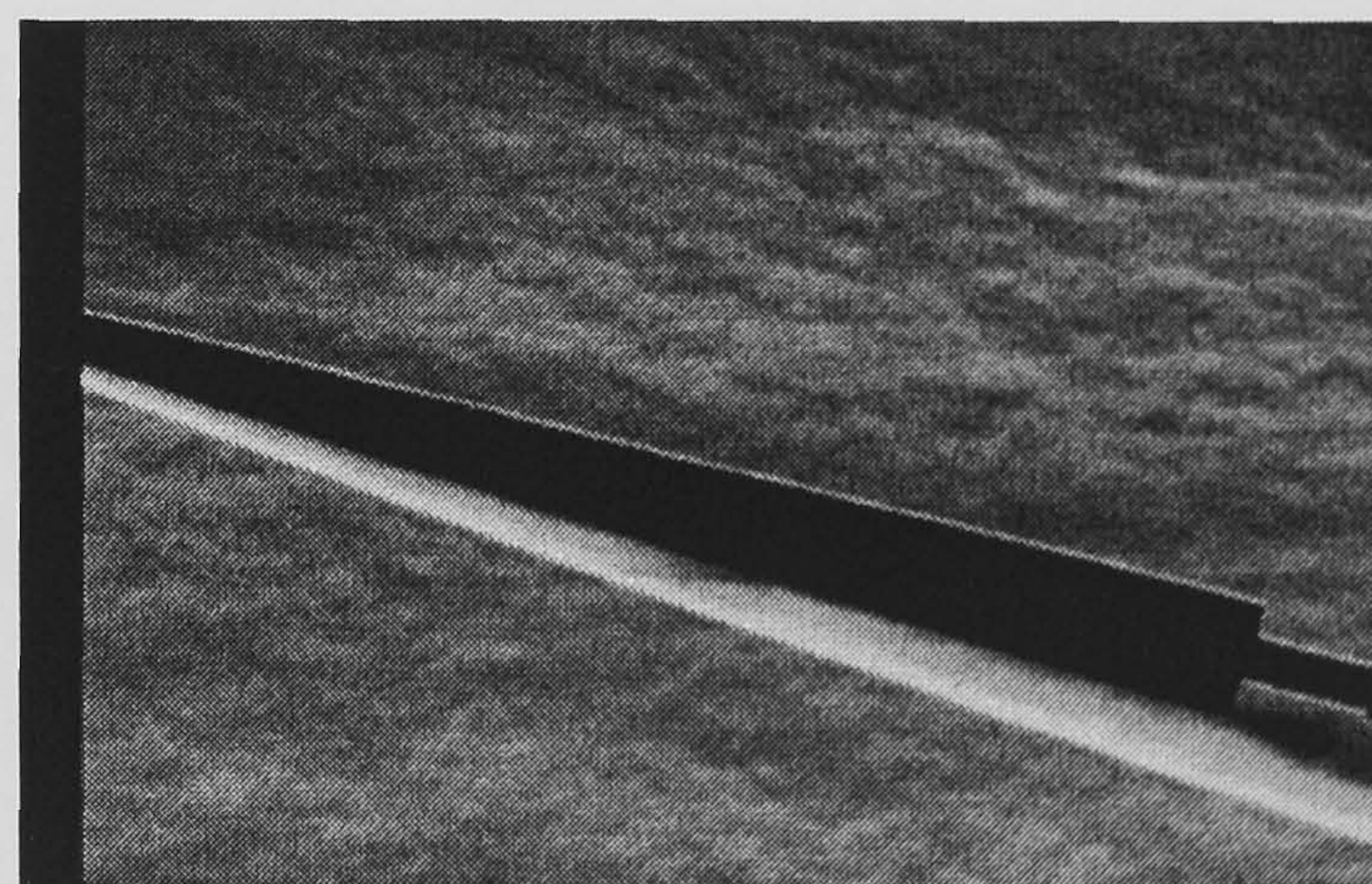
(b) $\alpha = 3^\circ$



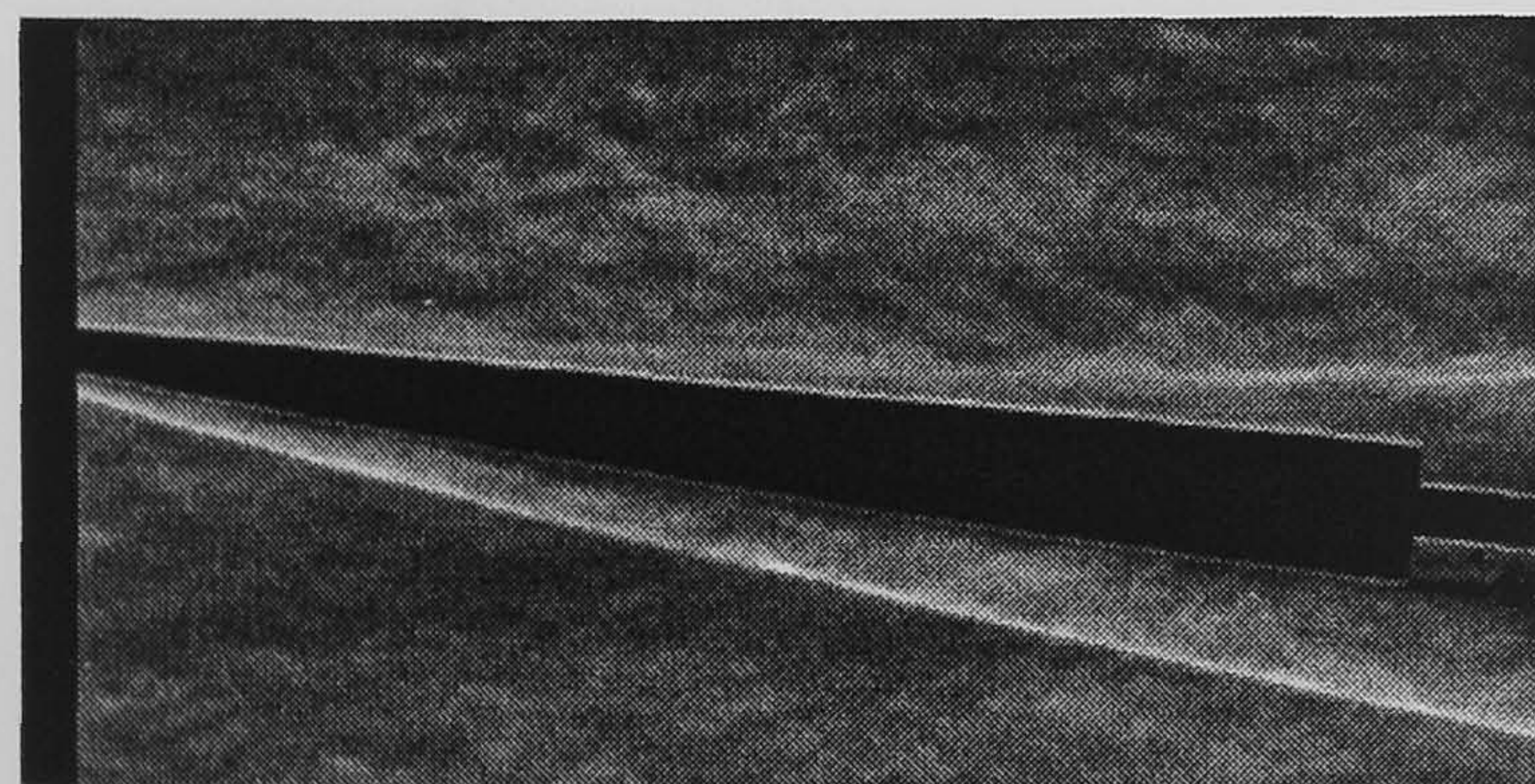
(f) $\alpha = 12^\circ$



(c) $\alpha = 5^\circ$



(g) $\alpha = 15^\circ$



(d) $\alpha = 6^\circ$

The flow is separated on the leeward side at an incidence of 3° and spreads further upstream with increasing incidence.

Figure 17. Schlieren photographs of power-law body with and without strakes.

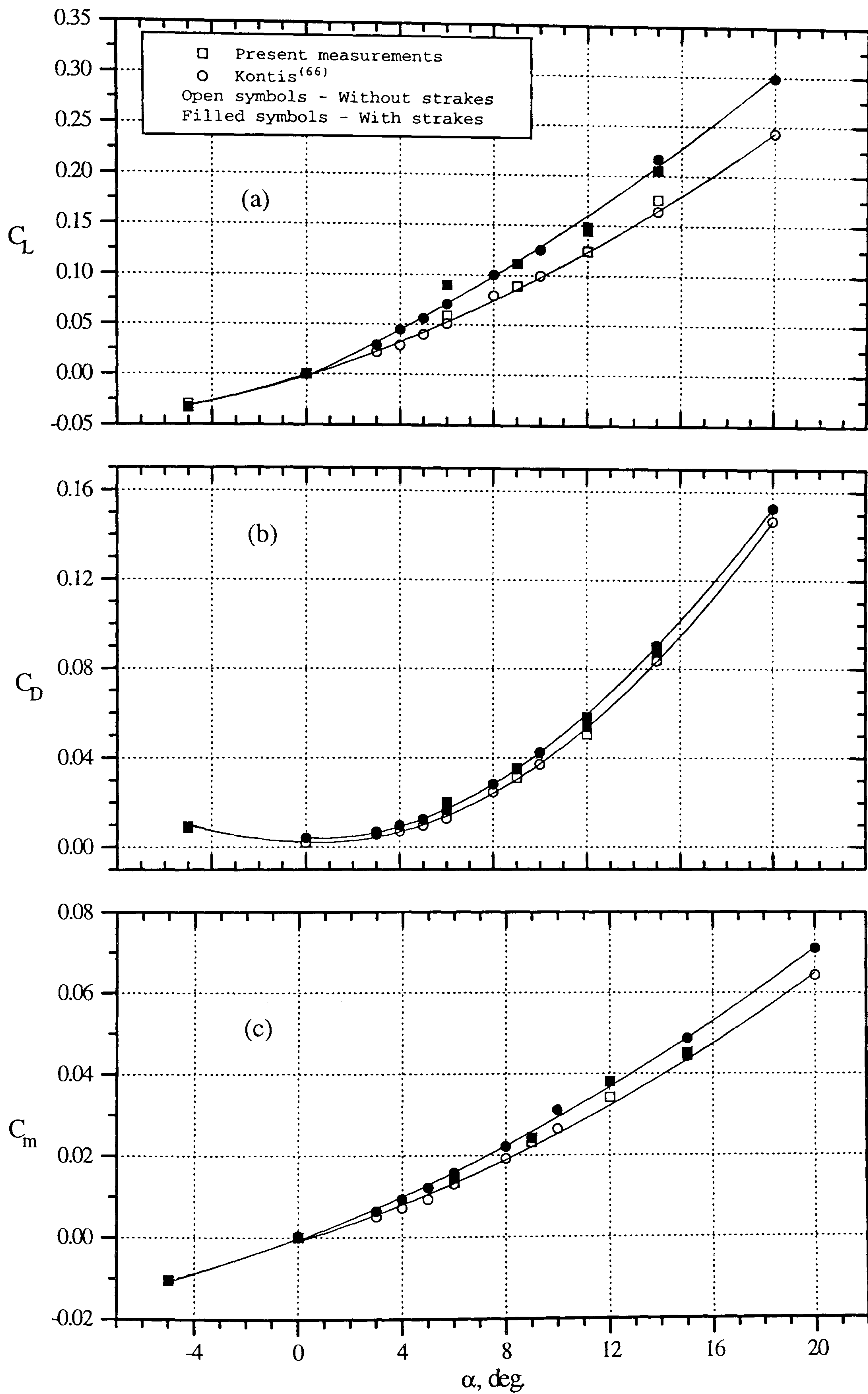


Figure 18. A comparison of the aerodynamic characteristics of the elliptic power-law body with and without strakes.

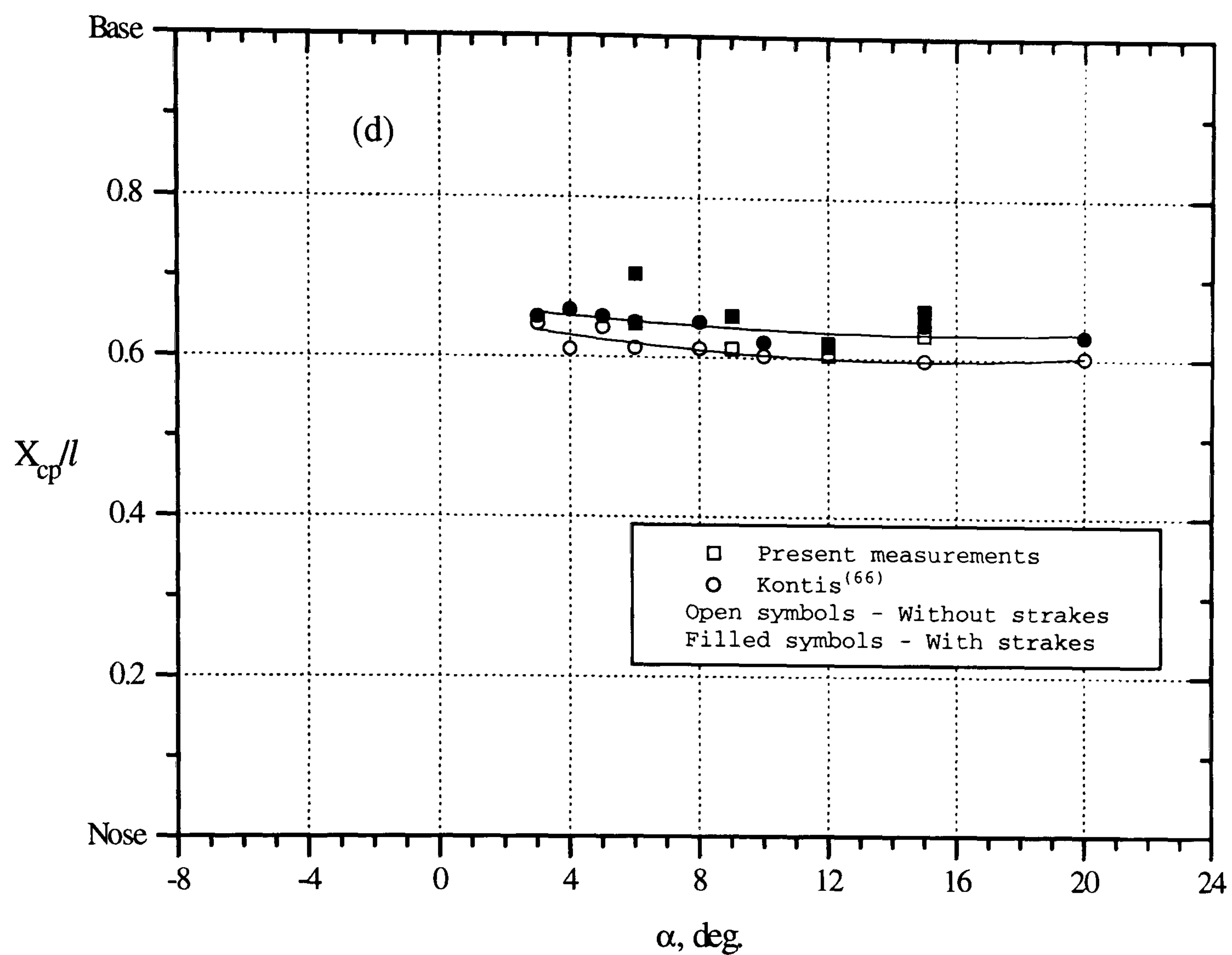
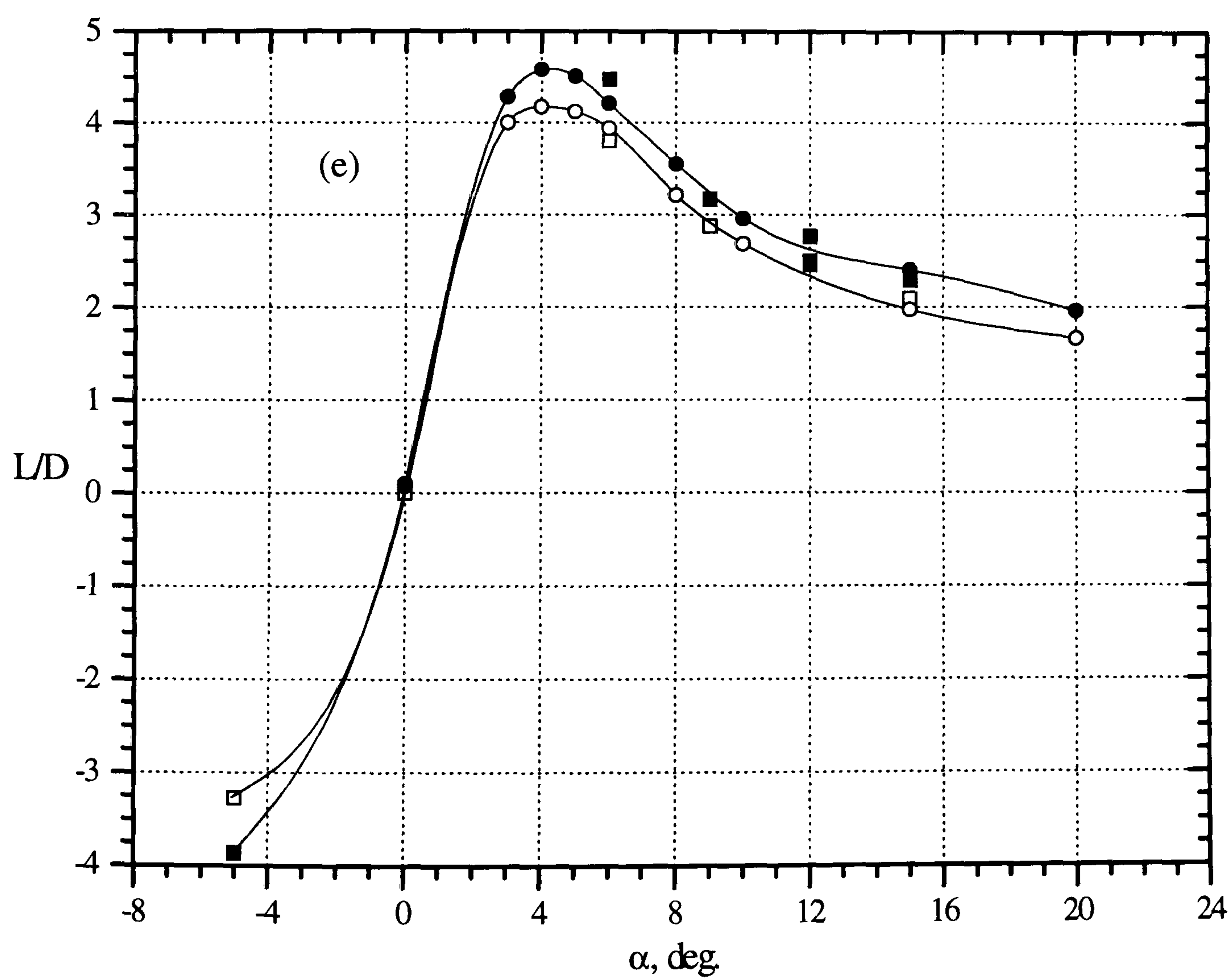


Figure 18. Continued.

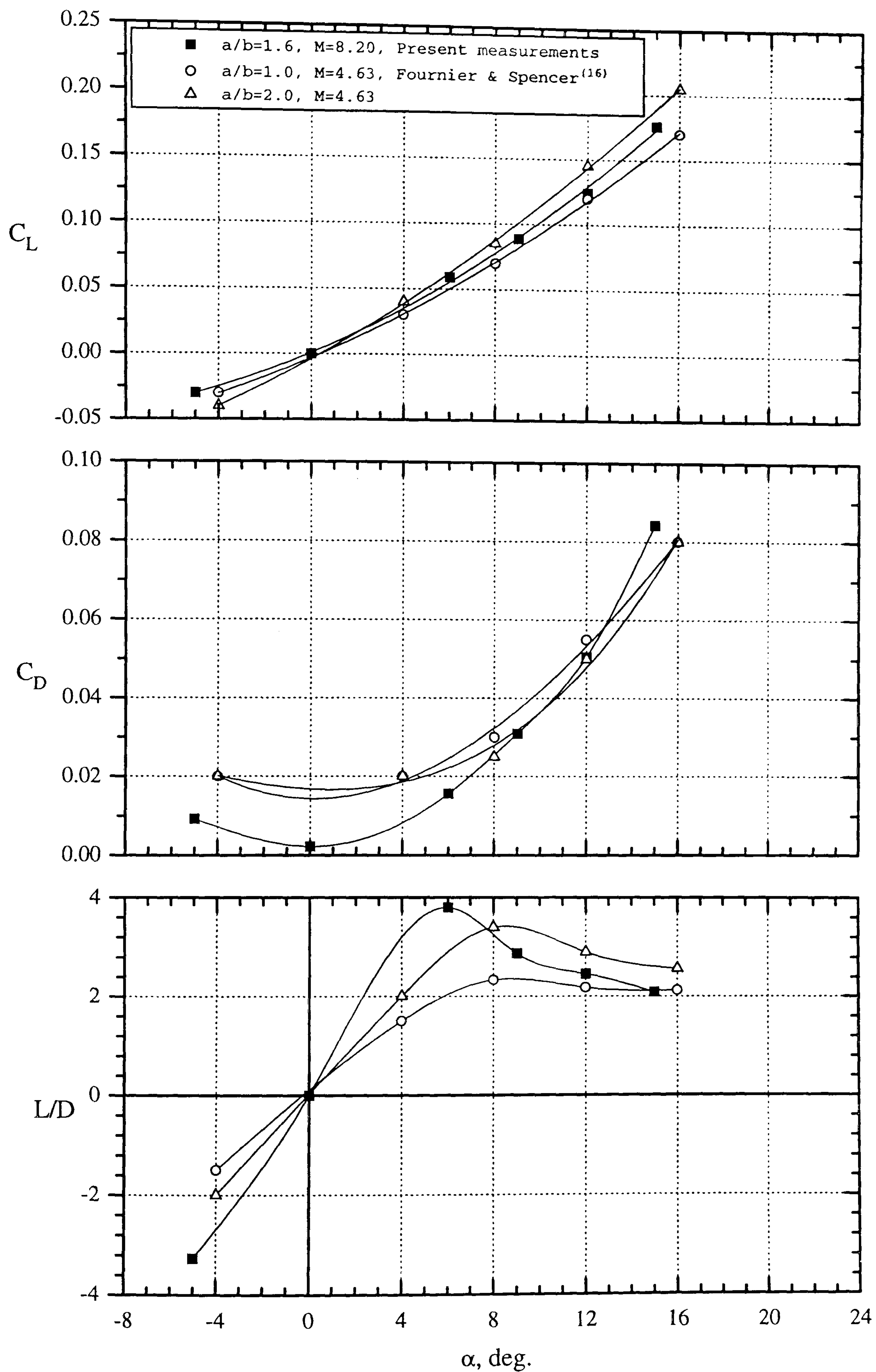


Figure 19(a). Comparison of present elliptical cone force measurements with other experimental results.

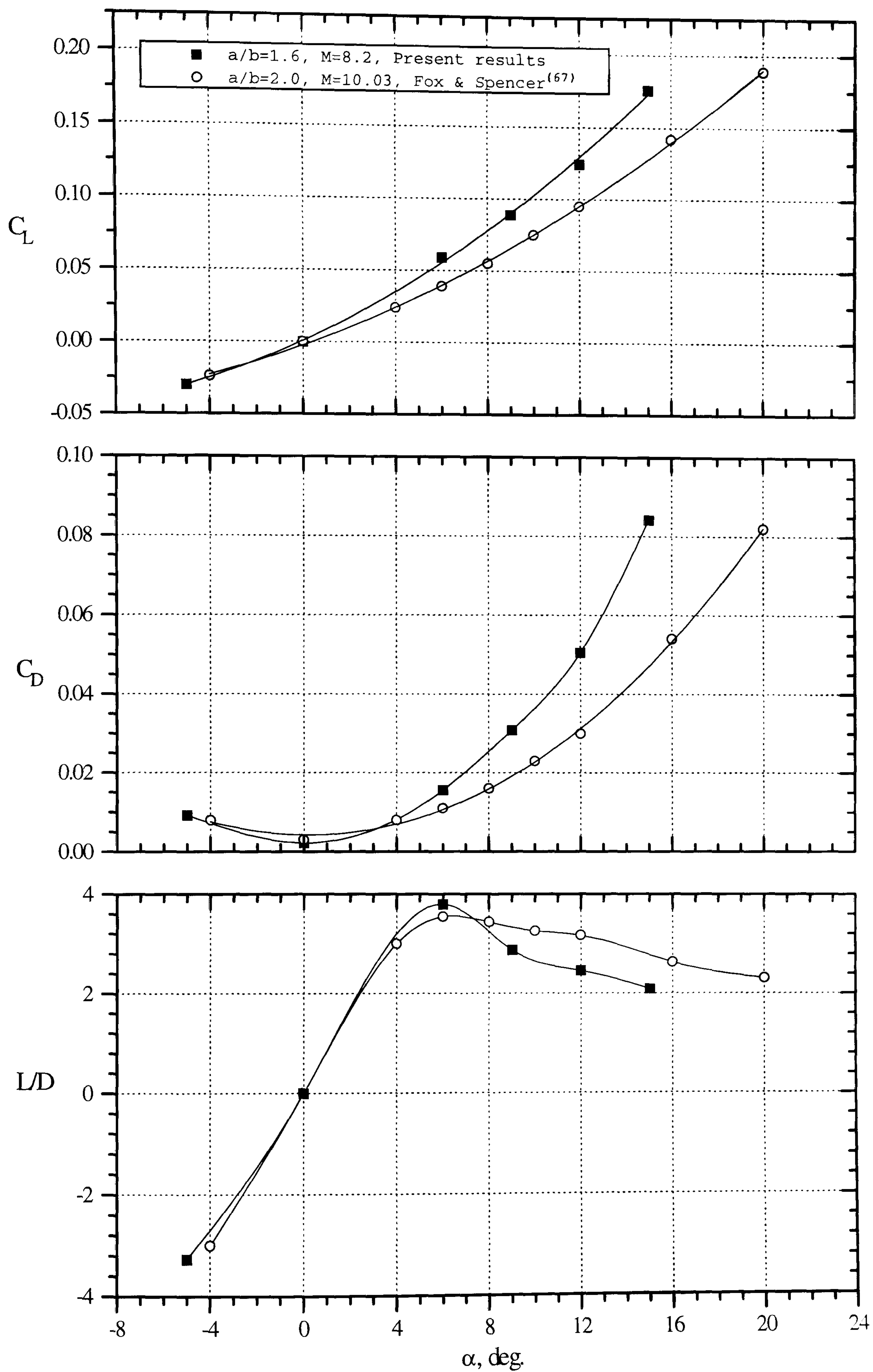


Figure 19(b). Comparison of present elliptical cone force measurements with other experimental results.

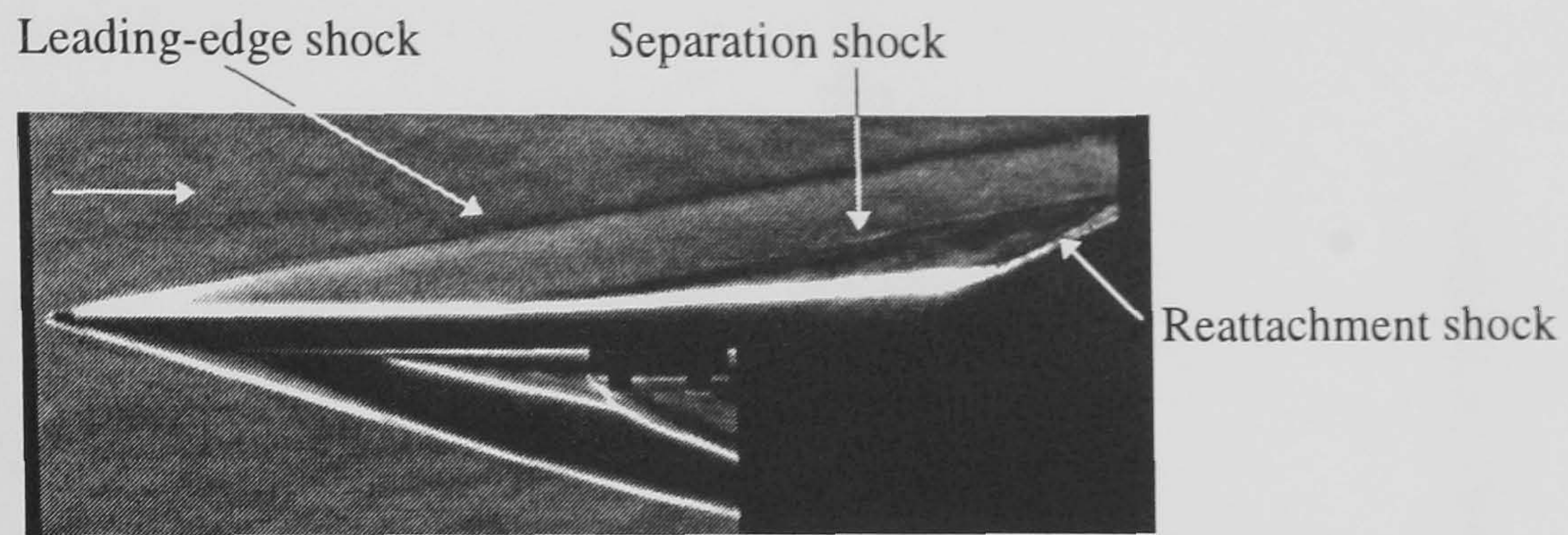


Figure 20. A schlieren picture showing separated flow over a sharp flat plate at zero incidence with 25° flap.

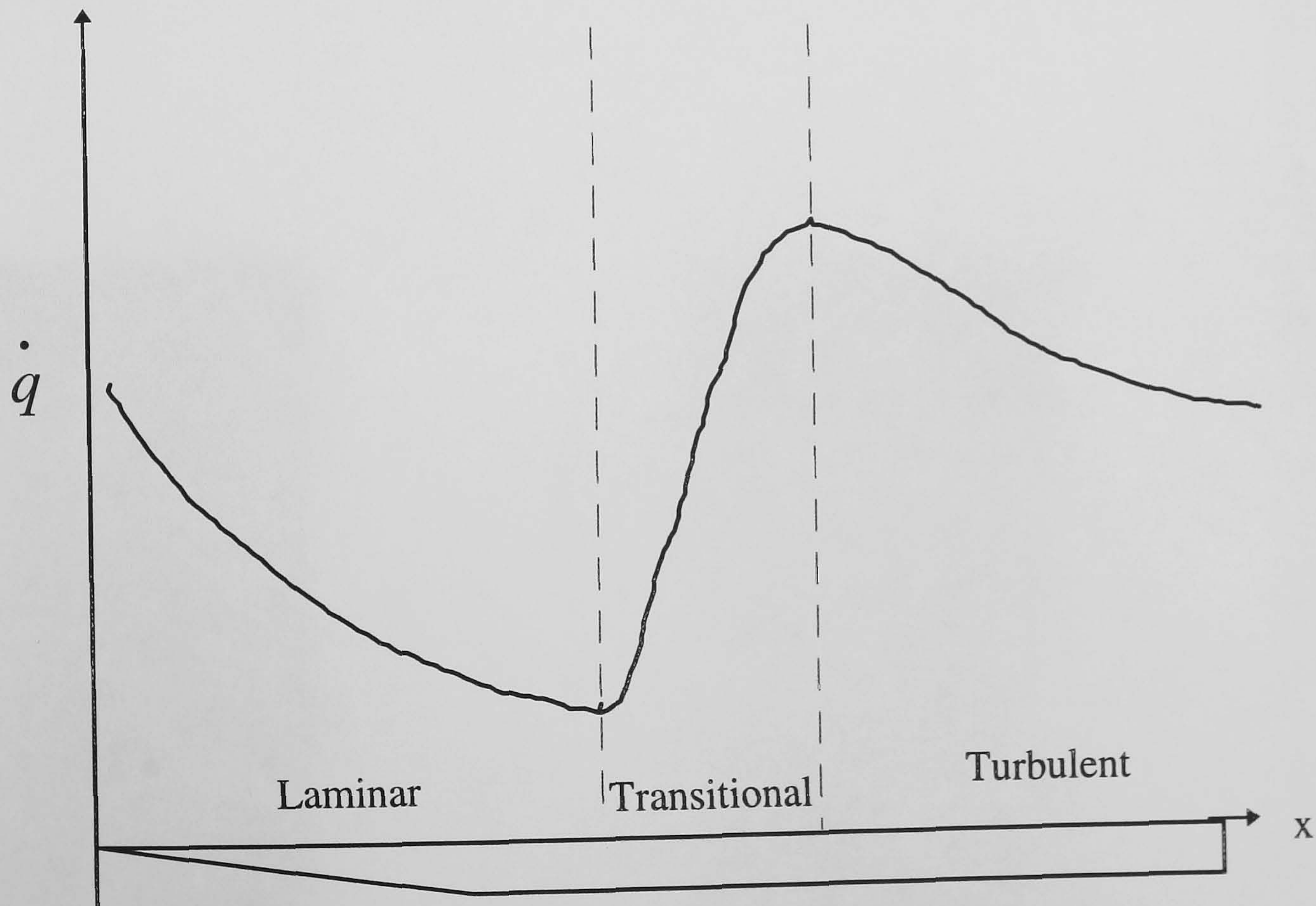
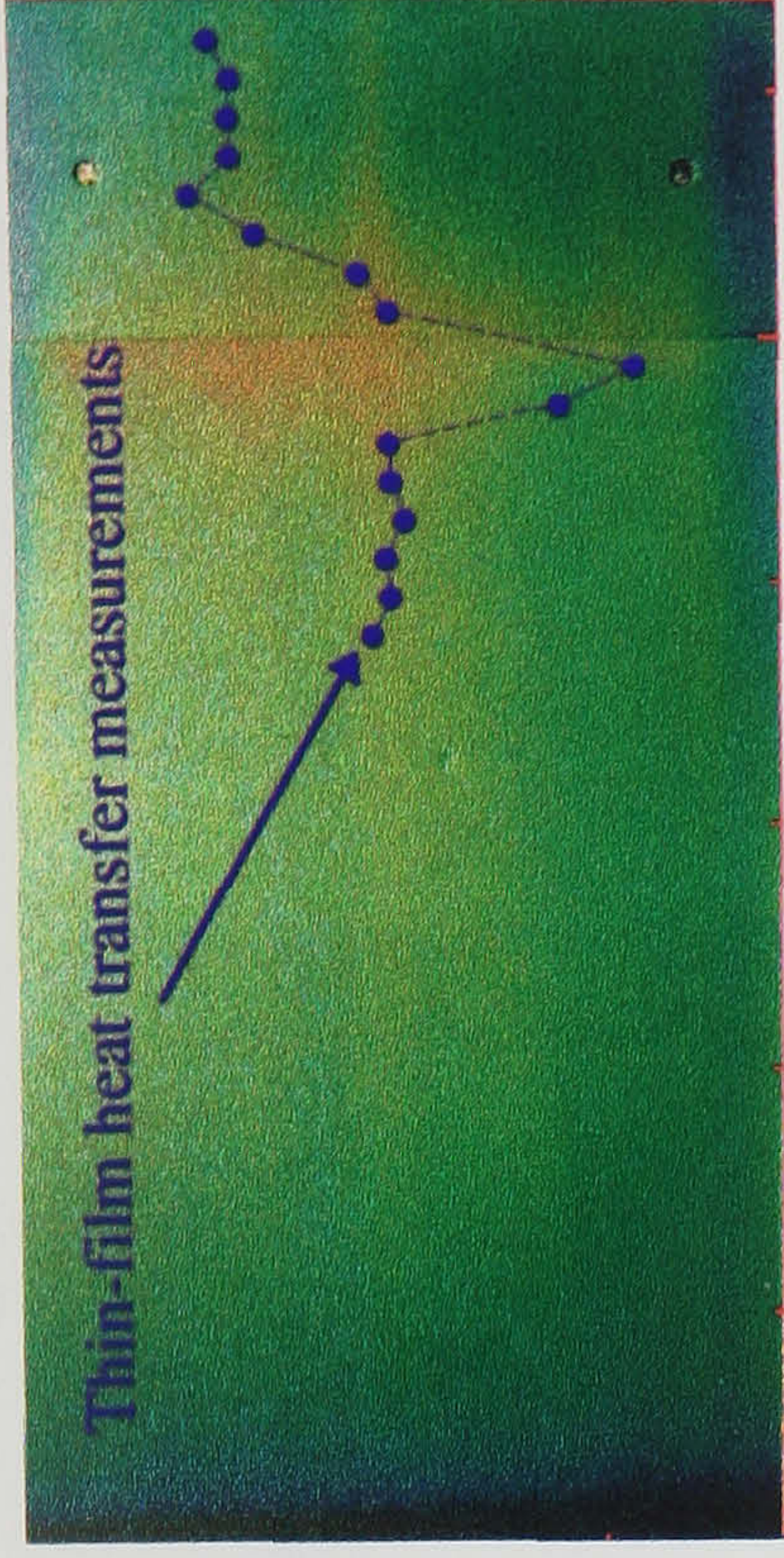


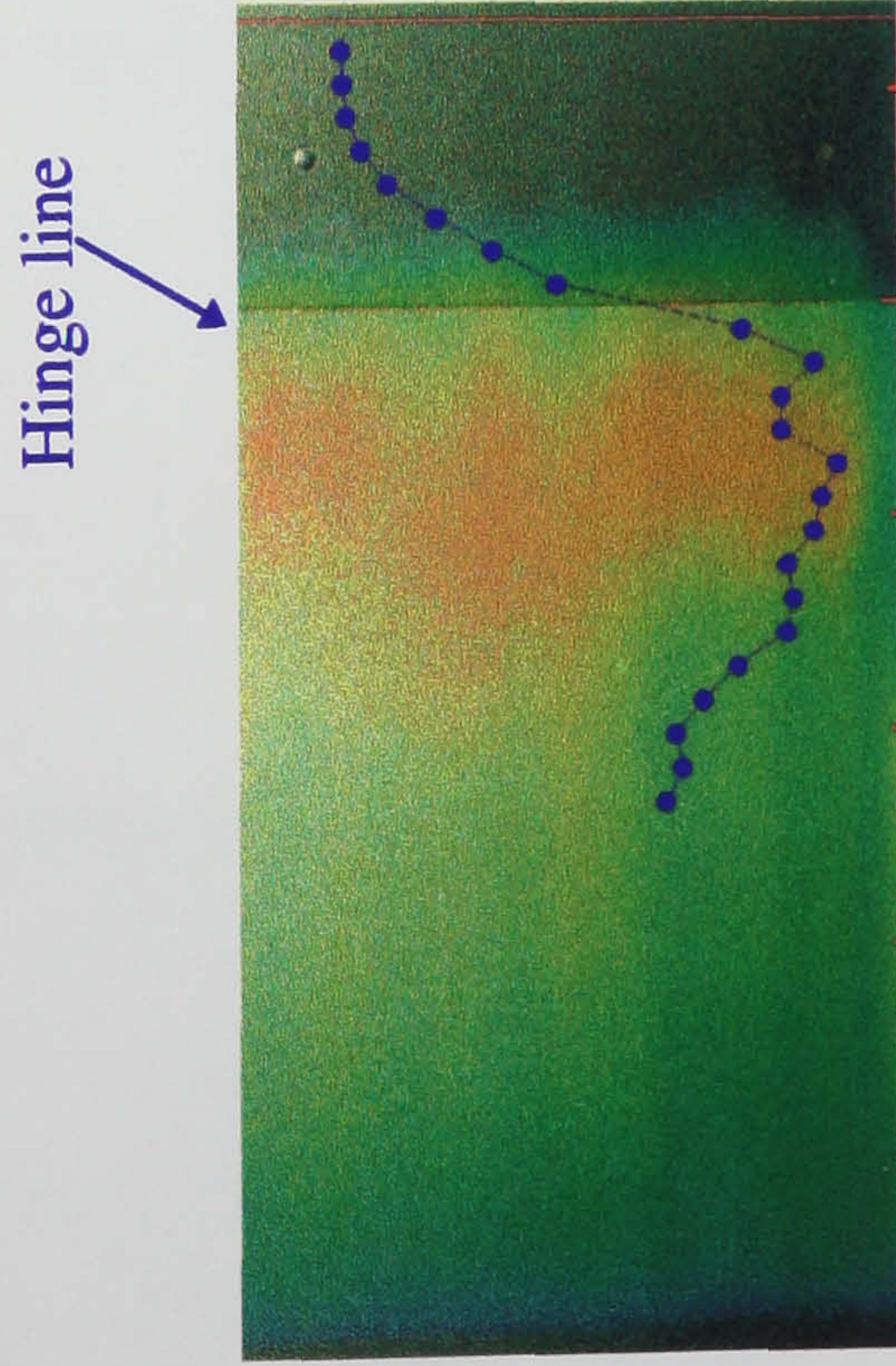
Figure 21. Heat transfer rate distribution over a sharp flat plate with natural transition from laminar to turbulent flow.



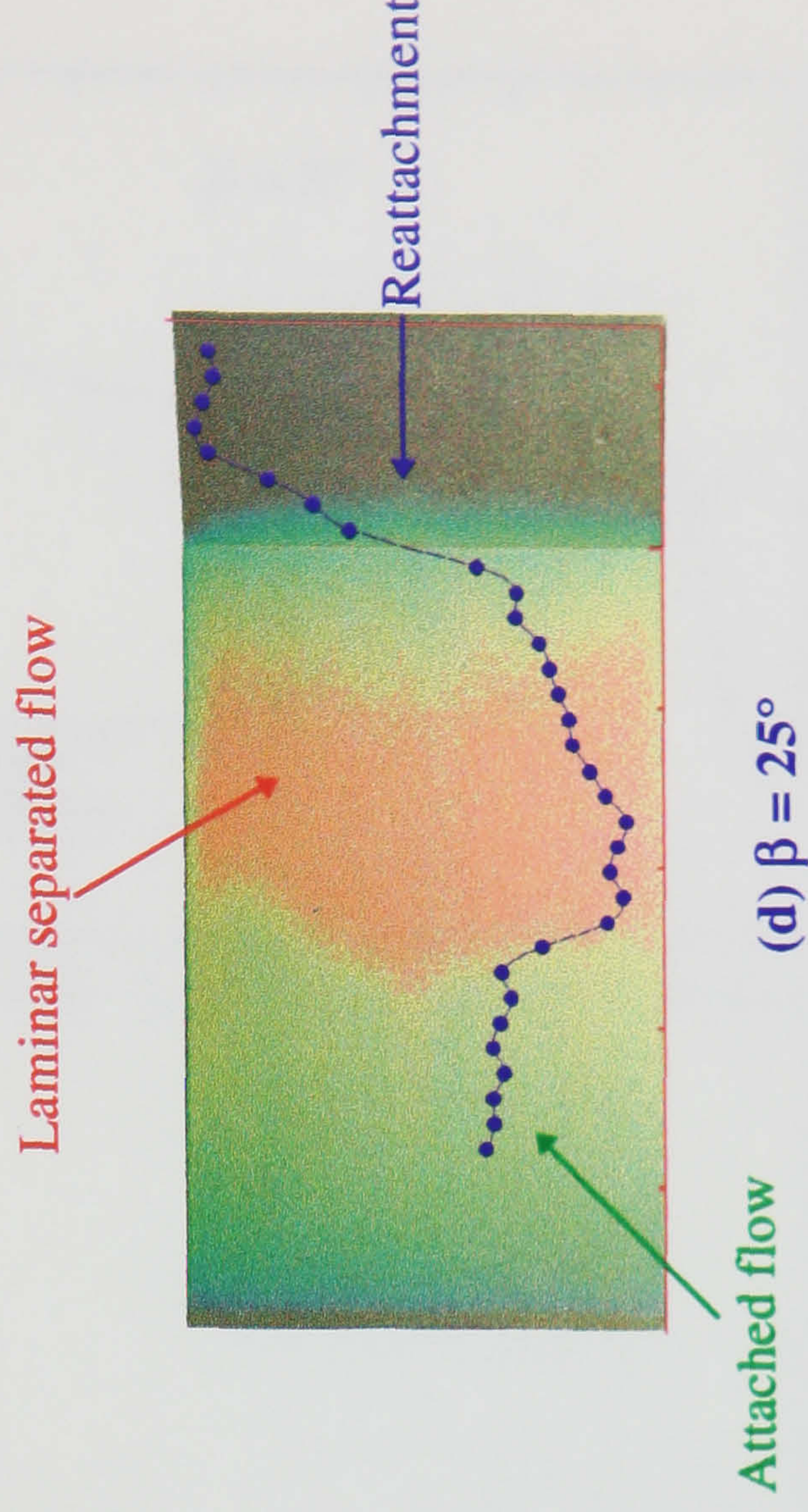
(a) $\beta = 0^\circ$



(b) $\beta = 5^\circ$



(c) $\beta = 15^\circ$



(d) $\beta = 25^\circ$

Figure 22. Liquid crystal photographs, with superimposed thin-film heat transfer measurements, showing variation in temperature distribution over the surface of a flat plate with increasing β .

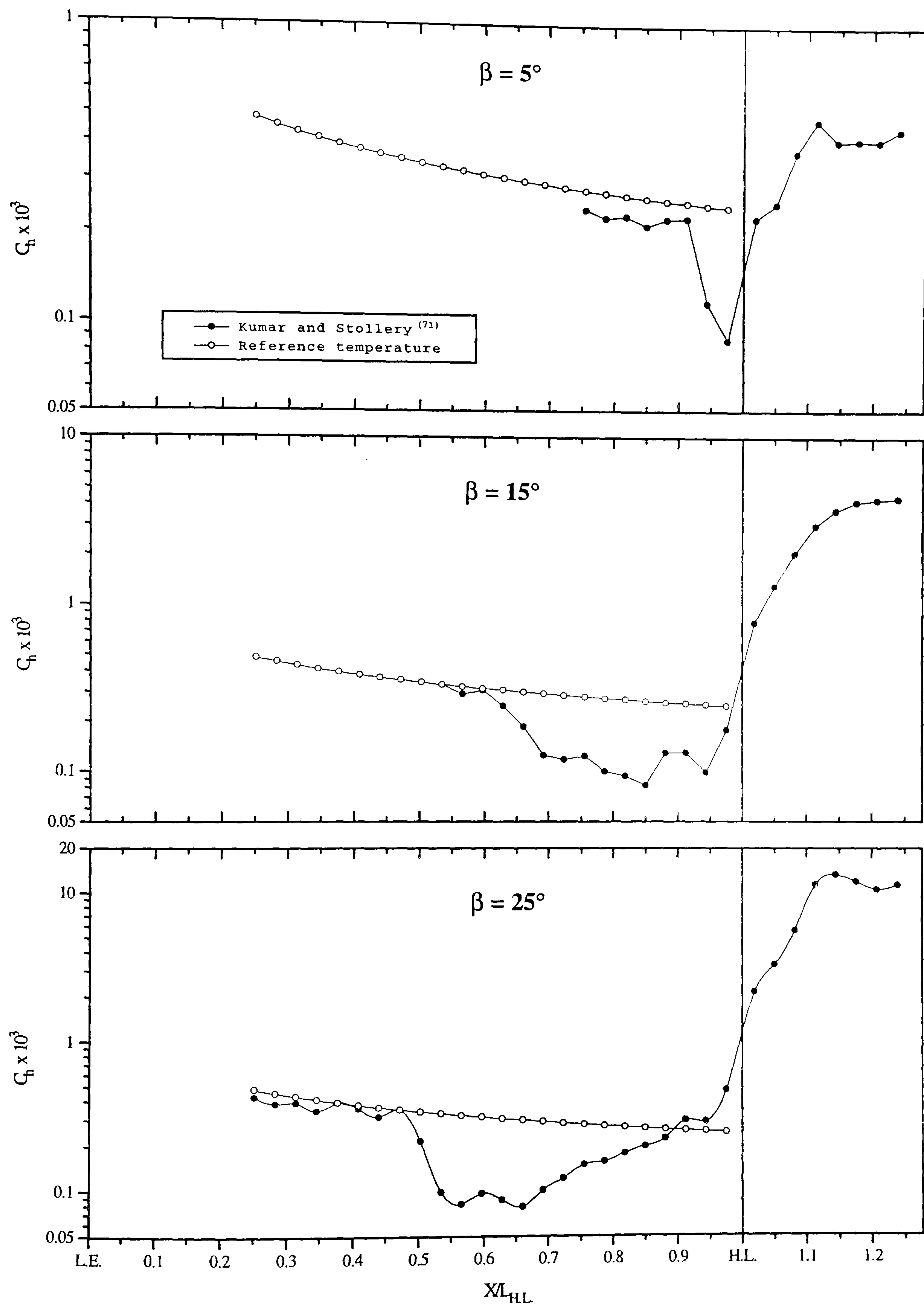


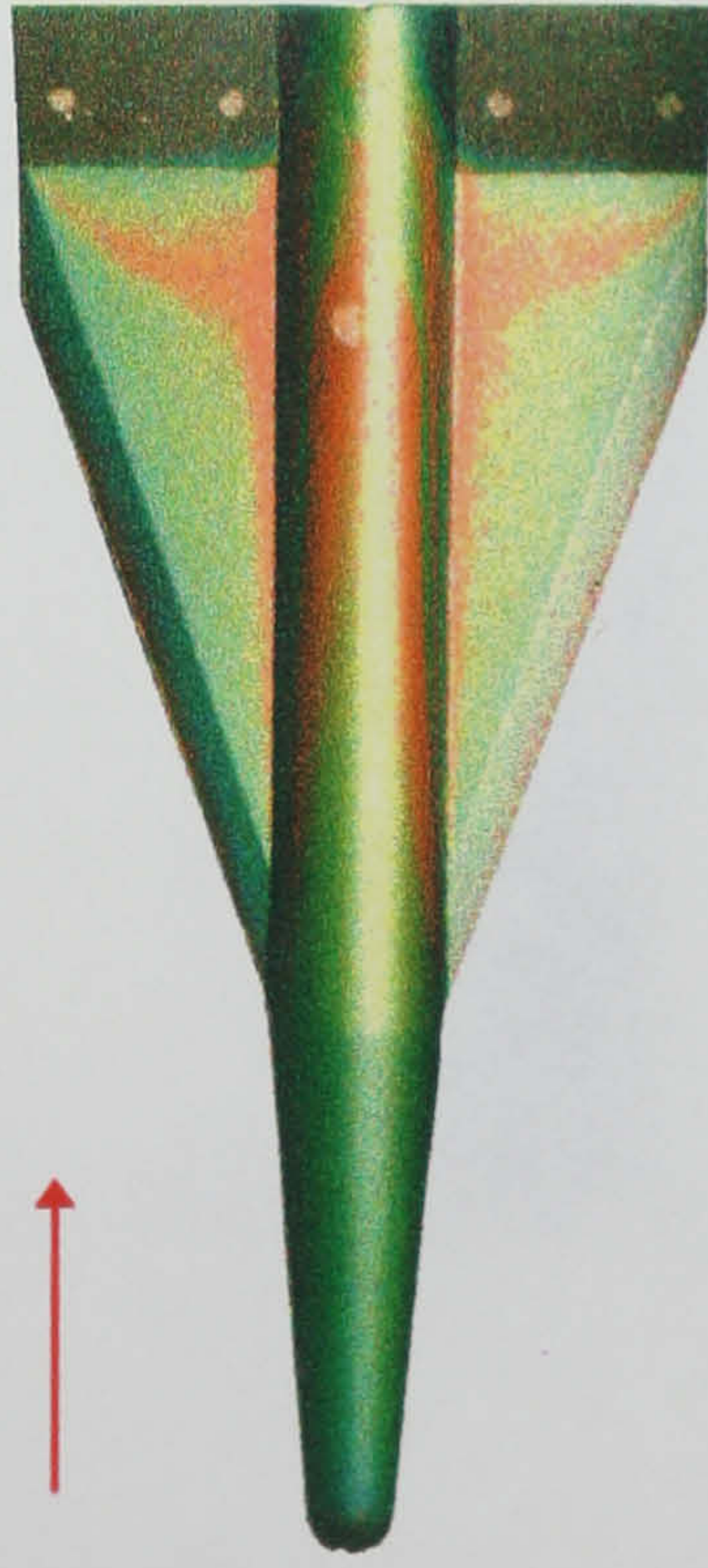
Figure 23. Variation in the heat transfer distribution, along the flat plate centre line, with increasing flap deflection angle.



