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**Three-dimensional flow characteristics downstream of a blade
endwall corner**

Altai, AbdulKarim A. J. Abdulla, Ph.D.

City University of New York, 1993

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Ann Arbor, MI 48106**

**THREE DIMENSIONAL FLOW CHARACTERISTICS DOWNSTREAM
OF A BLADE ENDWALL CORNER**

by

AbdulKarim A.J. Abdulla Altaii

A dissertation submitted to the Graduate Faculty in Engineering

in partial fulfillment of the requirements for the degree of

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The City University of New York

1993

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ABSTRACT

THREE DIMENSIONAL FLOW CHARACTERISTICS
DOWNSTREAM OF A BLADE ENDWALL CORNER

by

AbdulKarim A.J. Abdulla Altaii

Advisor: Professor Rishi Raj

The flow downstream of the corner formed by an airfoil and a flat plate was investigated experimentally. The region for this study was simulated by attaching two uncambered, unswept NACA 65-015 airfoils on either side of a flat plate. The airfoils had a semicircular leading edge. The initial 7 cm of the leading edge of flat plate was artificially roughened by gluing a strip of sand paper to promote a thicker turbulent boundary layer. Free stream turbulence of the order of 1.5% was introduced, by inserting a combination of two grids