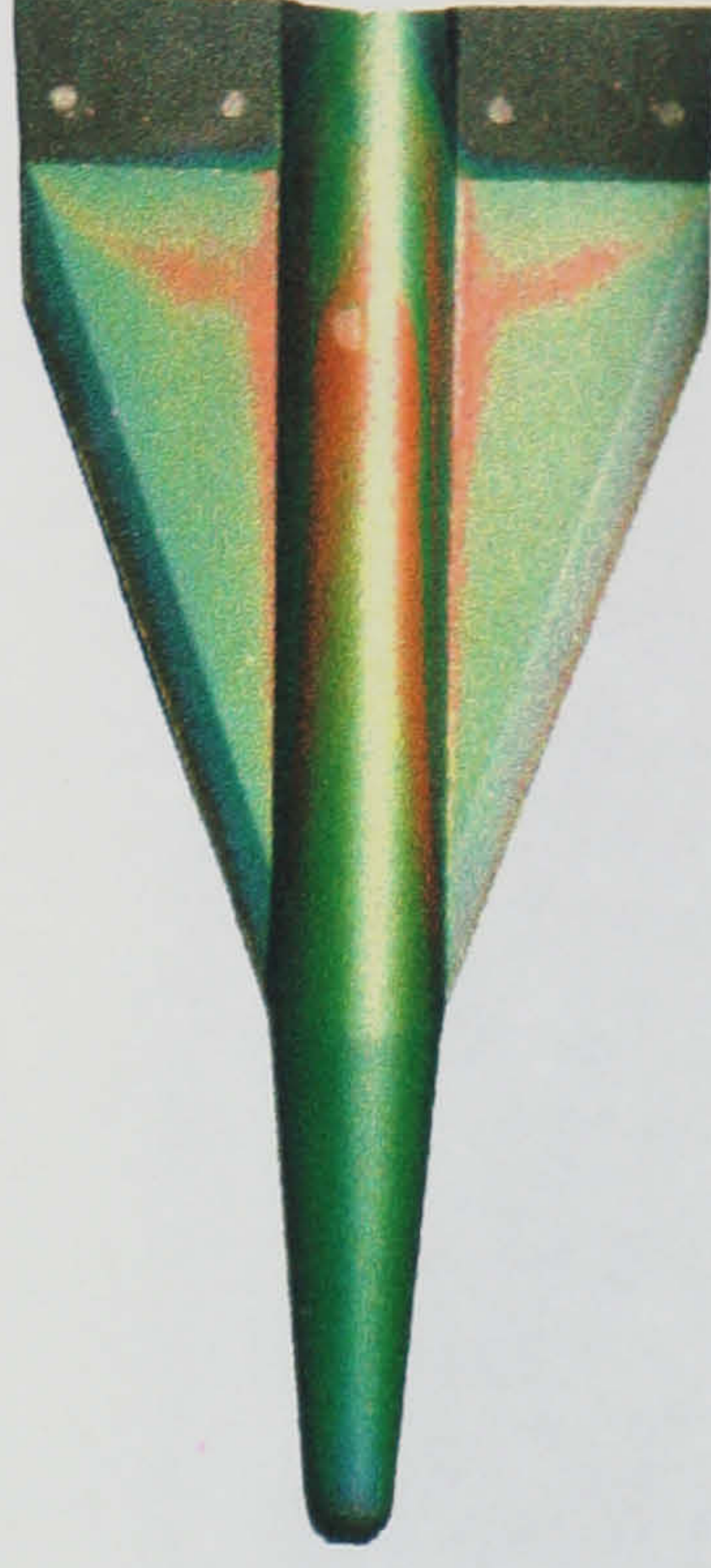
(a) Delay of 5 msec



(b) Delay of 10 msec



(c) Delay of 15 msec



(d) Delay of 25 msec.

Figure 24. Liquid crystal pictures indicating optimum time for a snap and the repeatability for complex flow patterns.
 $\alpha = 5^\circ$ and $\beta = 25^\circ$

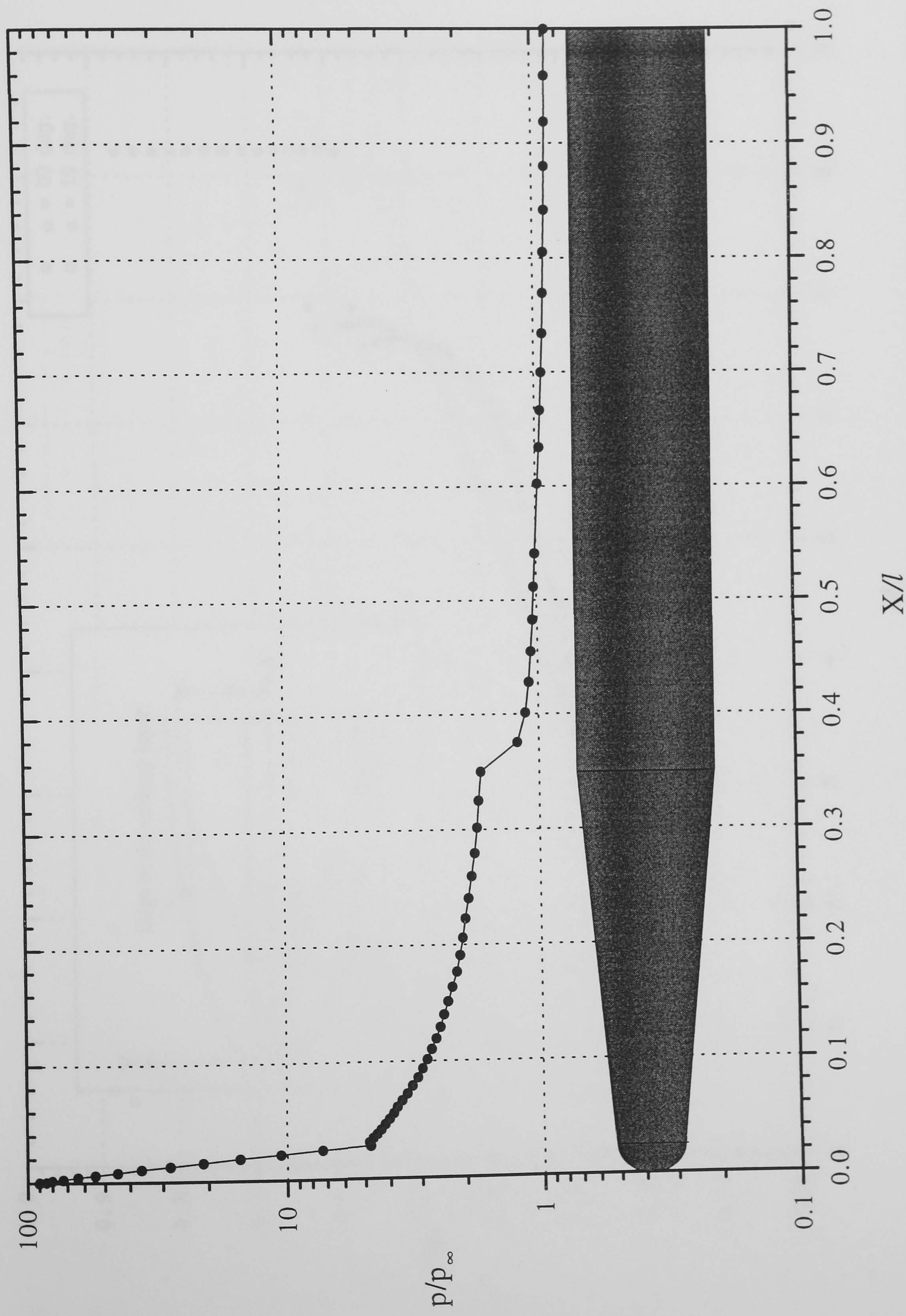


Figure 25. Variation of surface pressure with the axial distance for the cone-cylinder at $\alpha = 0^\circ$.

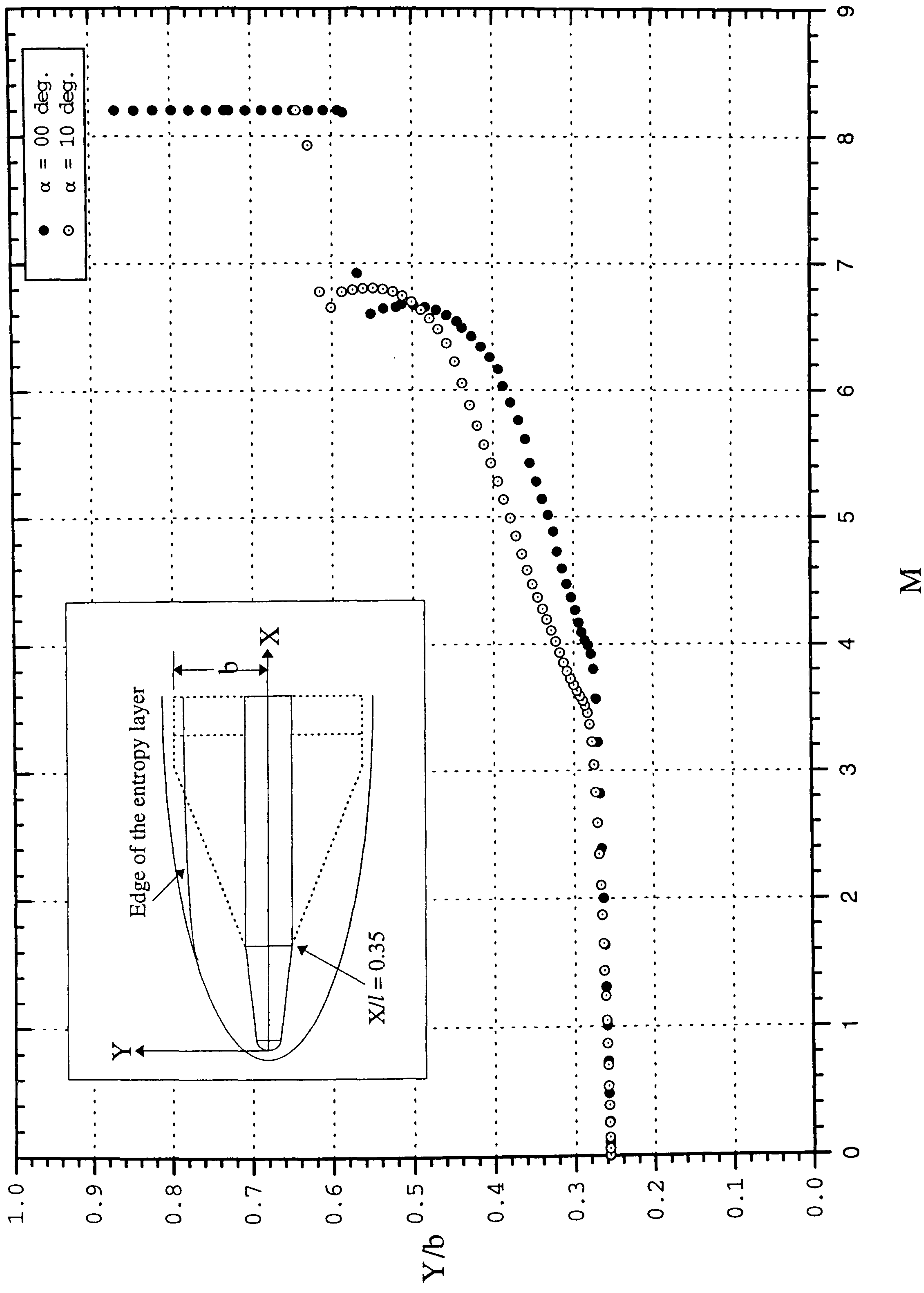


Figure 26(a). Mach number variation inside the shock layer of cone-cylinder body for $\alpha = 0^\circ$ and 10° at $X/l = 0.35$.

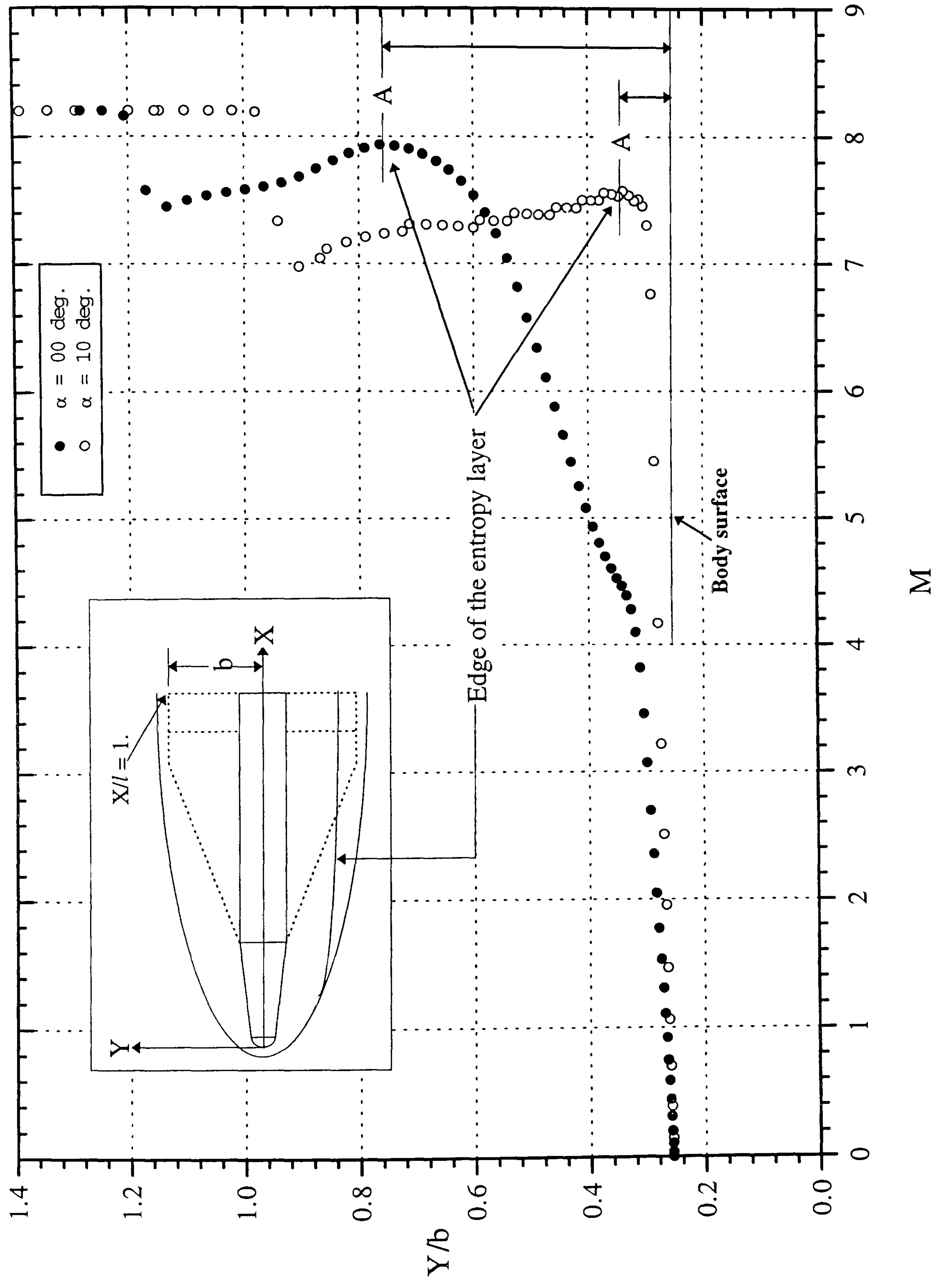


Figure 26(b). Mach number variation inside the shock layer of cone-cylinder body for $\alpha = 0^\circ$ and 10° at $X/l = 1.0$.

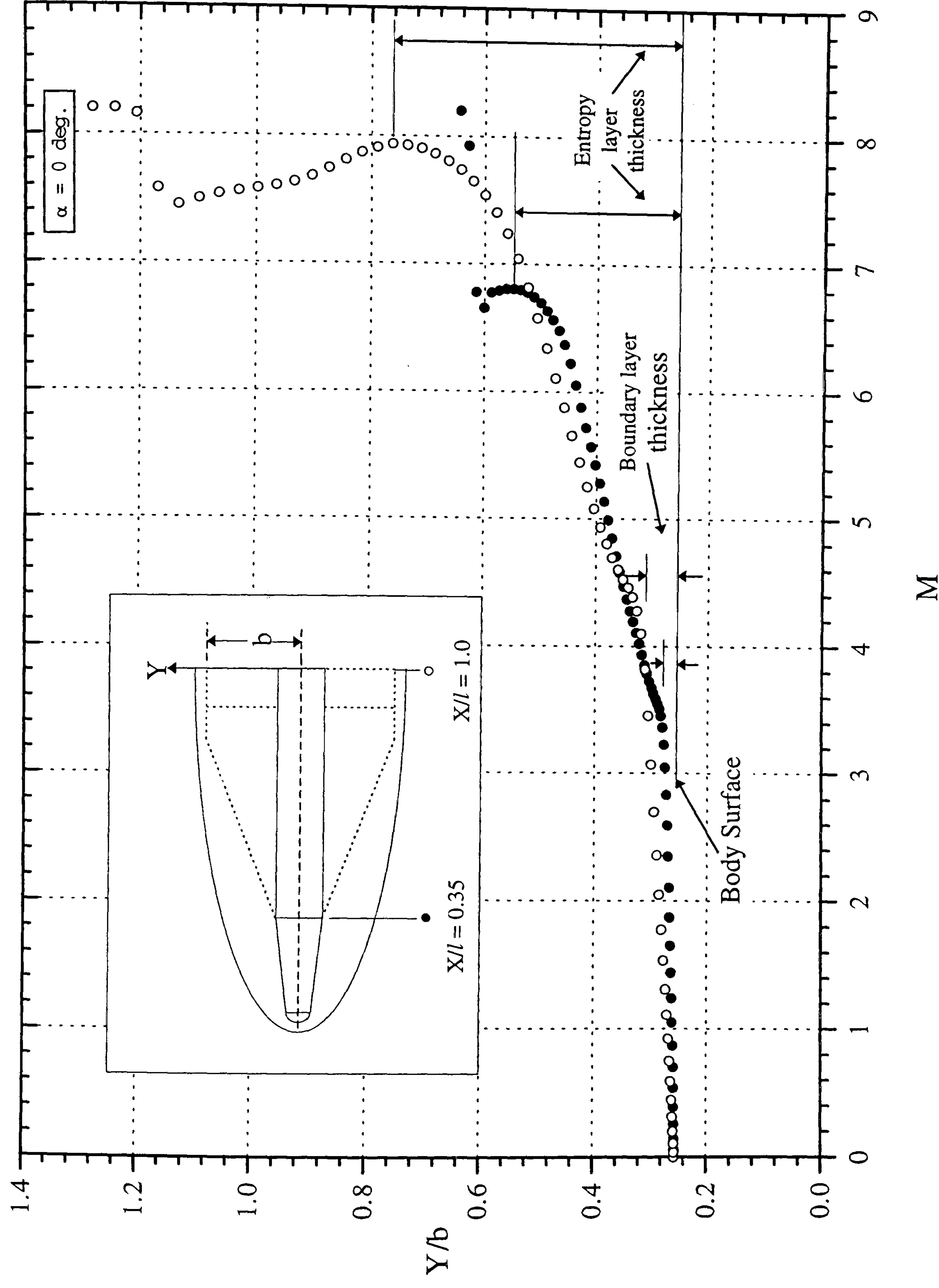


Figure 26(c). Mach number variation inside the shock layer of cone-cylinder body for $\alpha = 0^\circ$ at $X/l = 0.35$ and $X/l = 1.0$.

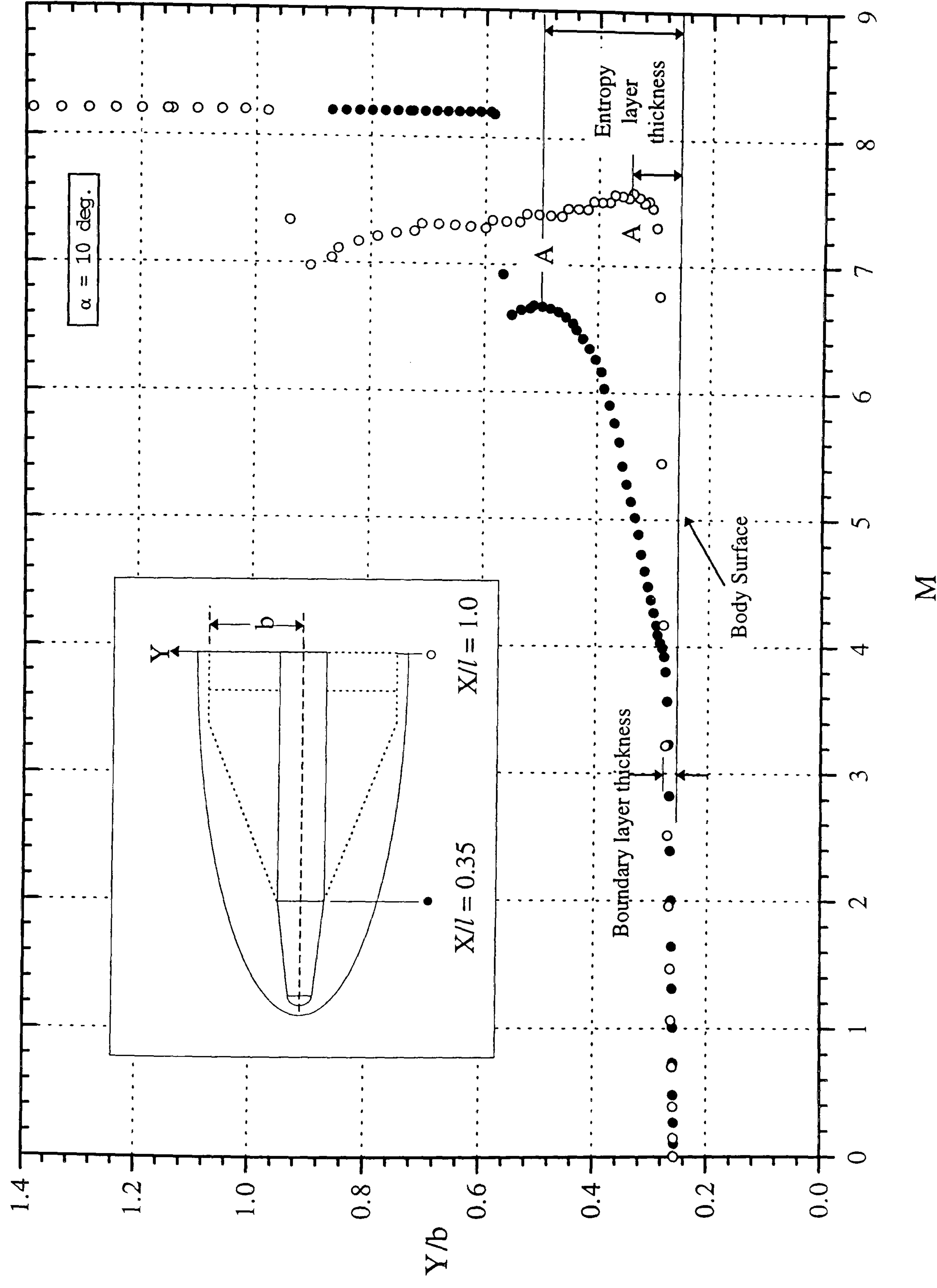
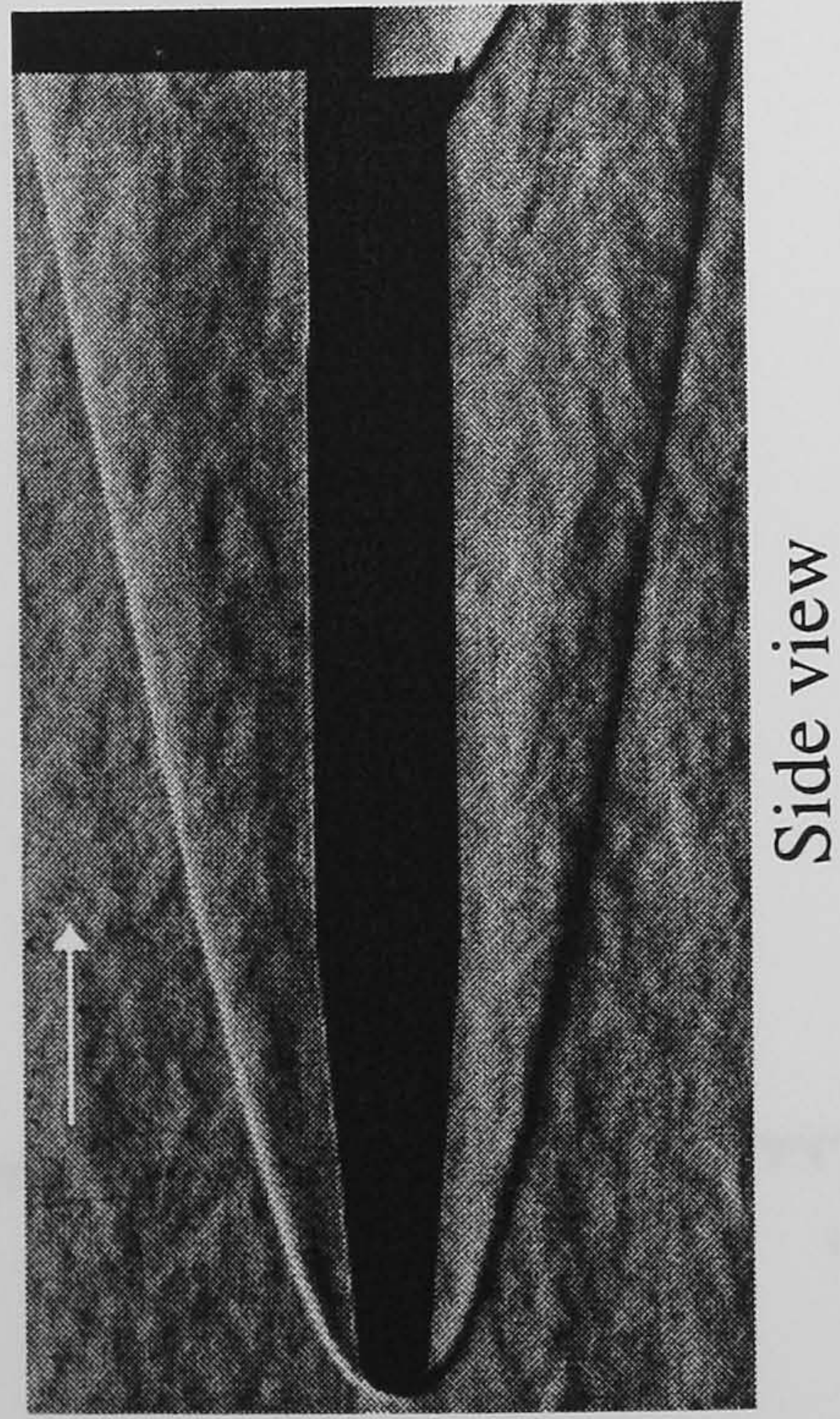
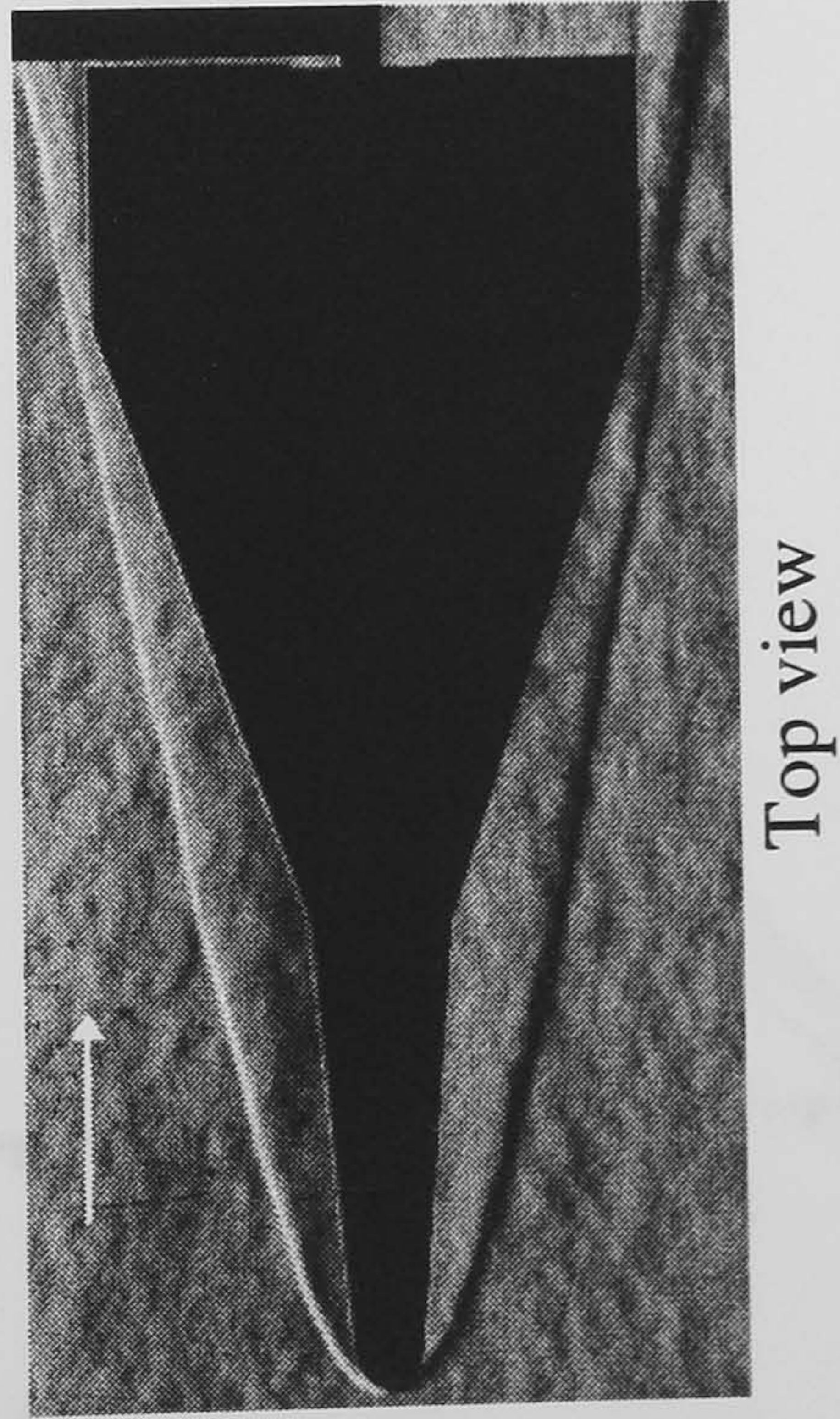
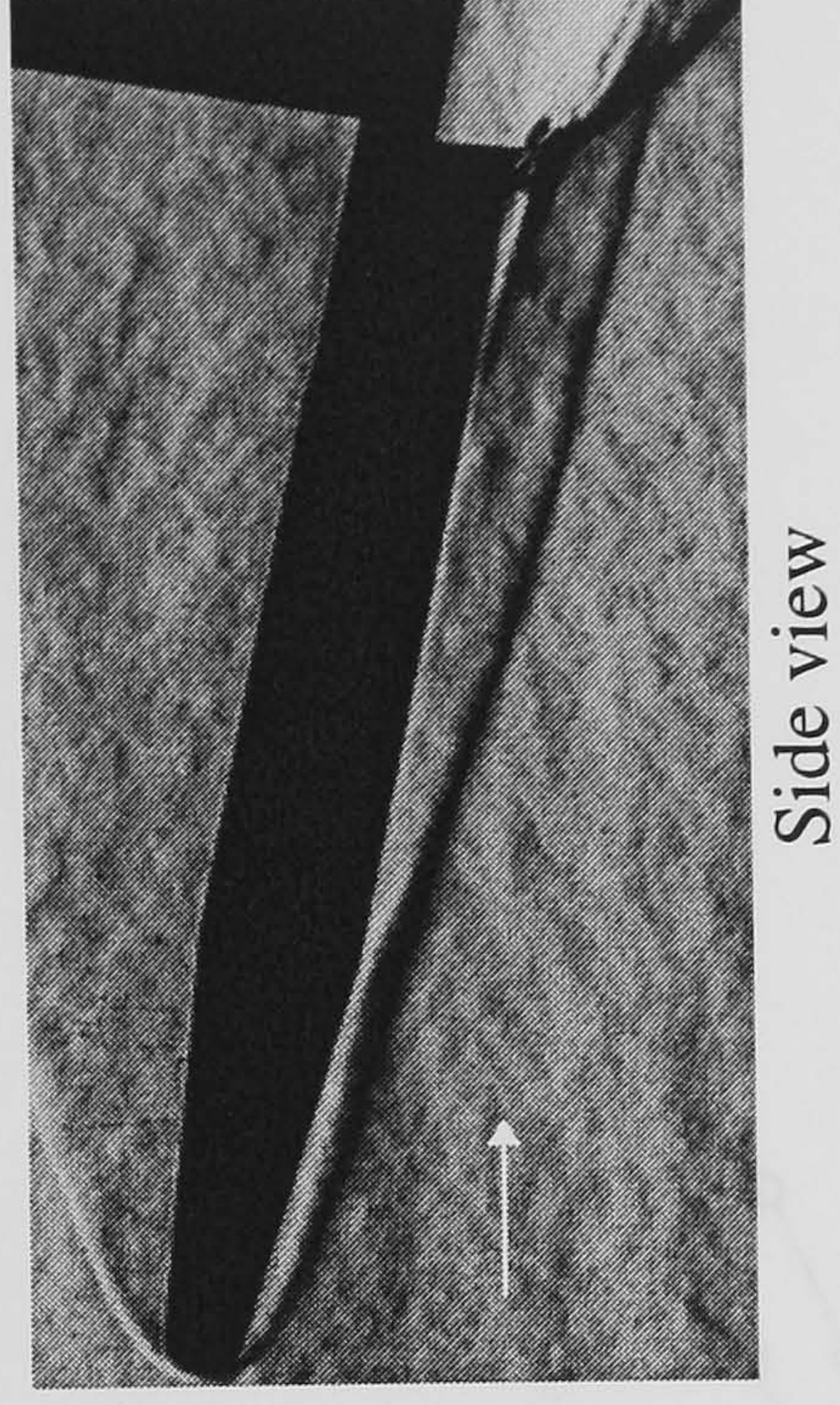
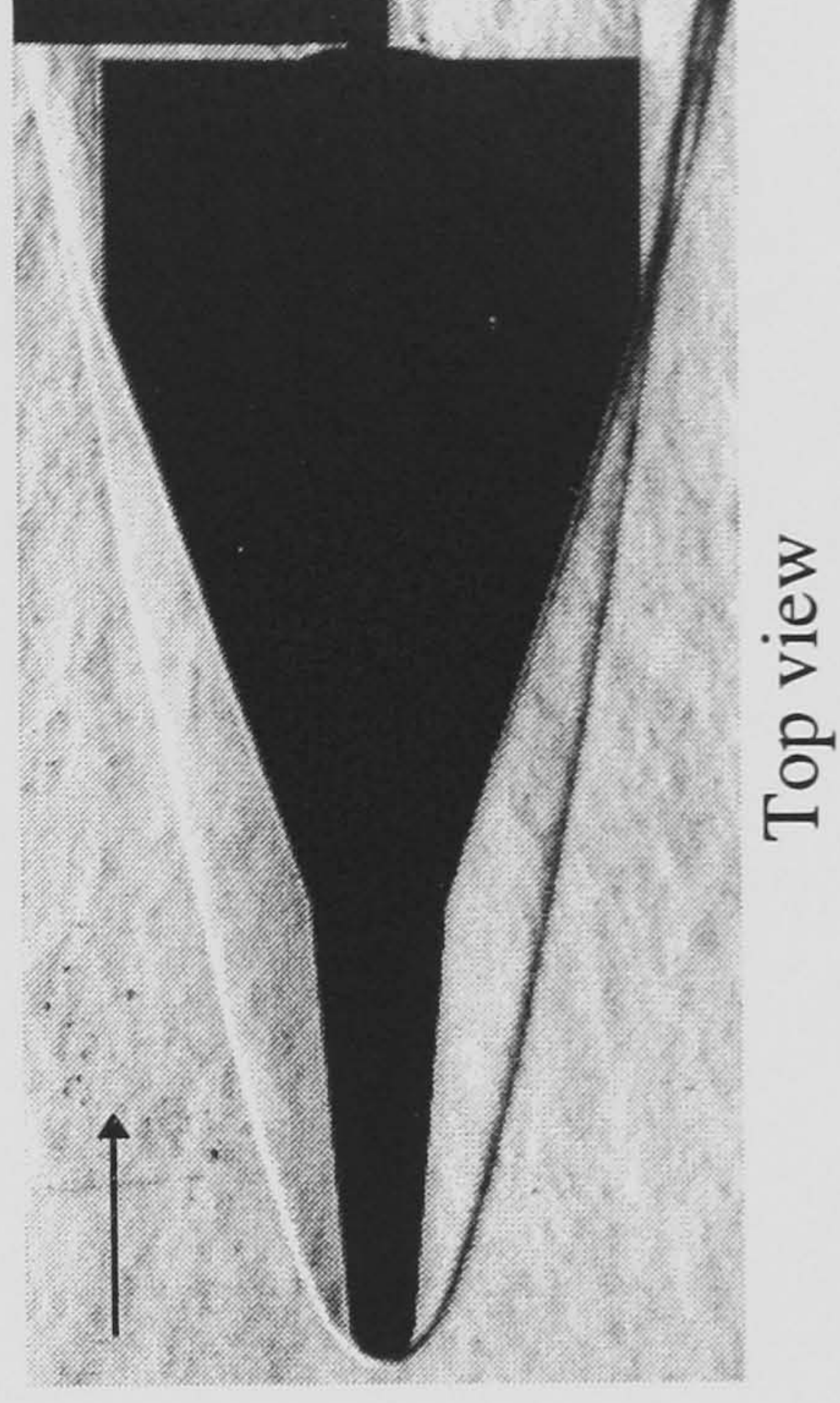


Figure 26(d). Mach number variation inside the shock layer of cone-cylinder body for $\alpha = 10^\circ$ at $X/l = 0.35$ and $X/l = 1.0$.



(a) $\alpha = 0^\circ$ and $\beta = 25^\circ$



(b) $\alpha = 10^\circ$ and $\beta = 25^\circ$

Figure 27. Schlieren pictures showing the top and side view of the wing-body at 0° and 10° incidence.

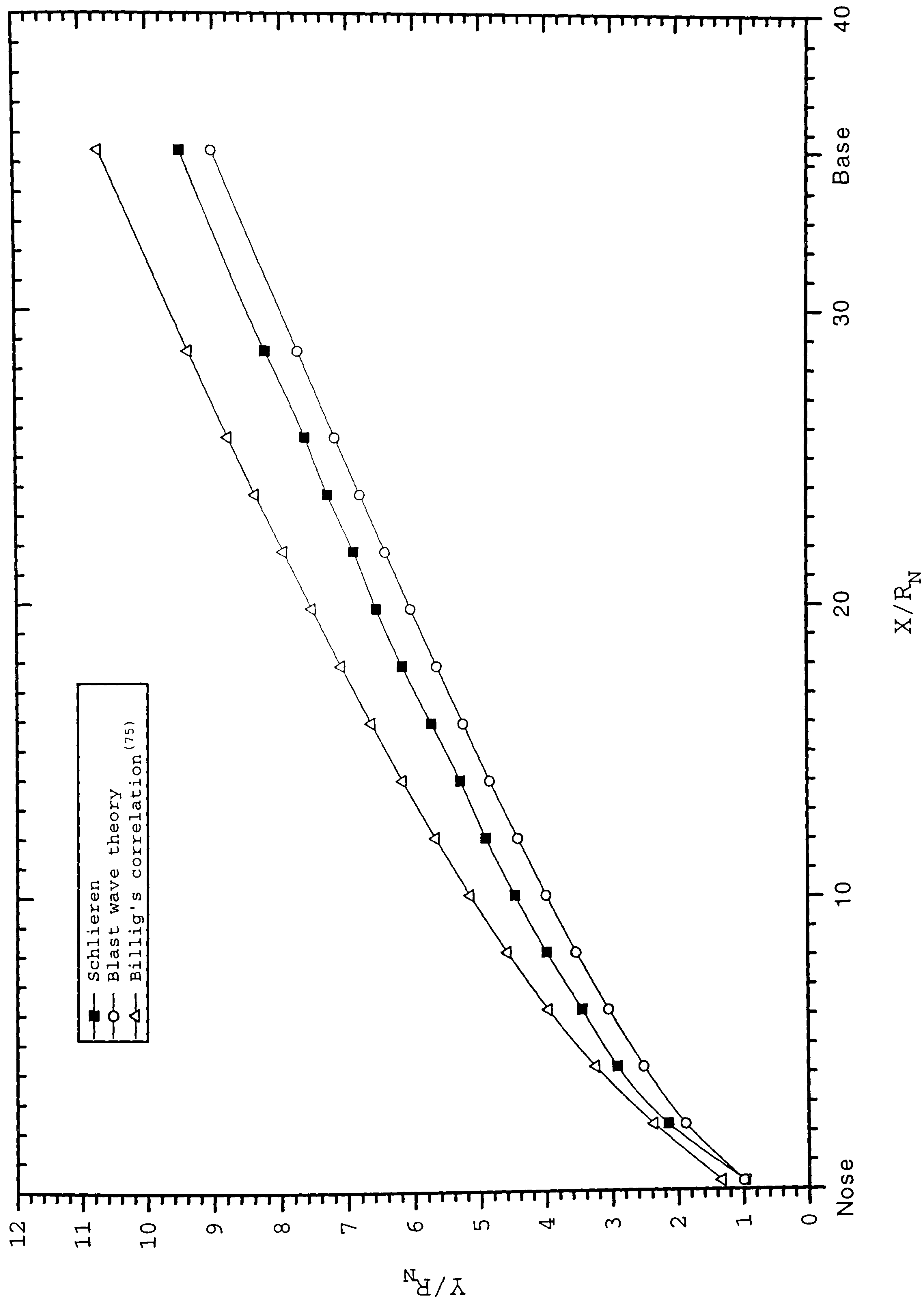
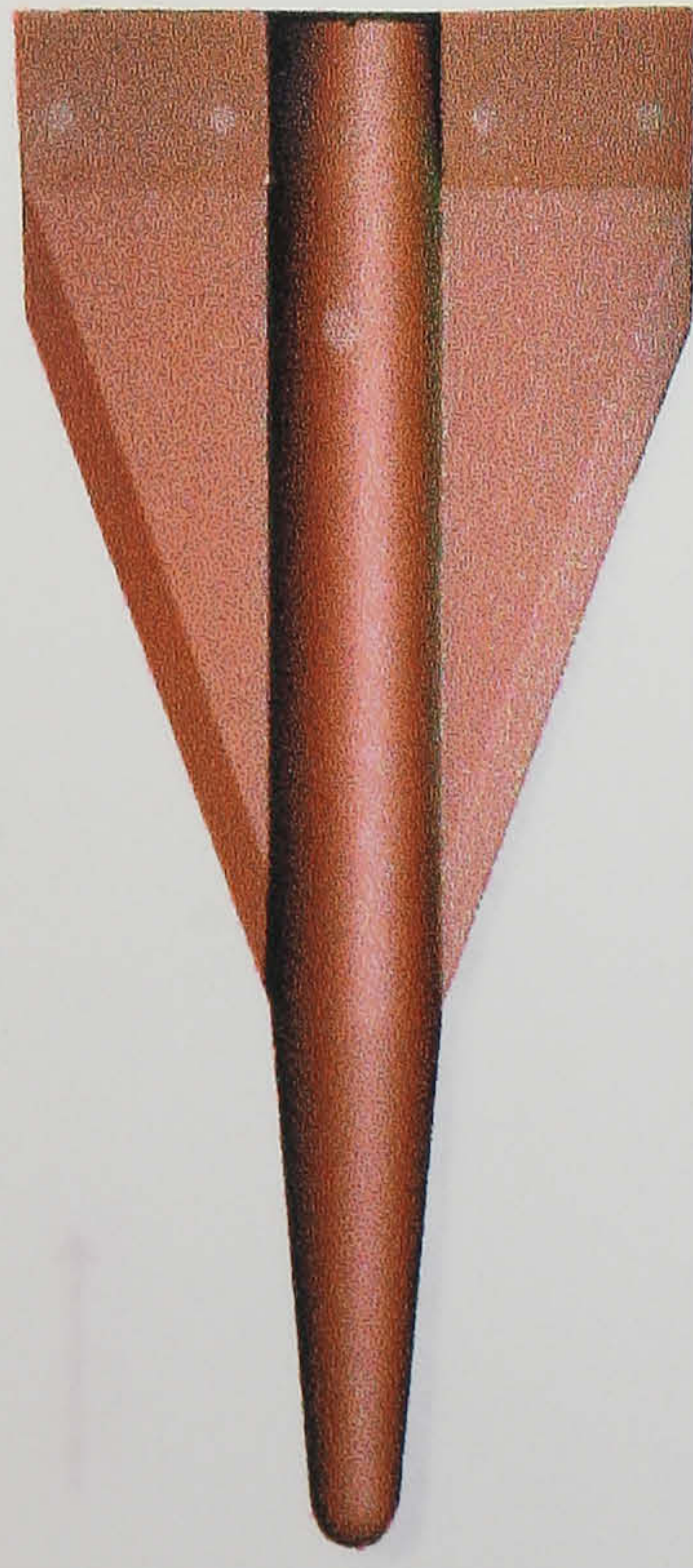
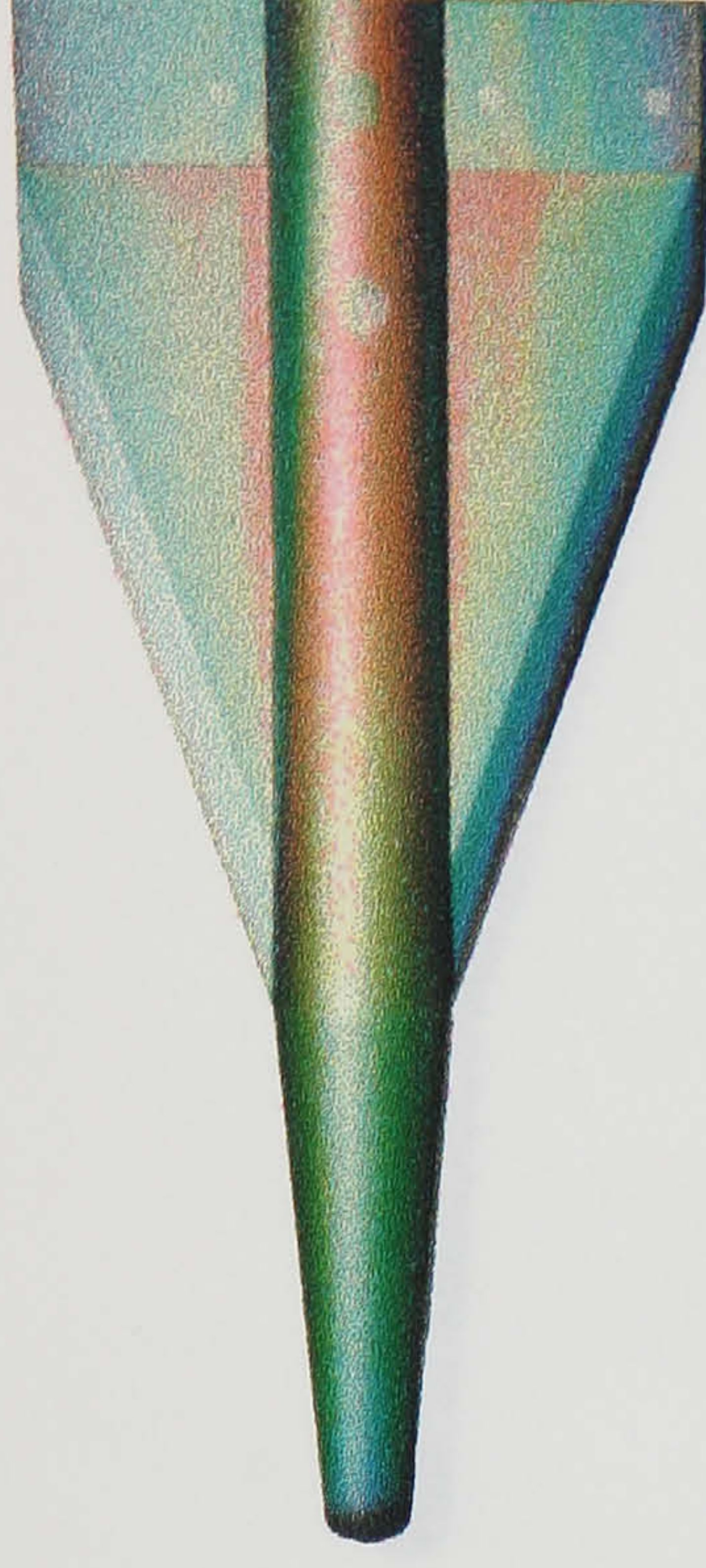


Figure 28. A comparison of experimental and theoretical shape of wing-body bow shock at zero degree incidence.



(a) Before tunnel firing



(b) $\alpha = 0^\circ$ and $\beta = 5^\circ$

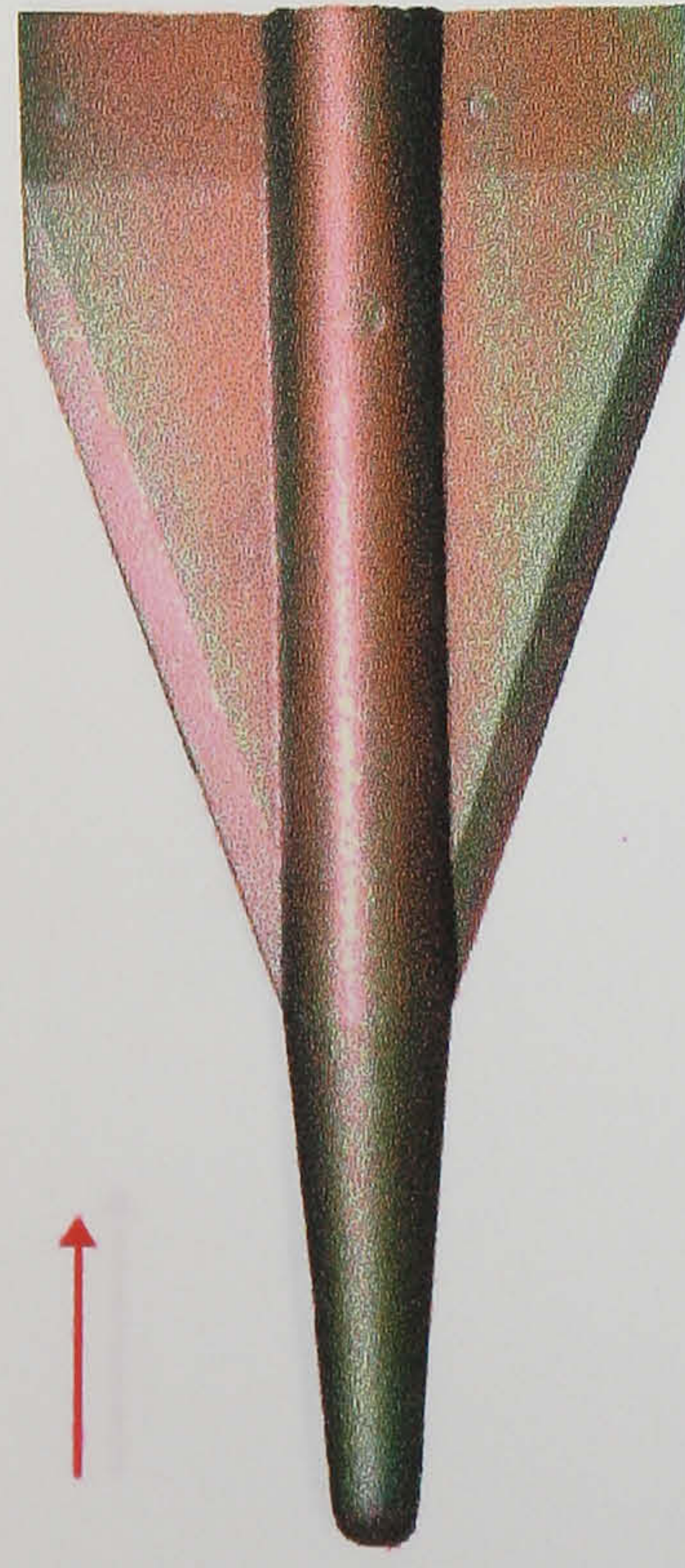


(c) $\alpha = 5^\circ$ and $\beta = 5^\circ$

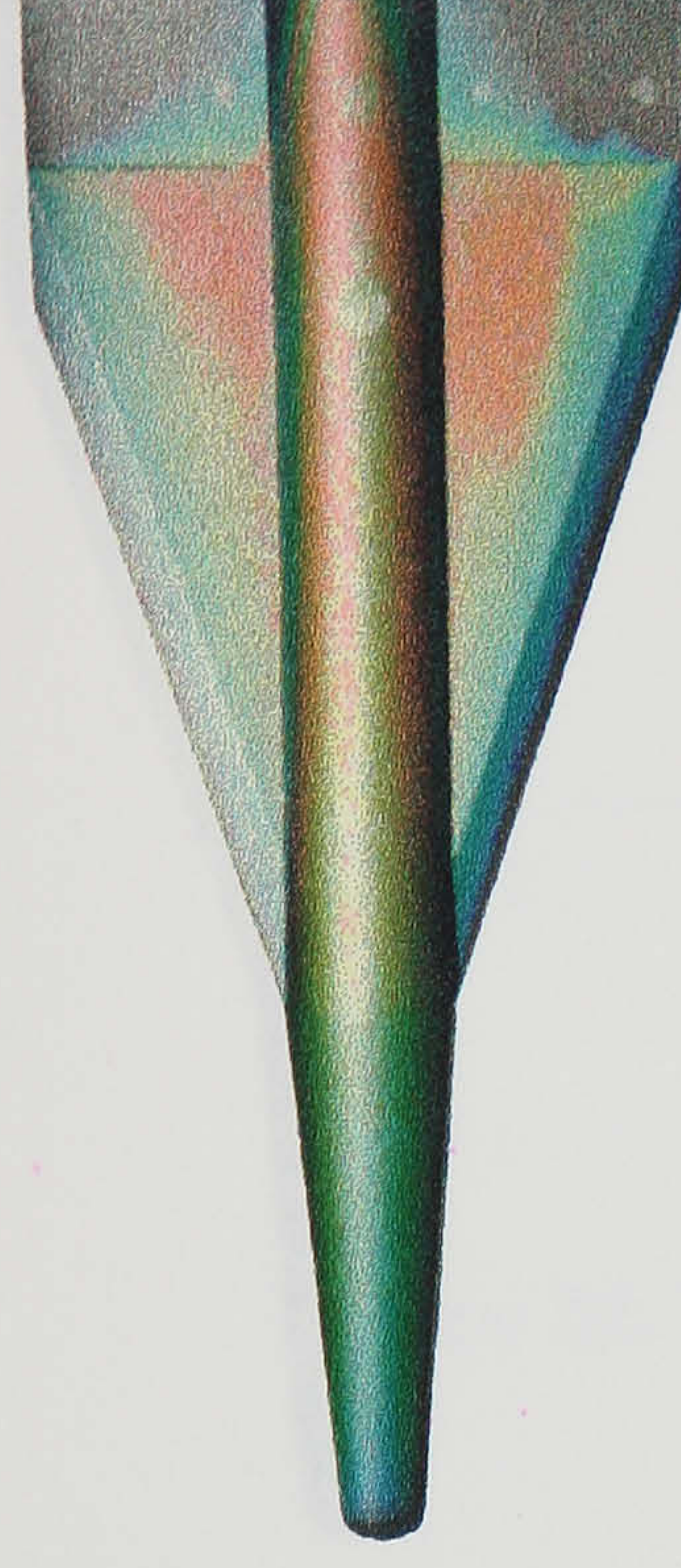


(d) $\alpha = 10^\circ$ and $\beta = 5^\circ$

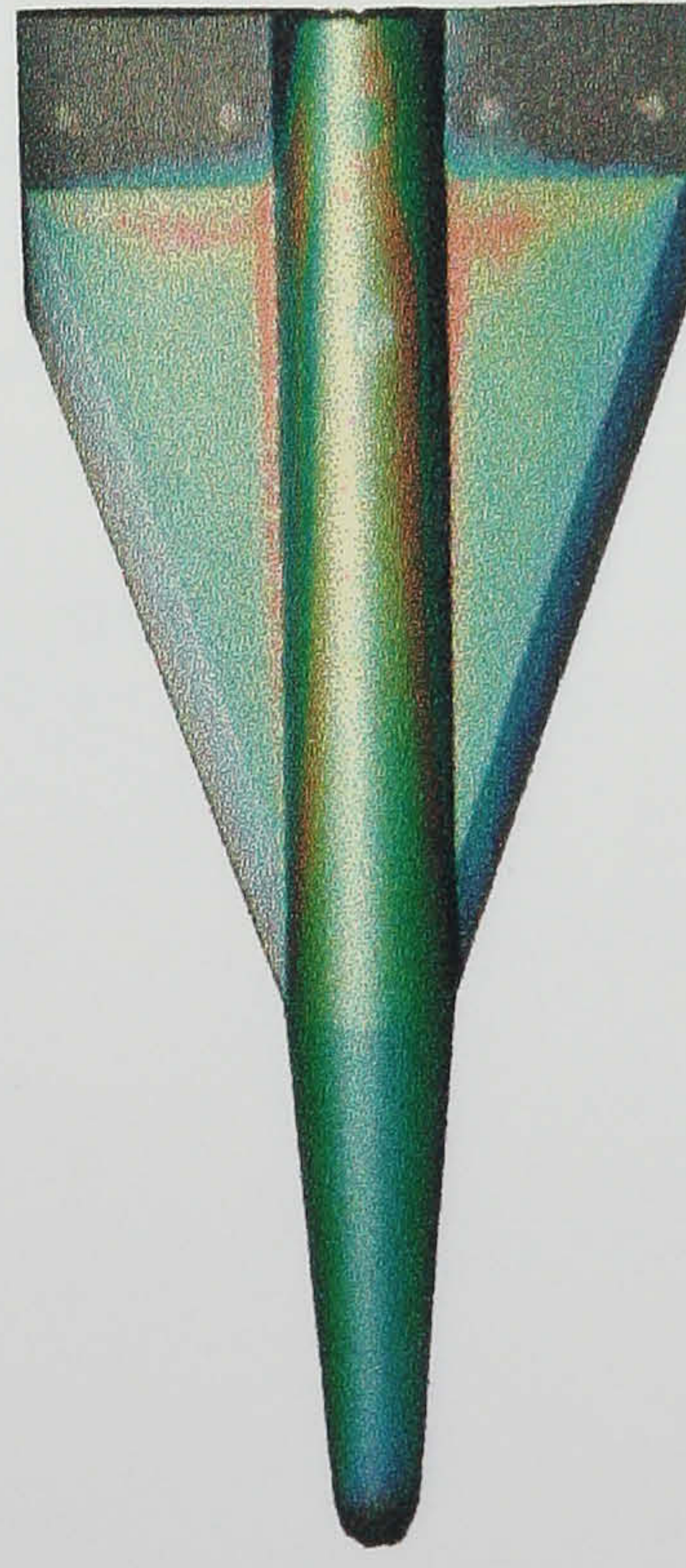
Figure 29. Liquid crystal pictures indicating variation of surface temperature distribution over windward side of the wing-body before and after the tunnel firing.



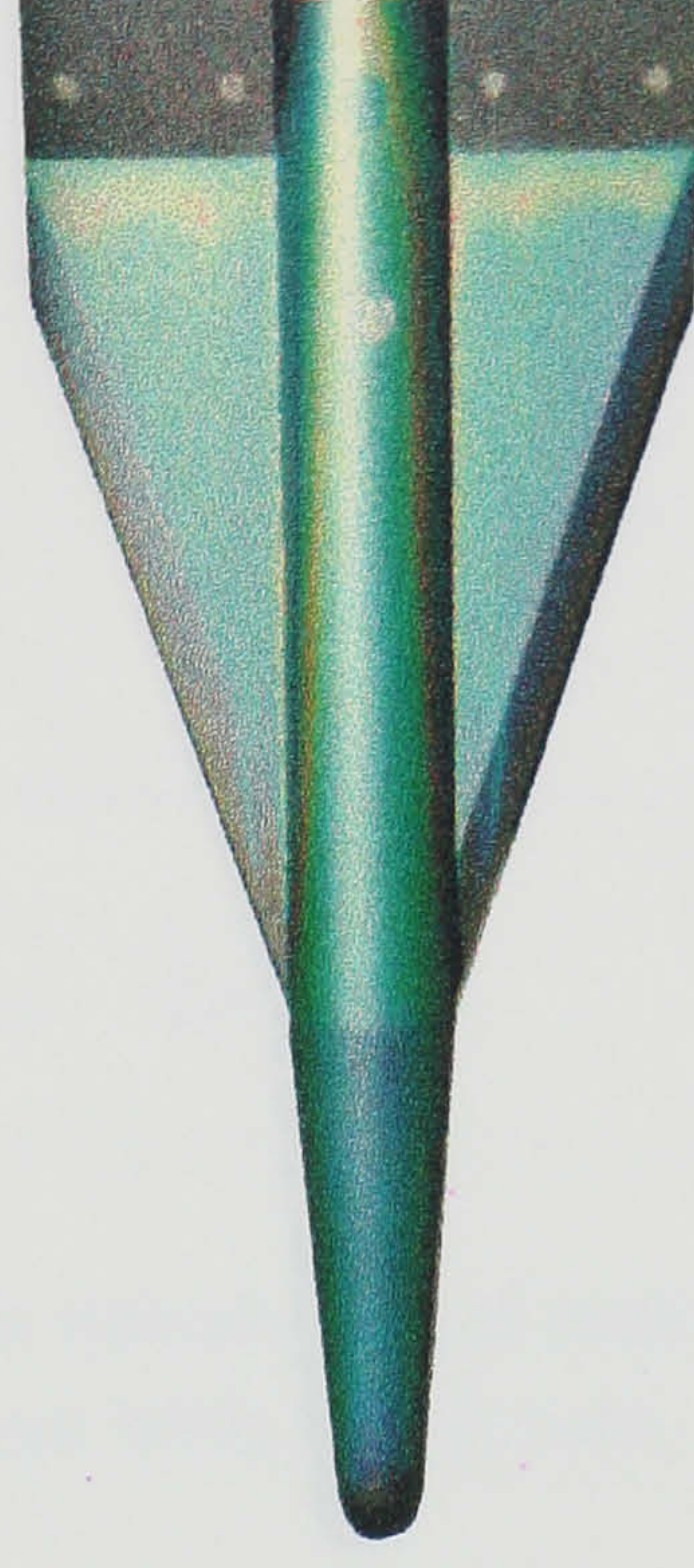
(a) $\alpha = 0^\circ$ and $\beta = 0^\circ$



(b) $\alpha = 0^\circ$ and $\beta = 15^\circ$

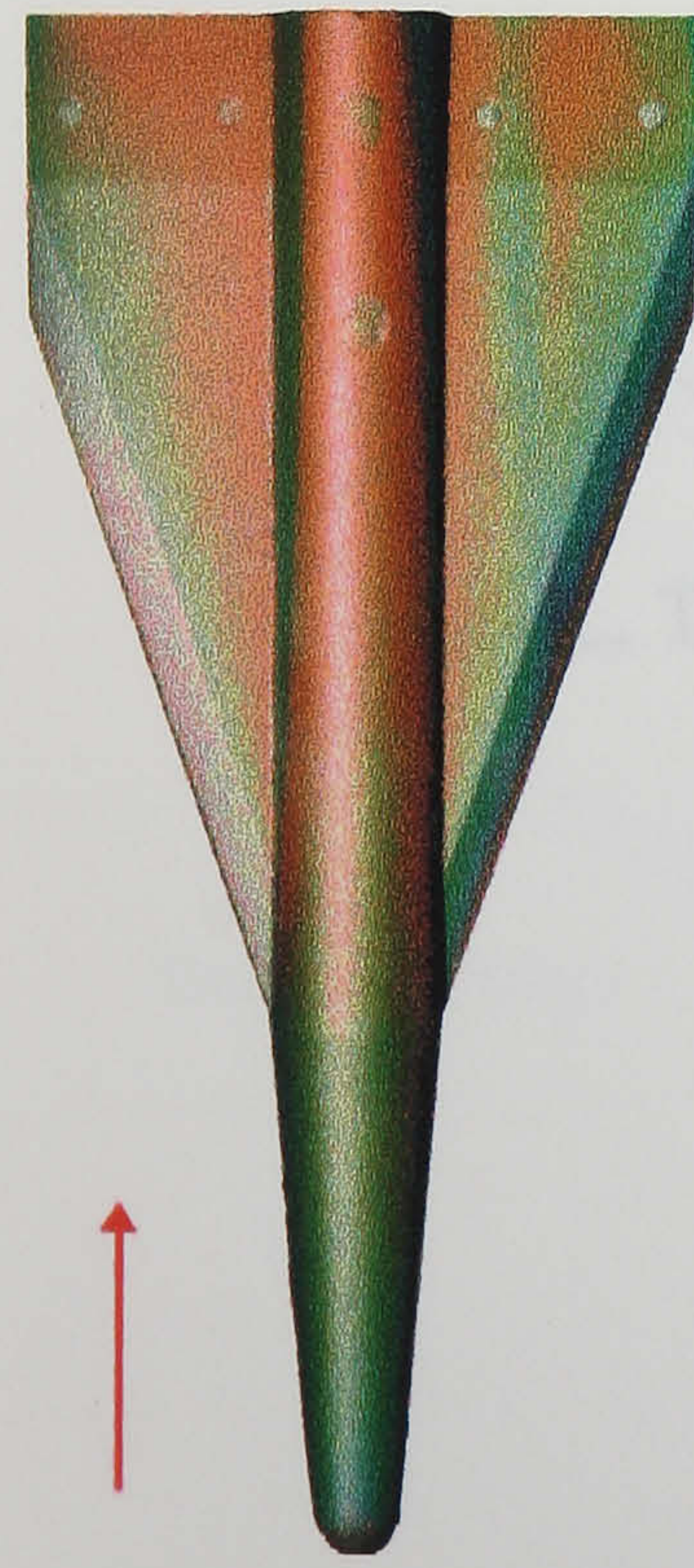


(c) $\alpha = 5^\circ$ and $\beta = 15^\circ$

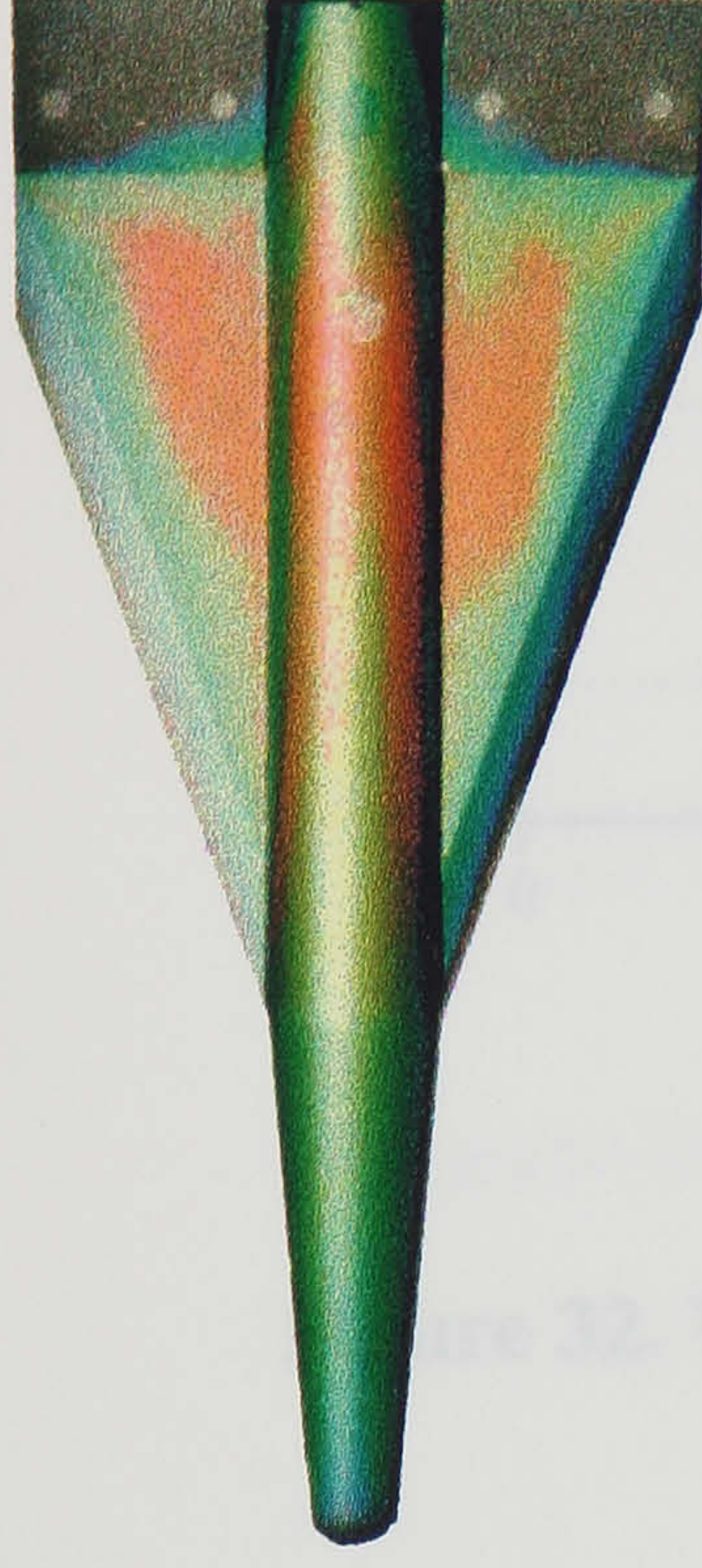


(d) $\alpha = 10^\circ$ and $\beta = 15^\circ$

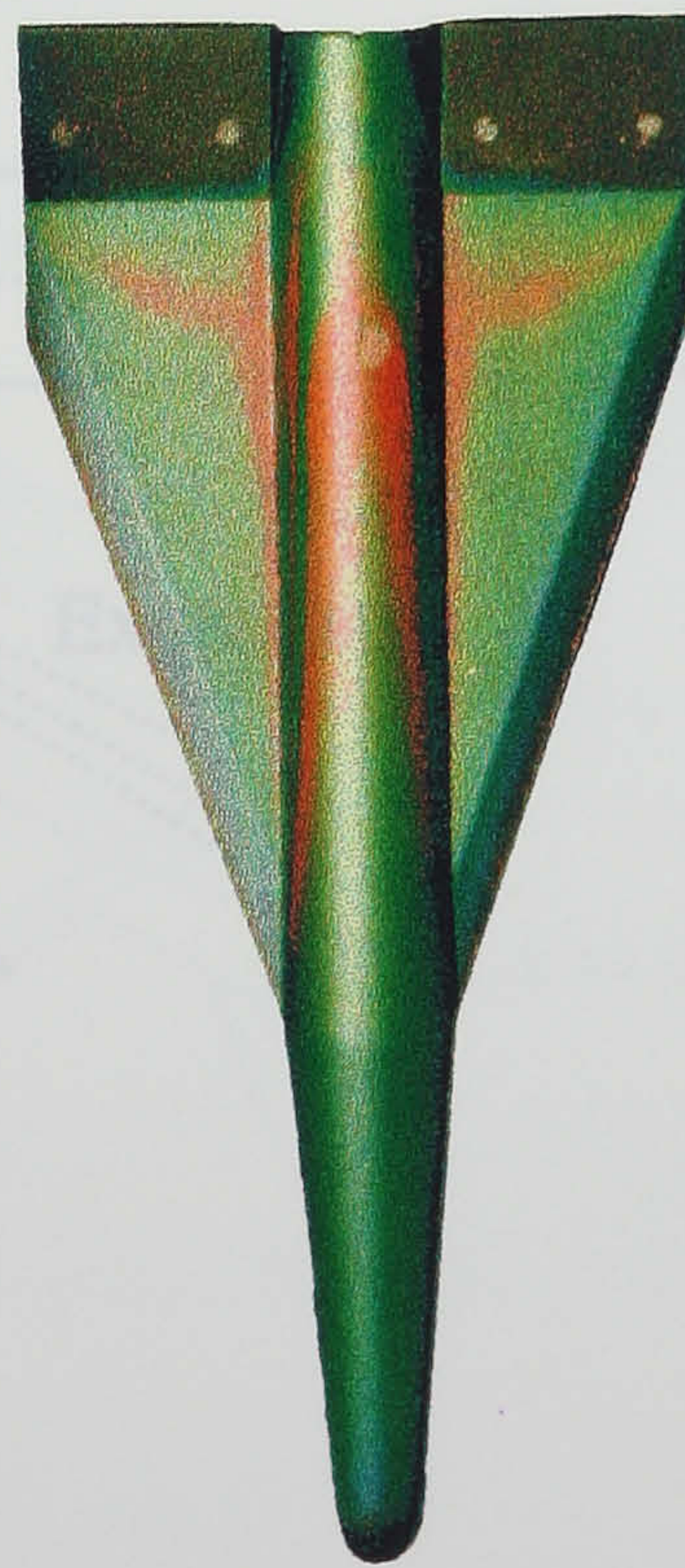
Figure 30. Liquid crystal pictures indicating variation of surface temperature distribution over windward side of the wing-body with a change in α for $\beta = 15^\circ$.



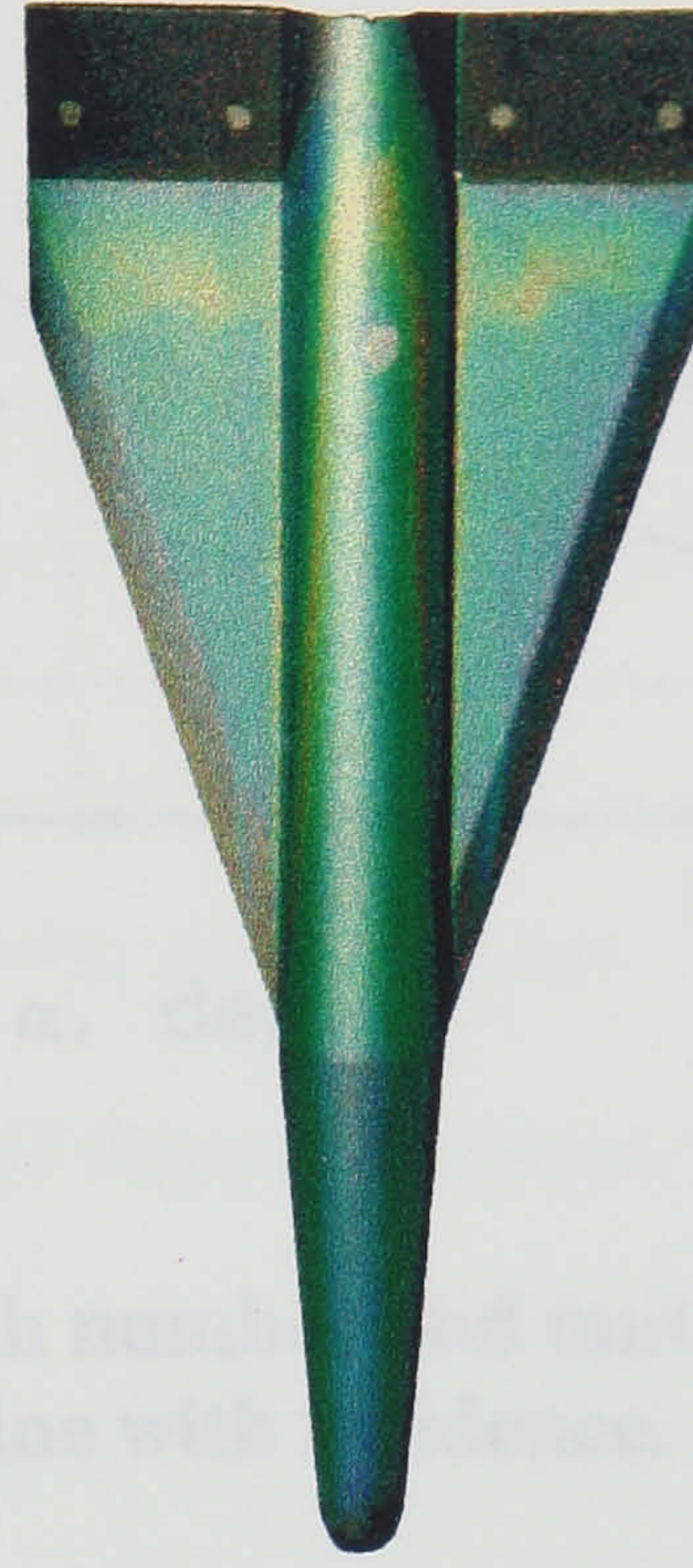
(a) $\alpha = 0^\circ$ and $\beta = 0^\circ$



(b) $\alpha = 0^\circ$ and $\beta = 25^\circ$



(c) $\alpha = 5^\circ$ and $\beta = 25^\circ$



(d) $\alpha = 10^\circ$ and $\beta = 25^\circ$

Figure 31. Liquid crystal pictures indicating variation of surface temperature distribution over windward side of the wing-body with a change in α for $\beta = 25^\circ$.

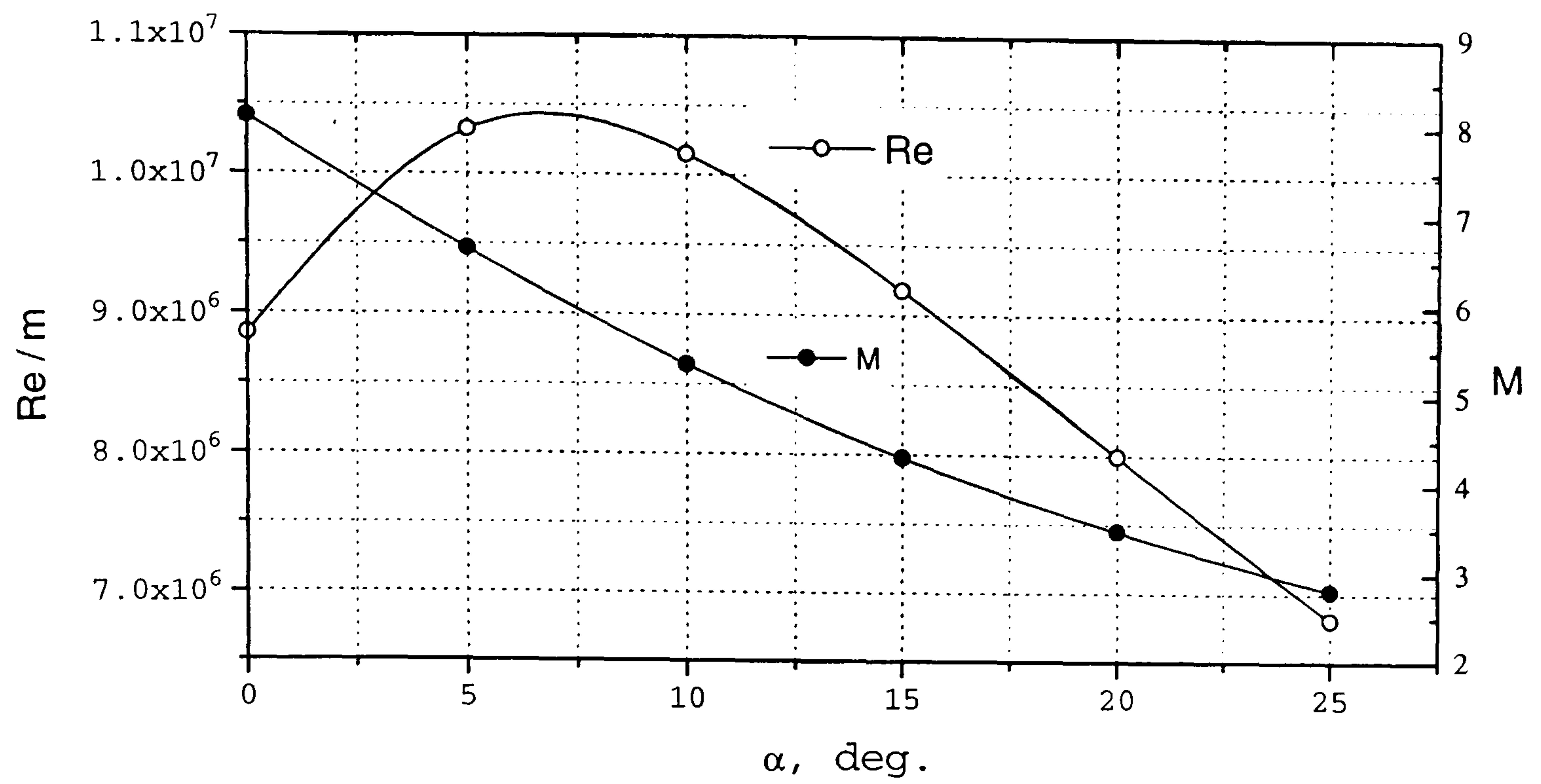


Figure 32. Variation of local Mach number and unit Reynolds number at the hinge-line with incidence.

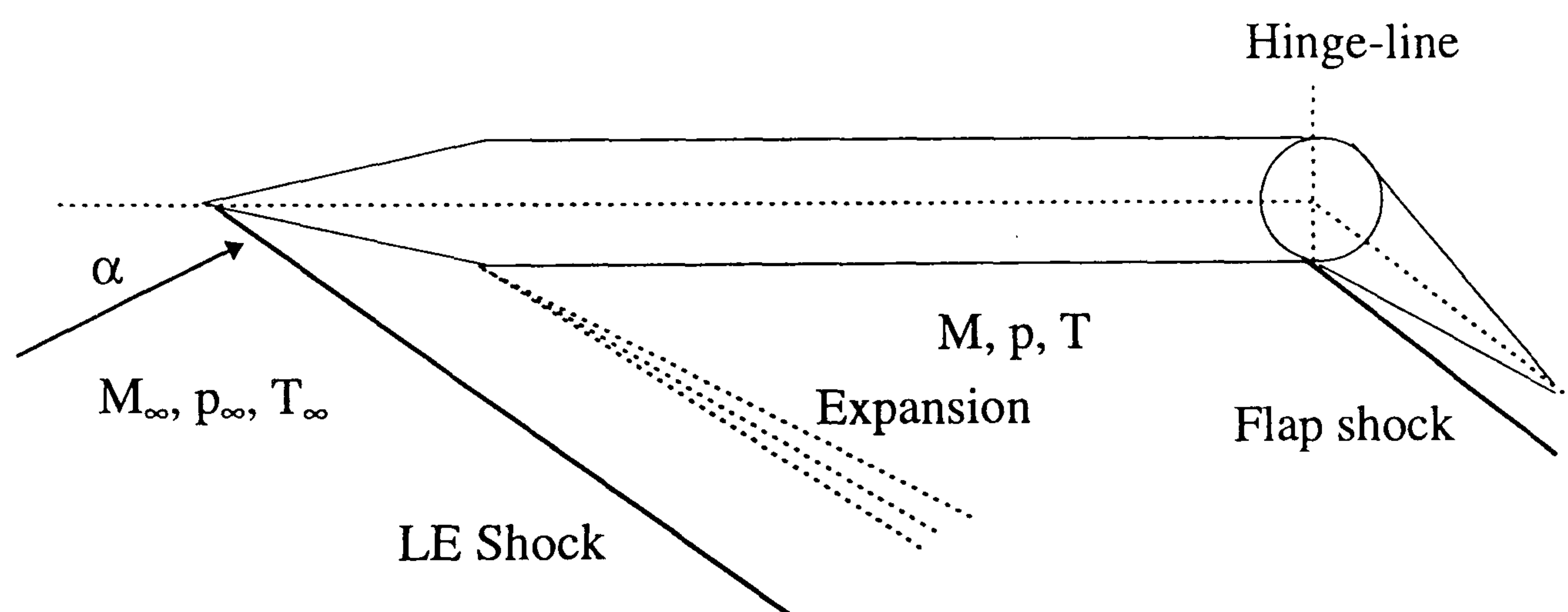


Figure 33. Flow field for calculating inviscid flow properties ahead of hinge-line.

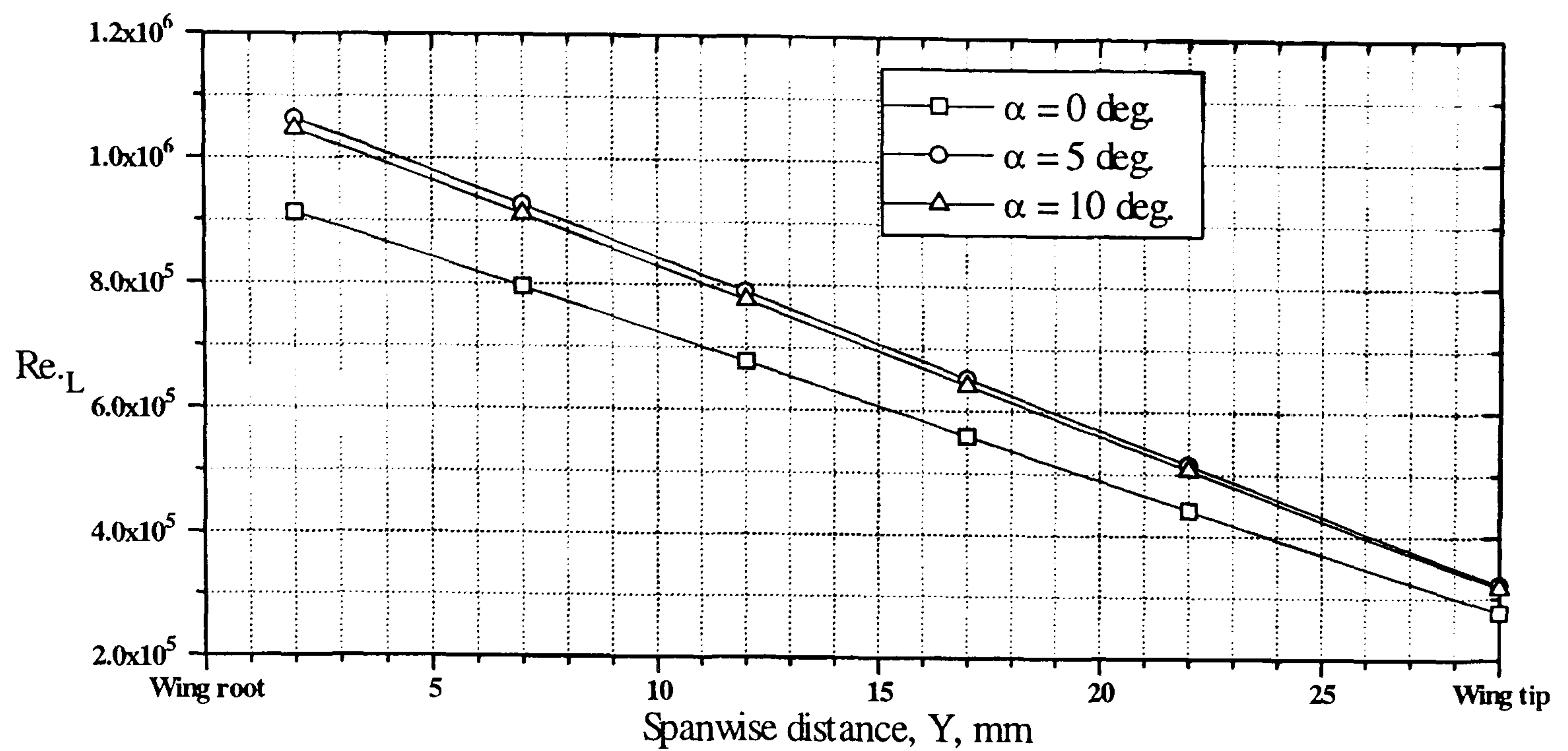


Figure 34. Spanwise variation of Reynolds number with incidence.

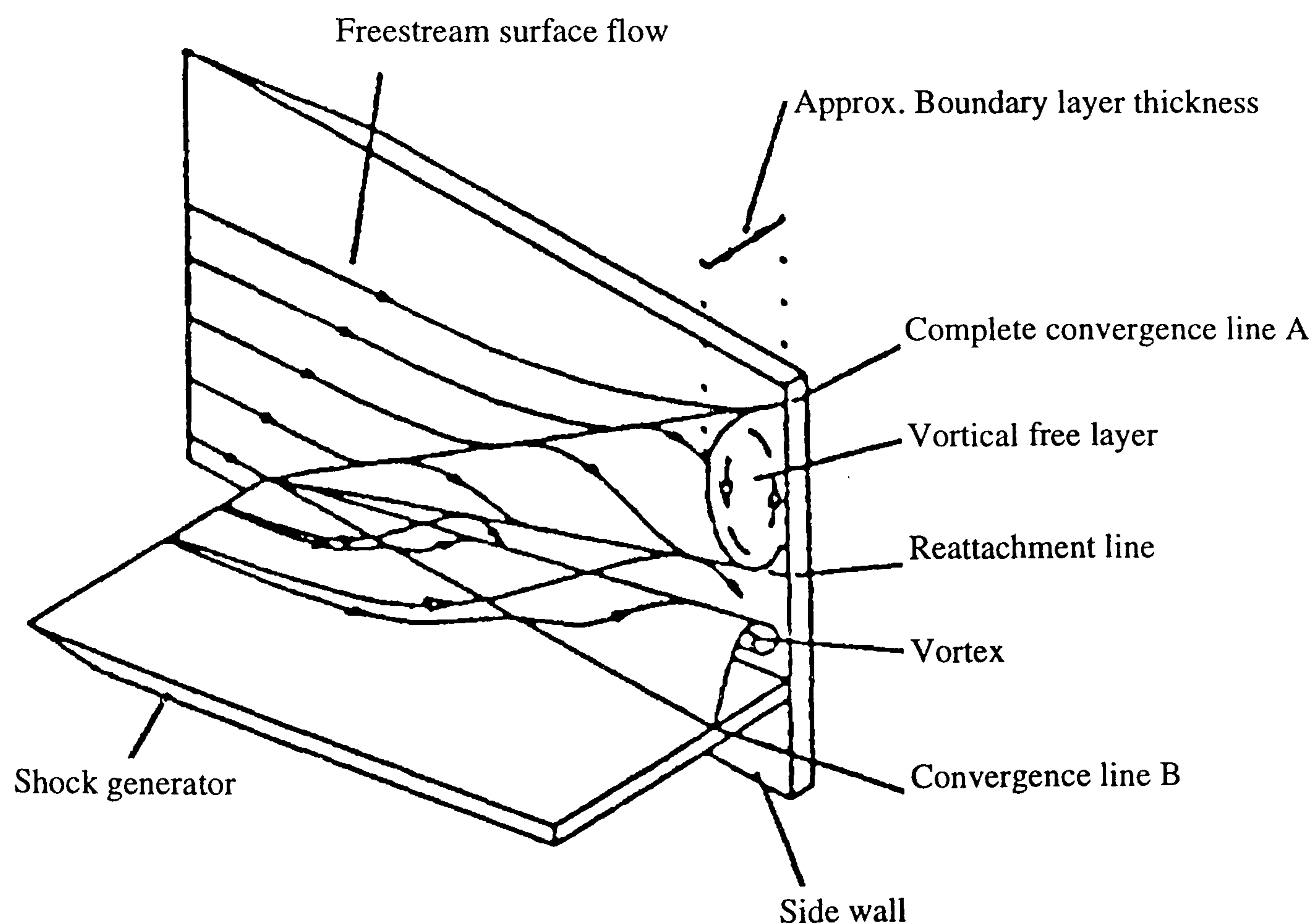
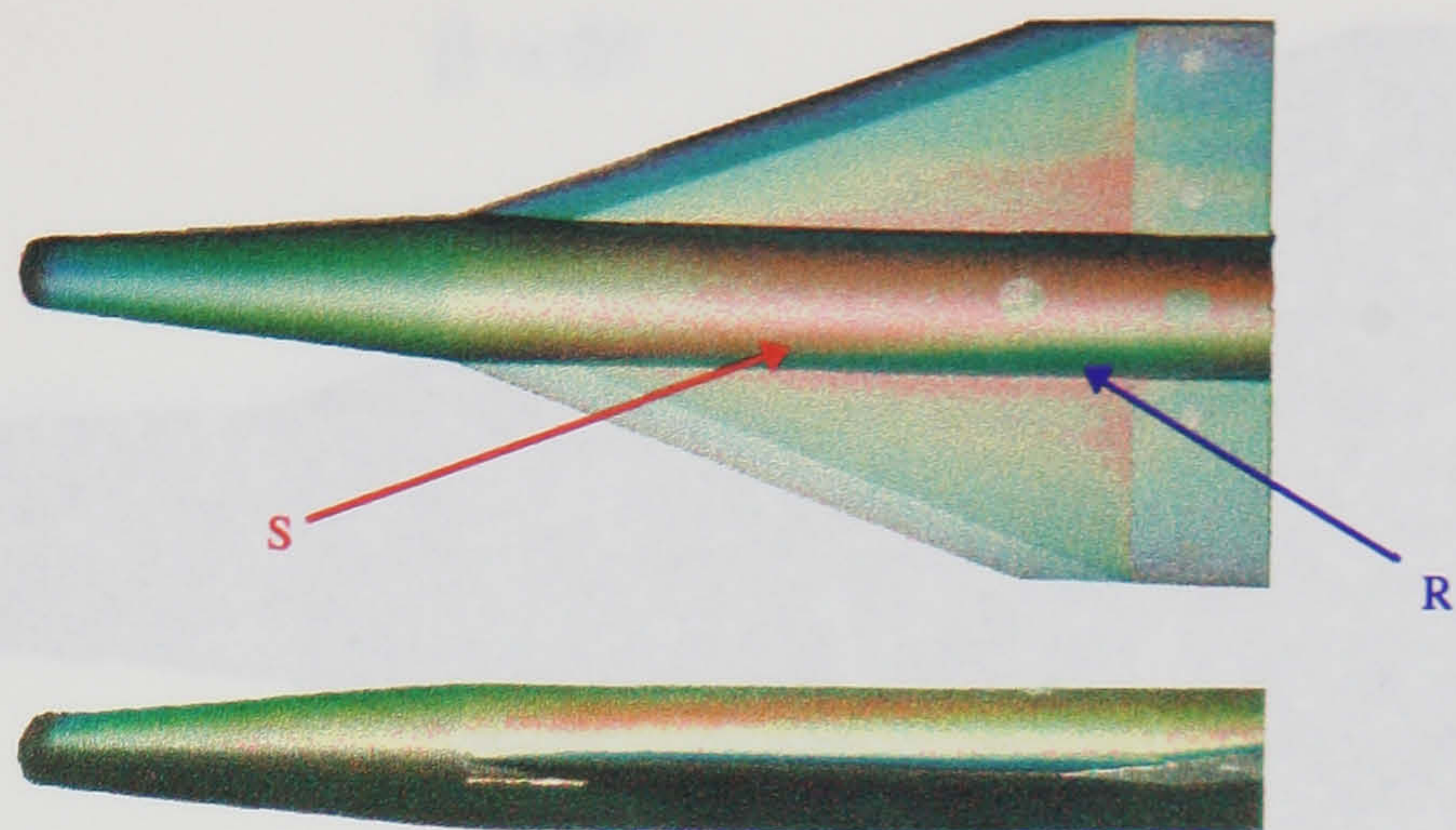
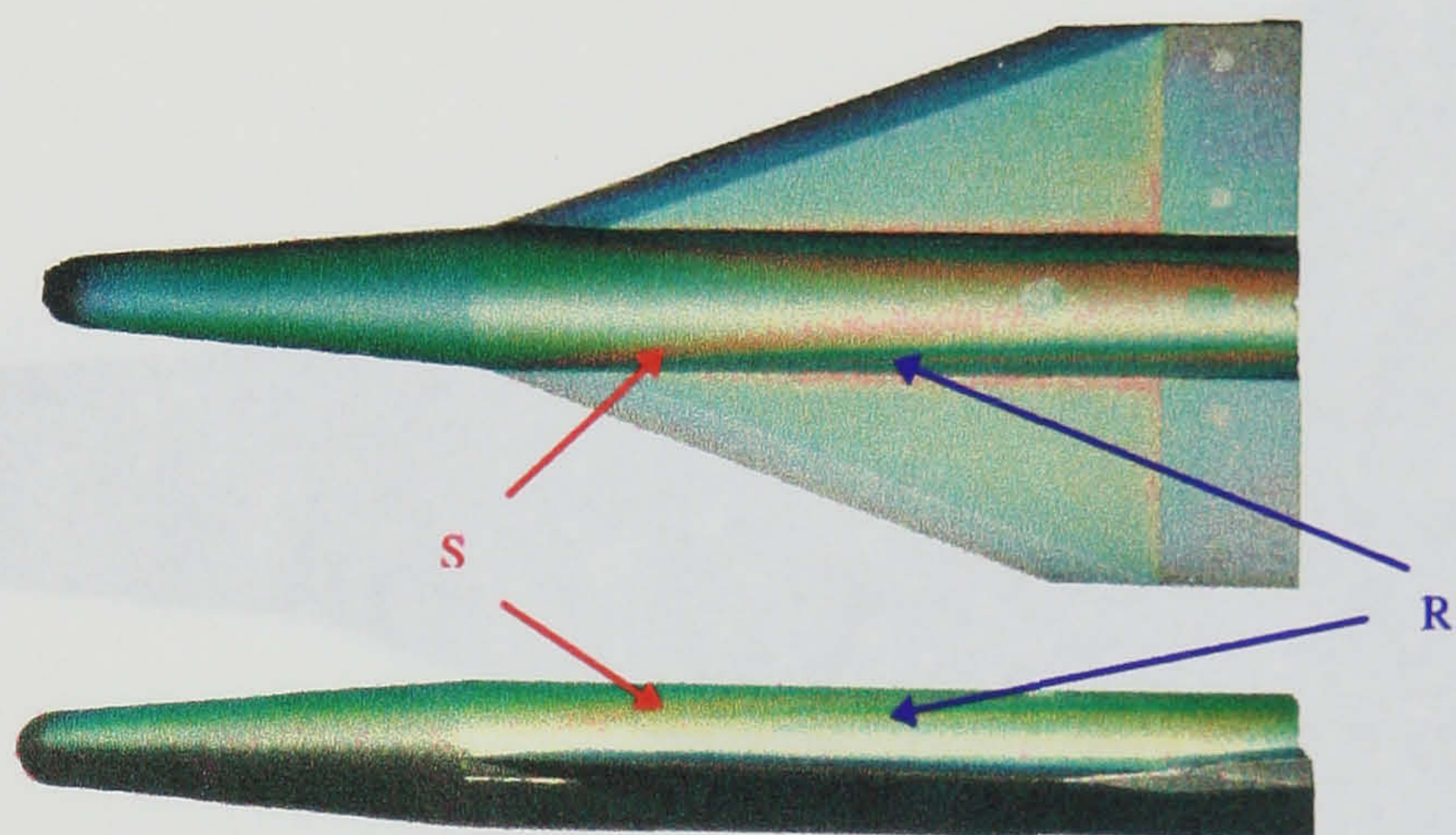


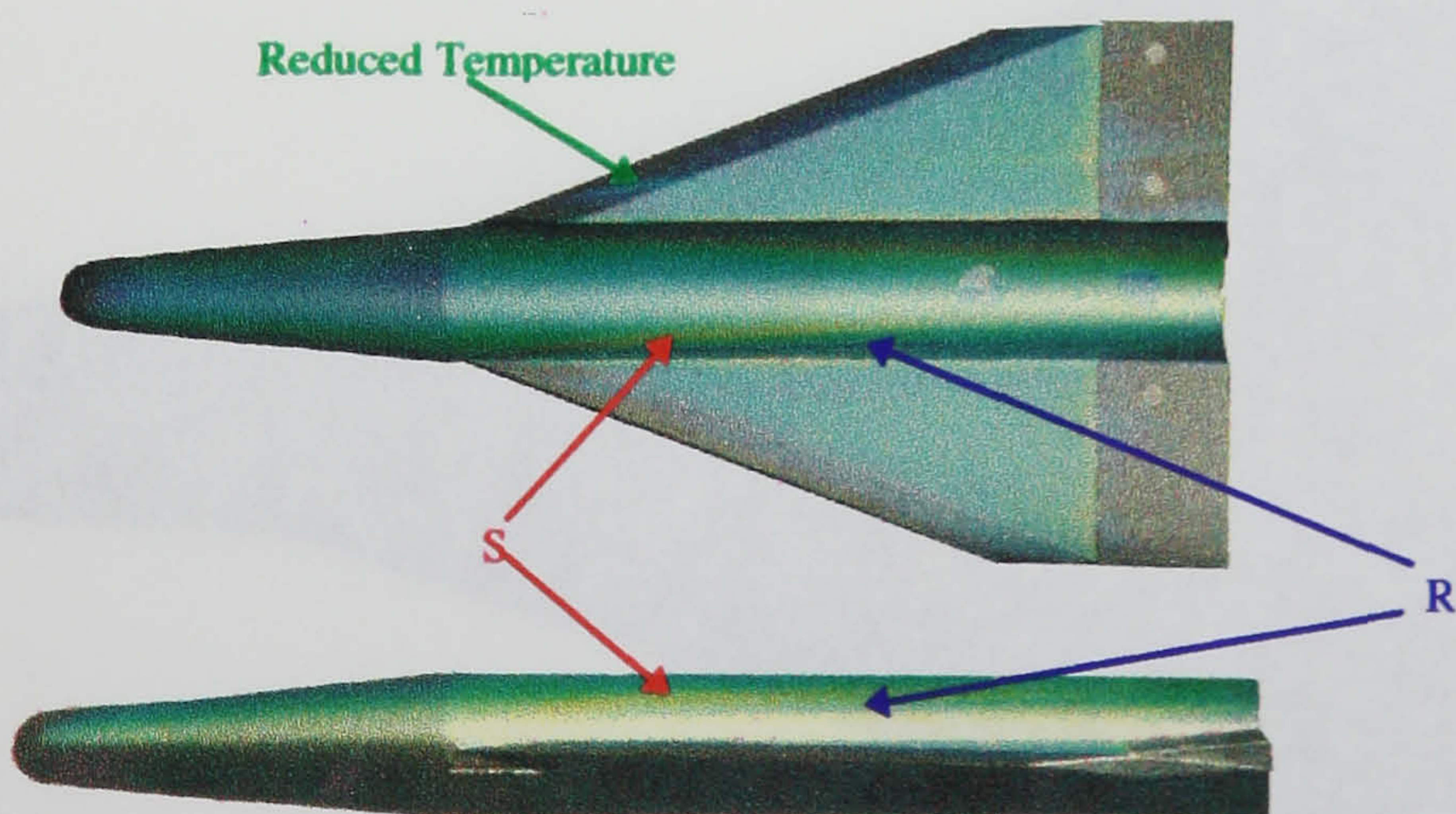
Figure 35. Glancing interaction from a sharp wedge, Kubota and Stollery⁽⁸⁴⁾.



(a) $\alpha = 0^\circ$, and $\beta = 5^\circ$



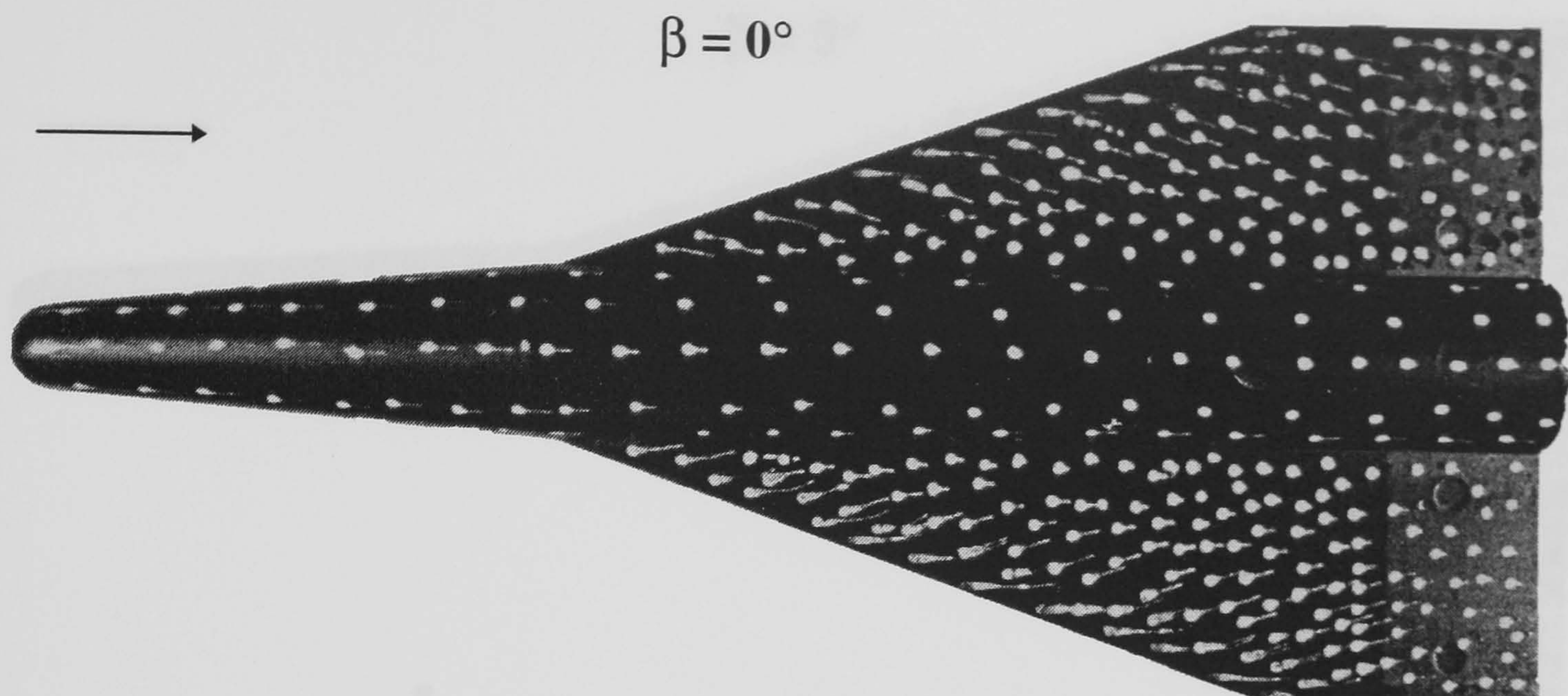
(b) $\alpha = 5^\circ$, and $\beta = 5^\circ$



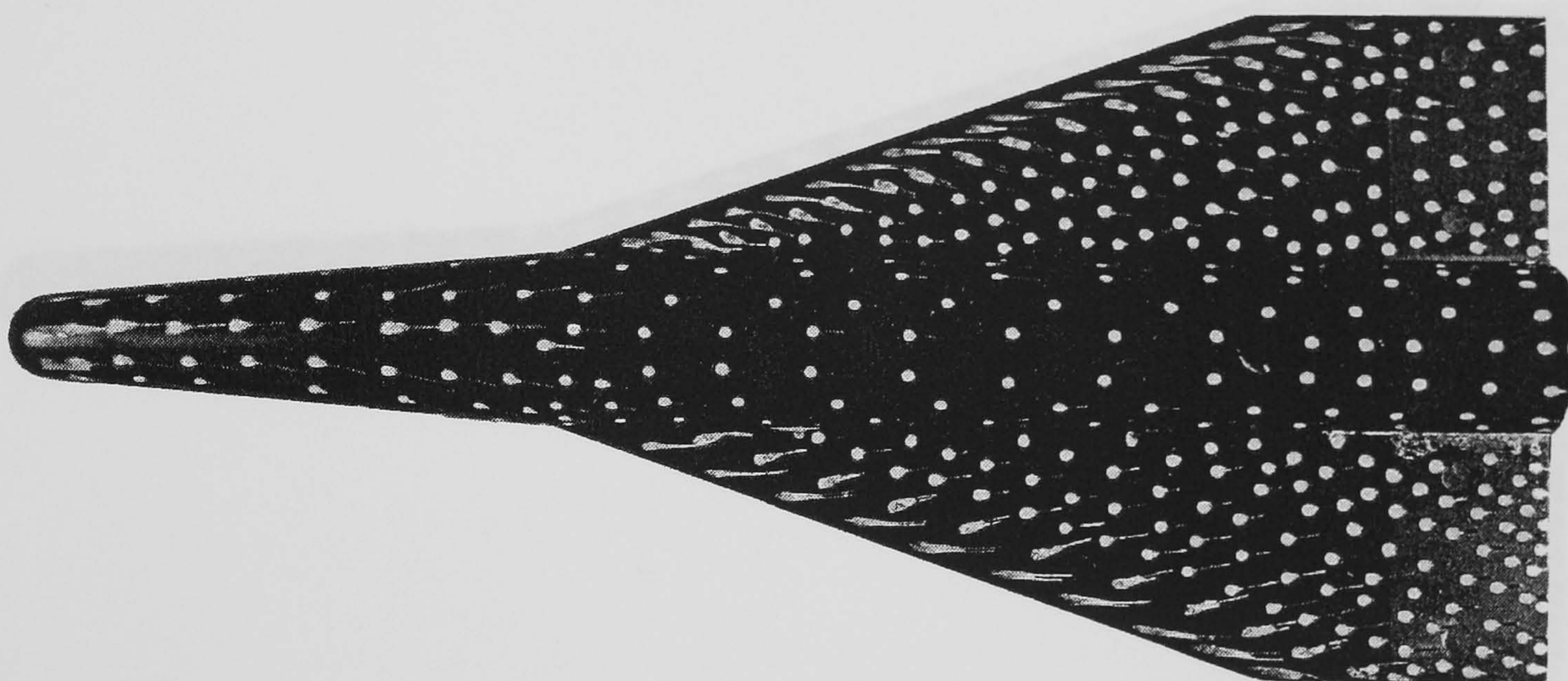
(c) $\alpha = 10^\circ$, and $\beta = 5^\circ$

S - Separation and **R- Reattachment**

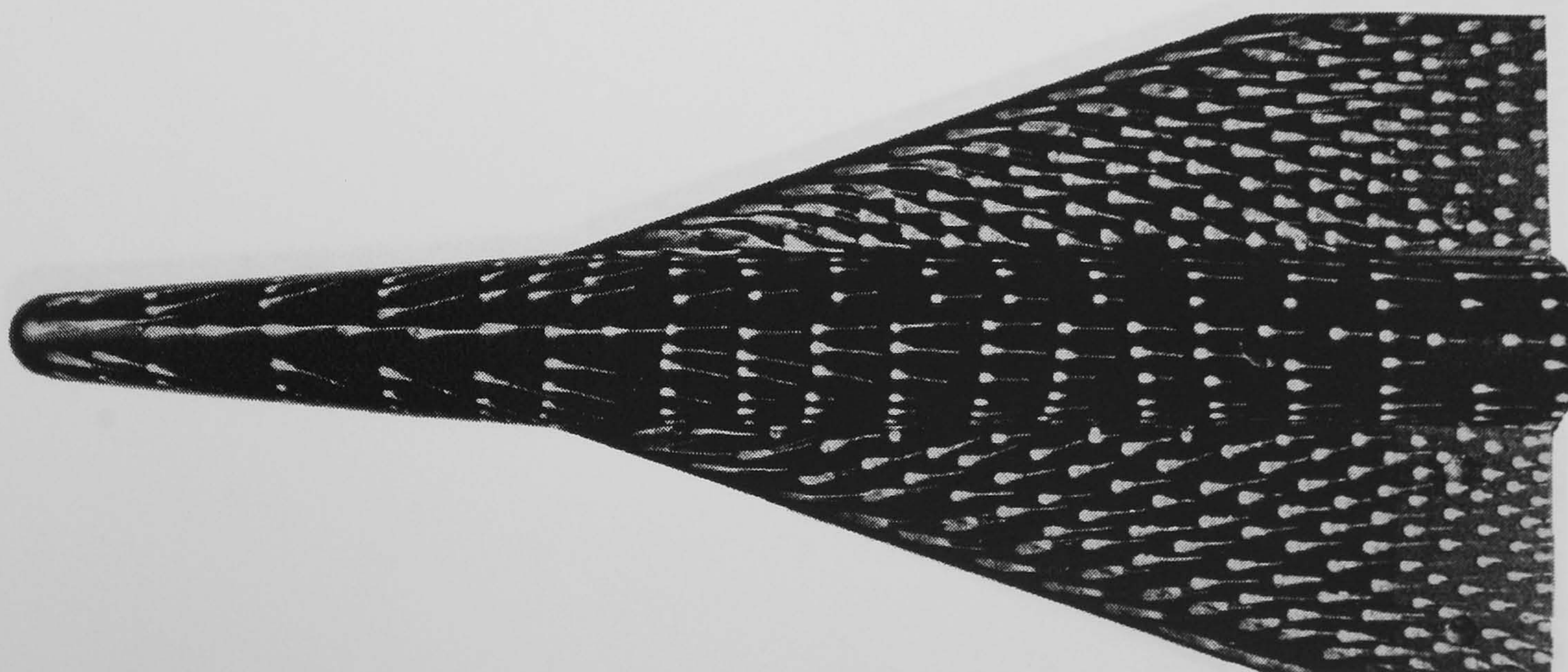
Figure 36. Liquid crystal pictures showing glancing interaction effects.



(a) $\alpha = 0^\circ$

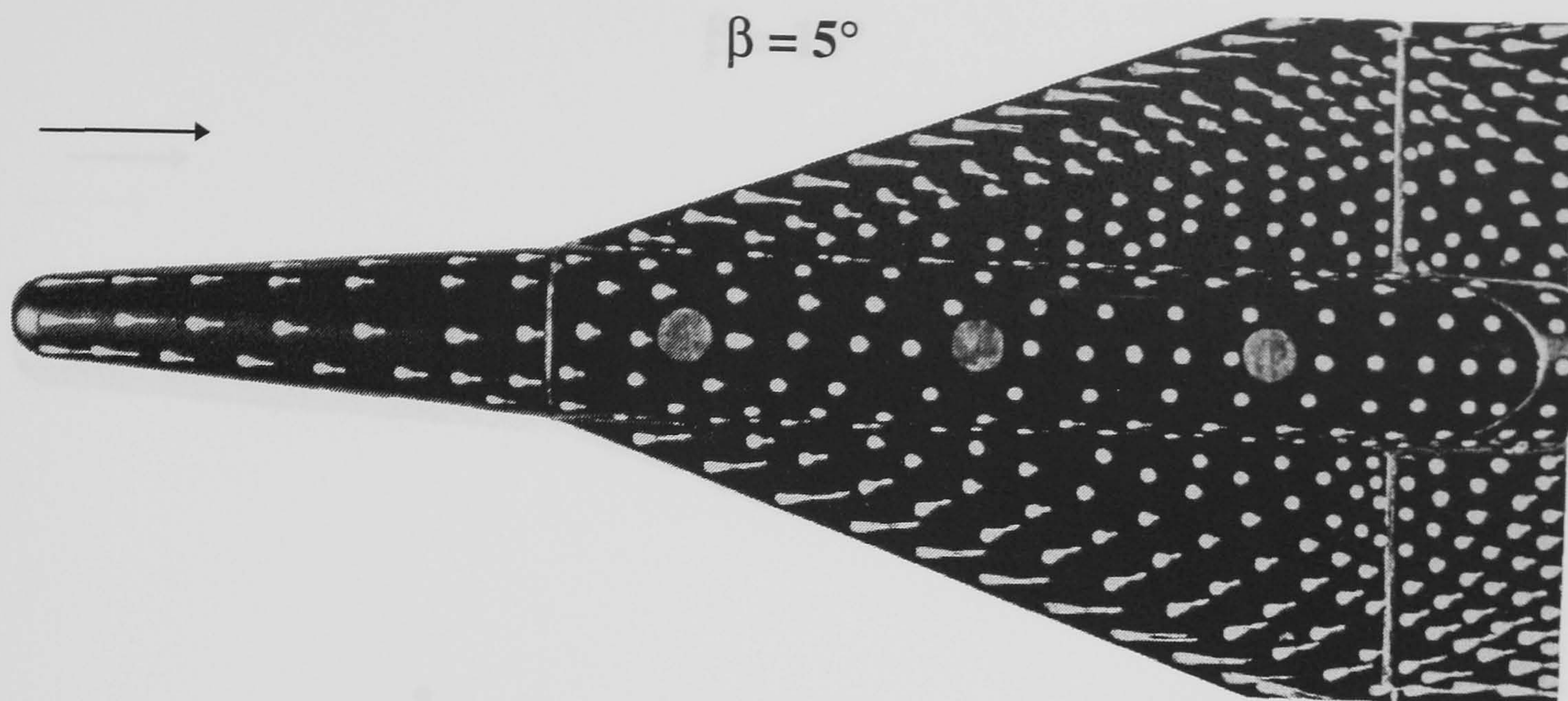


(b) $\alpha = 5^\circ$

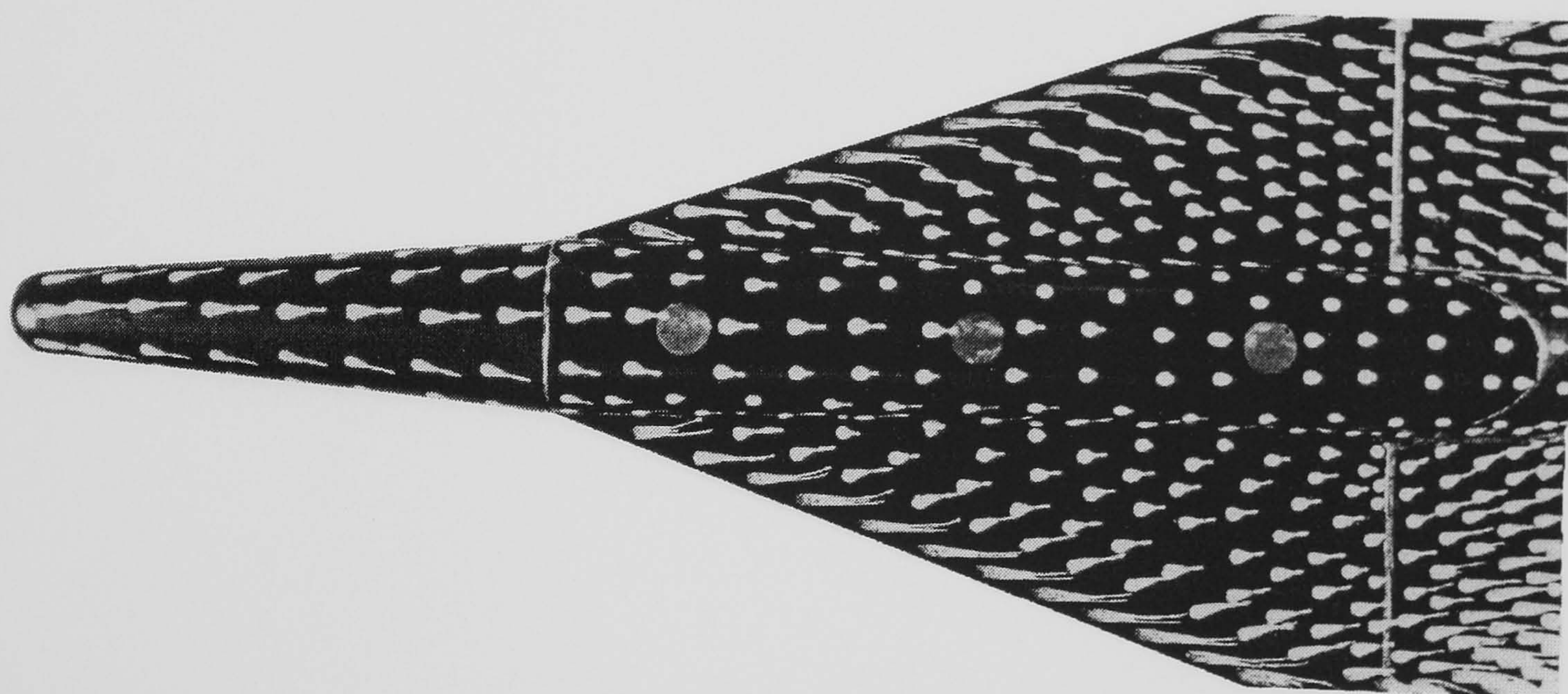


(c) $\alpha = 10^\circ$

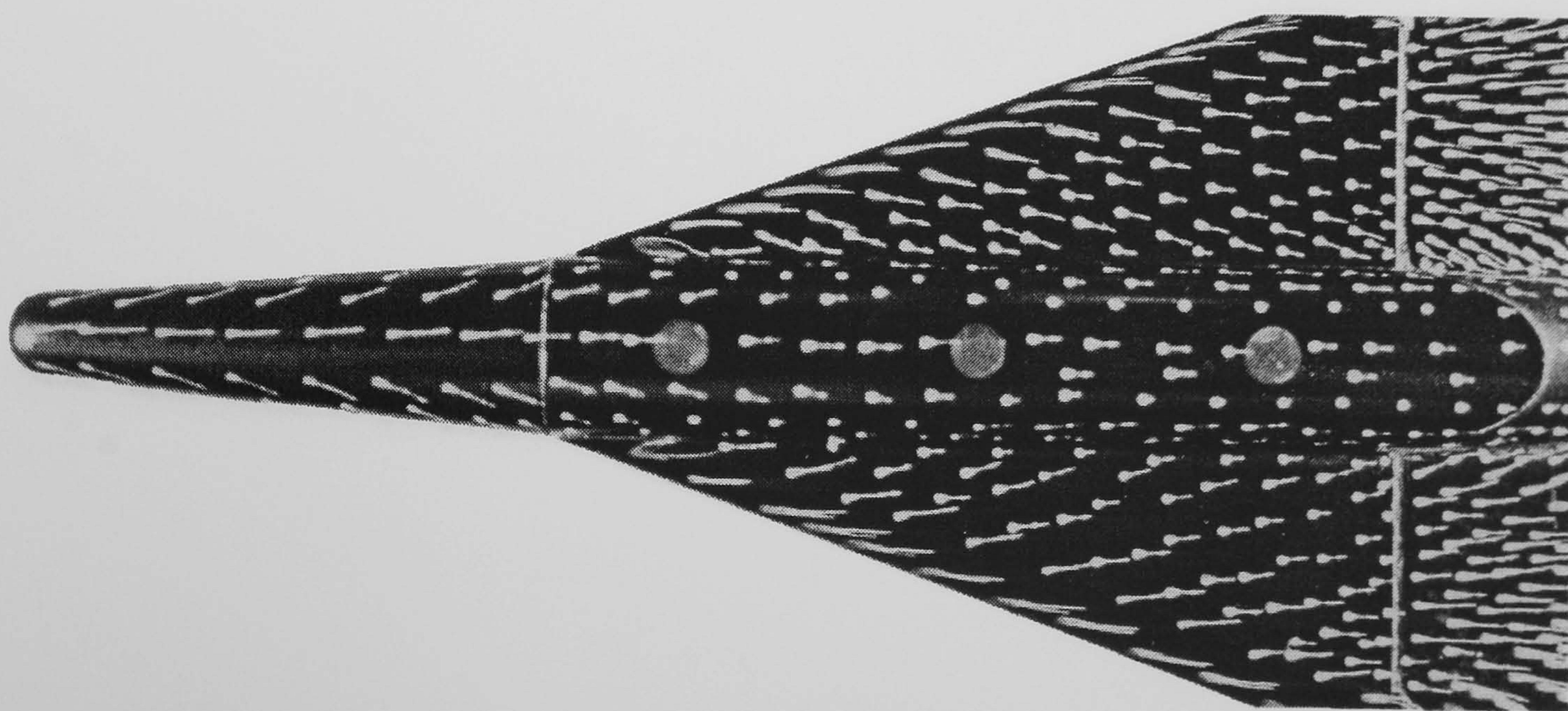
Figure 37. Oil-dot flow pictures showing attached flow over the lower surface with 0° flap deflection at incidence, $\alpha = 0^\circ$, 5° , and 10° .



(a) $\alpha = 0^\circ$

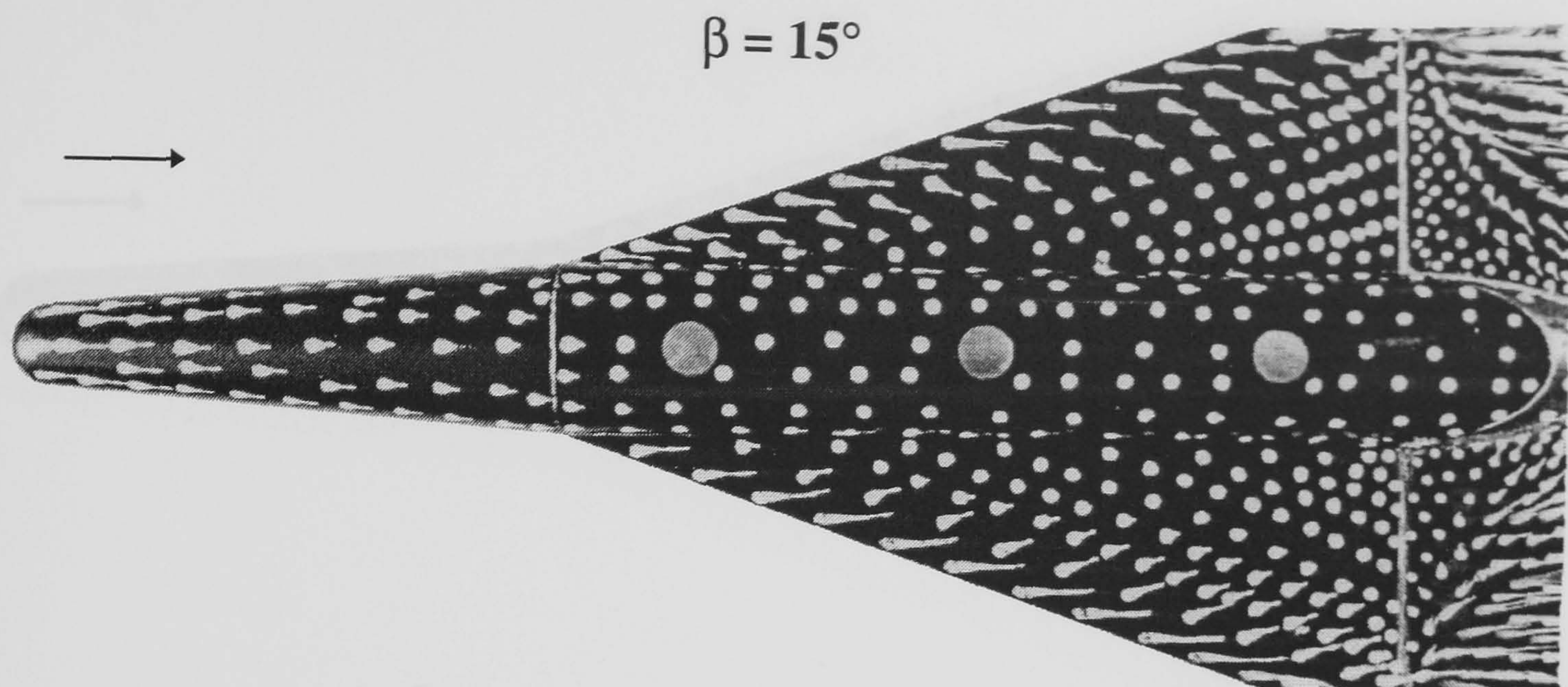


(b) $\alpha = 5^\circ$

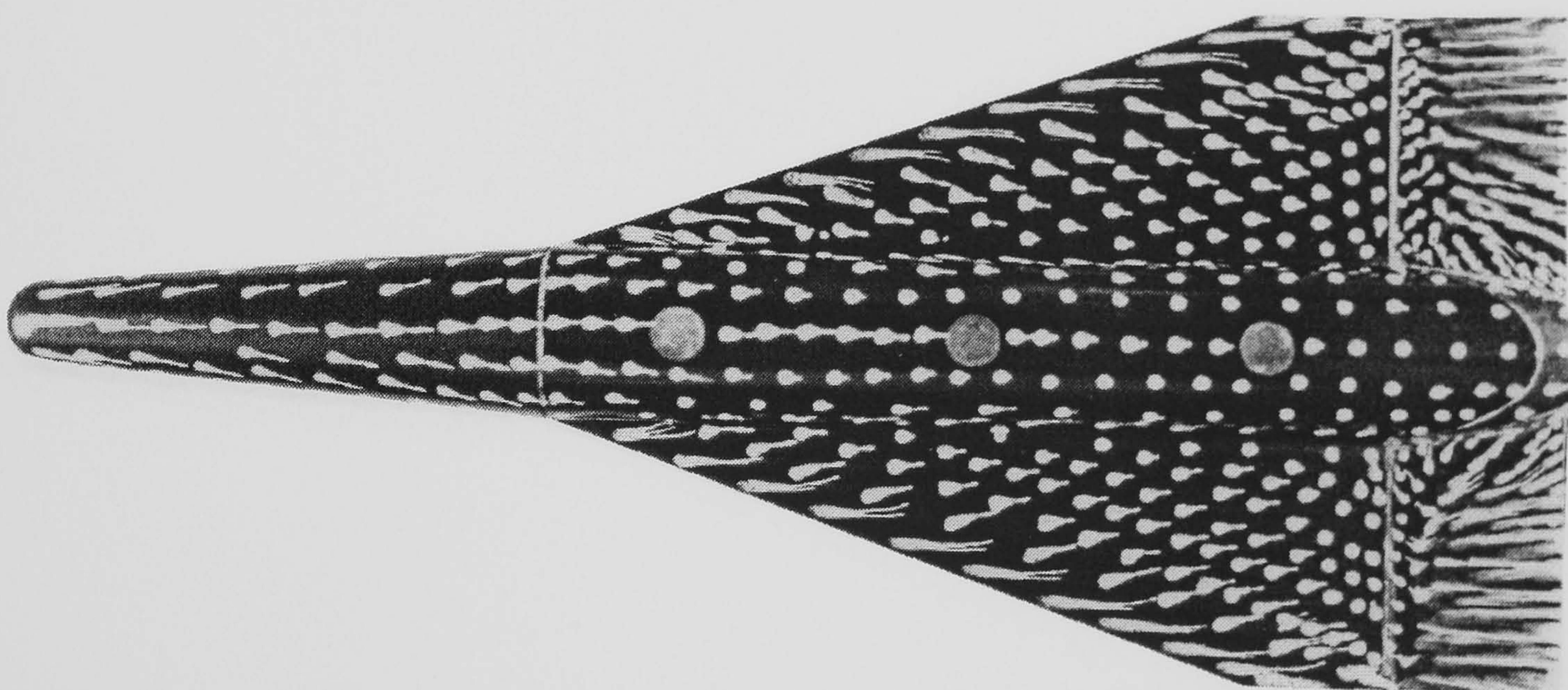


(c) $\alpha = 10^\circ$

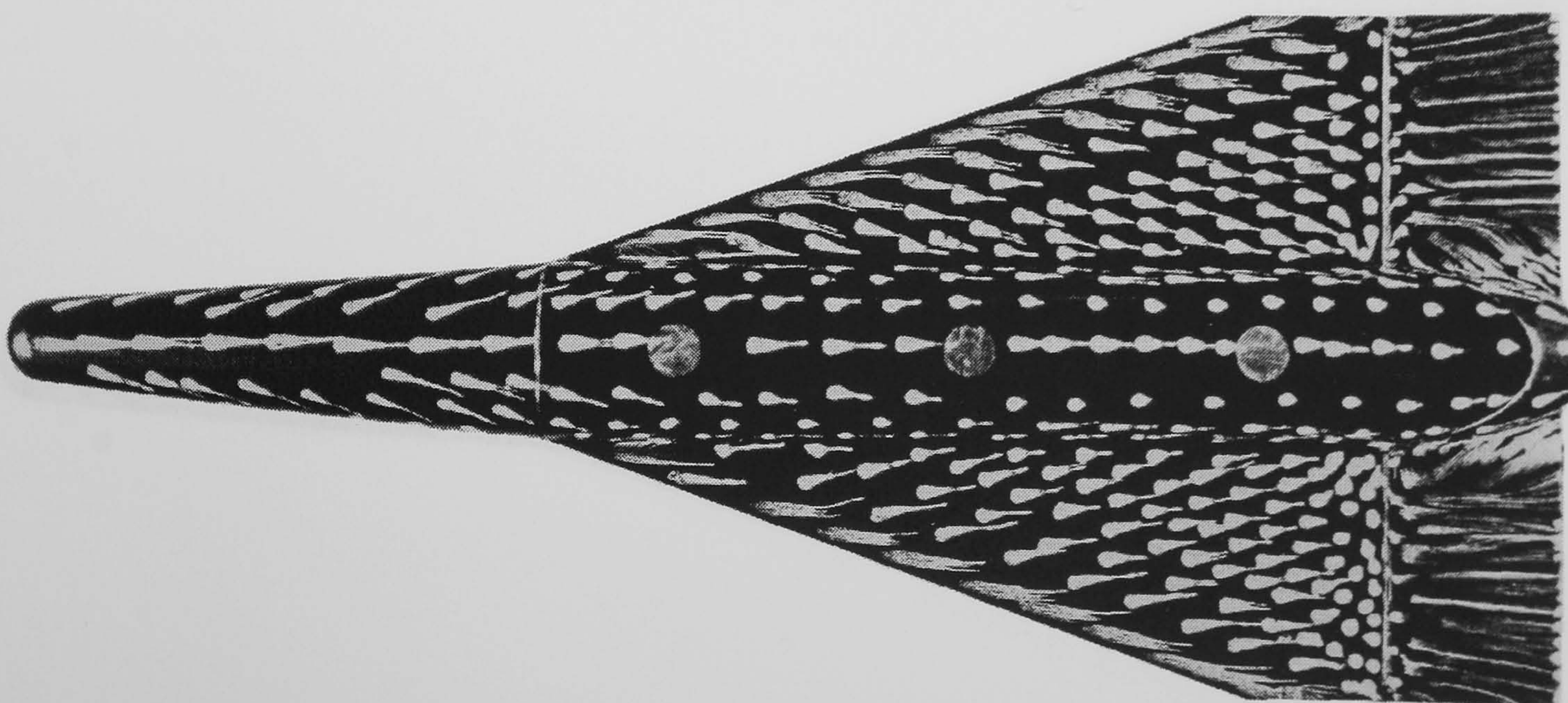
Figure 38. Oil-dot flow pictures showing attached flow with 5° flap deflection at incidence, $\alpha = 0^\circ$, 5° , and 10° .



(a) $\alpha = 0^\circ$

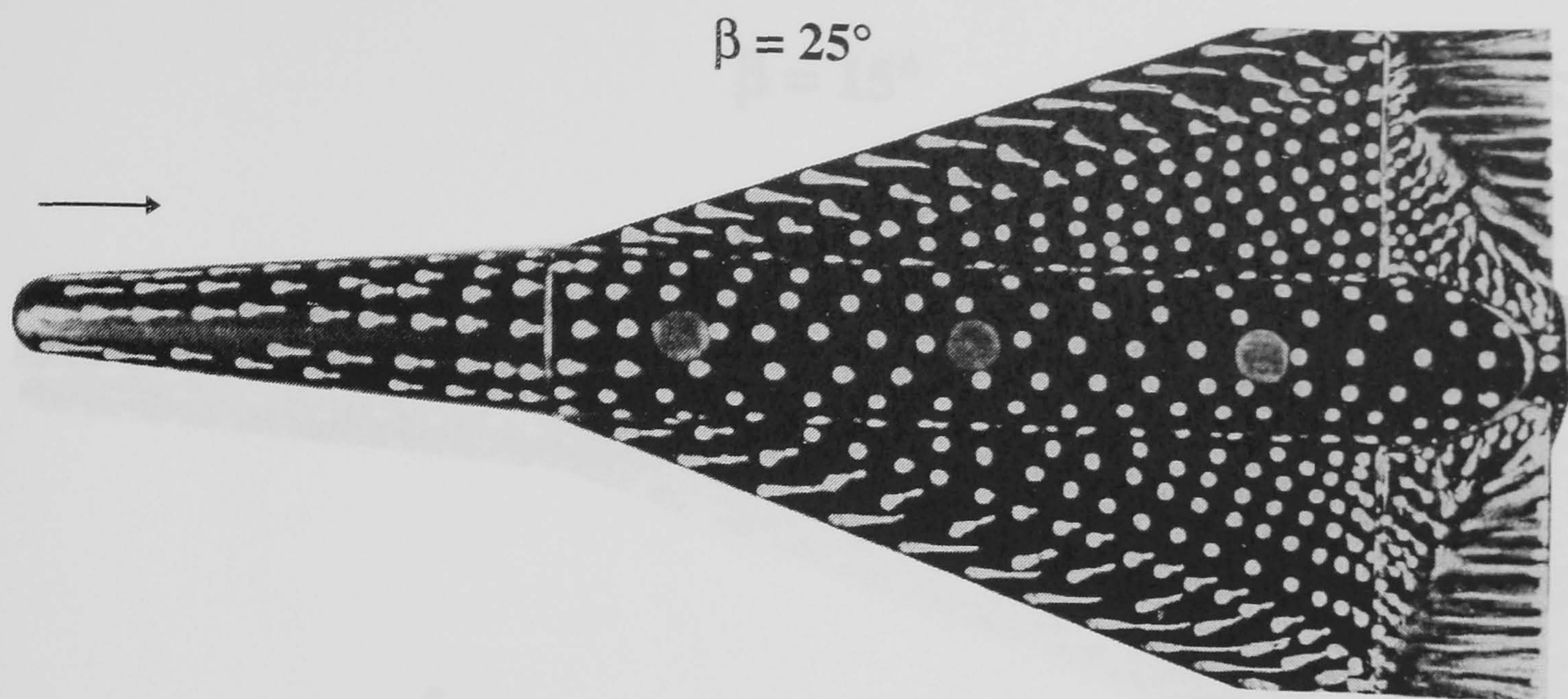


(b) $\alpha = 5^\circ$

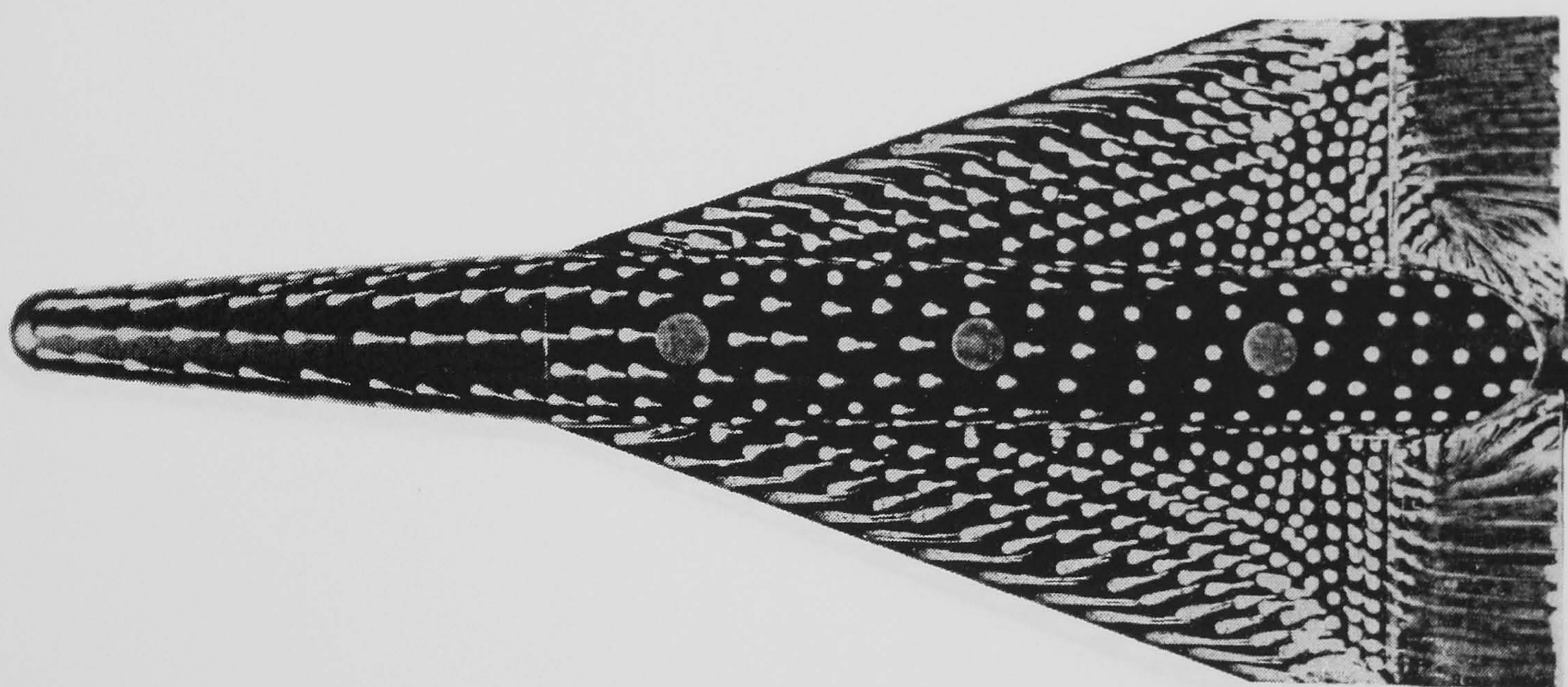


(c) $\alpha = 10^\circ$

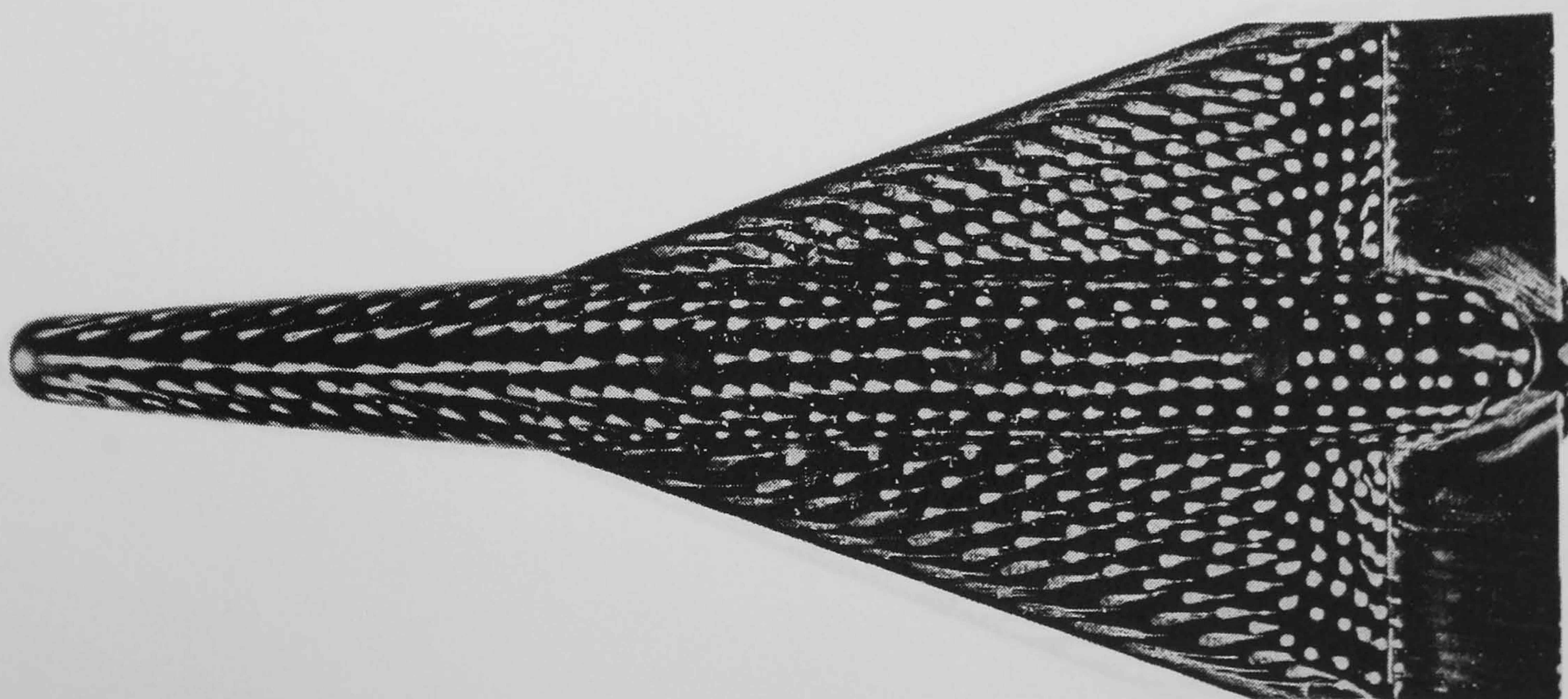
Figure 39. Oil-dot flow pictures showing reduction in the separated flow region for 15° flap deflection as the incidence increases from 0° to 10° .



(a) $\alpha = 0^\circ$

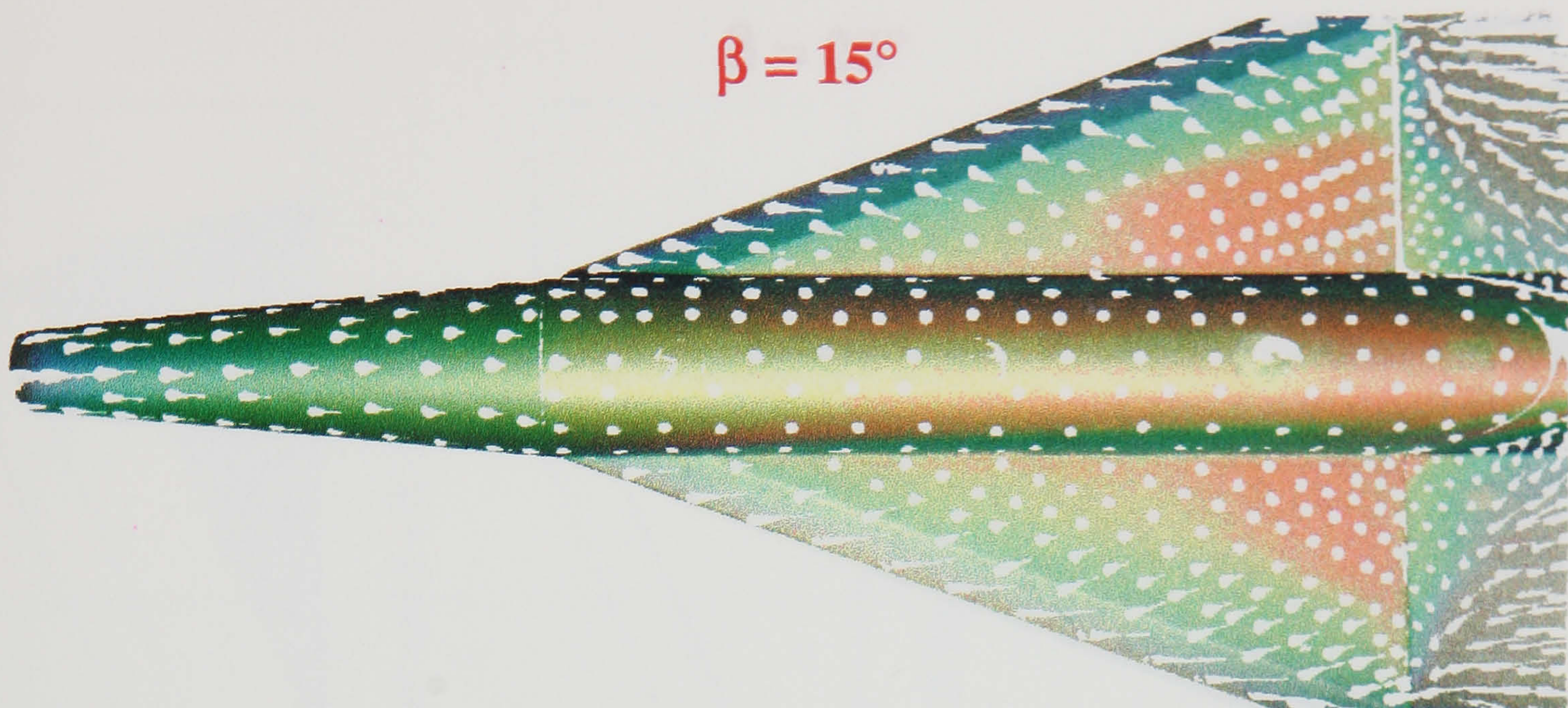


(b) $\alpha = 5^\circ$

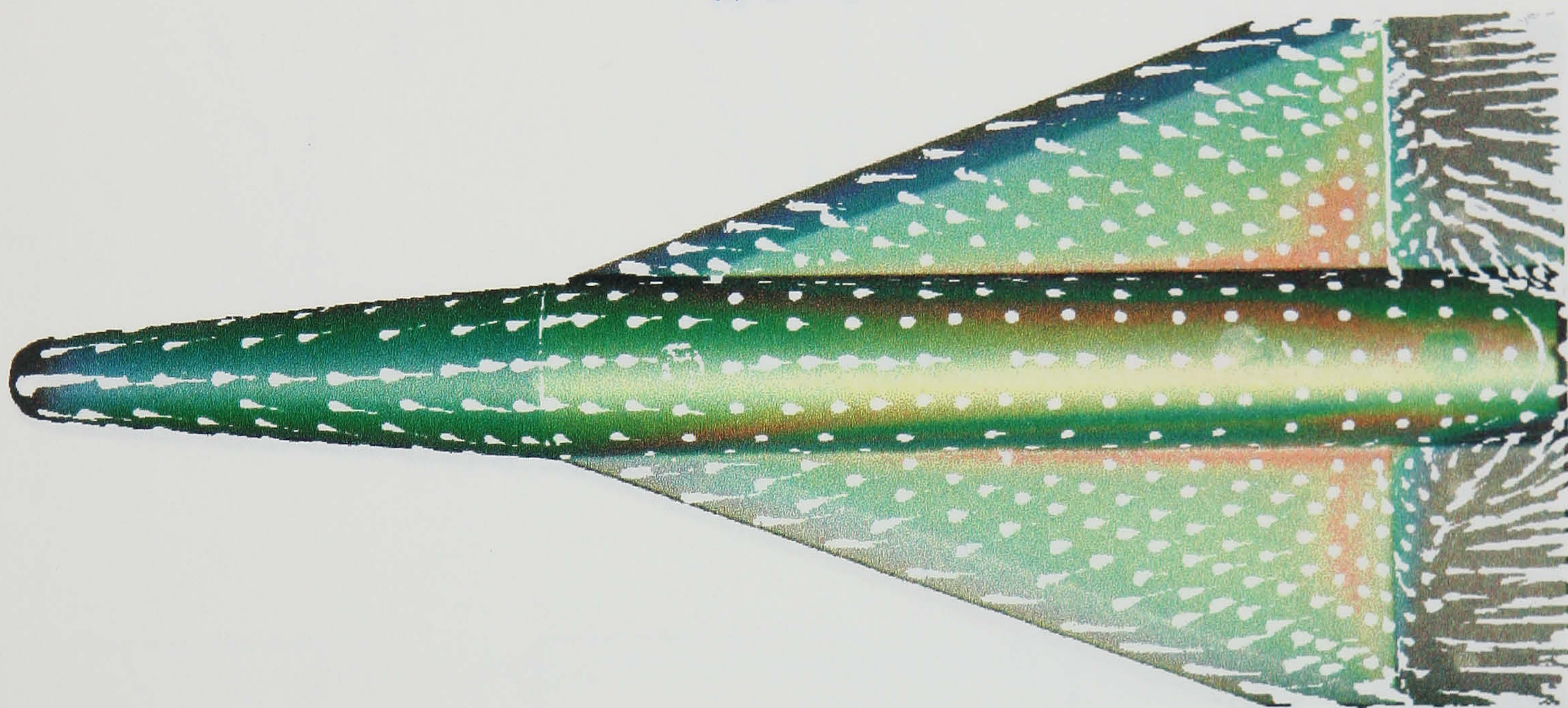


(c) $\alpha = 10^\circ$

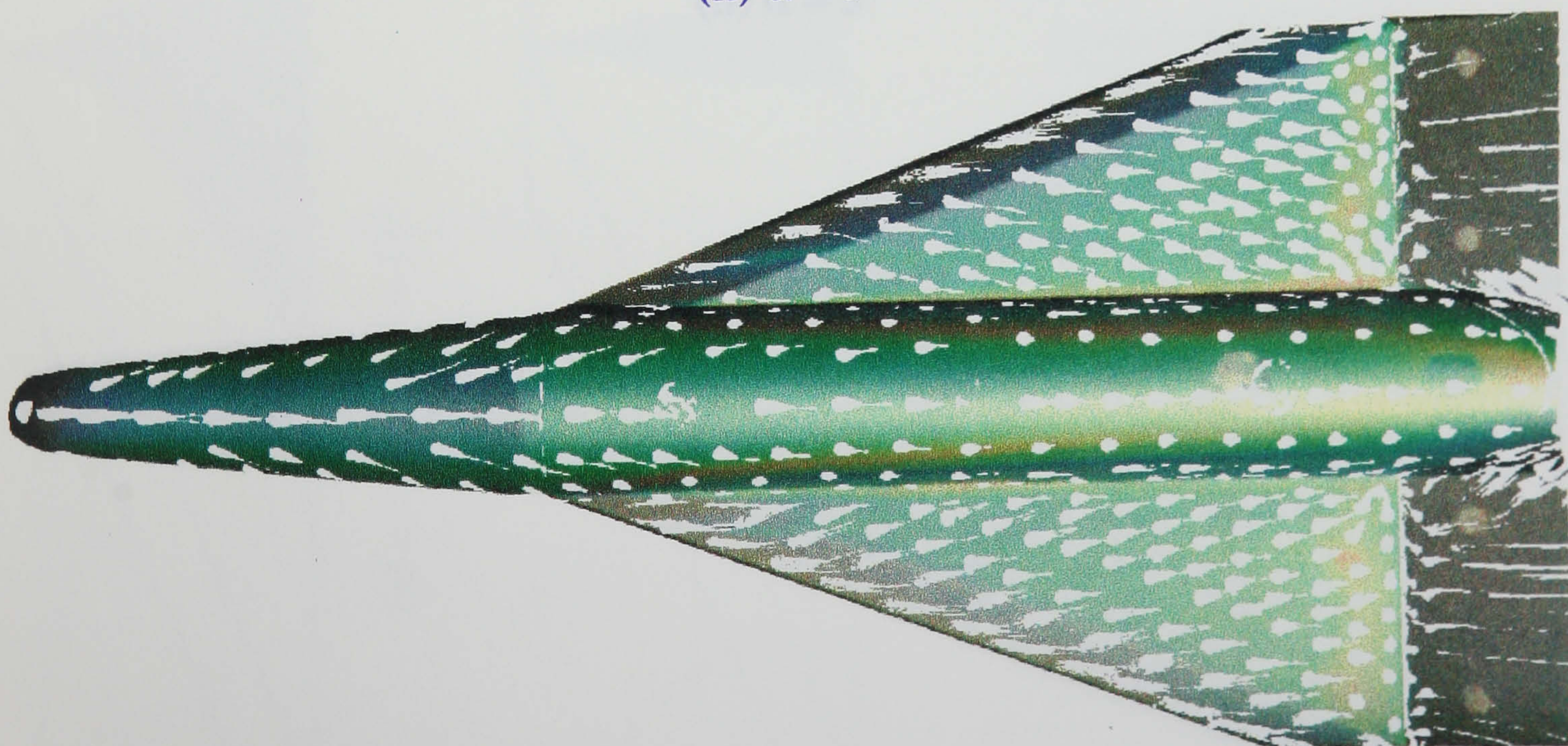
Figure 40. Oil-dot flow pictures showing reduction in the separated flow region for 25° flap deflection as the incidence increases from 0° to 10° .



(i) $\alpha = 0^\circ$



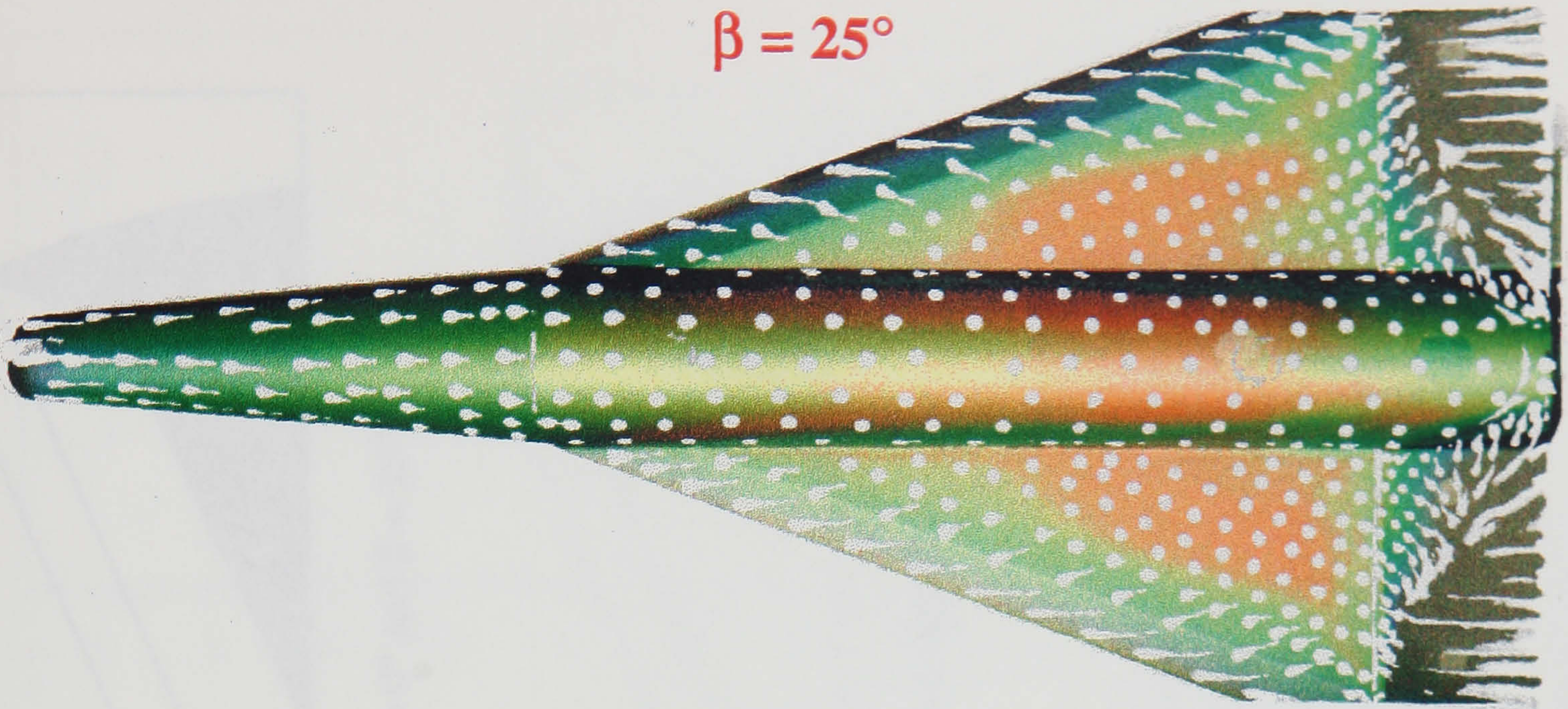
(ii) $\alpha = 5^\circ$



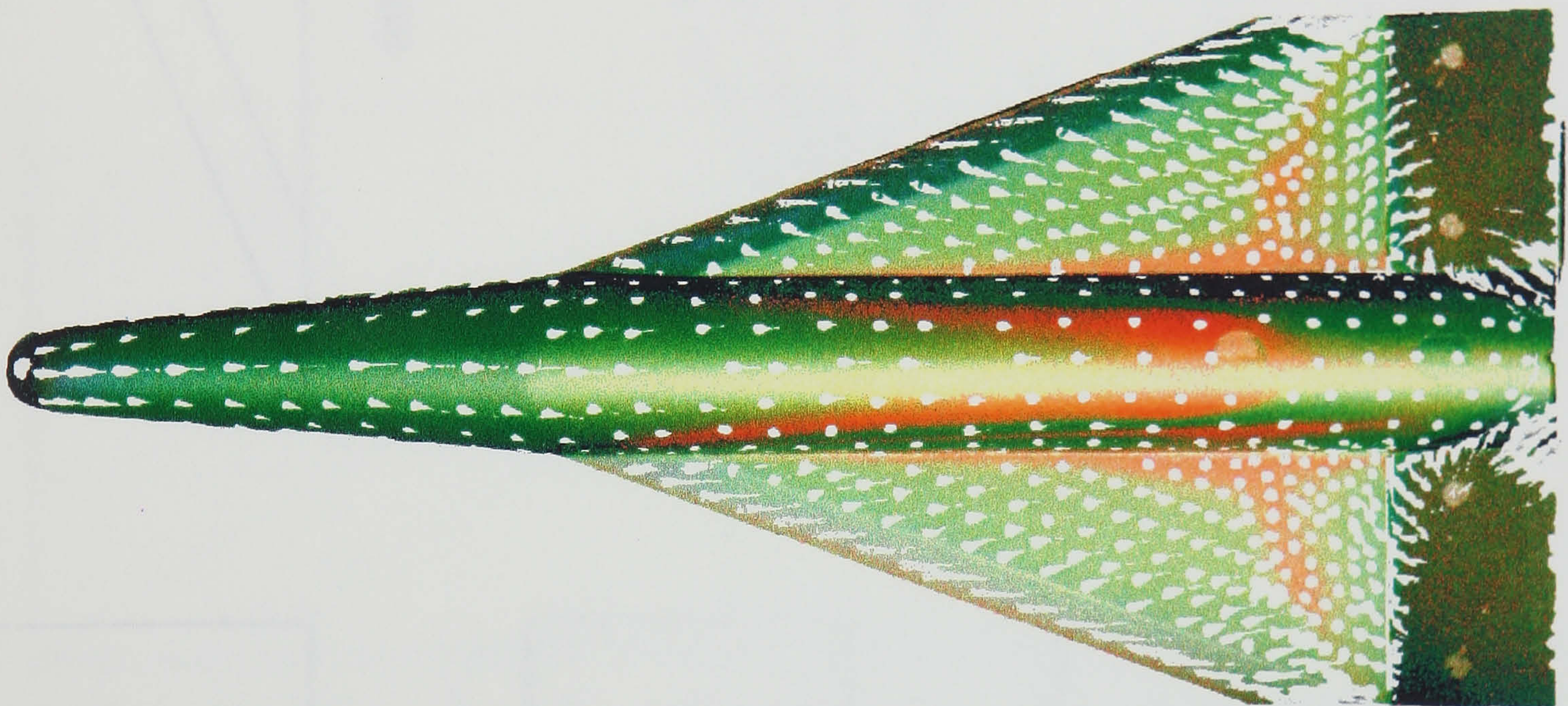
(iii) $\alpha = 10^\circ$

Figure 41(a). A comparison of liquid-crystal and oil-dot flow patterns over the lower side of the wing-body with $\beta = 15^\circ$.

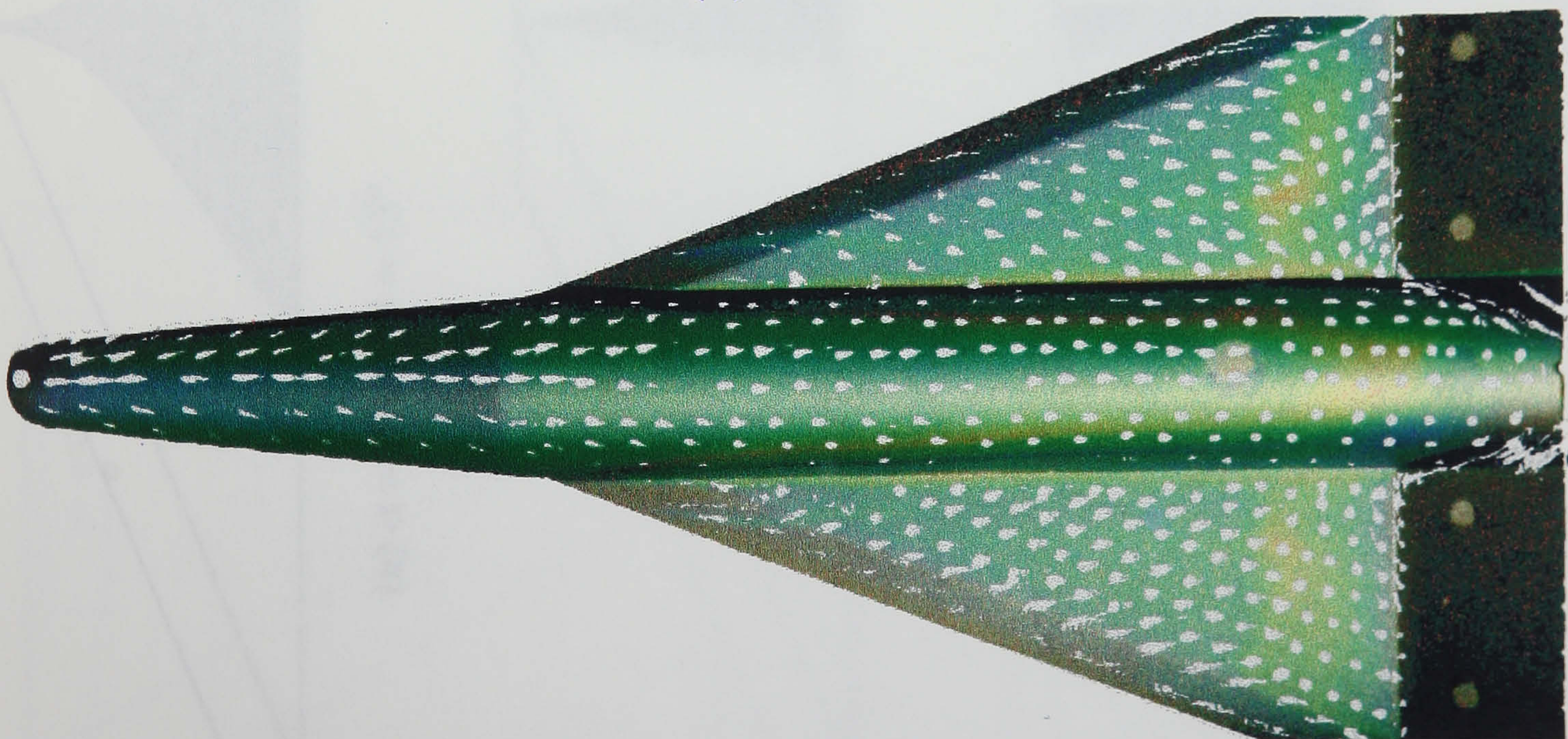
$$\beta = 25^\circ$$



$$(i) \alpha = 0^\circ$$



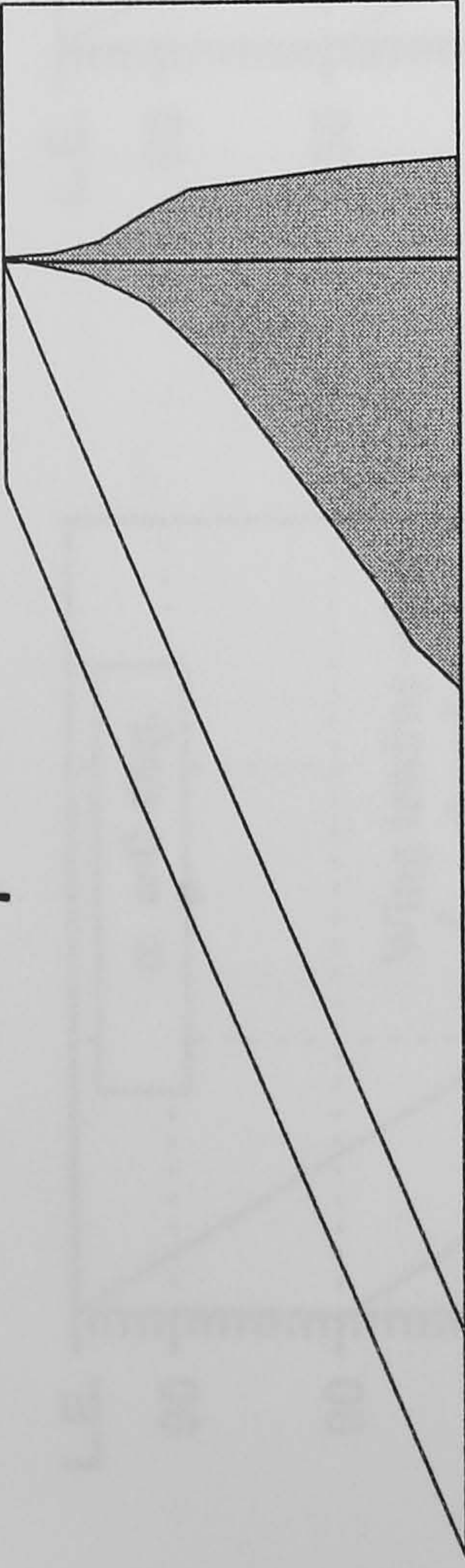
$$(ii) \alpha = 5^\circ$$



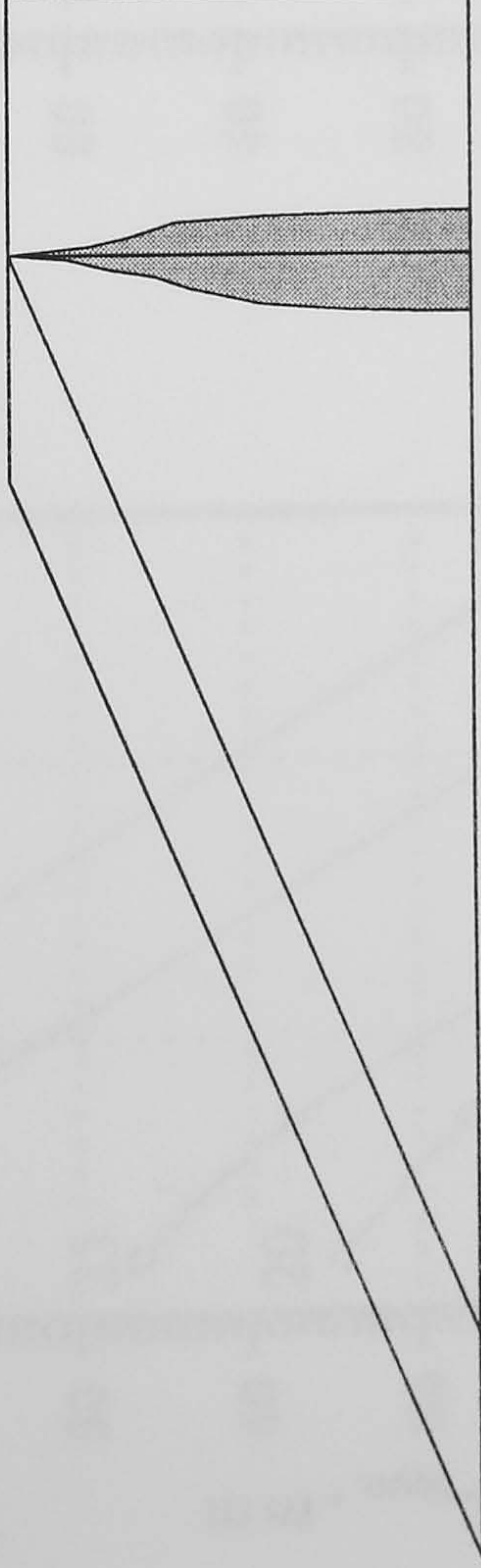
$$(iii) \alpha = 10^\circ$$

Figure 41(b). A comparison of liquid-crystal and oil-dot flow patterns over the lower side of the wing-body with $\beta = 25^\circ$.

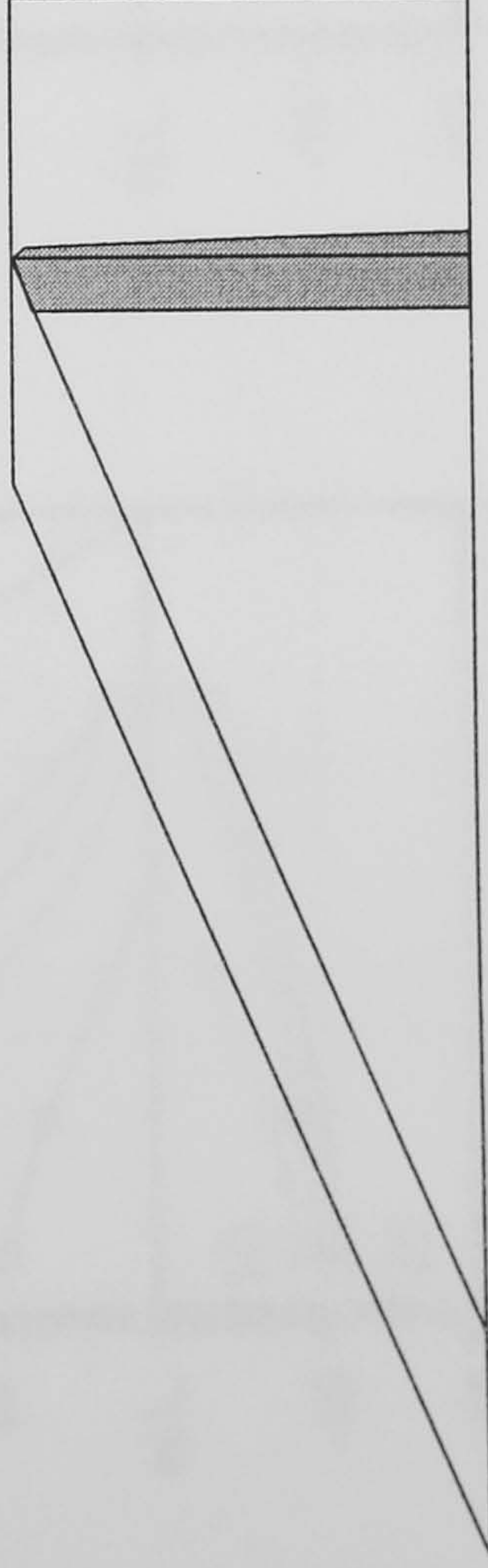
$\beta = 15^\circ$



(a) $\alpha = 0^\circ$ and $\beta = 15^\circ$

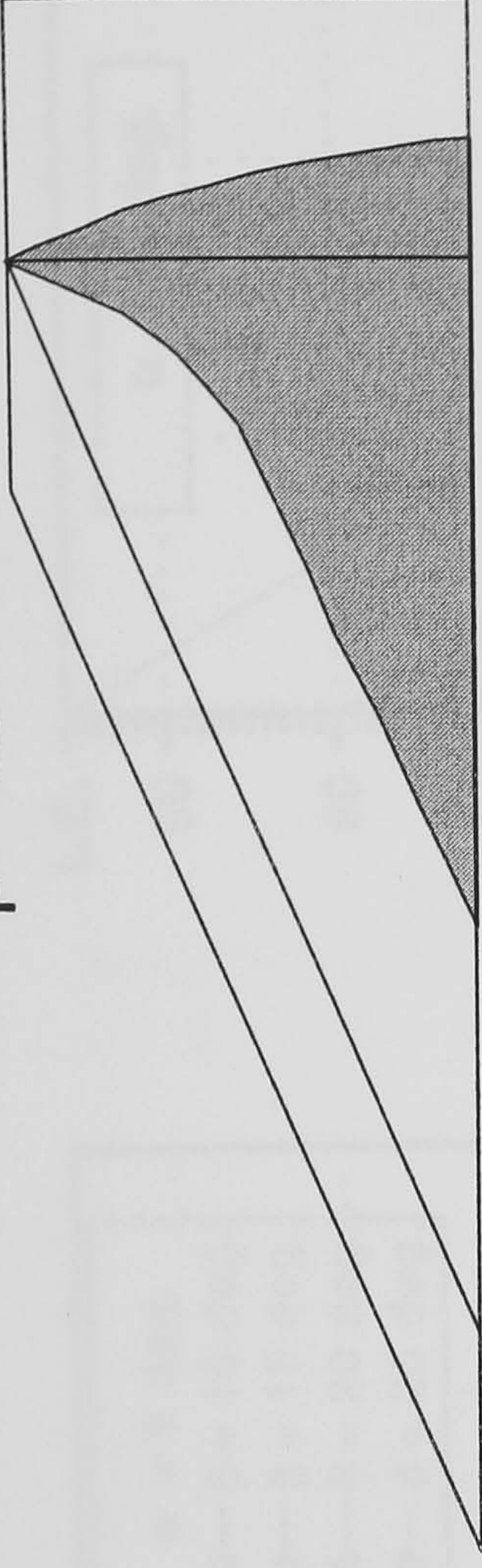


(c) $\alpha = 5^\circ$ and $\beta = 15^\circ$

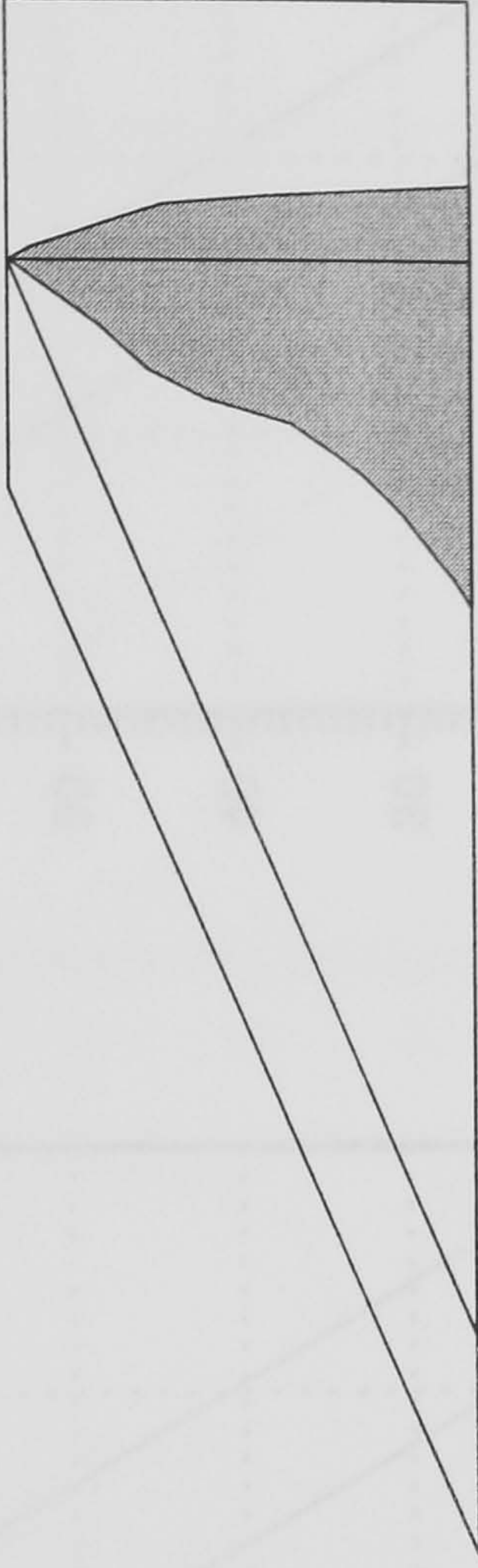


(e) $\alpha = 10^\circ$ and $\beta = 15^\circ$

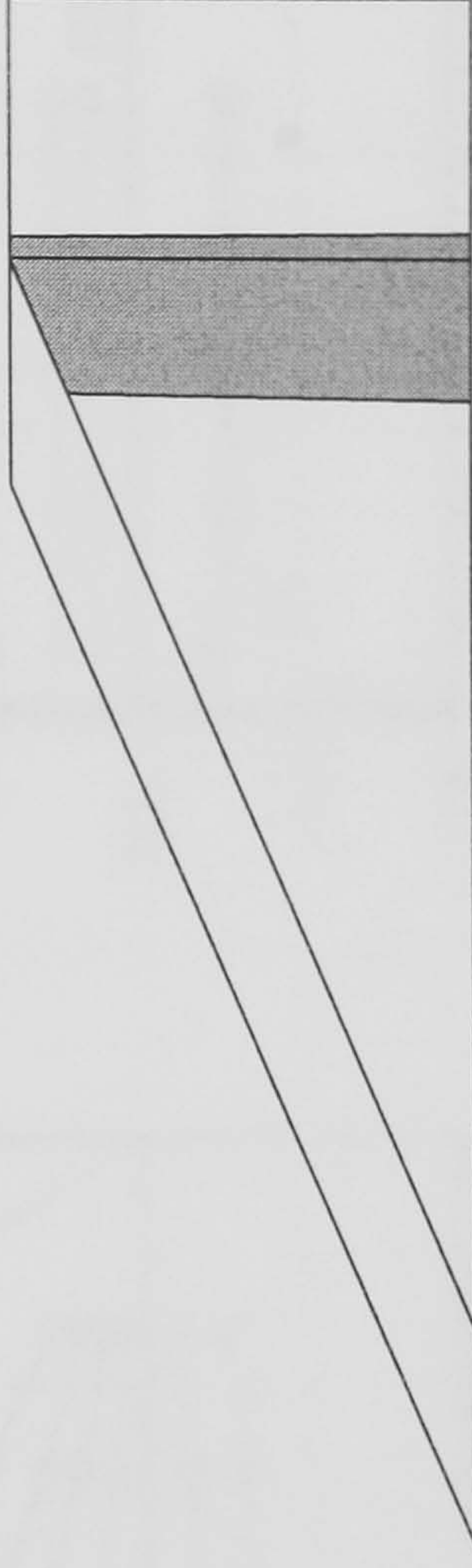
$\beta = 25^\circ$



(b) $\alpha = 0^\circ$ and $\beta = 25^\circ$



(d) $\alpha = 5^\circ$ and $\beta = 25^\circ$



(f) $\alpha = 10^\circ$ and $\beta = 25^\circ$

Figure 41(c). Sketch showing separated flow regions over the lower side of the wing as α is increased from 0 to 5 and 10° for $\beta = 15^\circ$ and 25° .

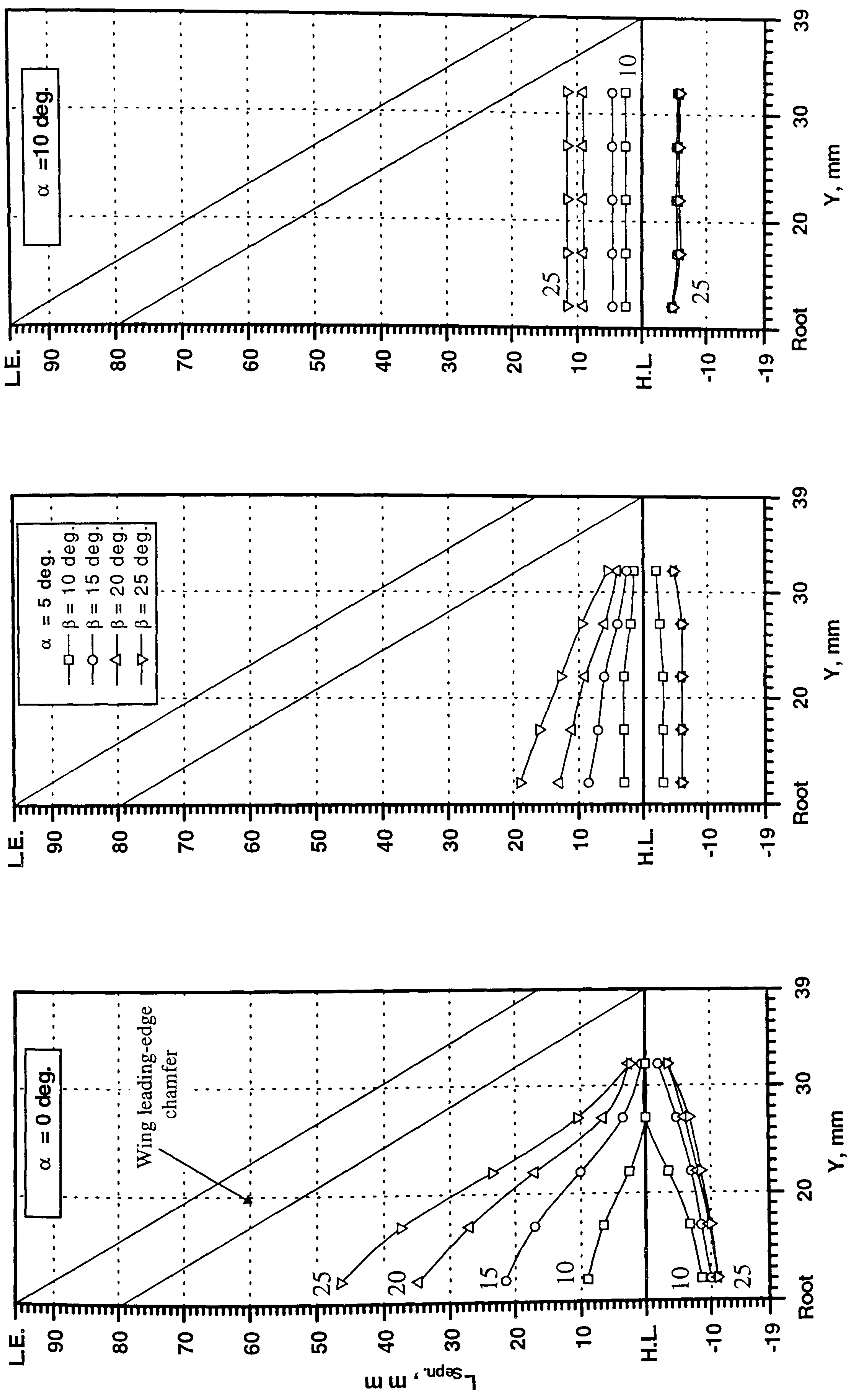


Figure 42. Variation in wing separation pattern with change in flap angle.

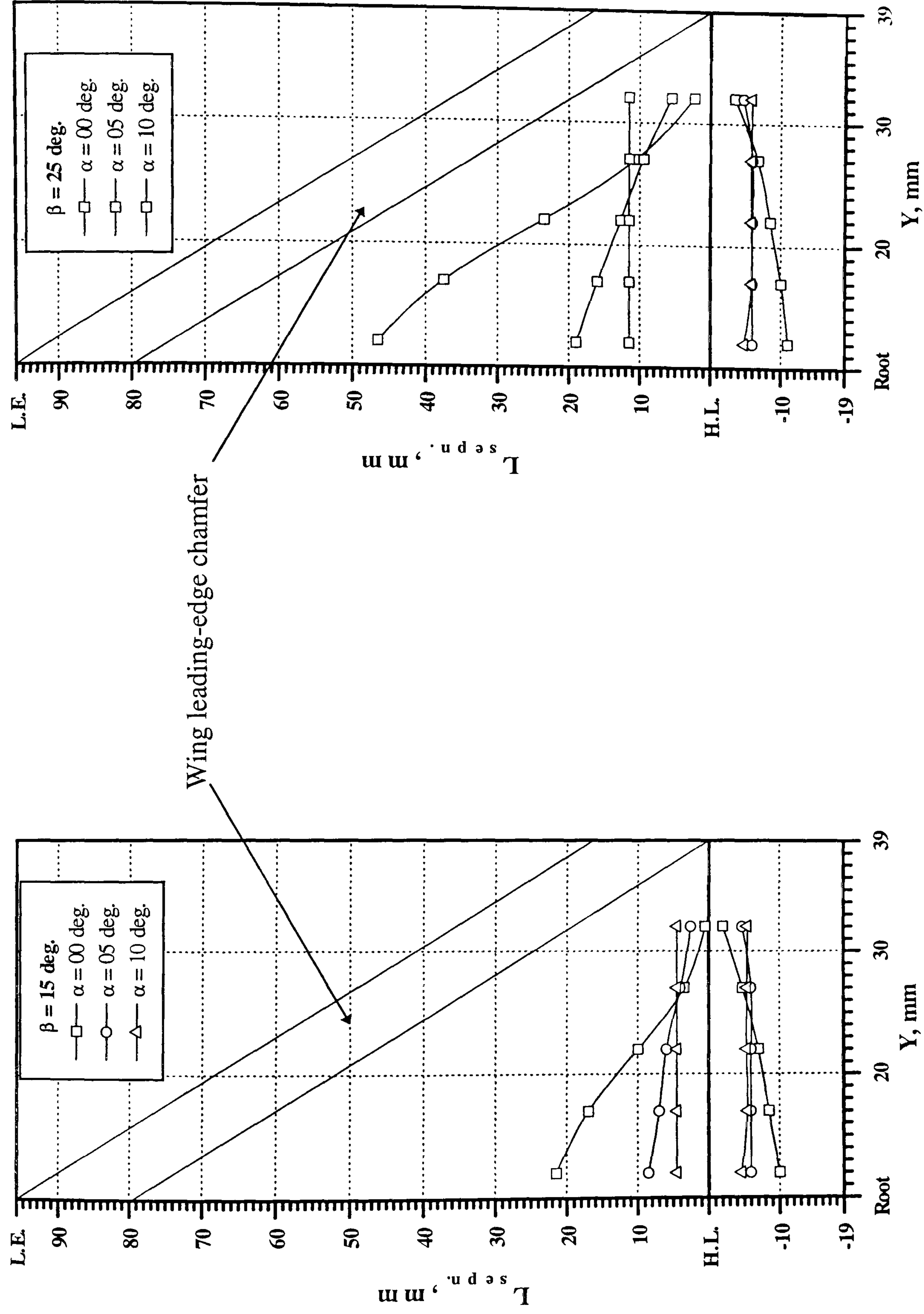


Figure 43. Variation in wing separation pattern with change in incidence.

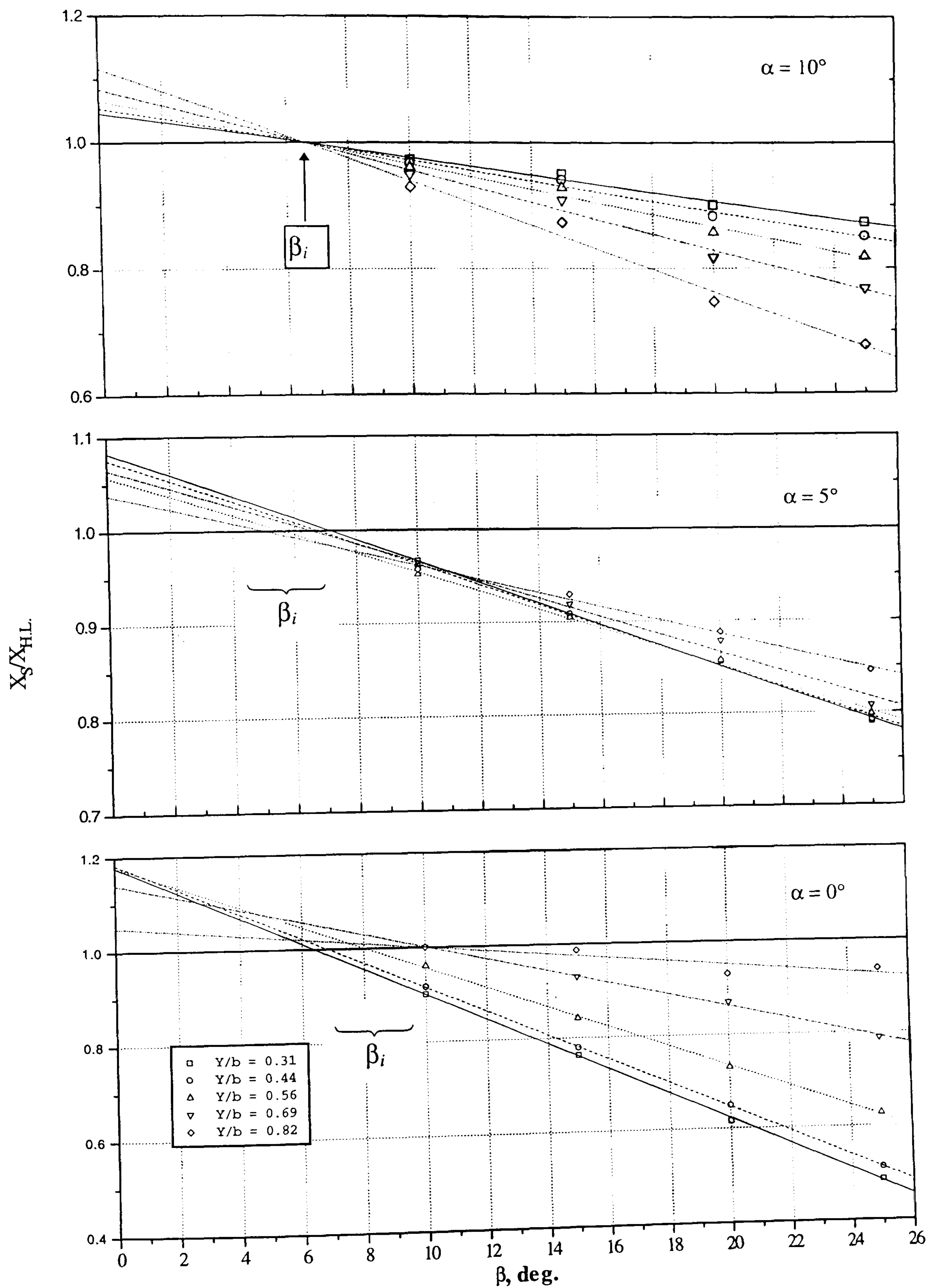


Figure 44. Incipient separation angle determined by extrapolating the separation length to zero.

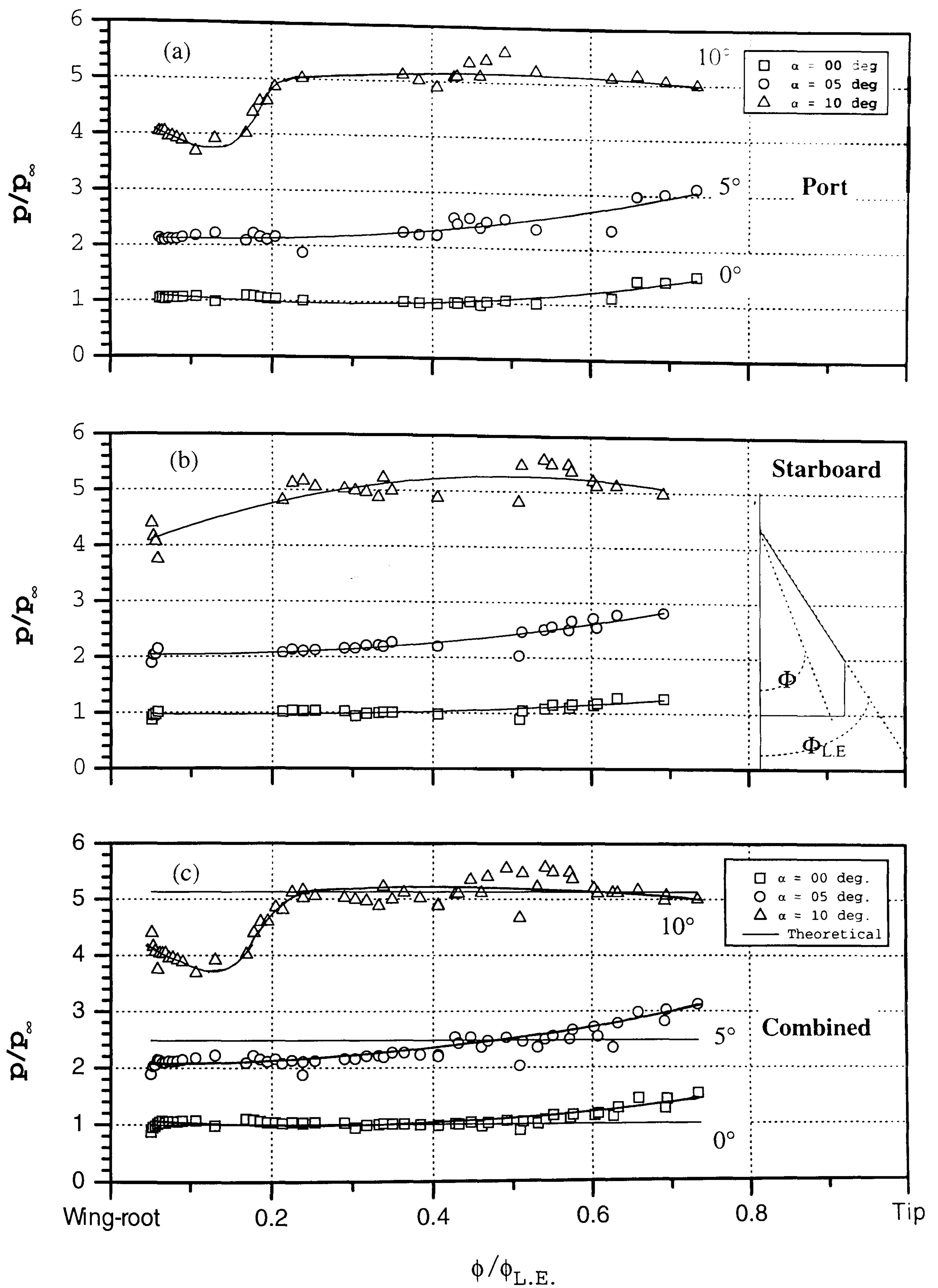


Figure 45. Pressure distribution over the surface of the wing along conical generators from the root of the wing leading-edge for $\beta = 0^\circ$.

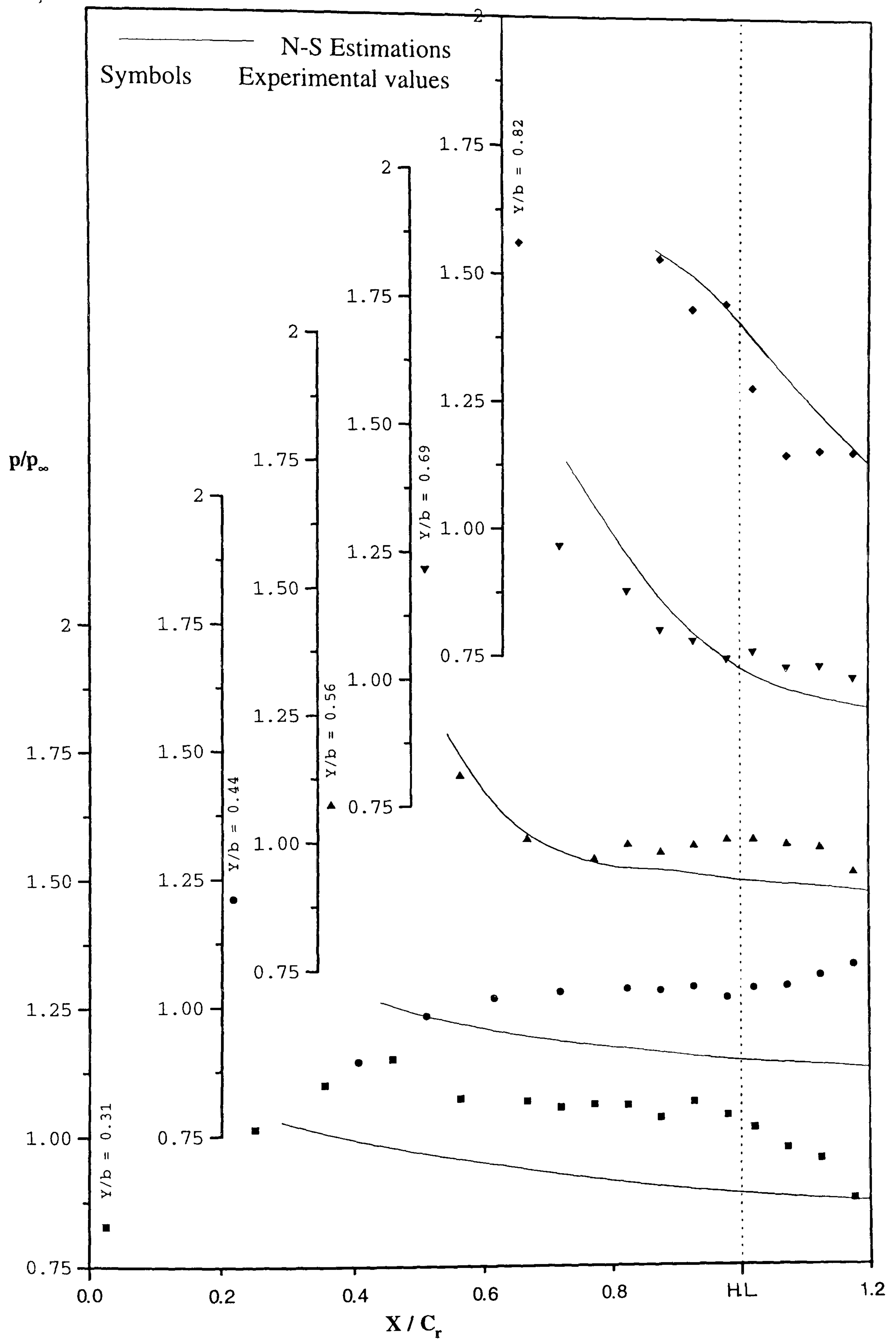


Figure 46(a). A comparison of the N-S estimations with the experimental pressure distribution over the lower side of wing for the wing-body configuration at $\alpha = 0^\circ$.

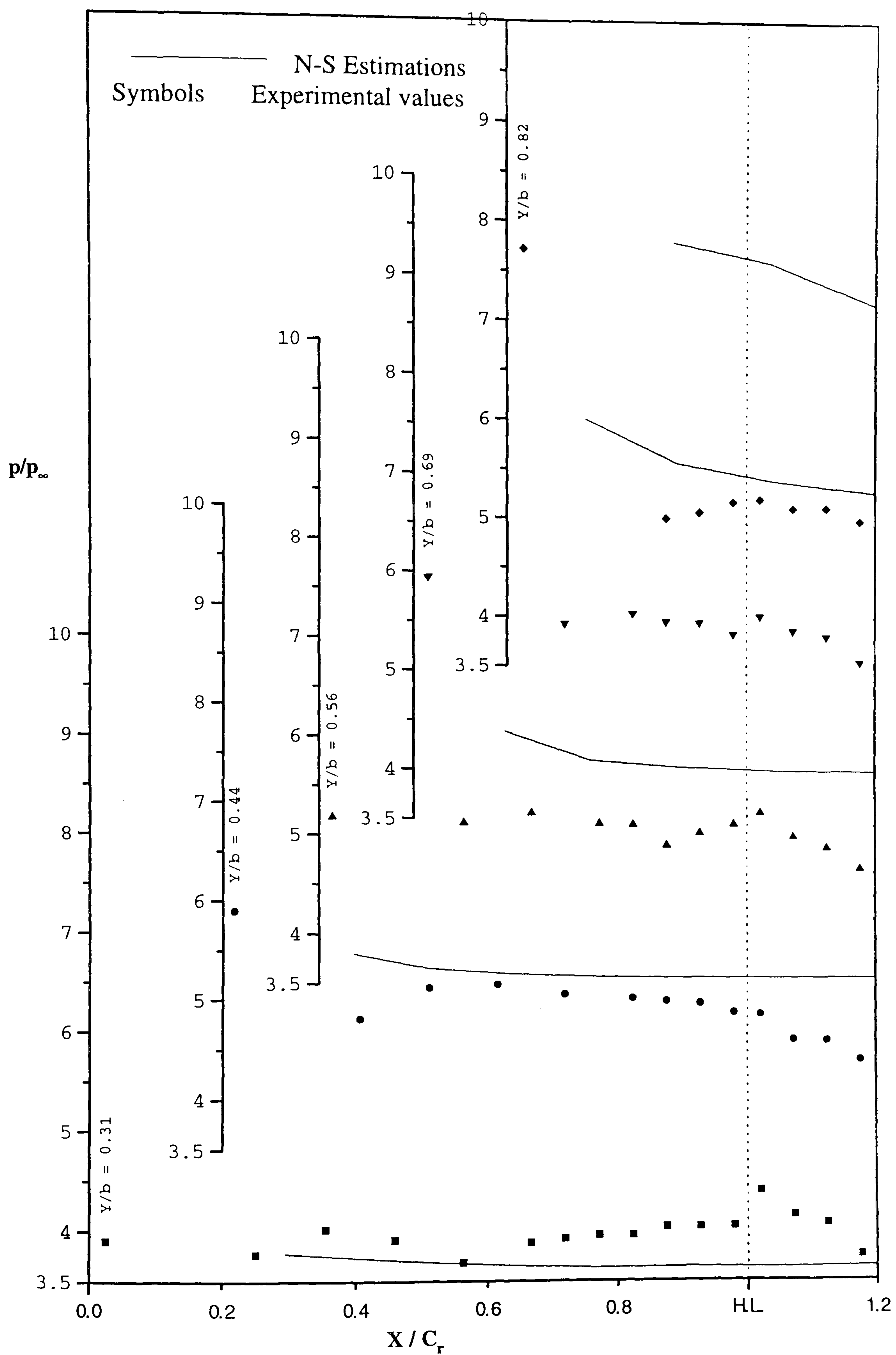
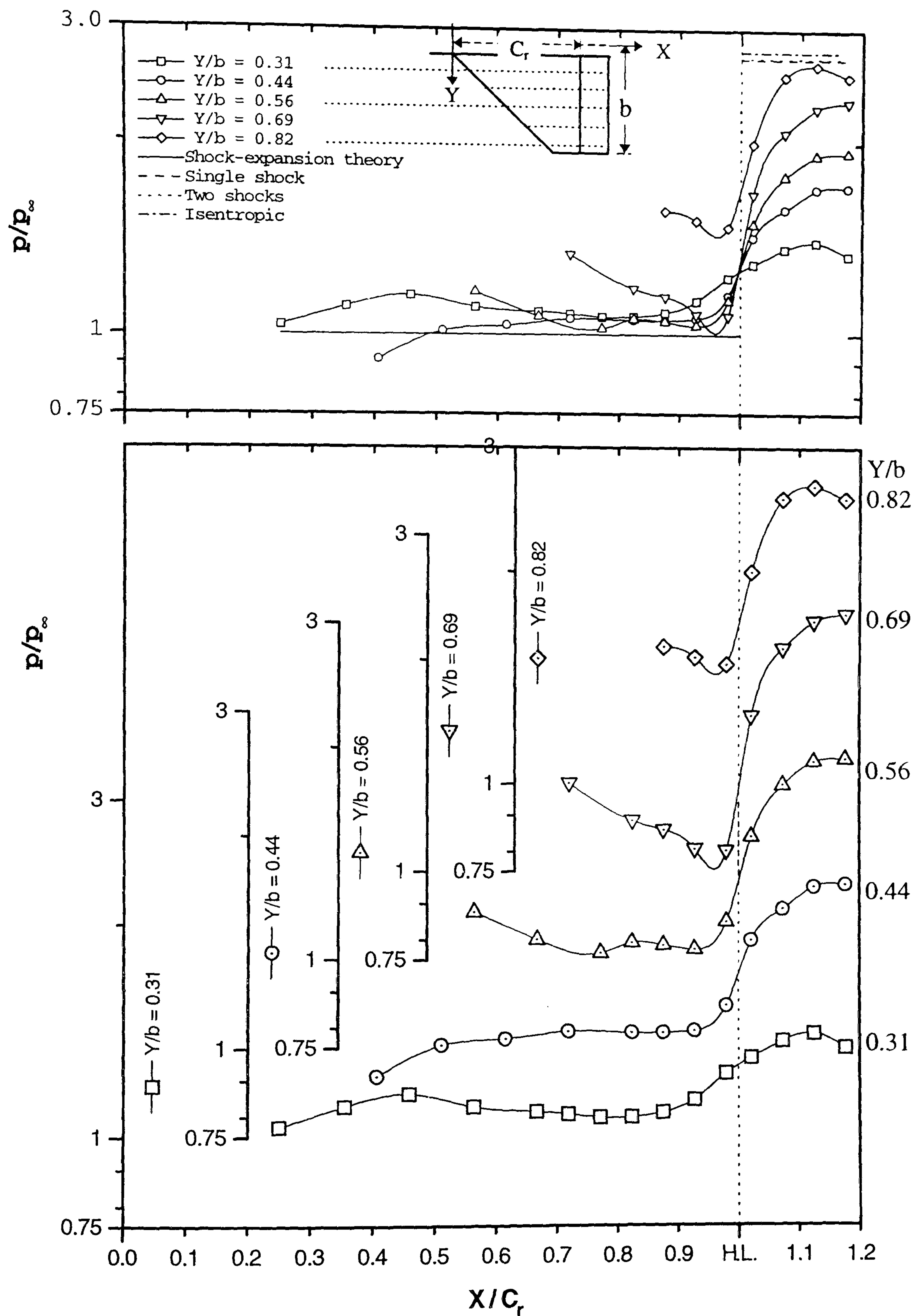
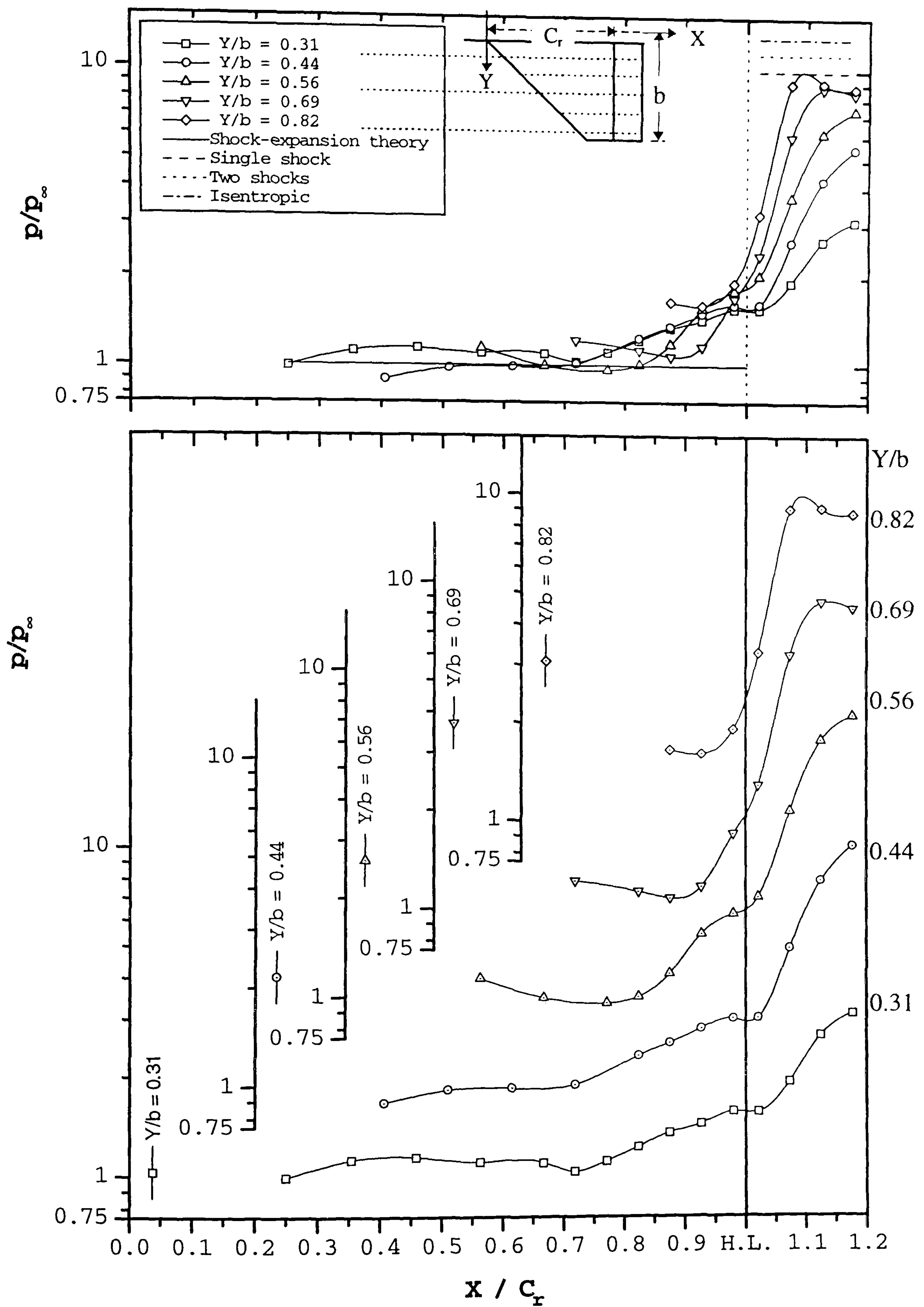


Figure 46(b). A comparison of the N-S estimations with the experimental pressure distribution over the lower side of wing for the wing-body configuration at $\alpha=10^\circ$.



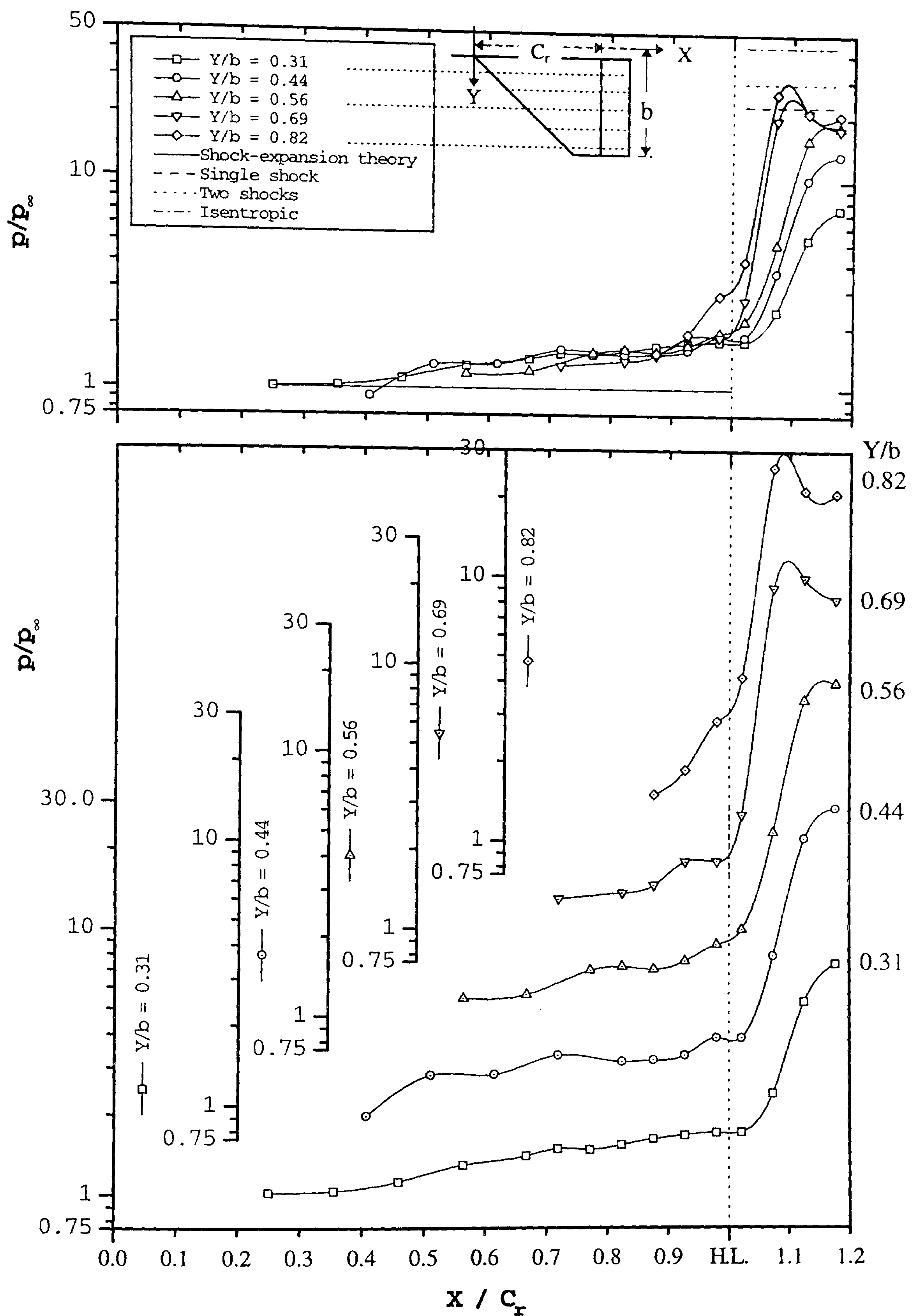
(a) $\beta = 5^\circ$

Figure 47. Chordwise pressure distribution on the lower side of the flapped delta wing at $\alpha = 0^\circ$.



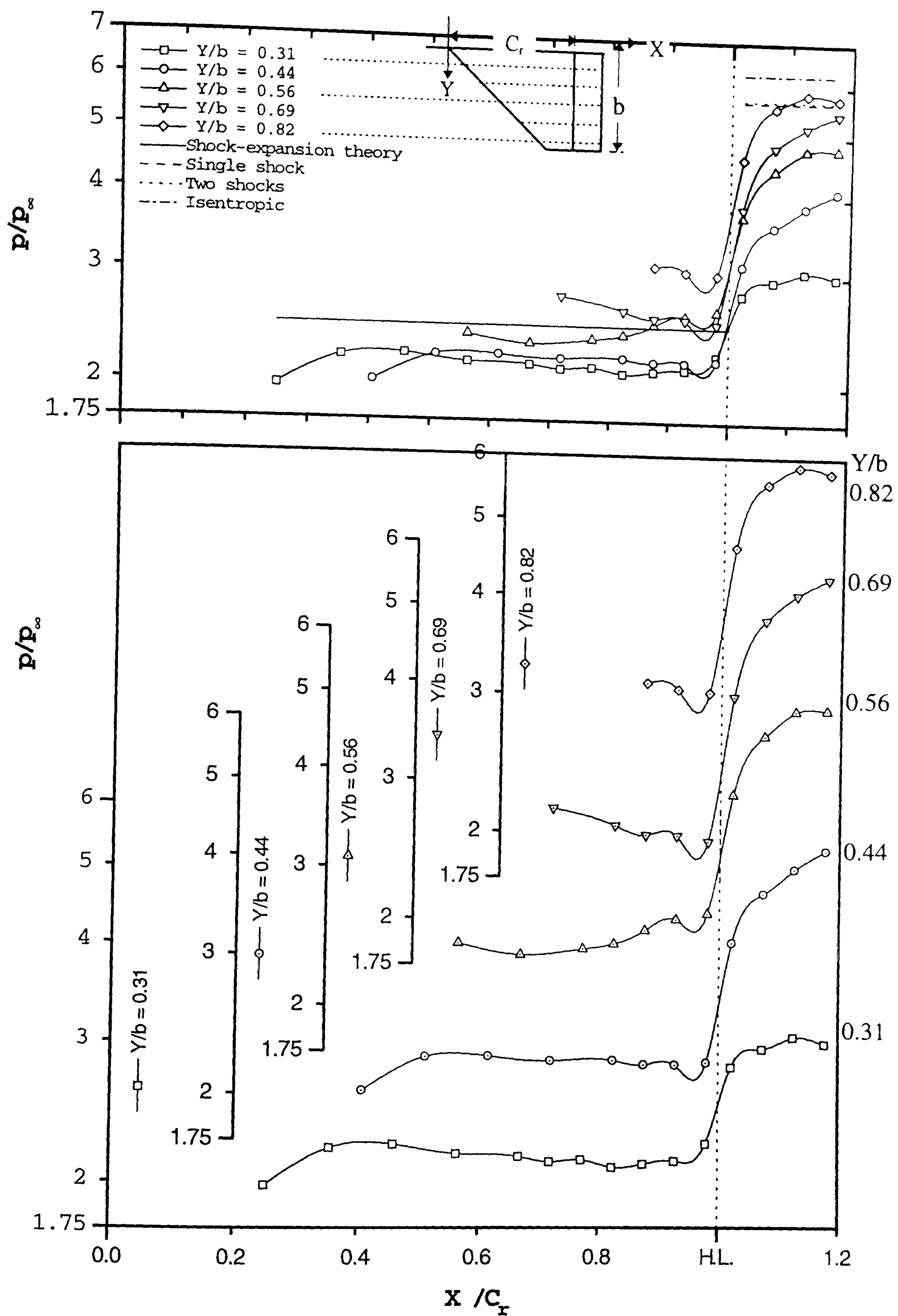
(b) $\beta = 15^\circ$

Figure 47. Continued.



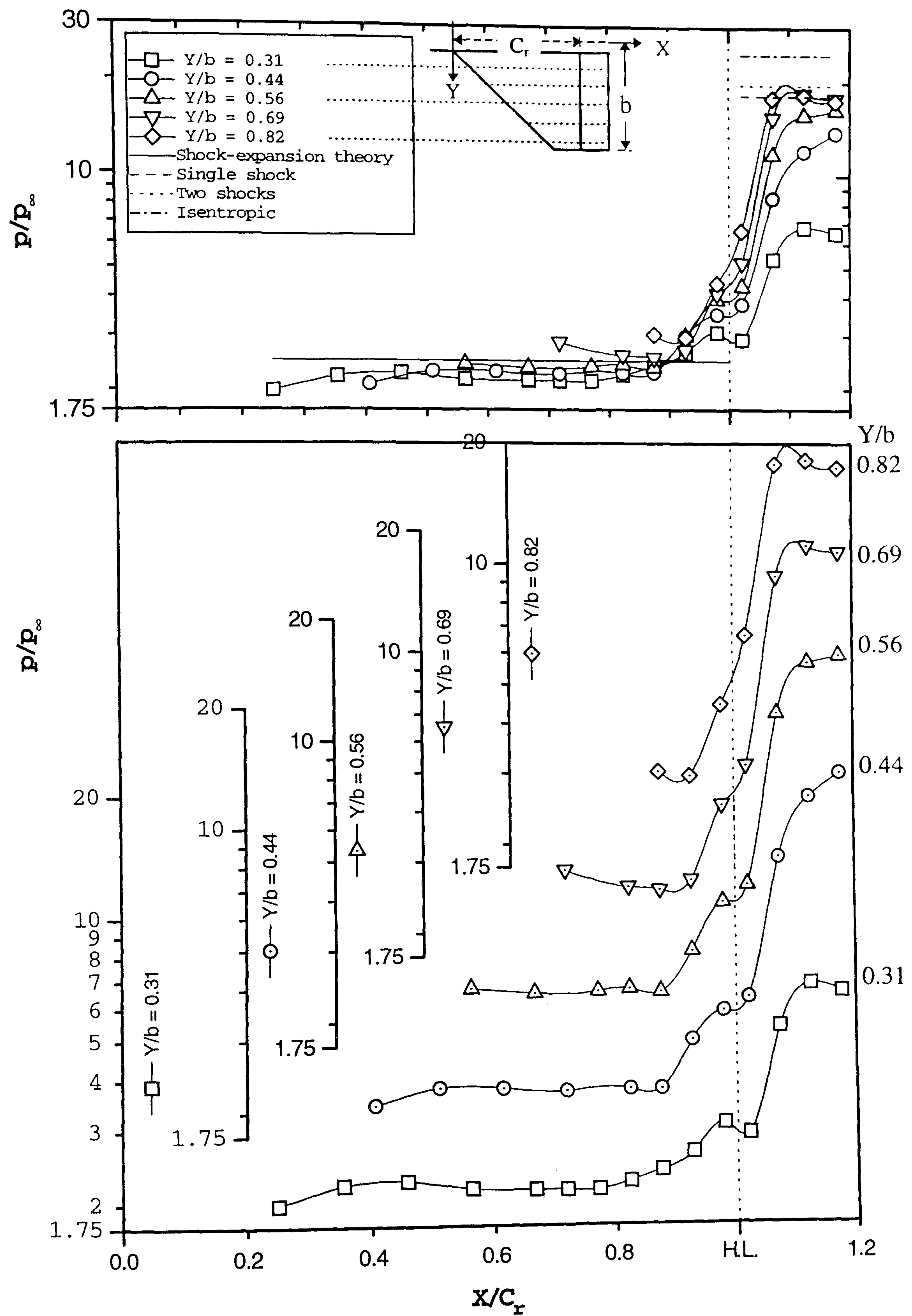
(c) $\beta = 25^\circ$

Figure 47. Continued.



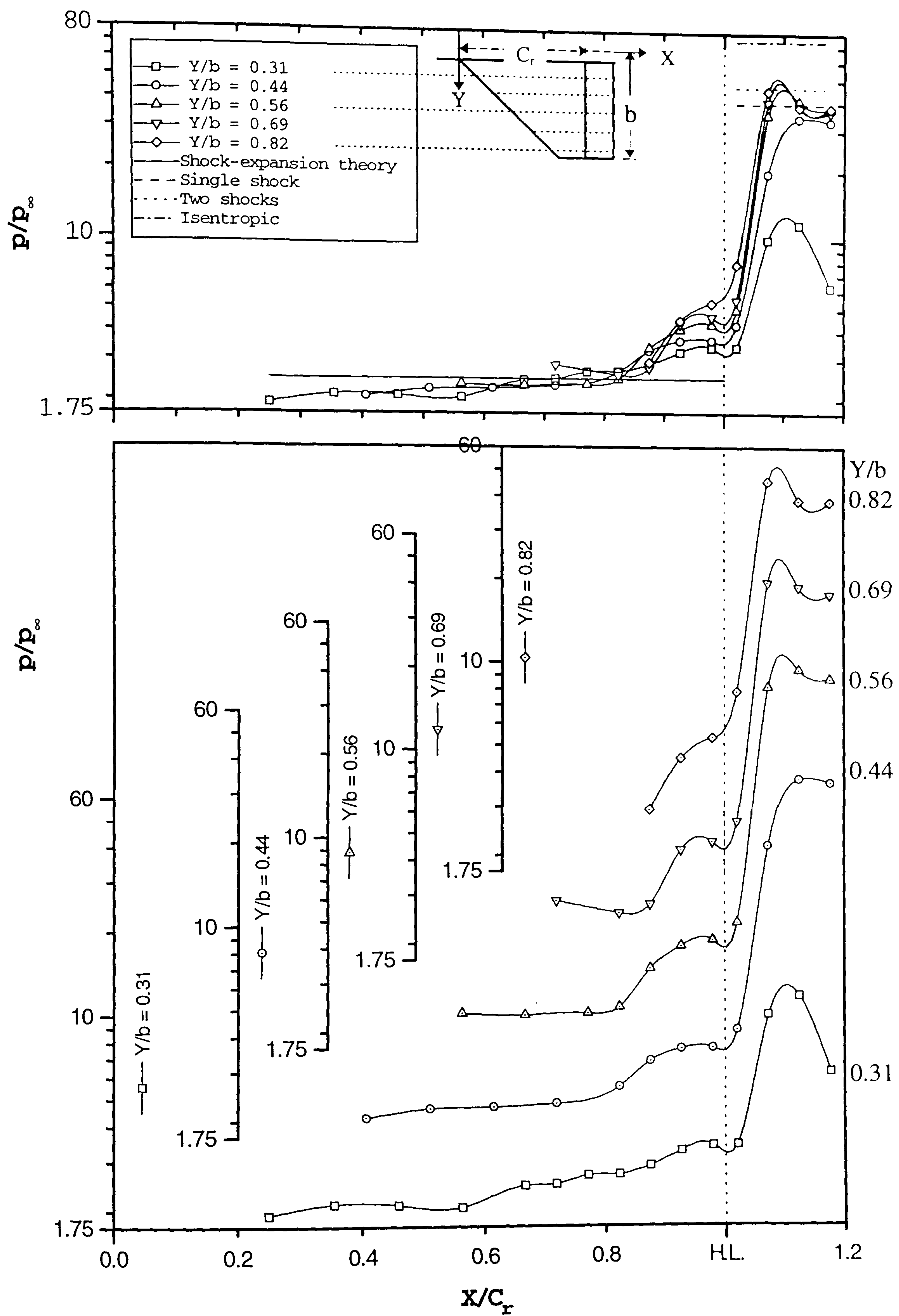
(a) $\beta = 5^\circ$

Figure 48. Chordwise pressure distribution on the lower side of the flapped delta wing at $\alpha = 5^\circ$.



(b) $\beta = 15^\circ$

Figure 48. Continued.



(c) $\beta = 25^\circ$

Figure 48. Continued.

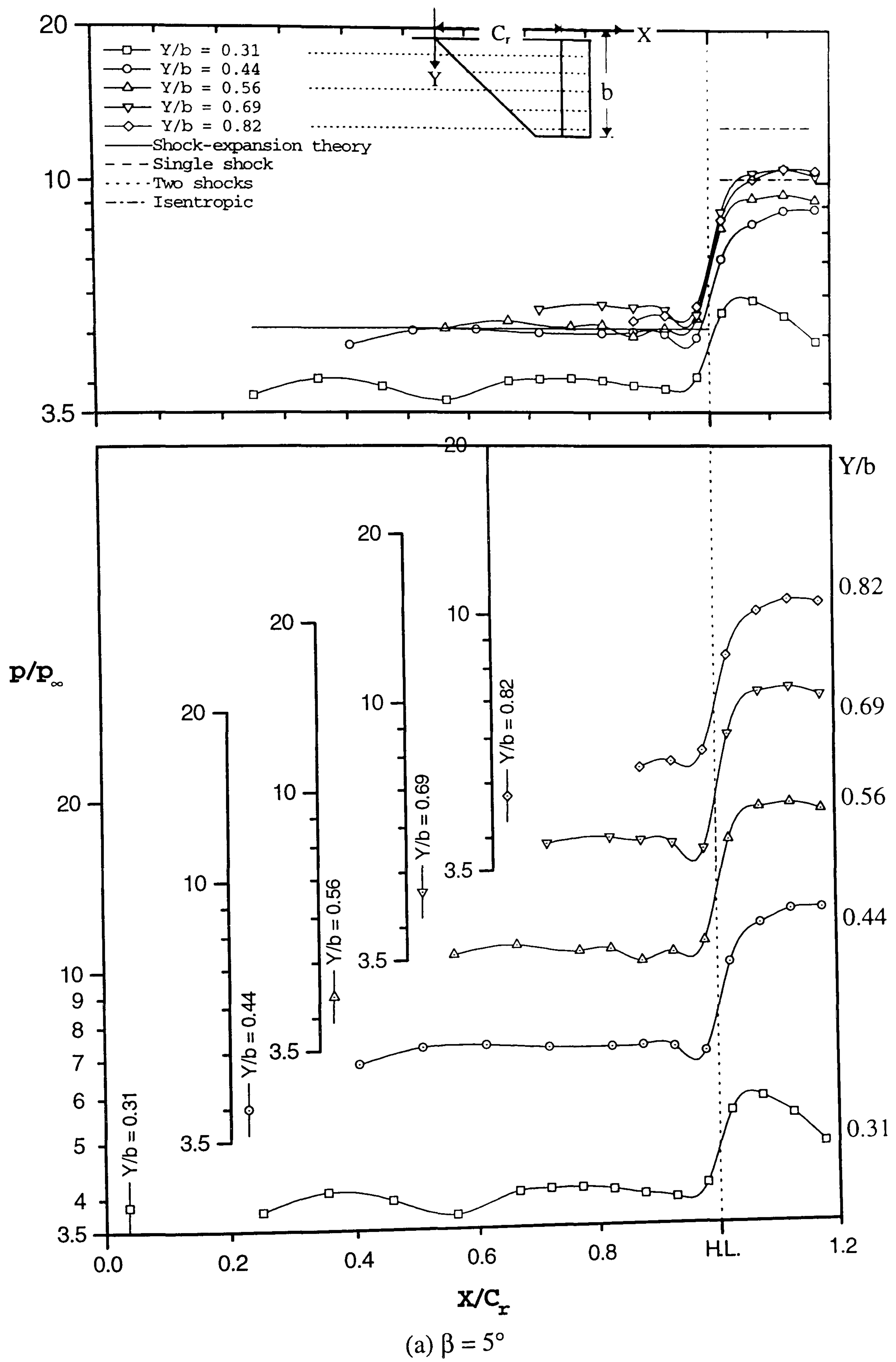
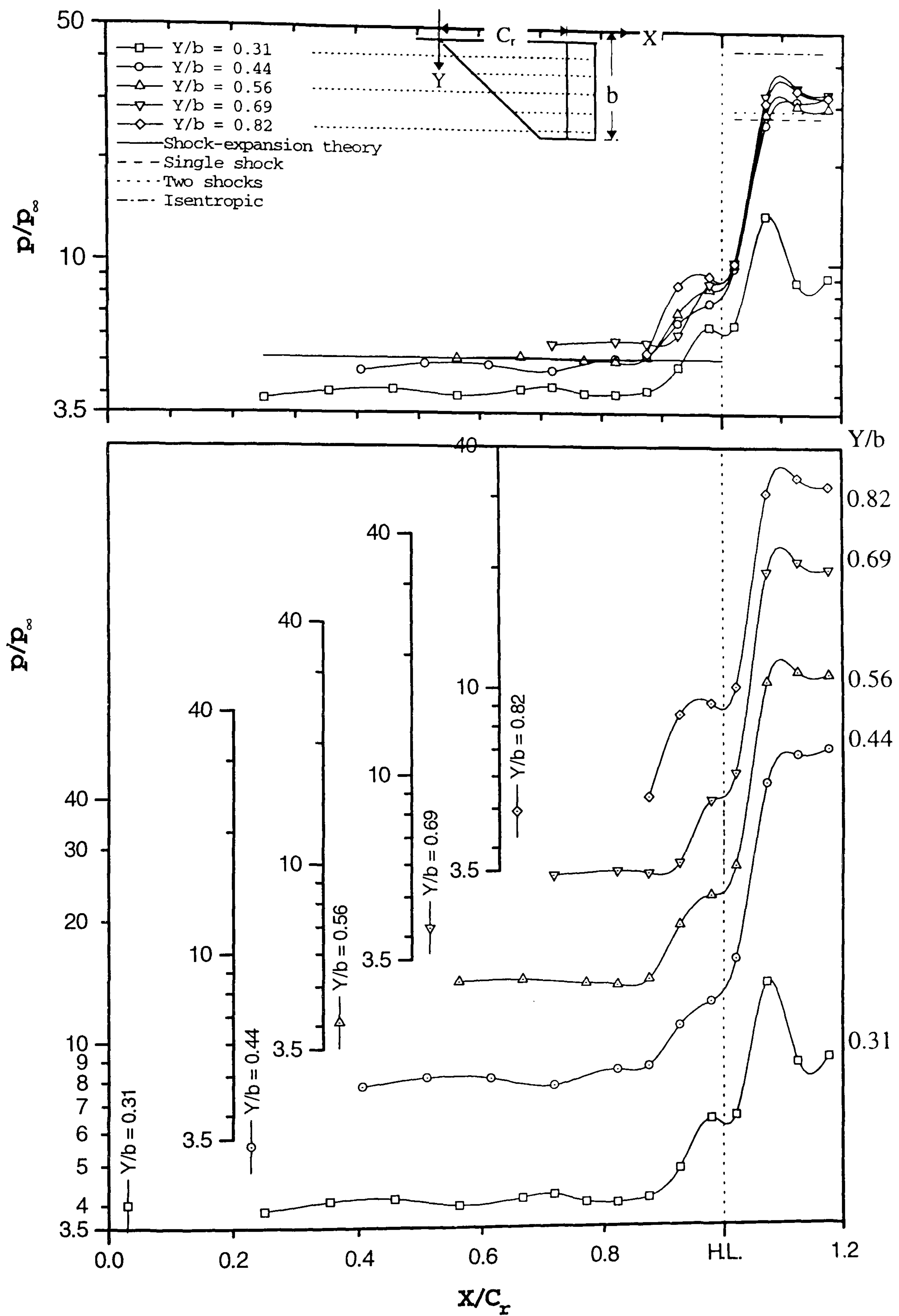
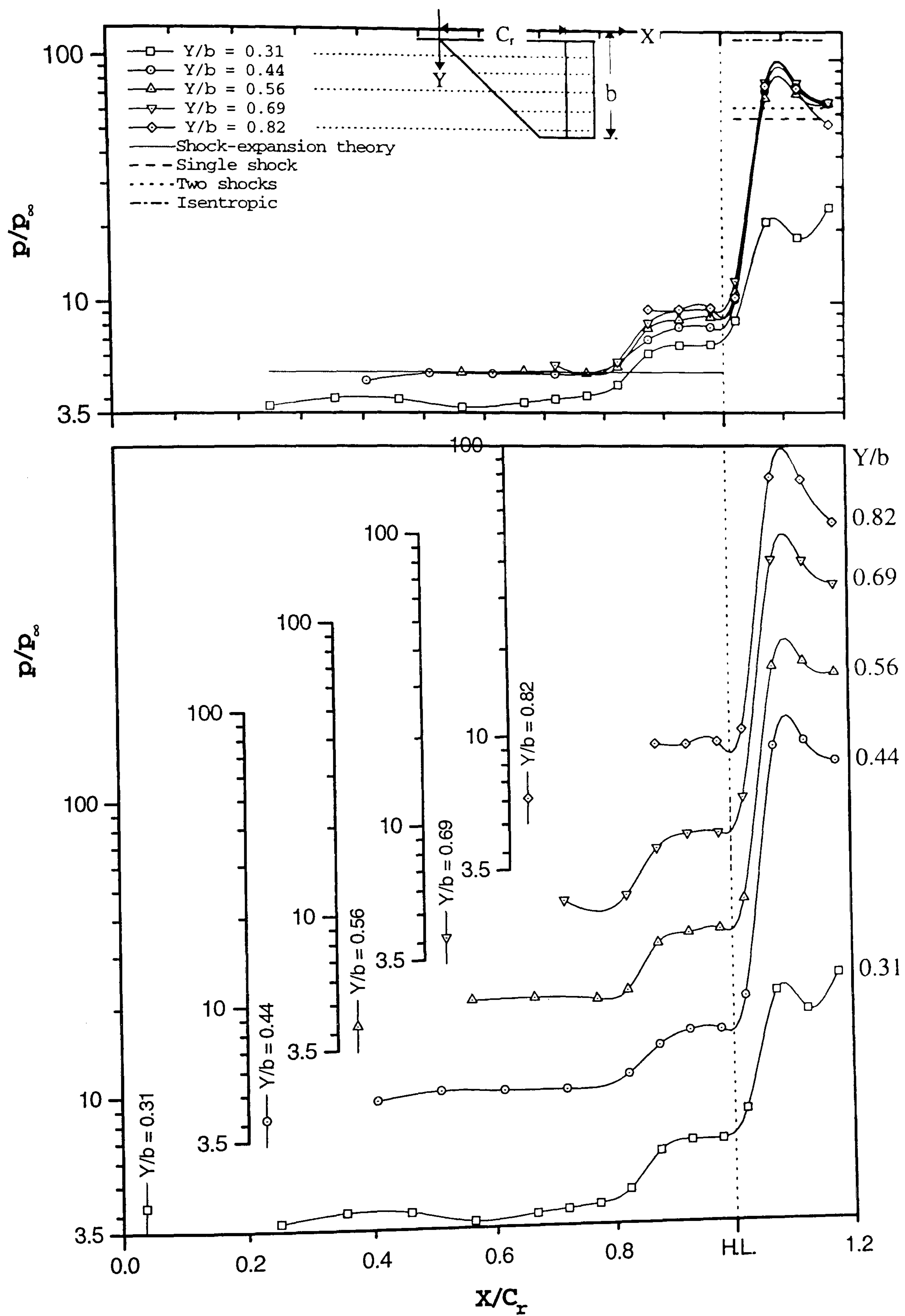


Figure 49. Chordwise pressure distribution on the lower side of the flapped delta wing at $\alpha = 10^\circ$.



(b) $\beta = 15^\circ$

Figure 49. Continued.



(c) $\beta = 25^\circ$

Figure 49. Continued.

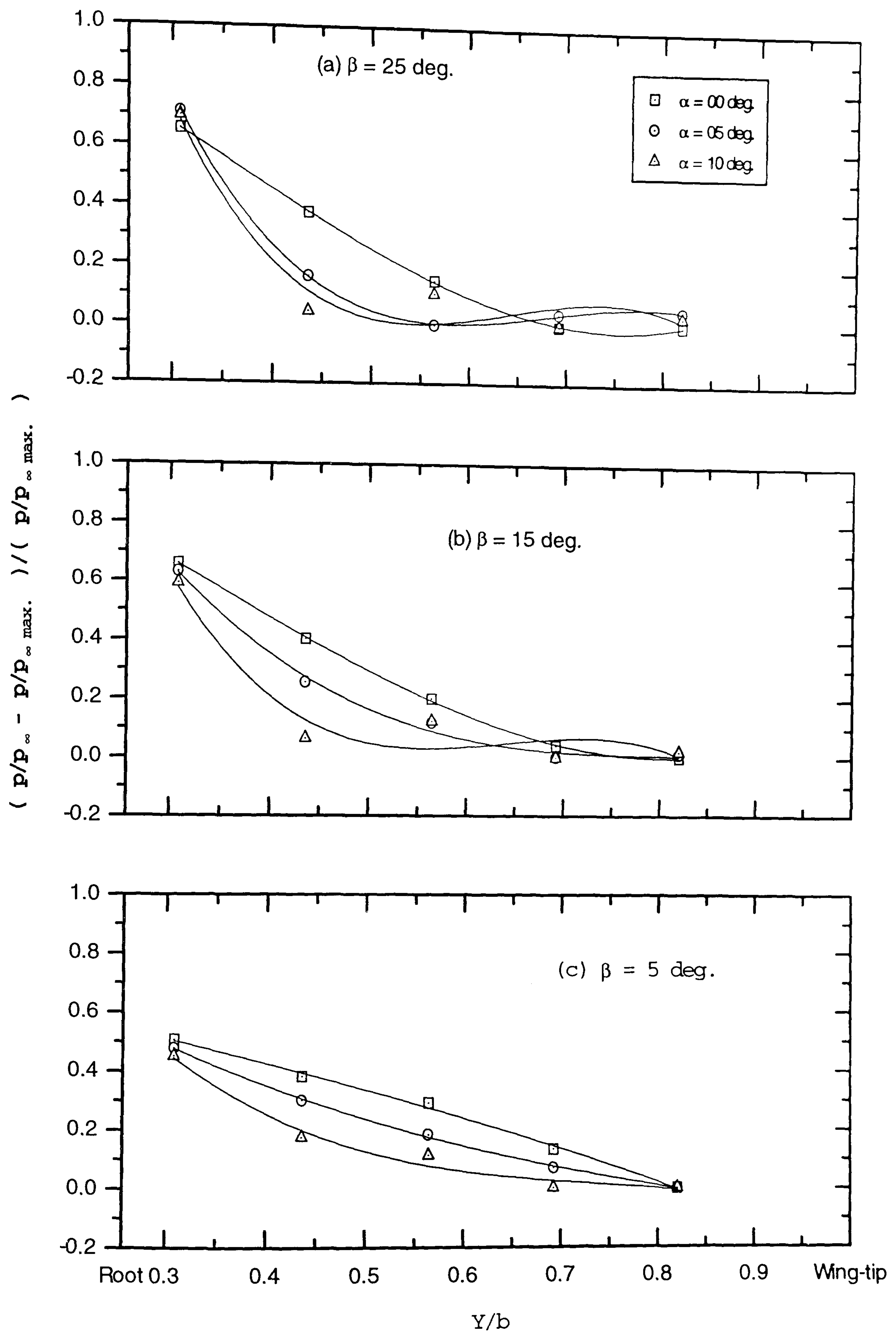
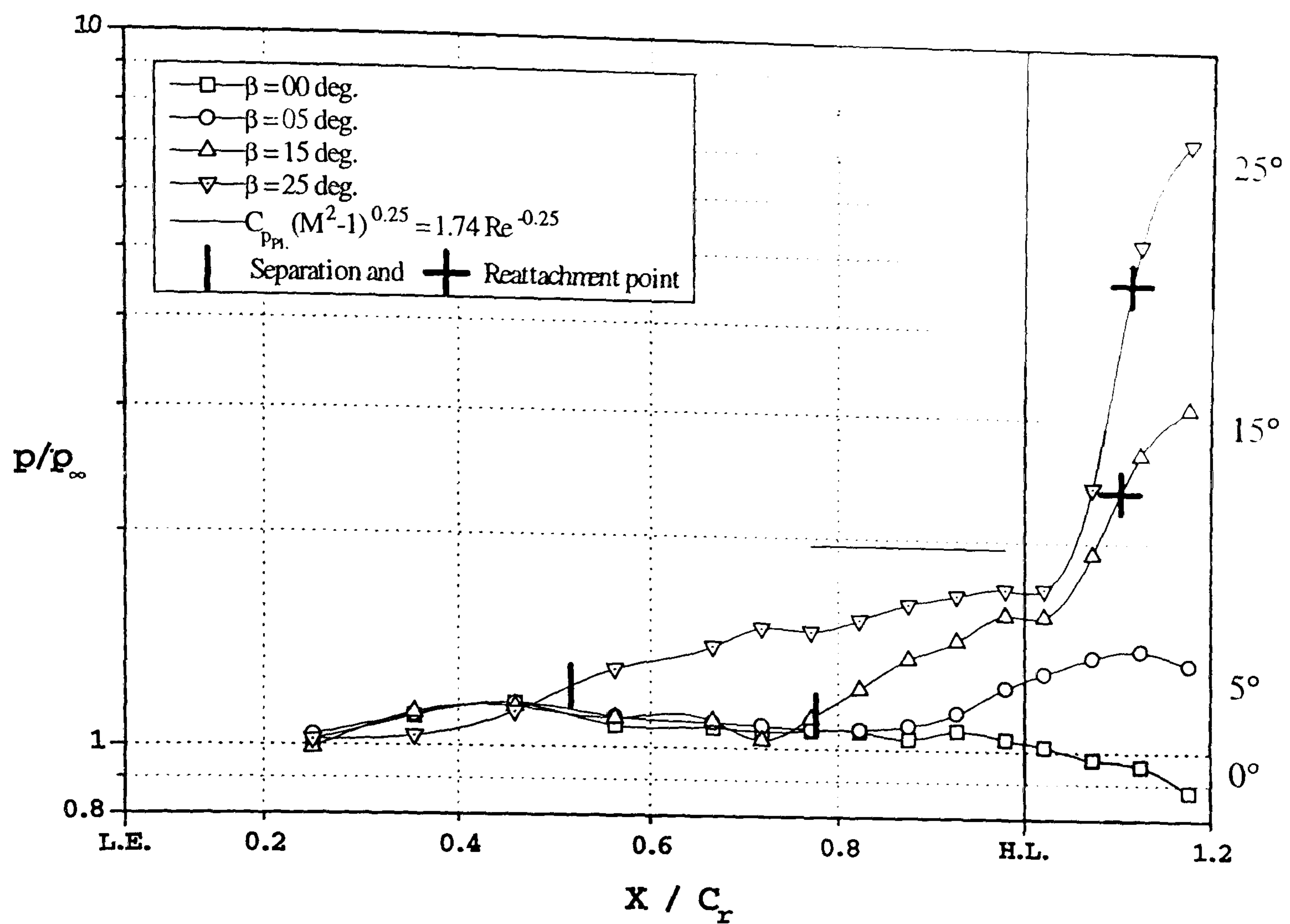
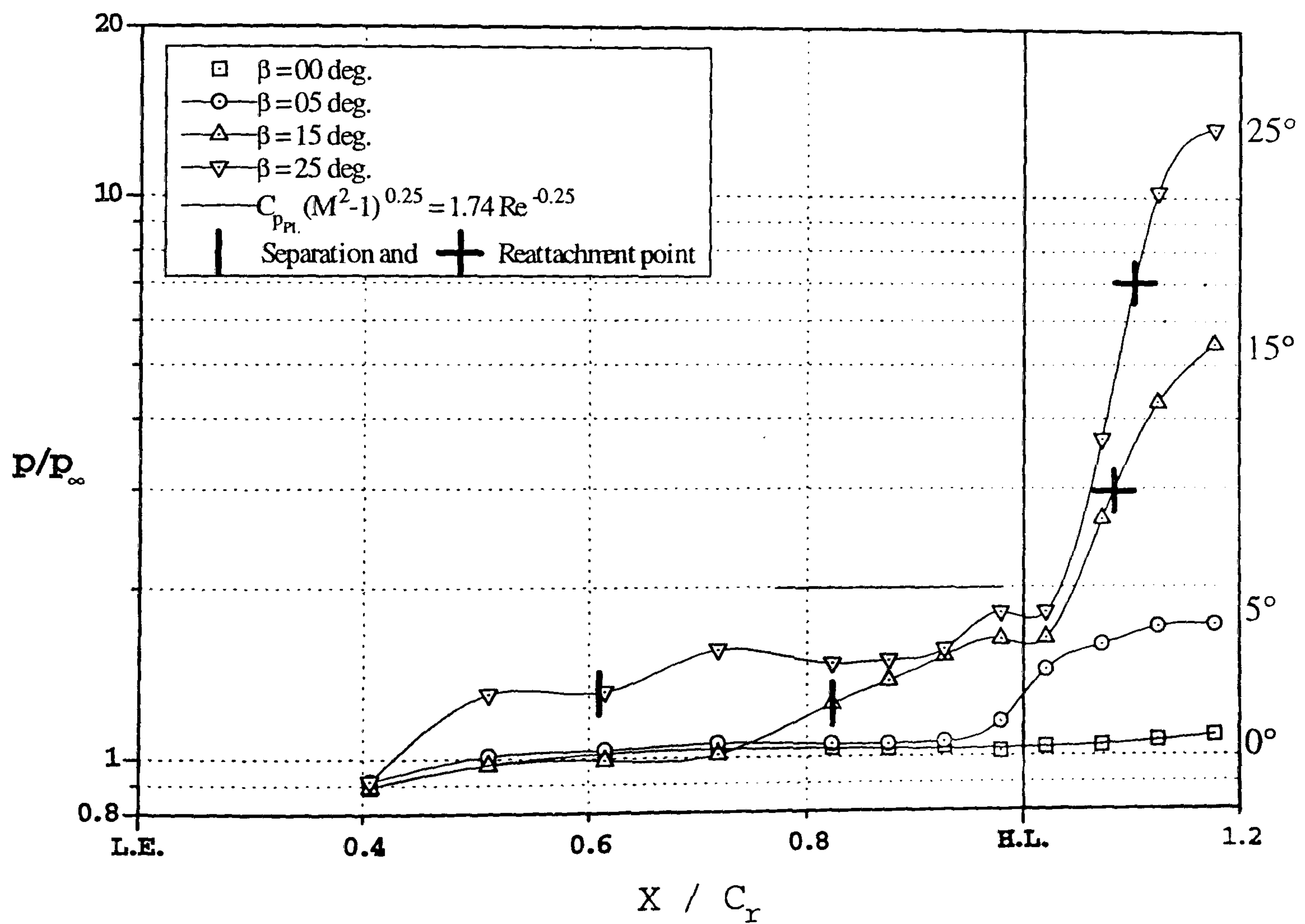


Figure 50. Effect of α and β on the flap peak pressure variations across the span.

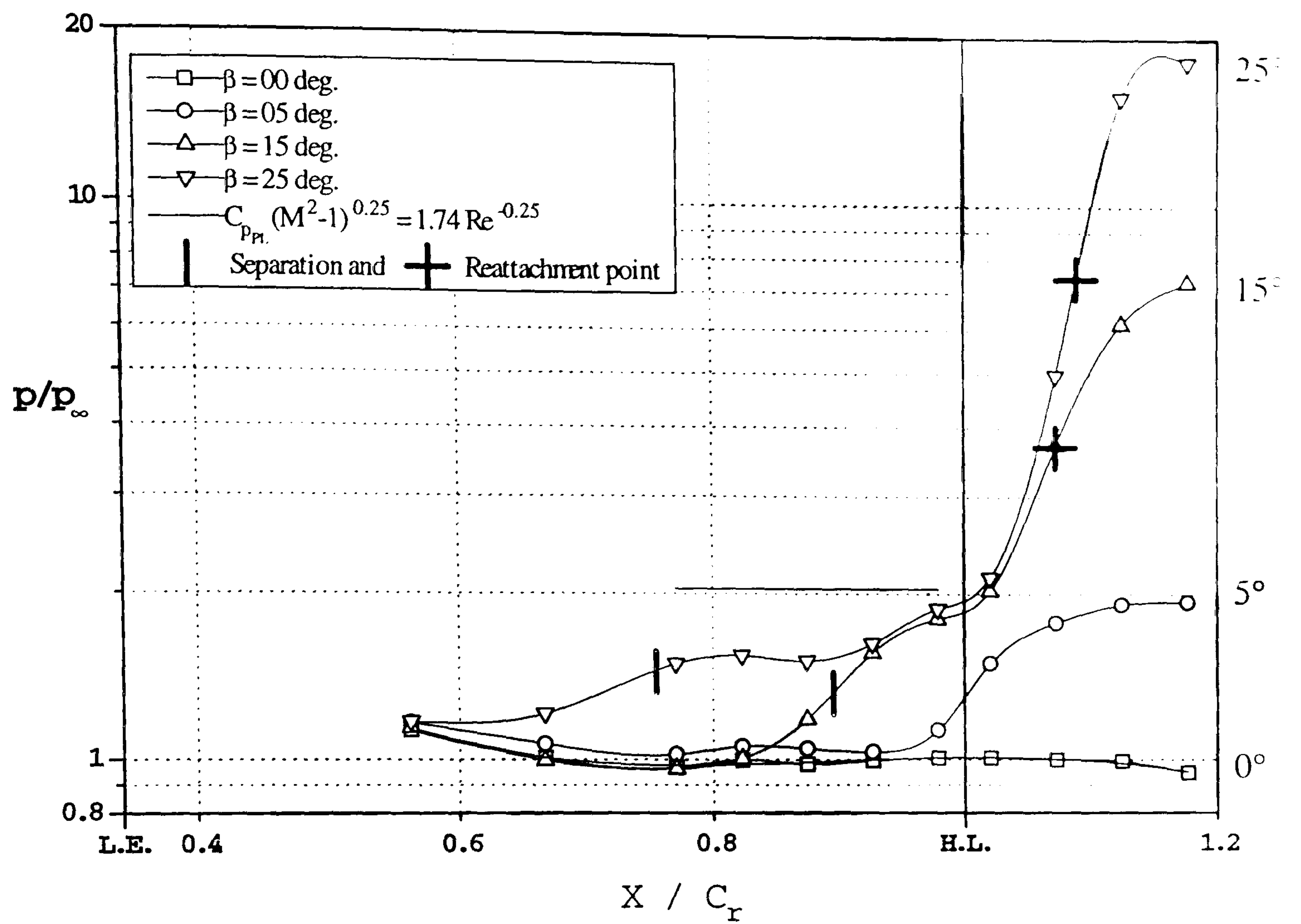


(a) $Y/b = 0.31$

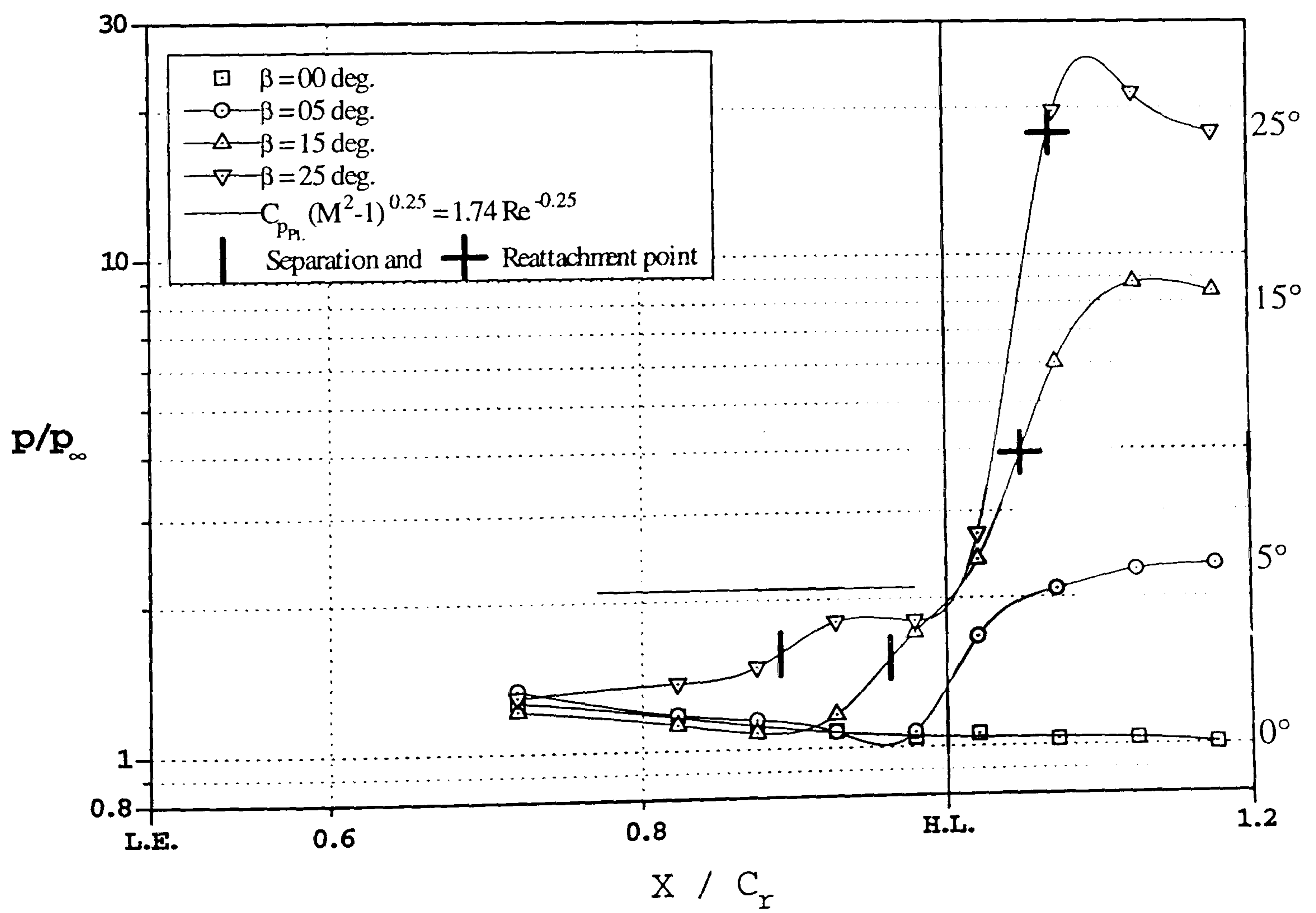


(b) $Y/b = 0.44$

Figure 51. Chordwise pressure distribution at different spanwise locations on the windward surface of the wing for a change in β from 0 to 25° and at $\alpha = 0^\circ$.

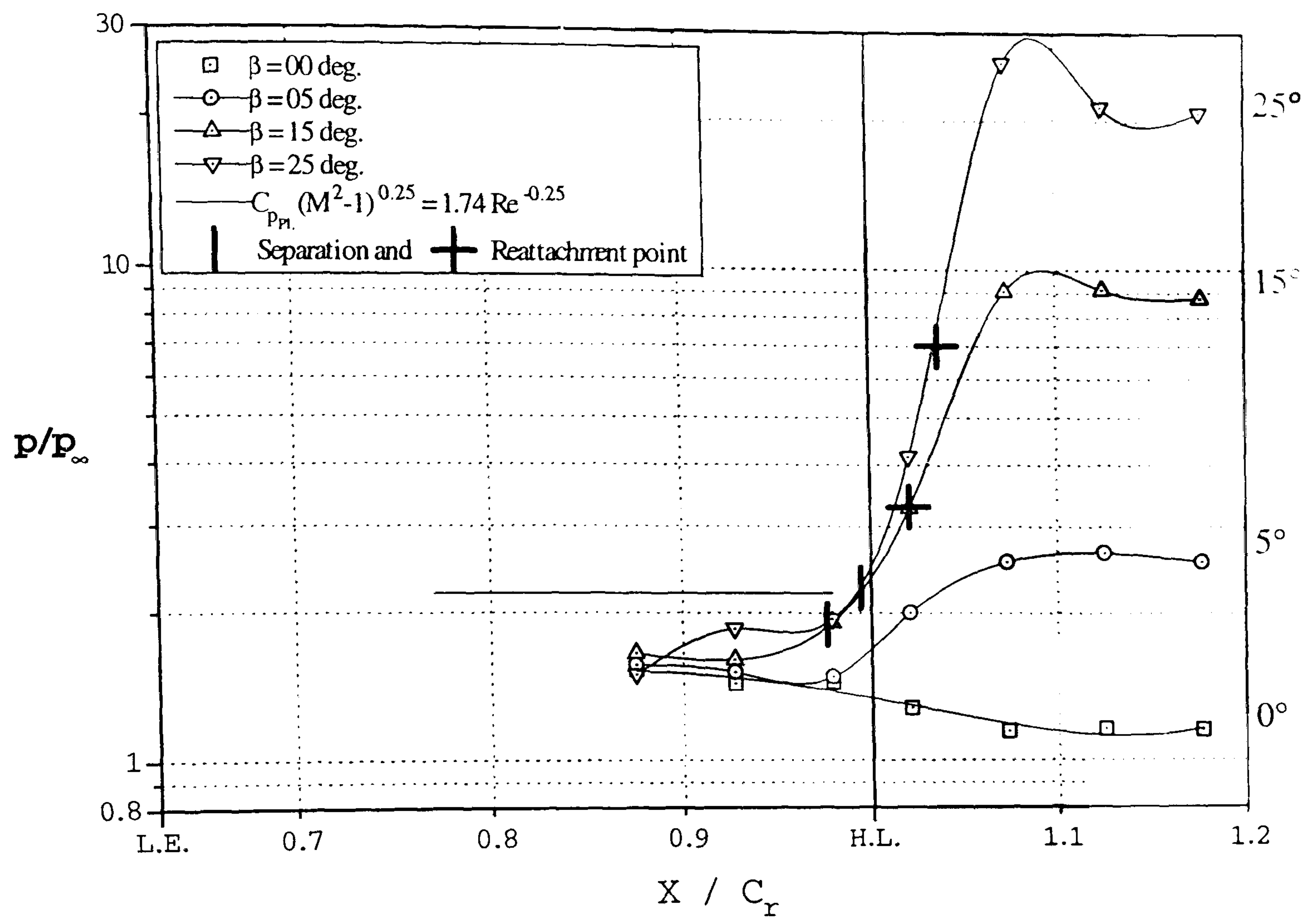


(c) $Y/b = 0.56$



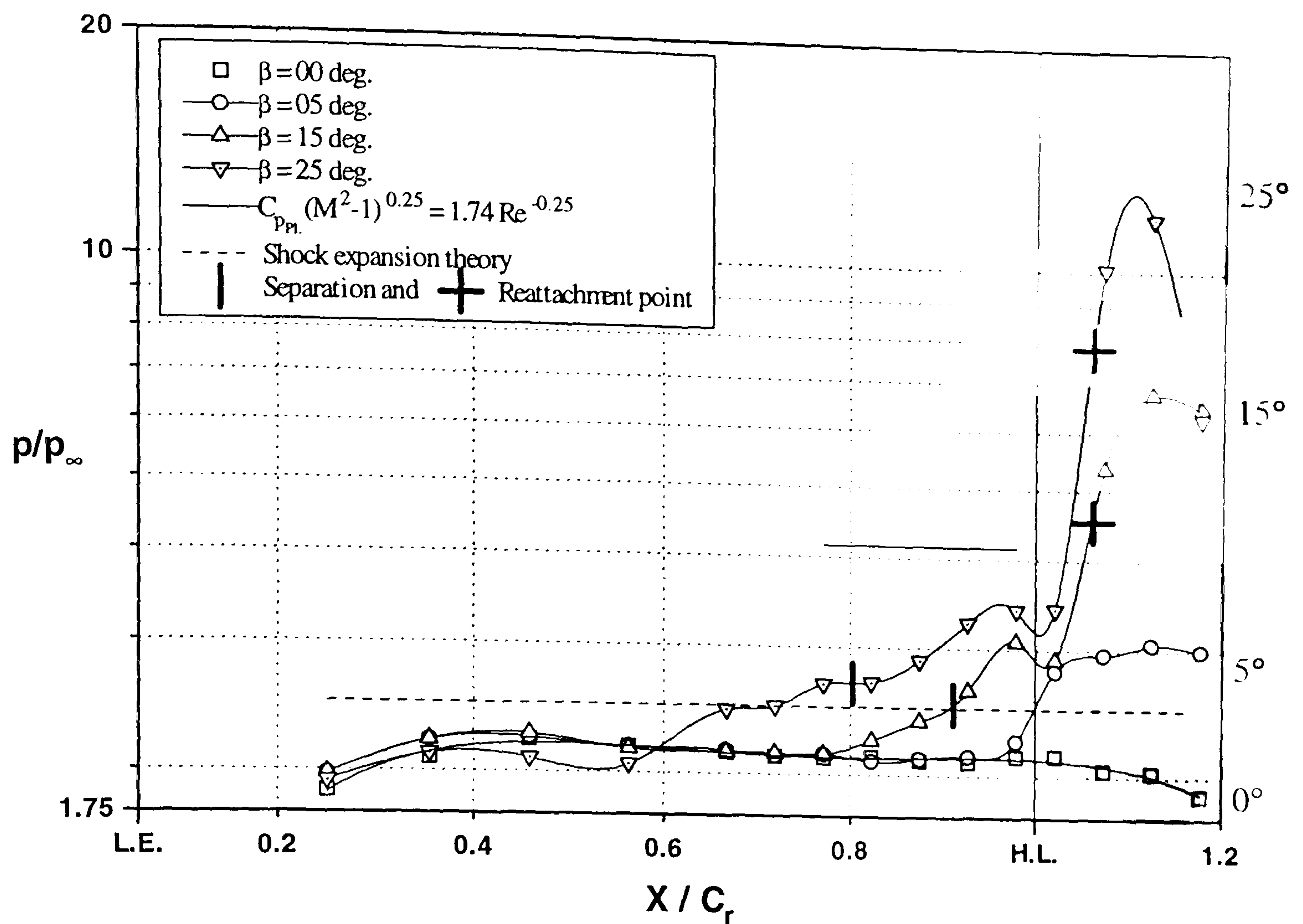
(d) $Y/b = 0.69$

Figure 51. Continued.

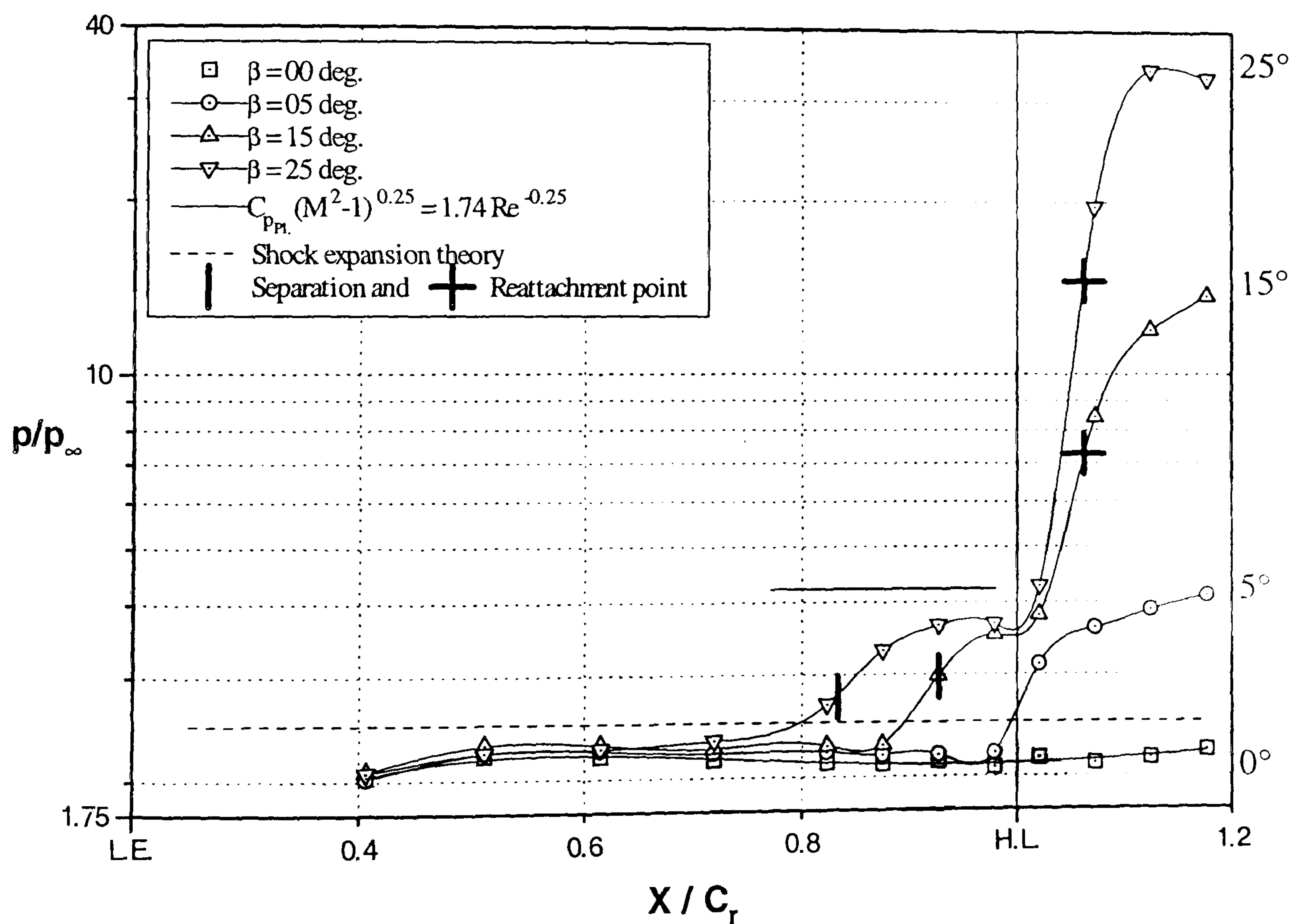


(e) $Y/b = 0.82$

Figure 51. Continued.

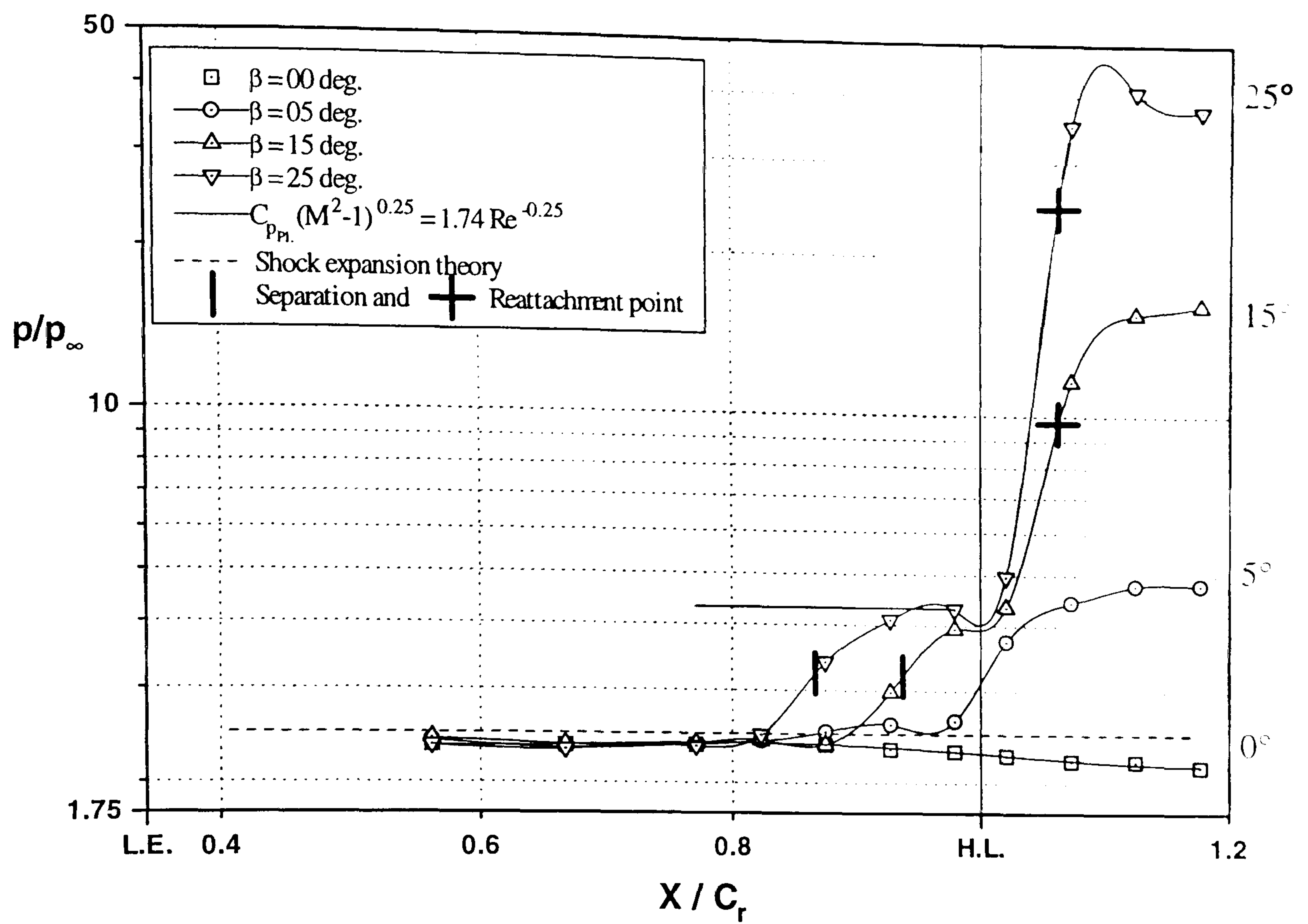


(a) $Y/b = 0.31$

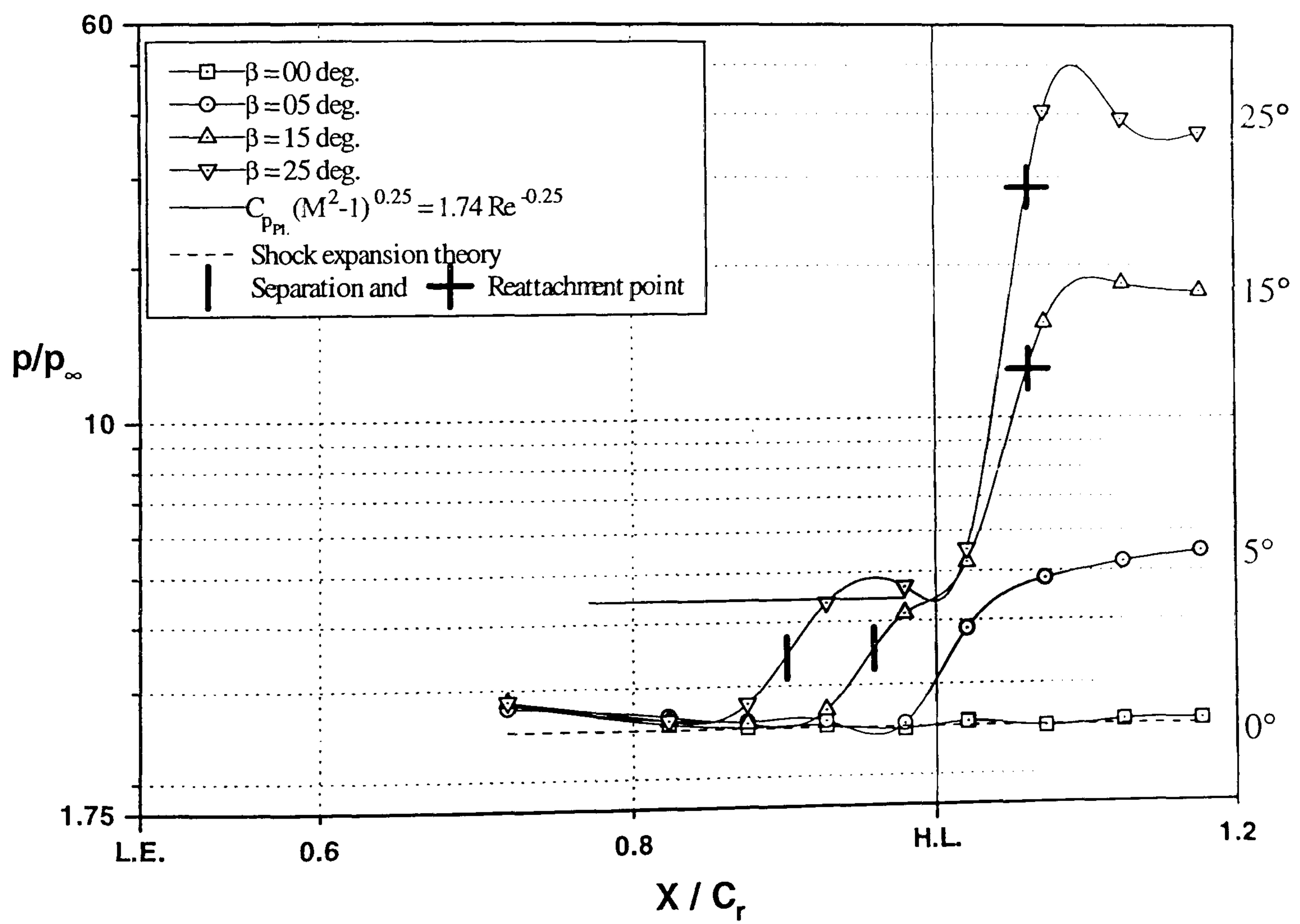


(b) $Y/b = 0.44$

Figure 52. Chordwise pressure distribution at different spanwise locations on the windward surface of the wing for a change in β from 0 to 25° and at $\alpha = 5^\circ$.

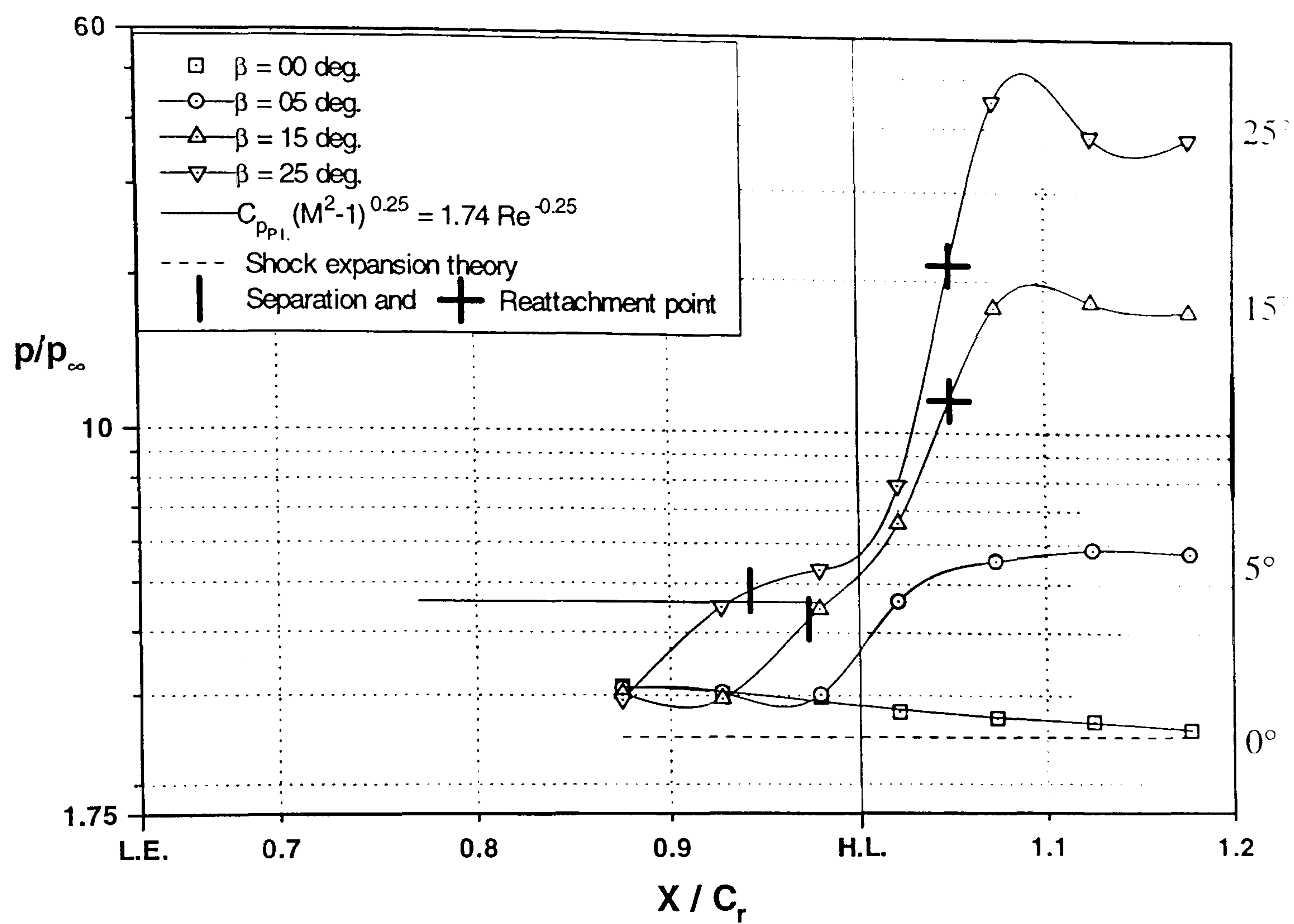


(c) $Y/b = 0.56$



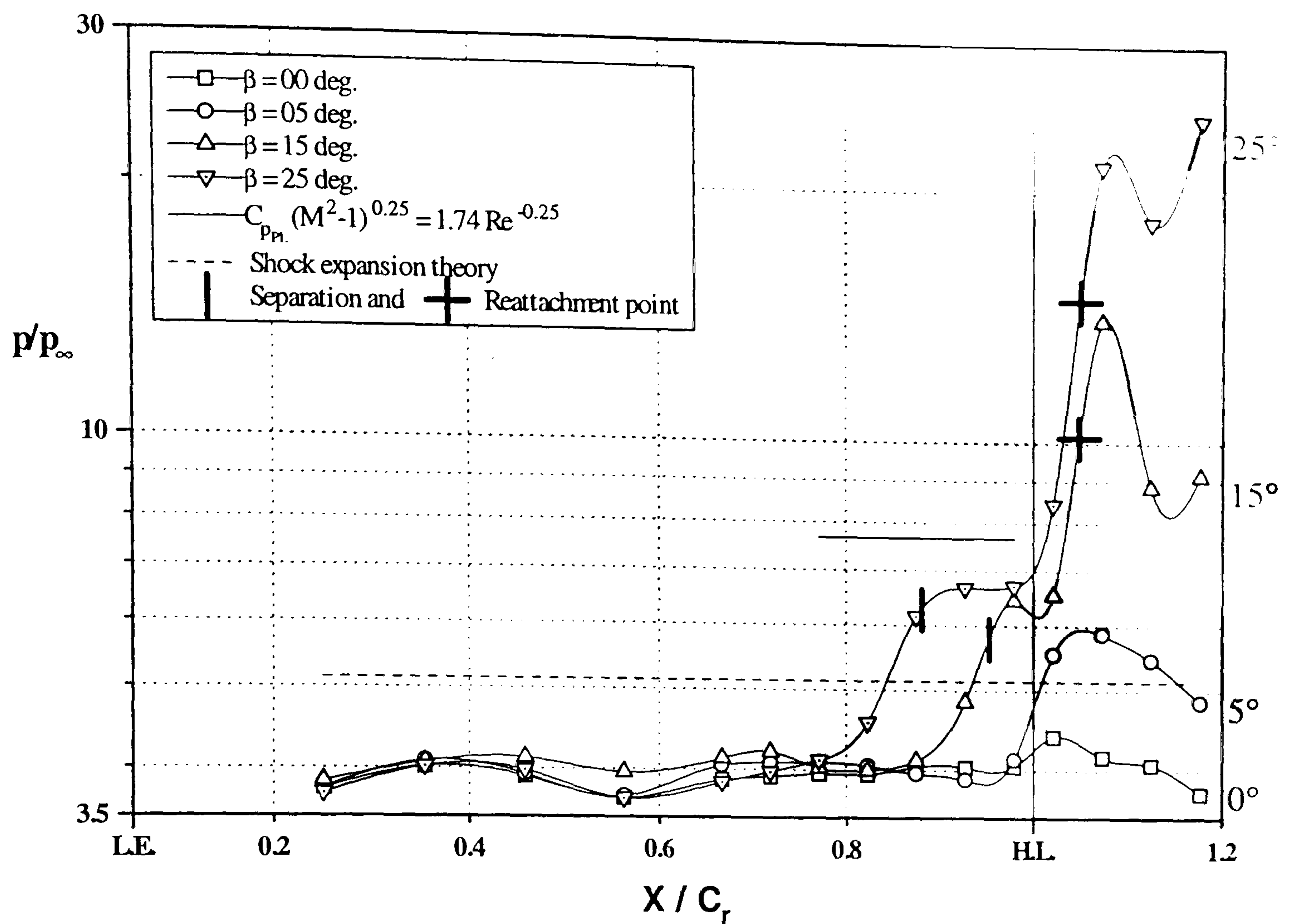
(d) $Y/b = 0.69$

Figure 52. Continued.

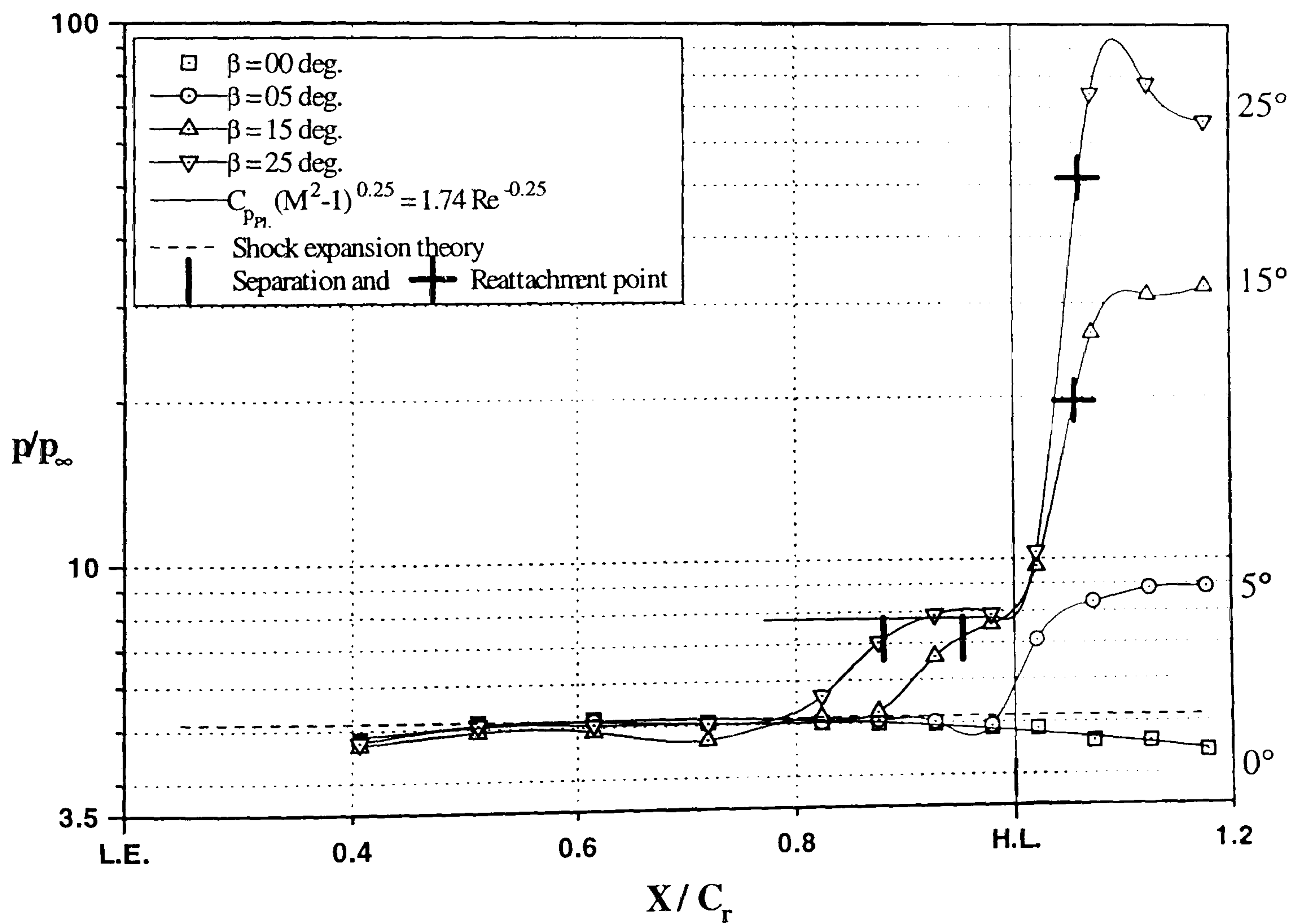


(e) $Y/b = 0.82$

Figure 52. Continued.

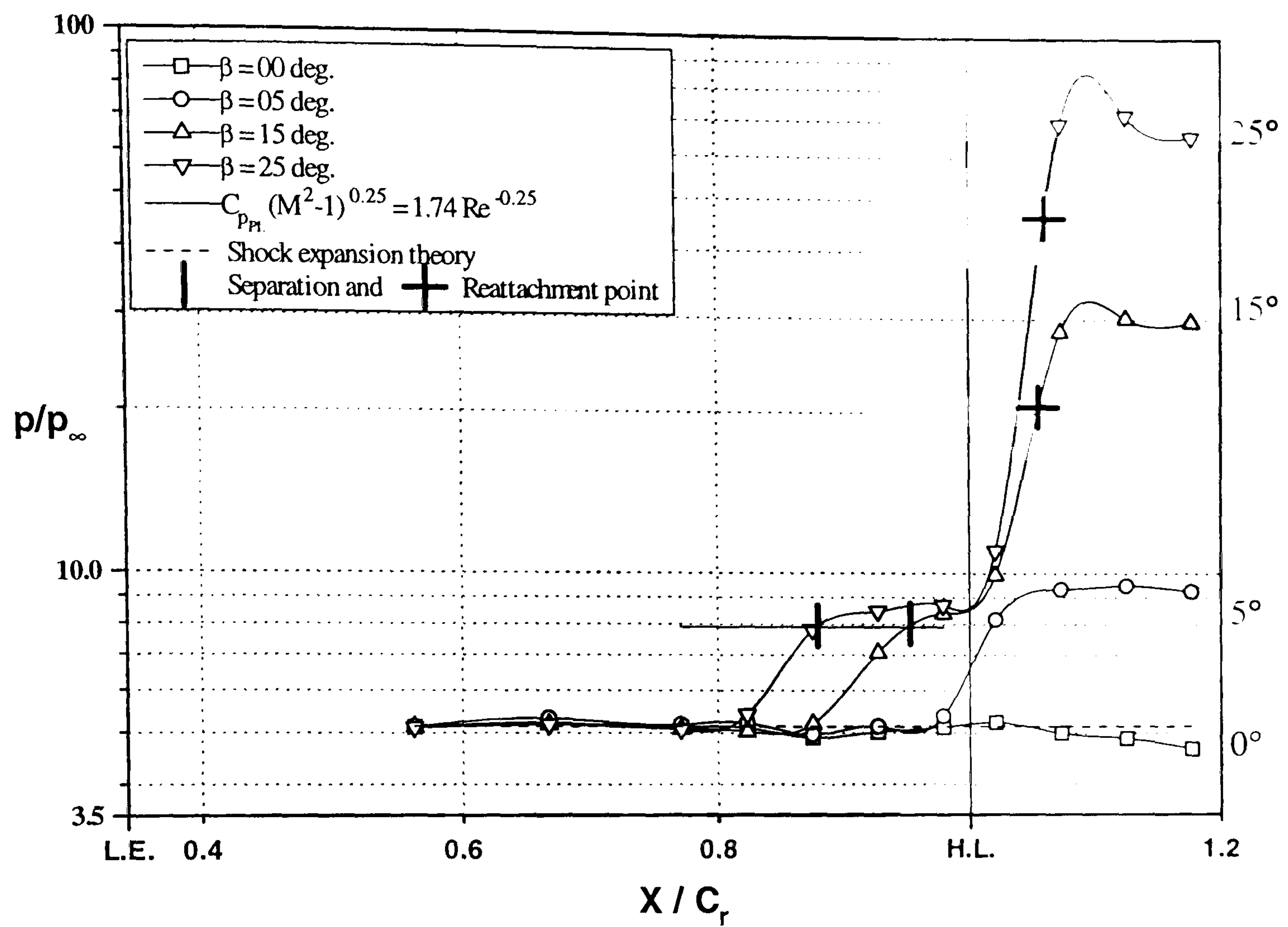


(a) $Y/b = 0.31$

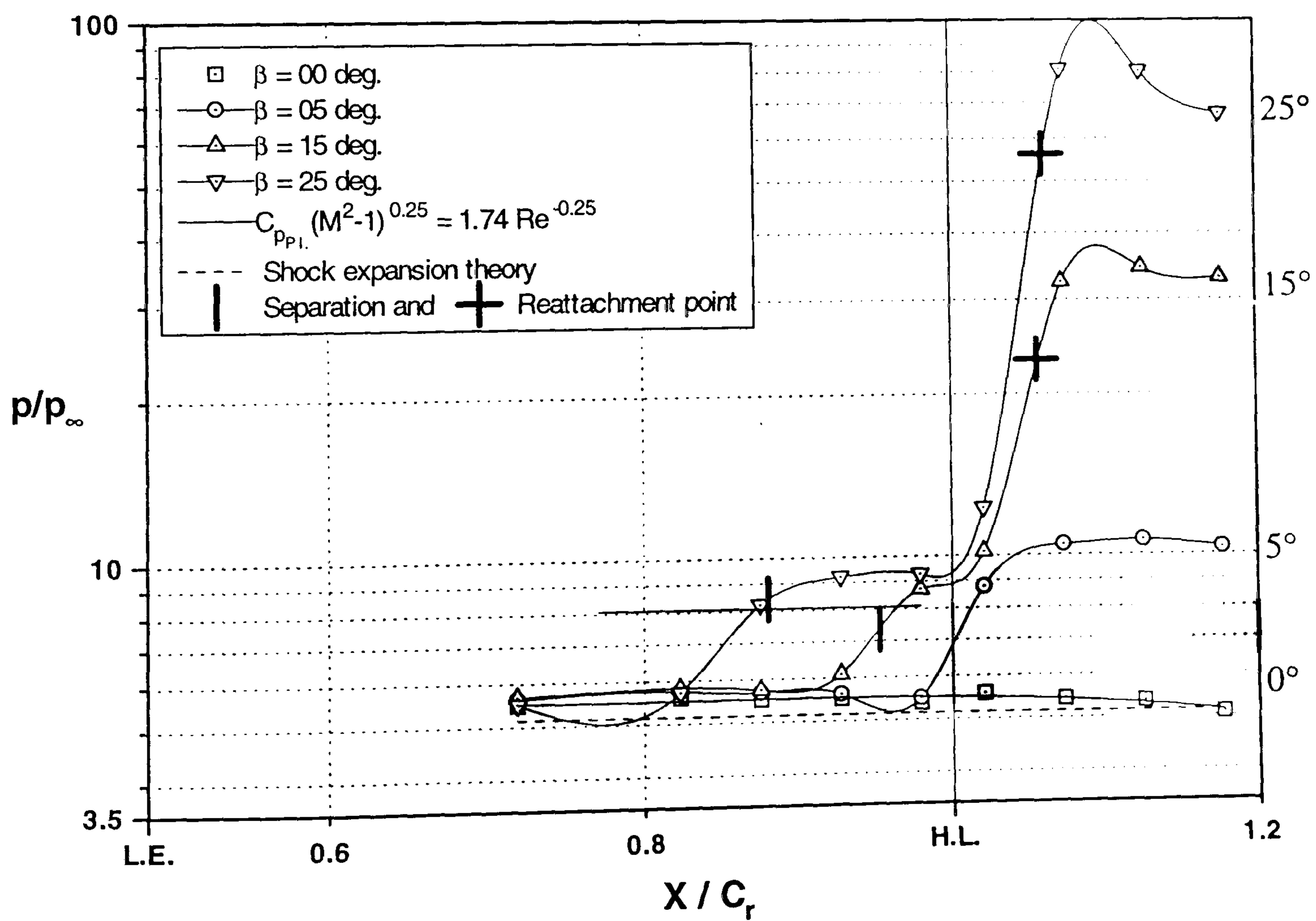


(b) $Y/b = 0.44$

Figure 53. Chordwise pressure distribution at different spanwise locations on the windward surface of the wing for a change in β from 0 to 25° and at $\alpha = 10^\circ$.

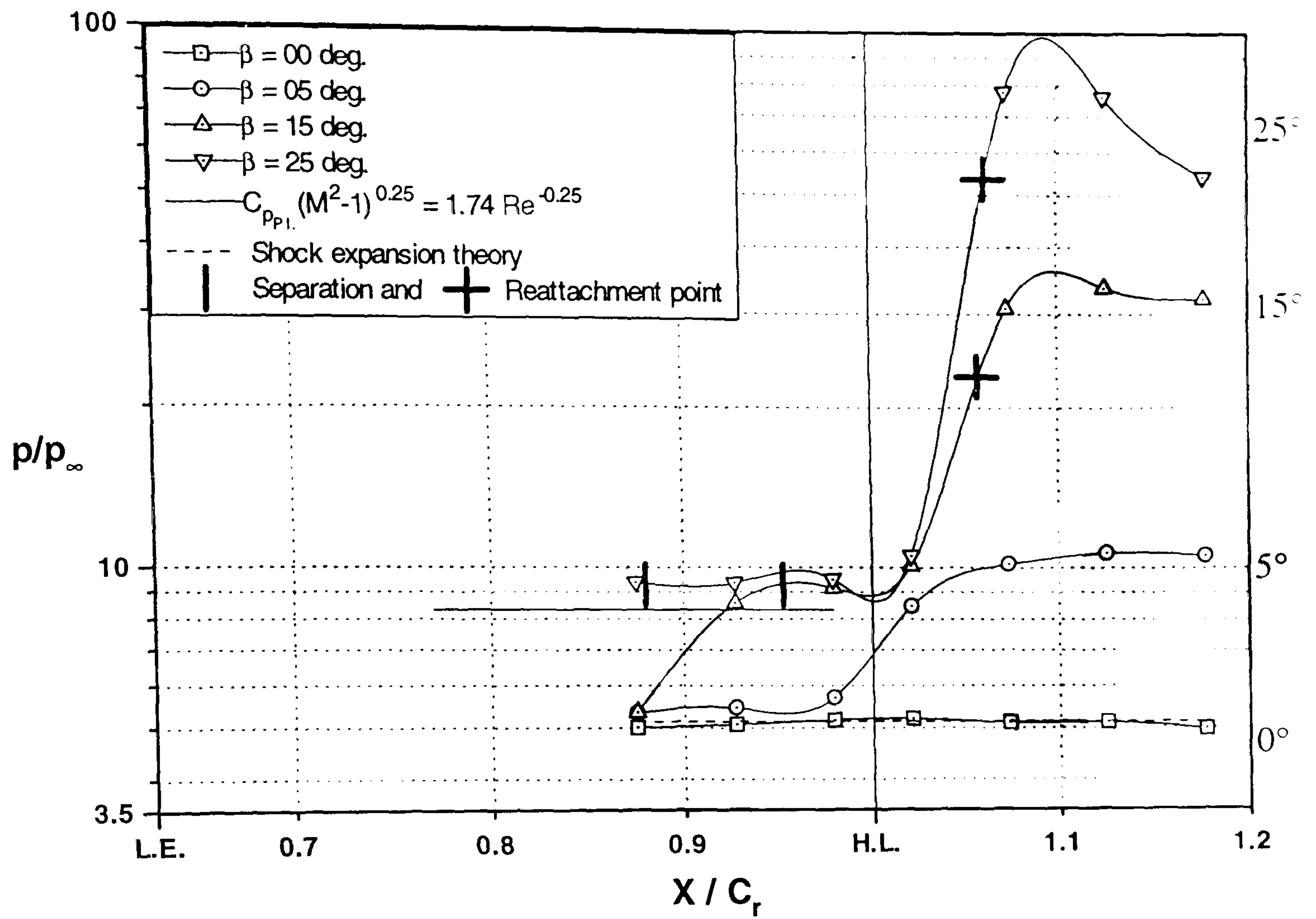


(c) $Y/b = 0.56$



(d) $Y/b = 0.69$

Figure 53. Continued.



(e) $Y/b = 0.82$

Figure 53. Continued.

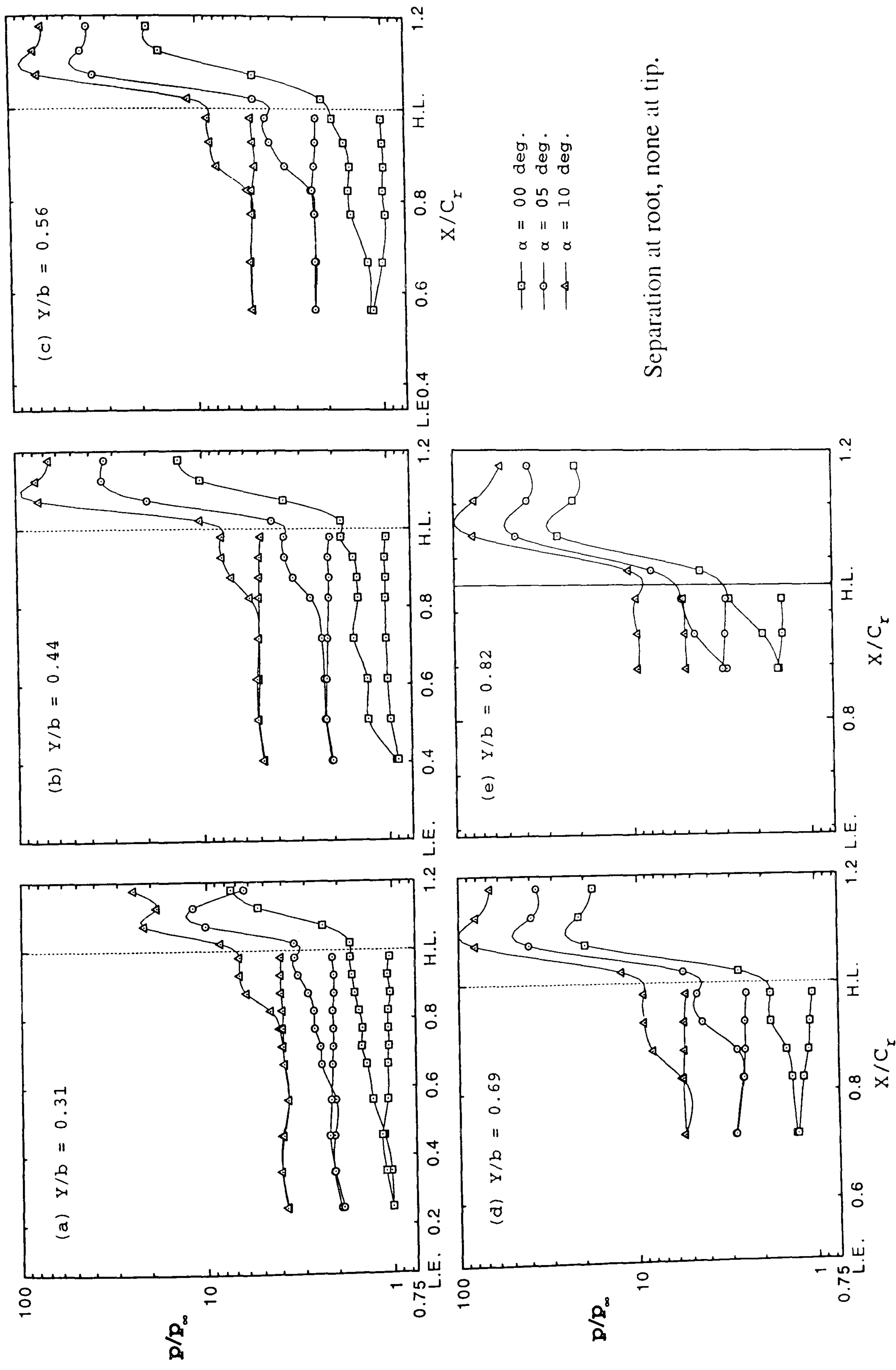


Figure 54. Variation in the chordwise pressure distribution with α at various spanwise locations for $\beta = 15^\circ$.

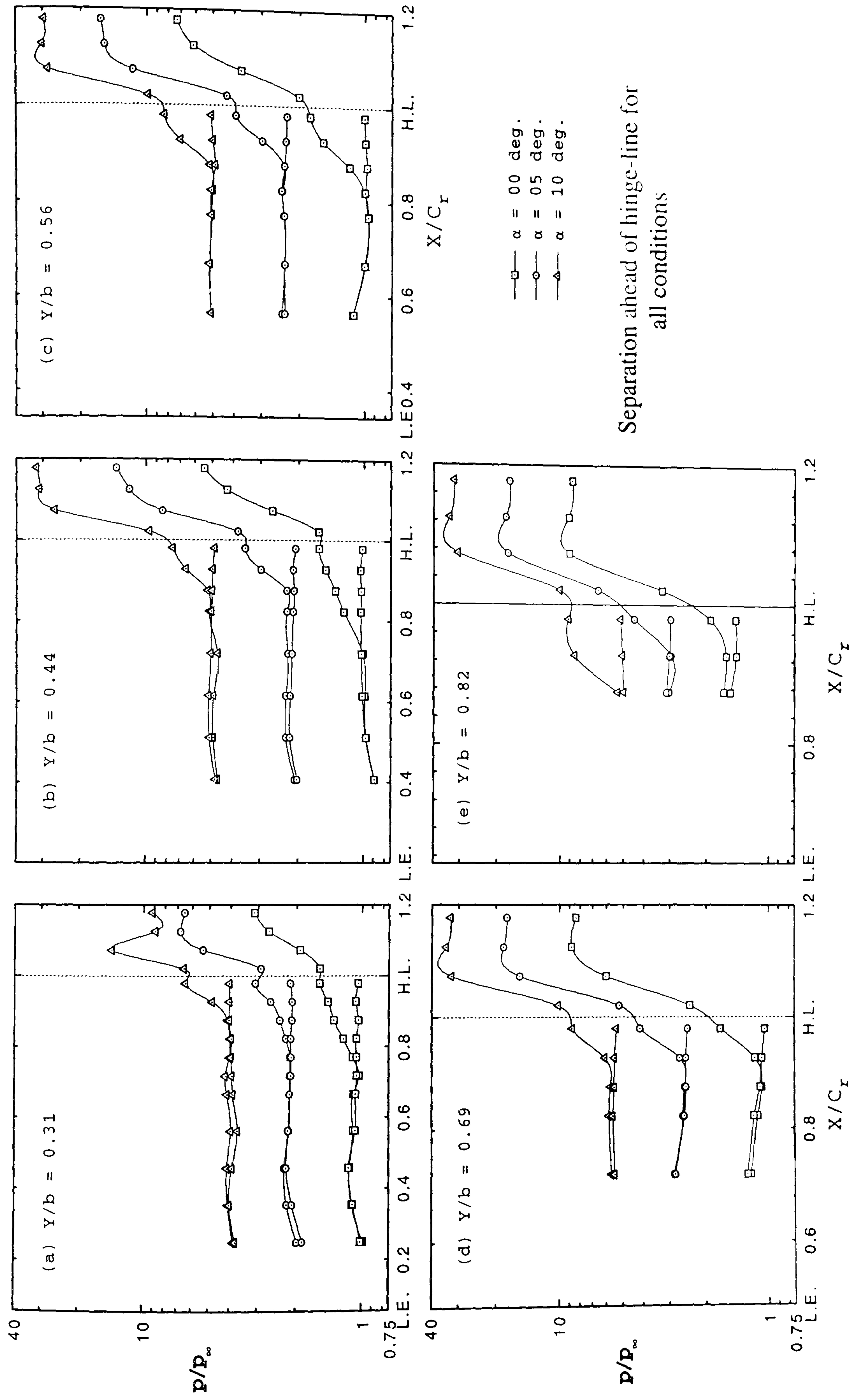
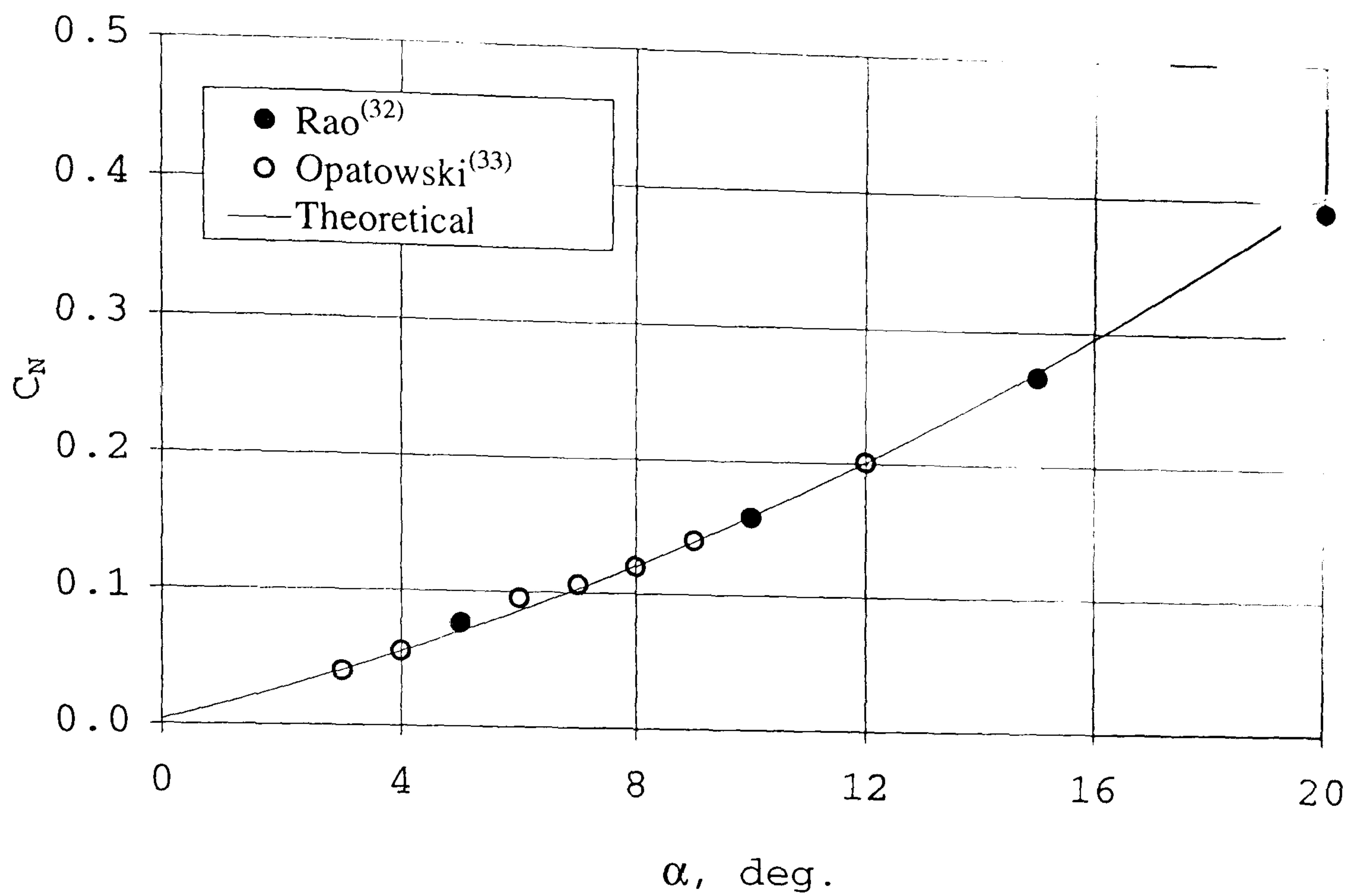
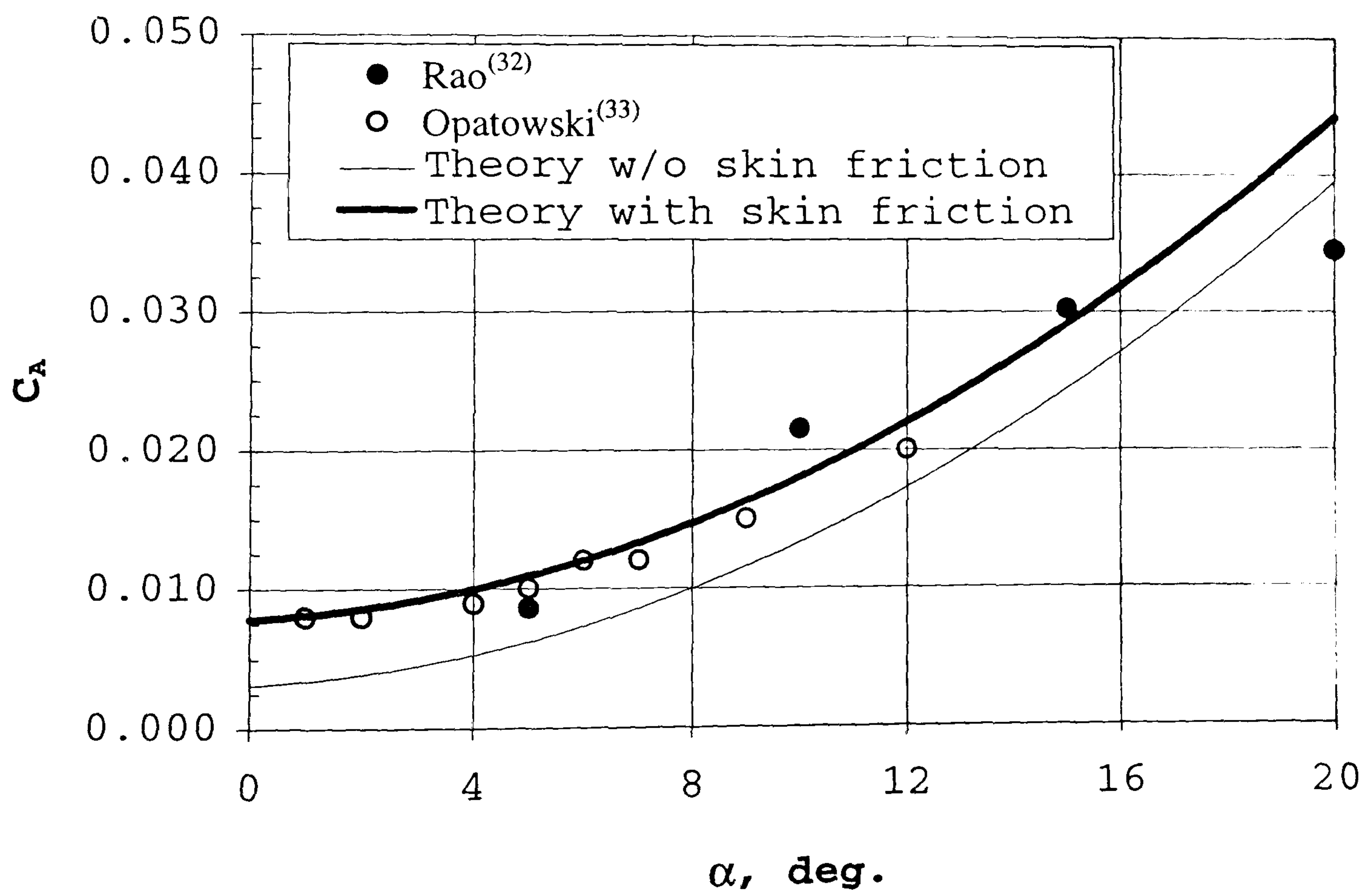


Figure 55. Variation in the chordwise pressure distribution with α at various spanwise locations for $\beta = 25^\circ$.



(a) Normal force coefficient



(b) Axial force coefficient

Figure 56. Aerodynamic characteristics of a caret delta wing.

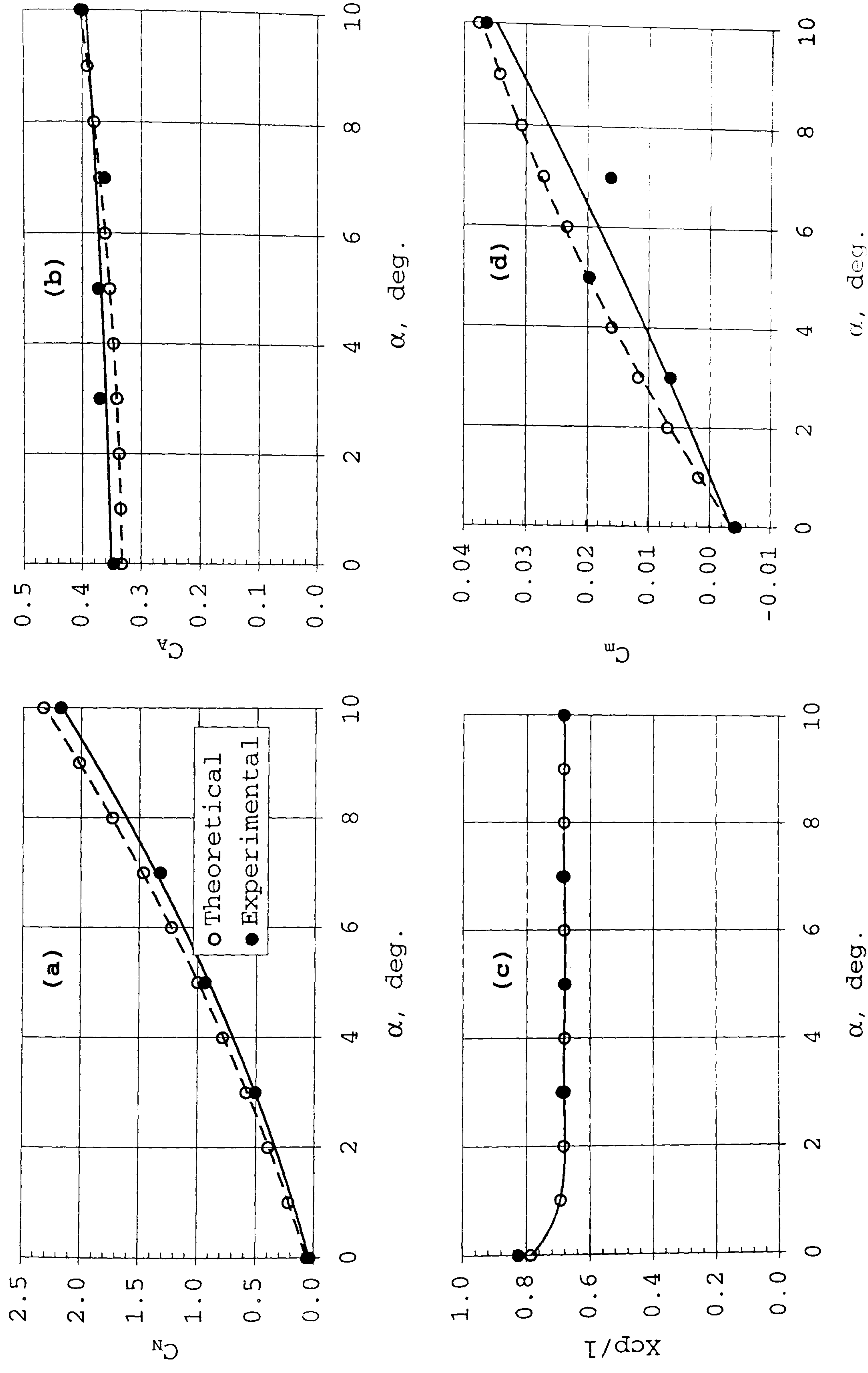


Figure 57. Aerodynamic characteristics of wing-body configuration for $\beta = 0^\circ$

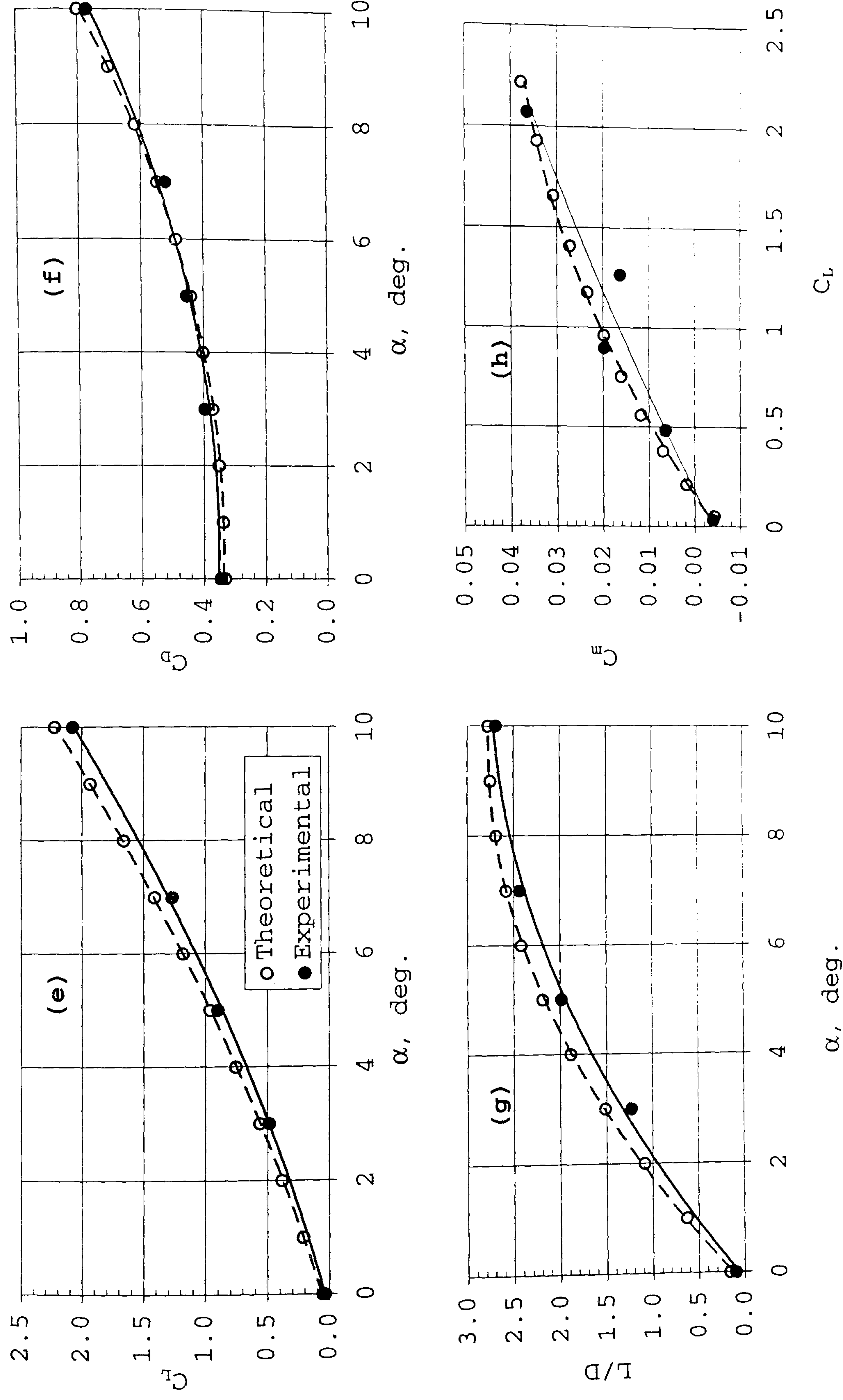


Figure 57. Continued.

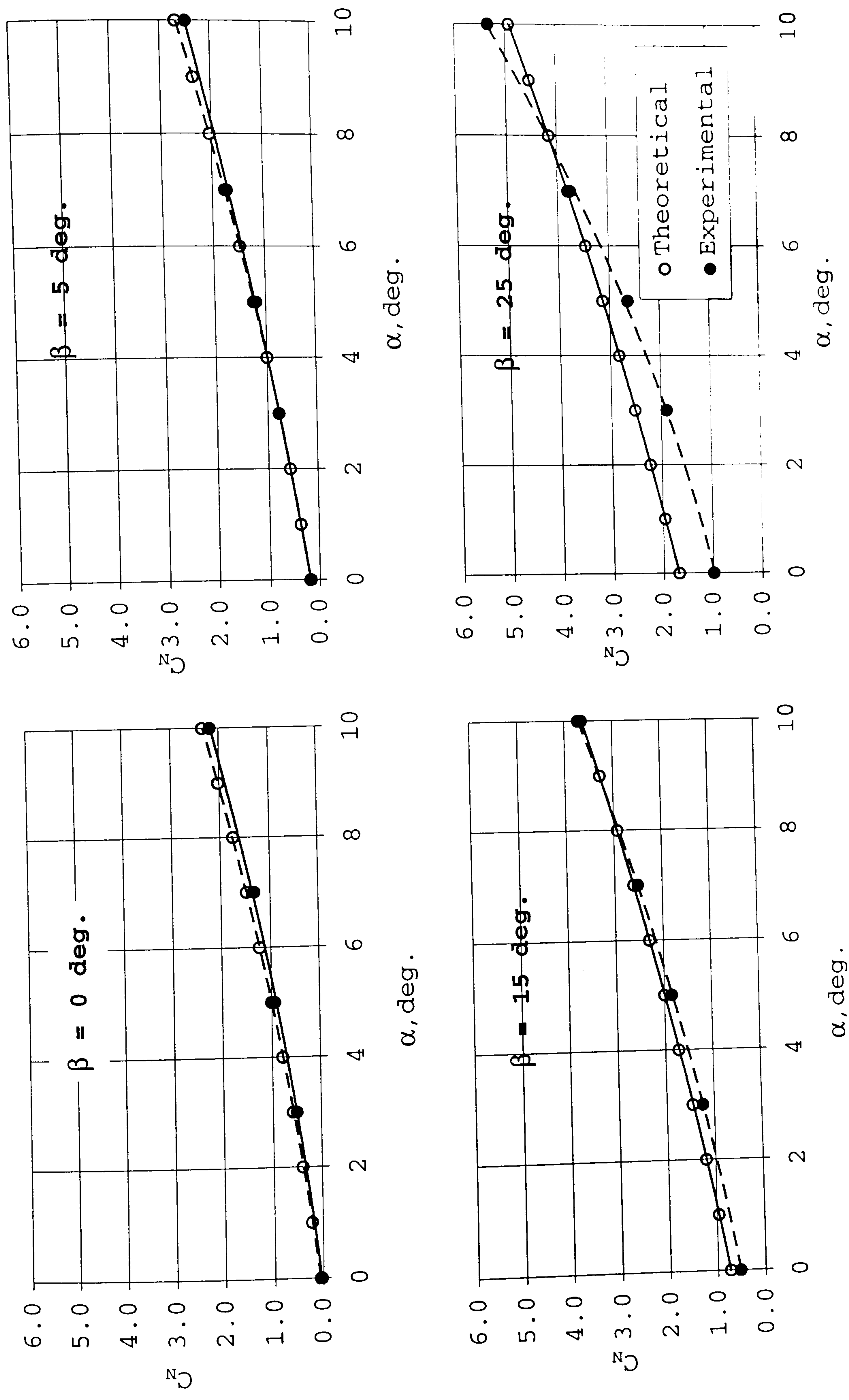


Figure 58(a). Variation of normal force coefficient with α for $\beta = 0, 5, 15, 25^\circ$.

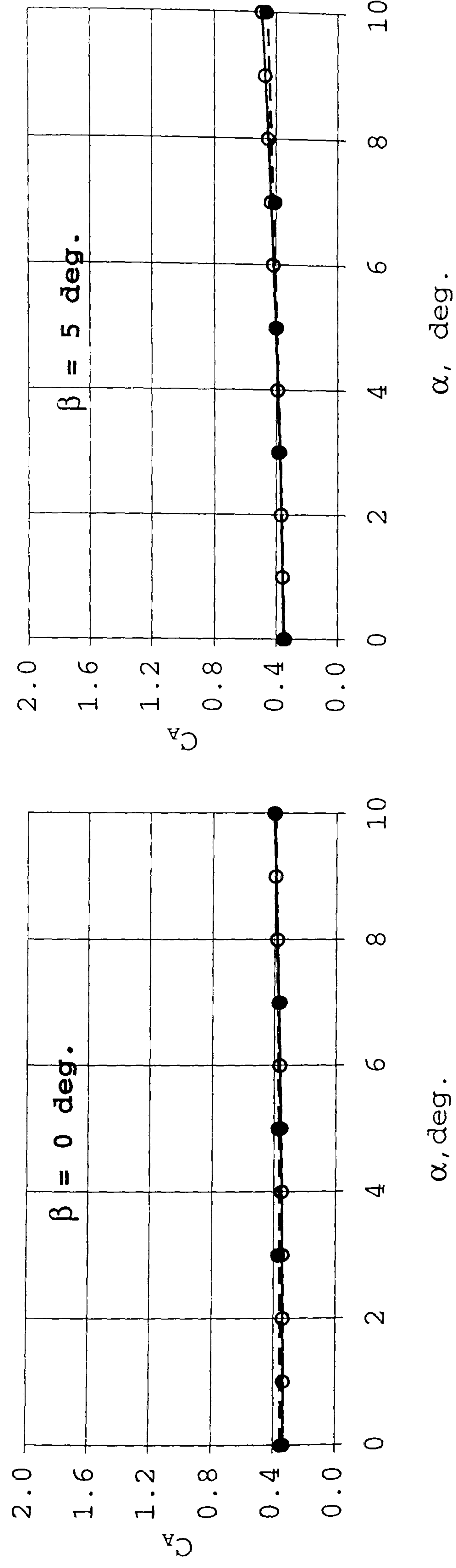
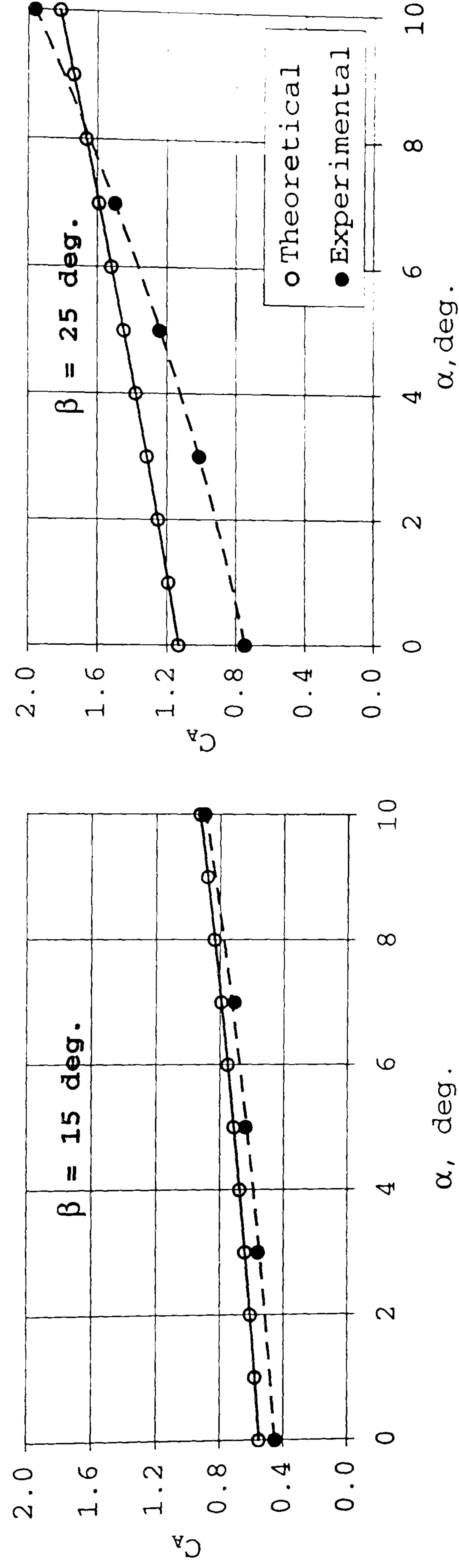


Figure 58(b). Variation of axial force coefficient with α for $\beta = 0, 5, 15, 25^\circ$.

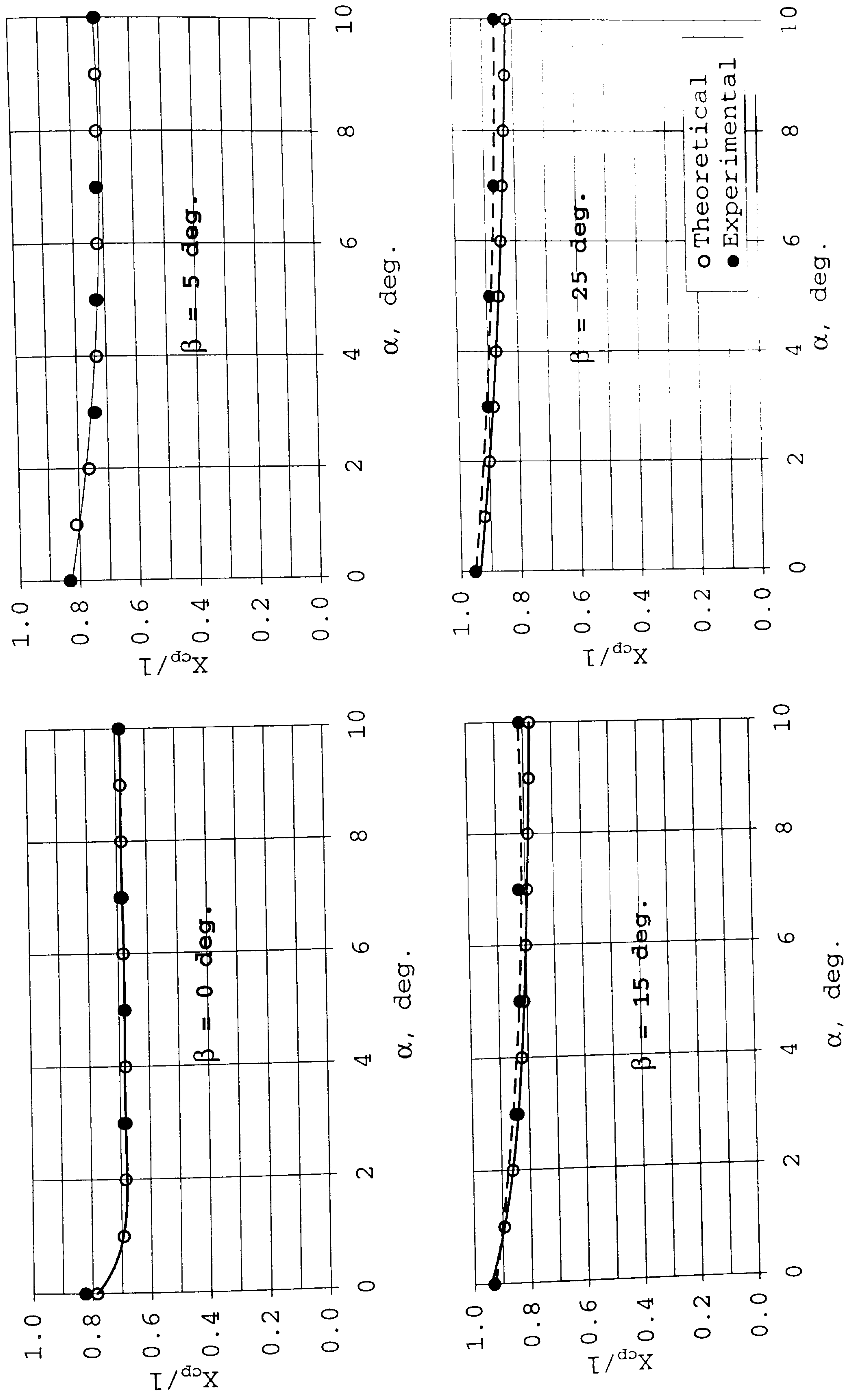


Figure 58(c). Variation of centre of pressure location with α for $\beta = 0, 5, 15, 25^\circ$.

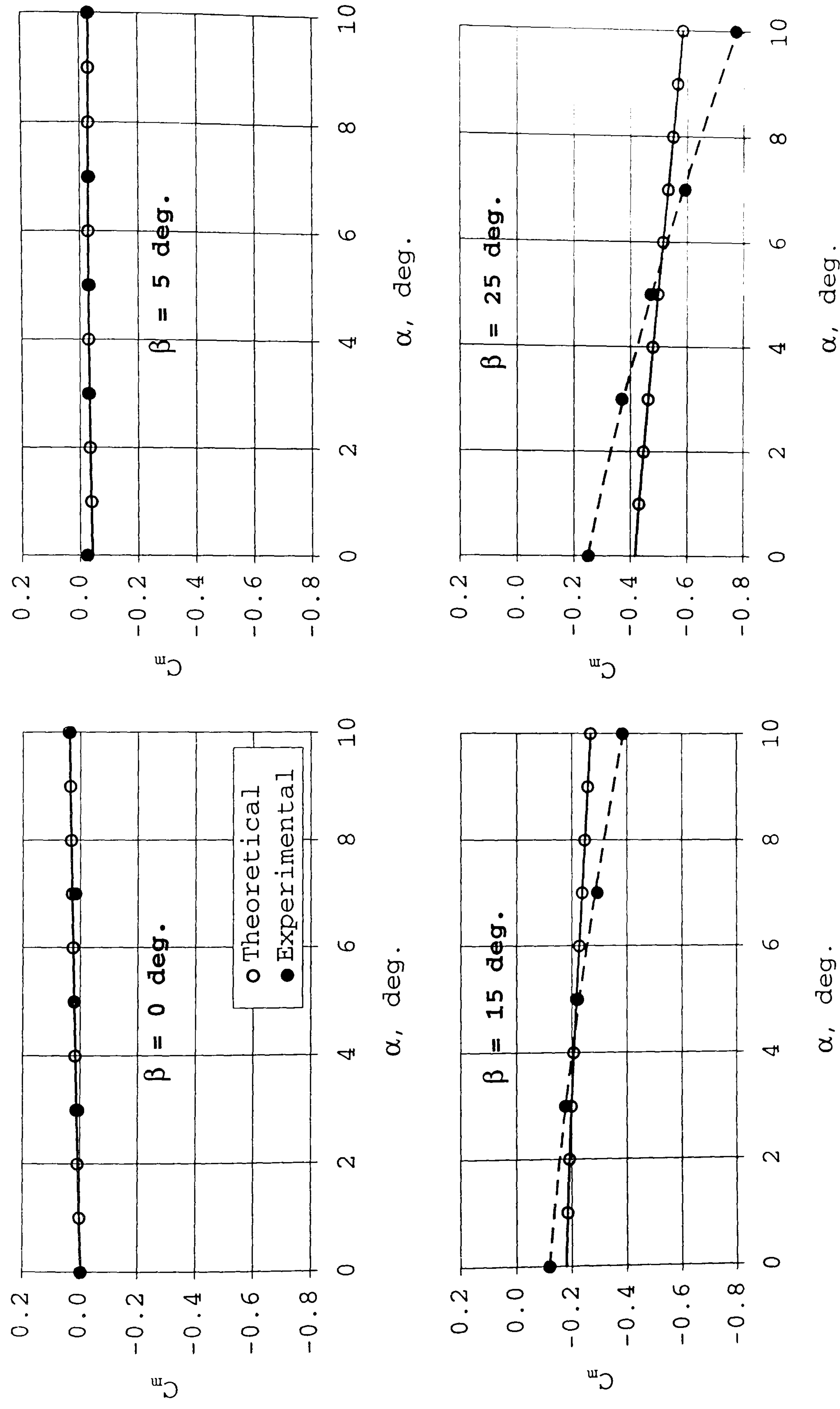


Figure 58(d). Variation of pitching moment coefficient with α for $\beta = 0, 5, 15, 25^\circ$.

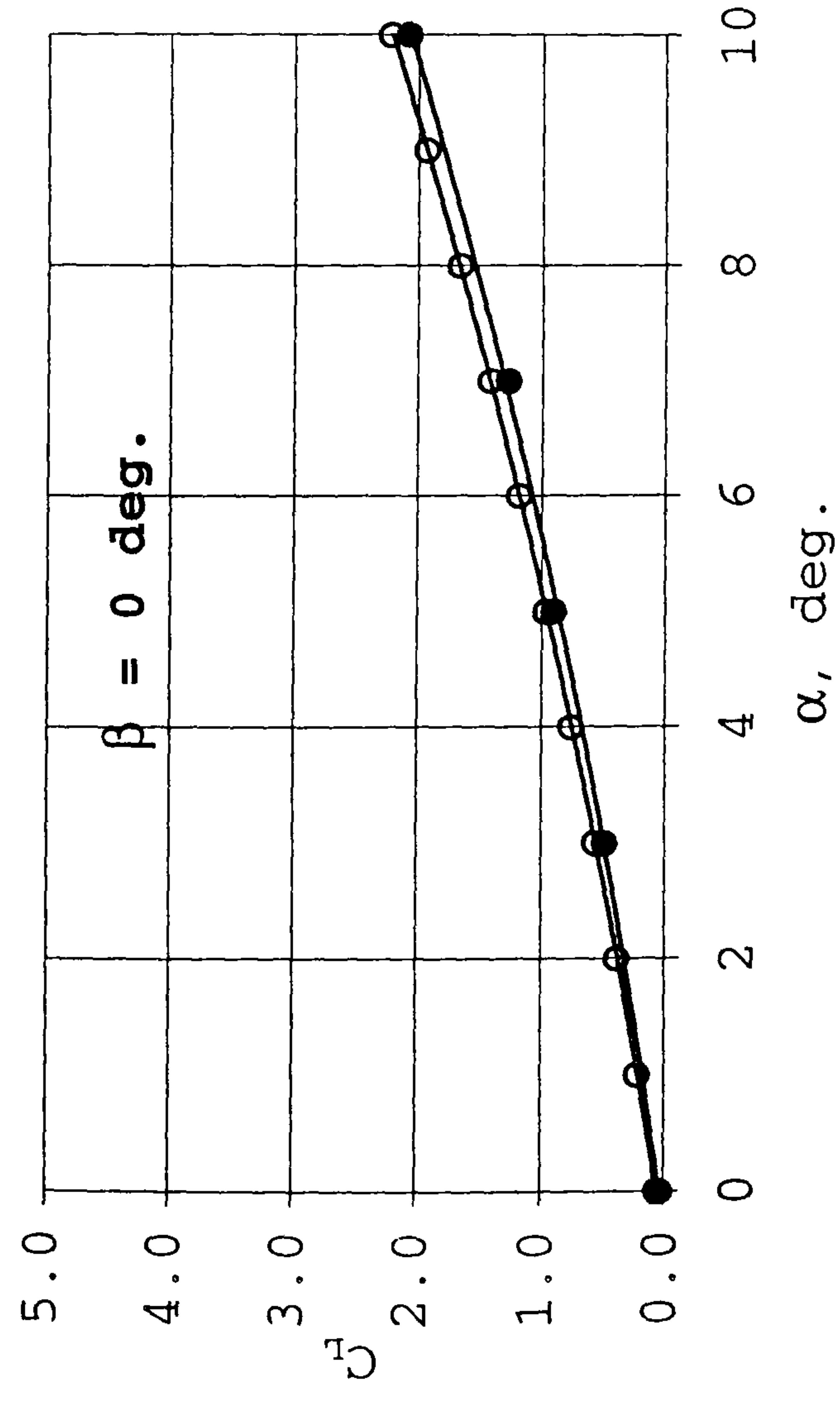
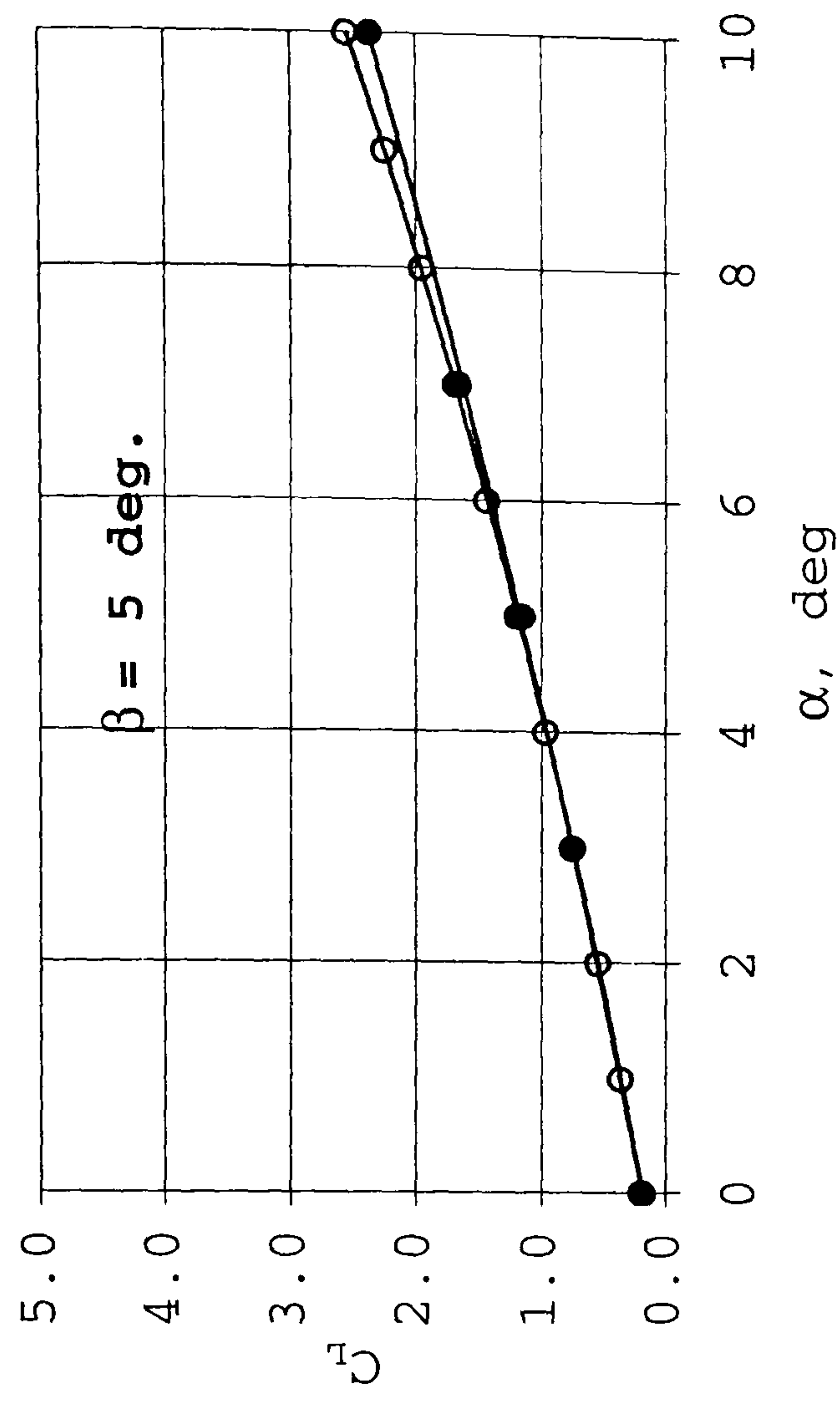
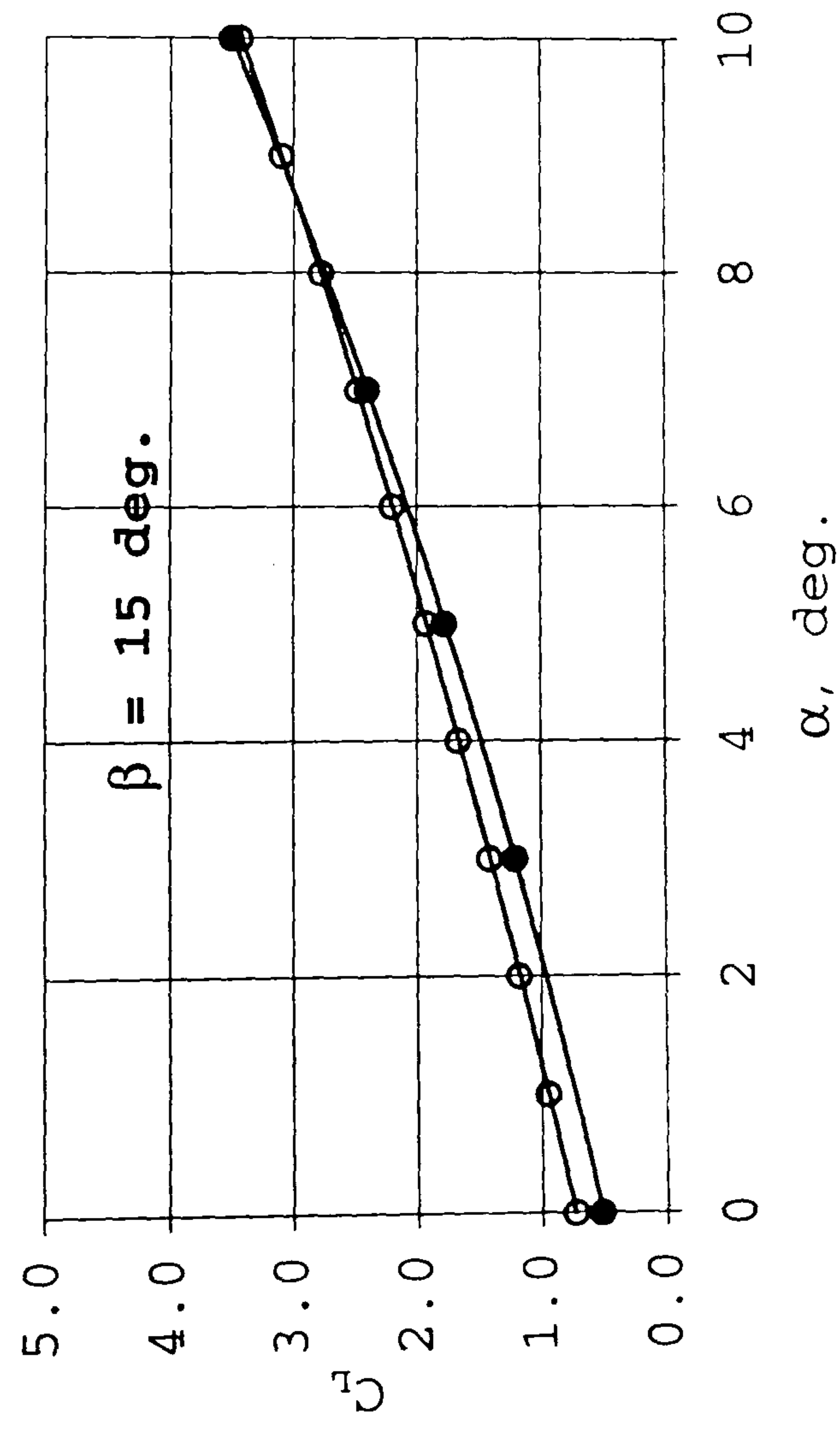
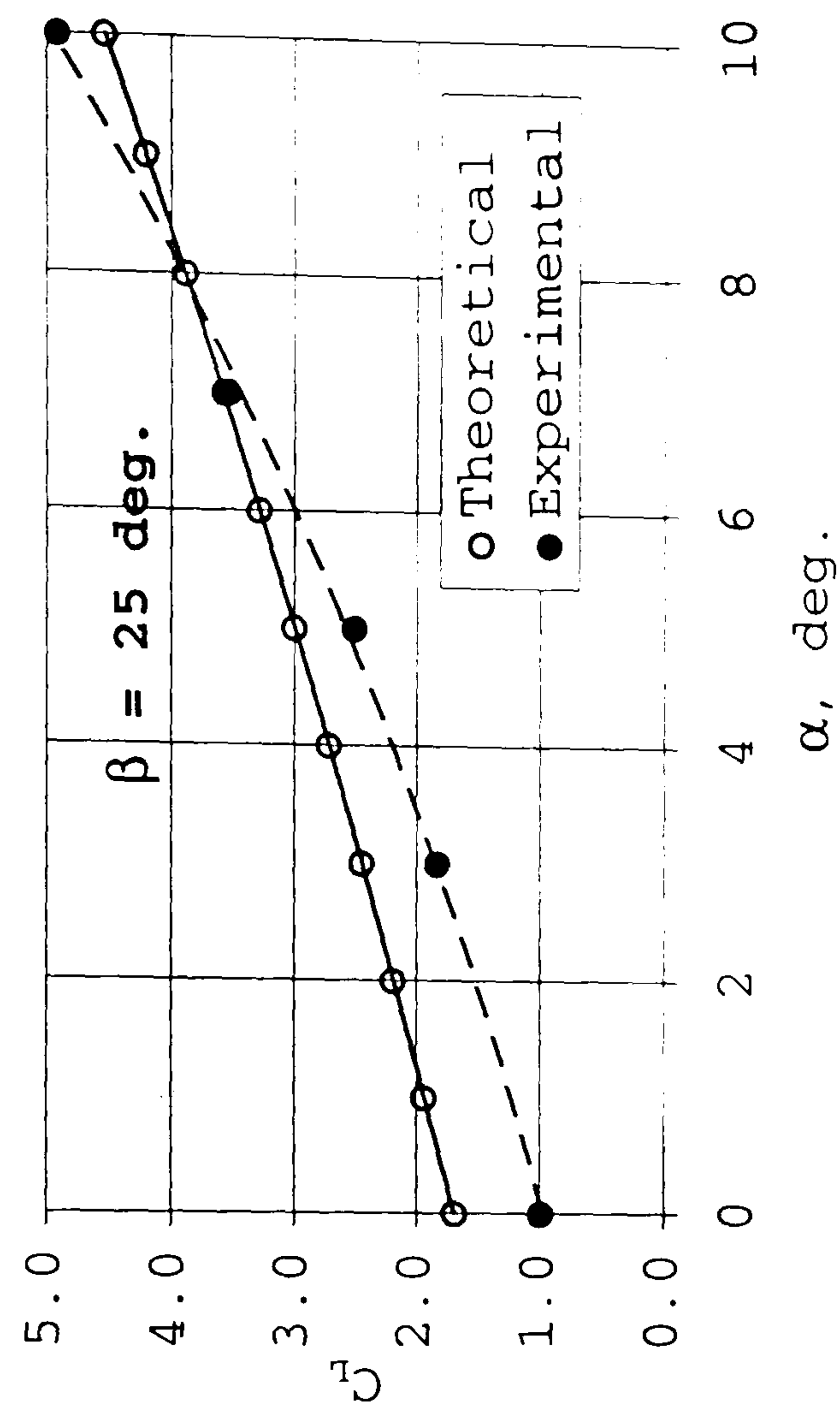


Figure 58(e). Variation of lift coefficient with α for $\beta = 0, 5, 15, 25^\circ$.

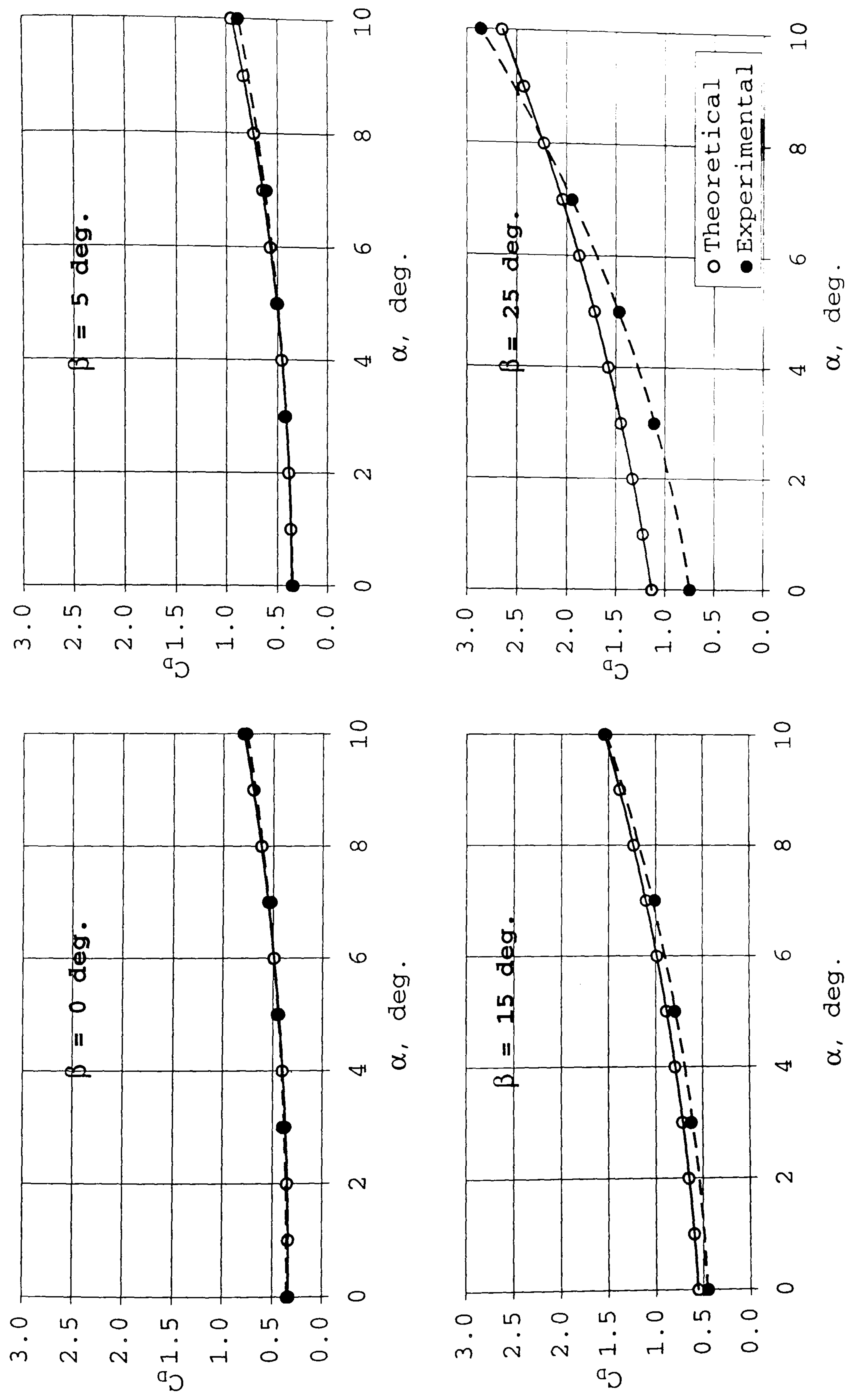


Figure 58(f). Variation of drag coefficient with α for $\beta = 0, 5, 15, 25^\circ$.

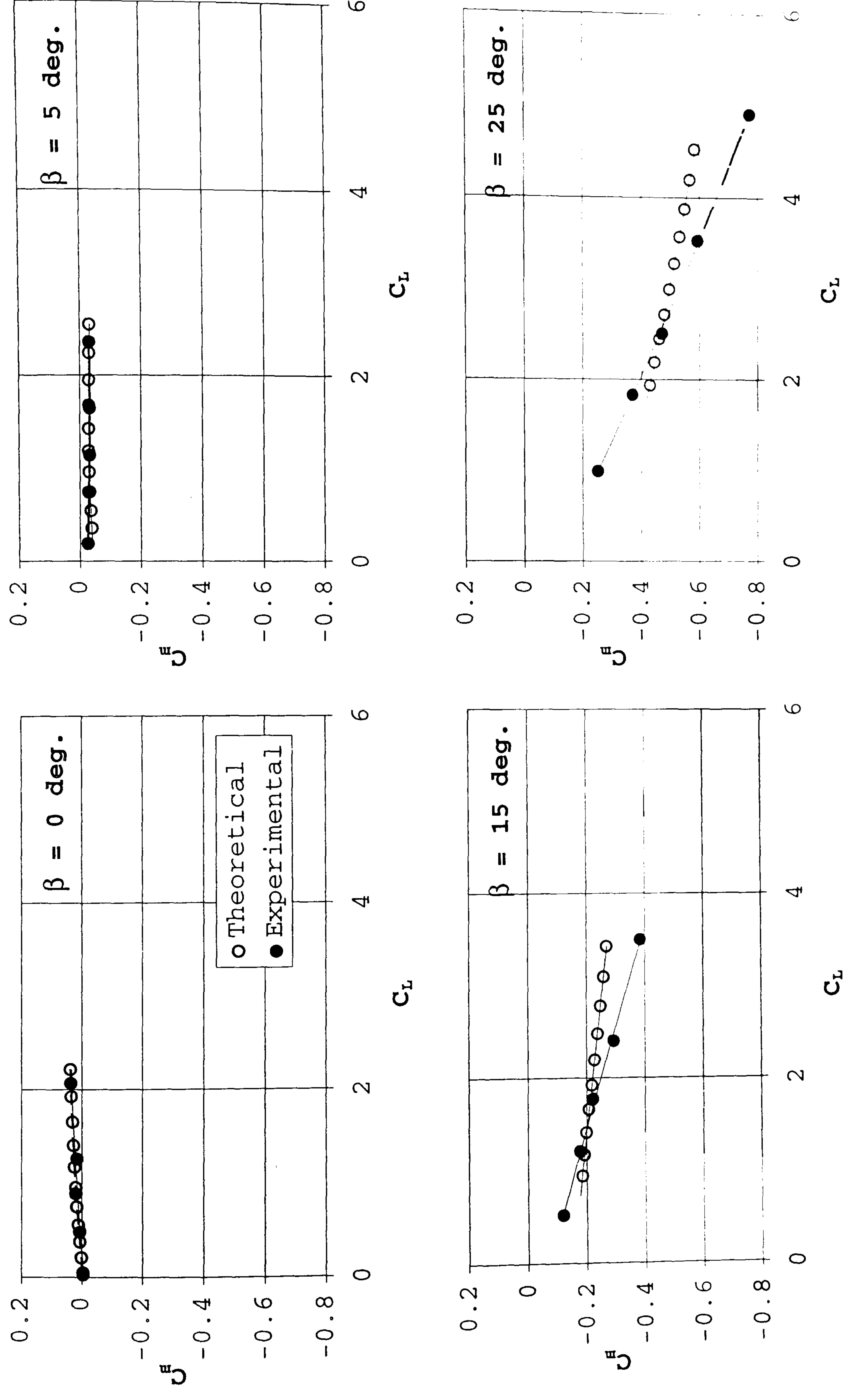


Figure 58(g). Variation of pitching moment coefficient with the lift coefficient with α for $\beta = 0, 5, 15, 25^\circ$.

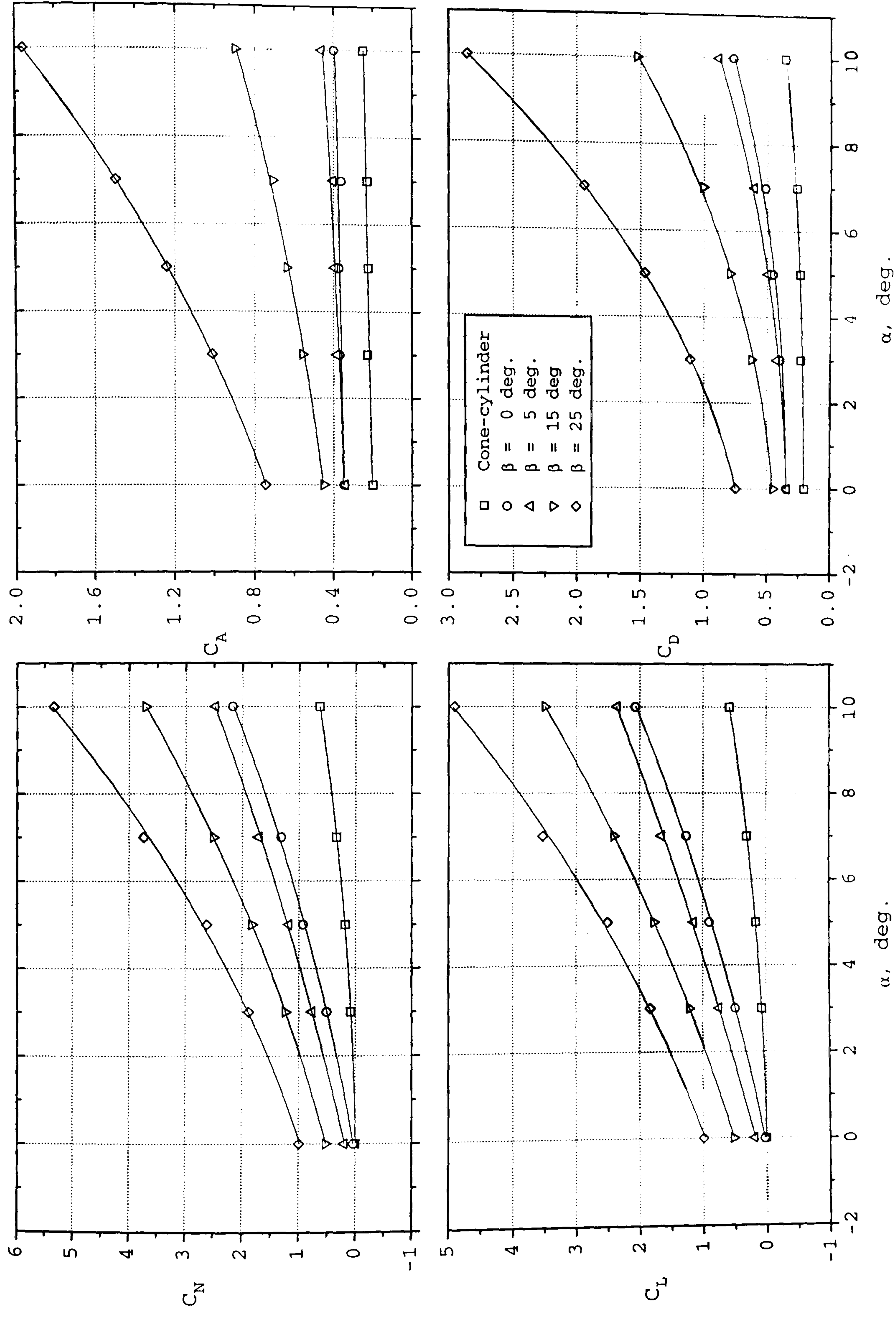


Figure 59. Experimental aerodynamic characteristics of the wing-body configuration.

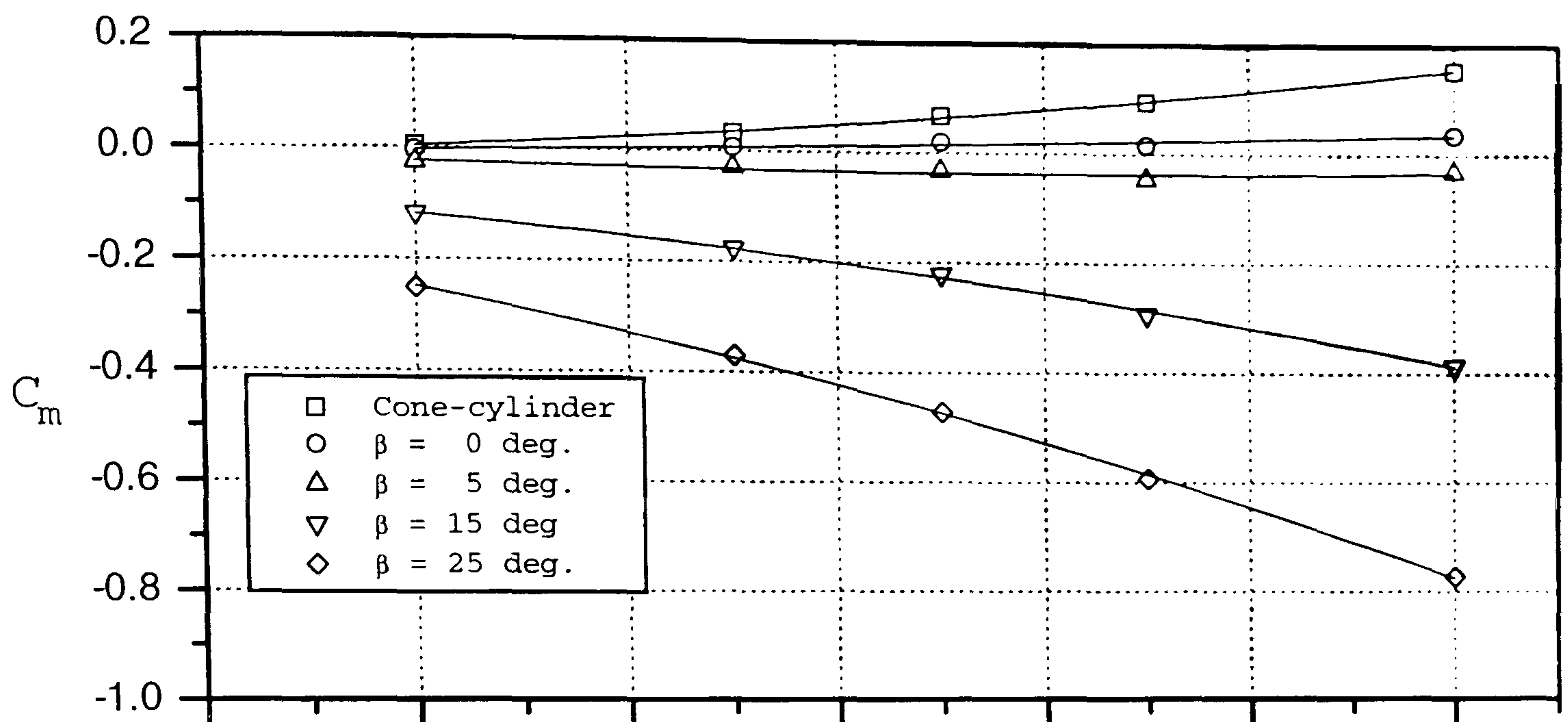
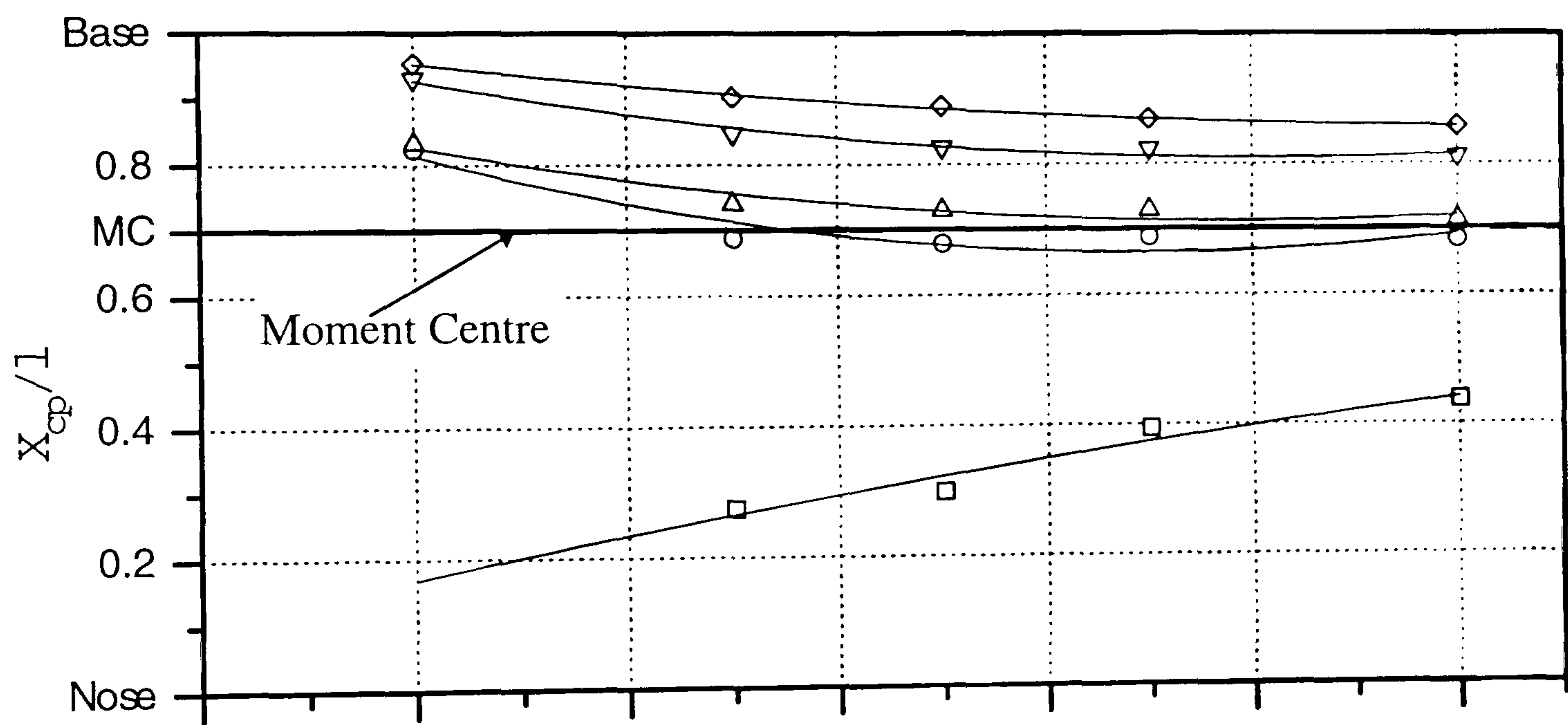
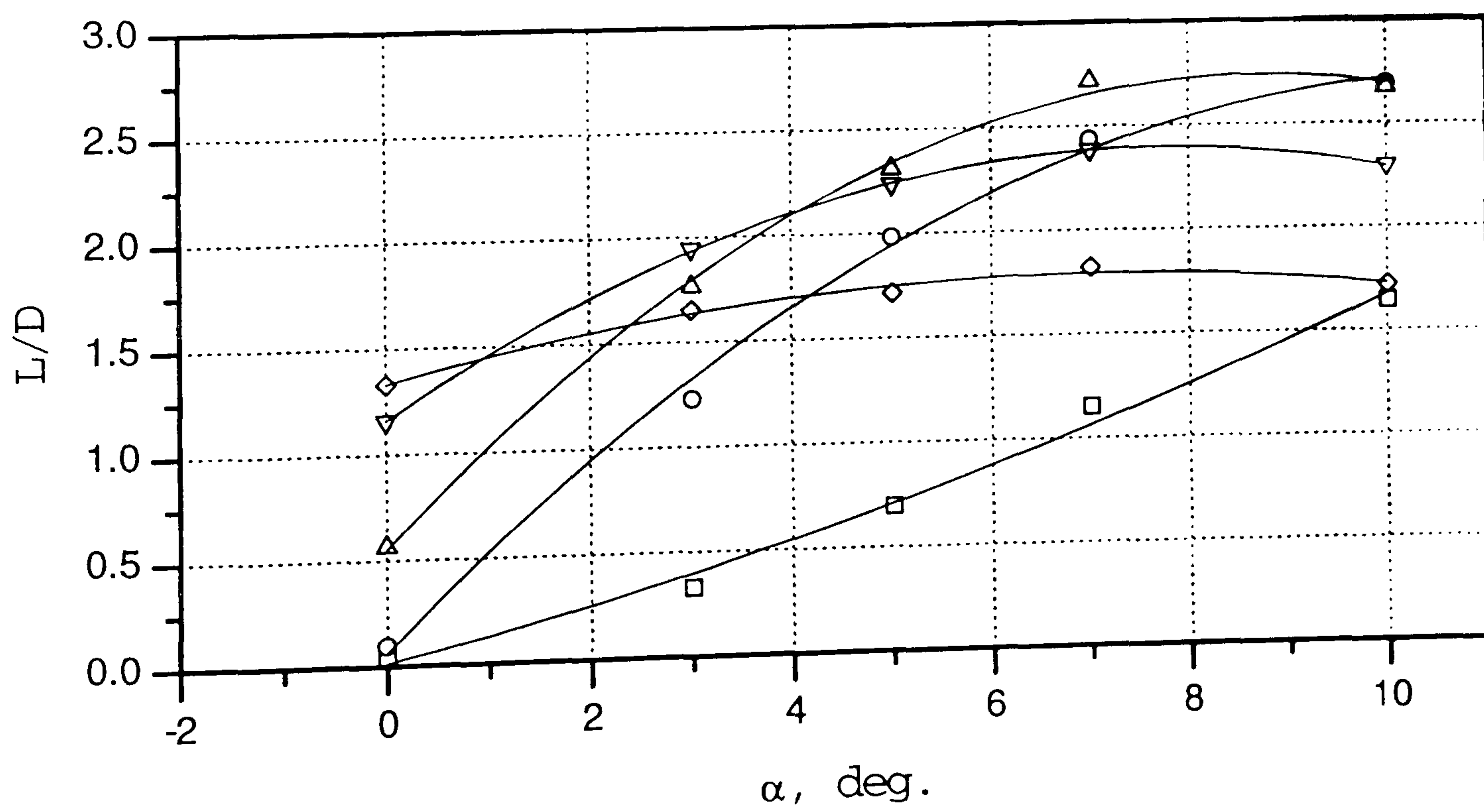


Figure 59. Continued.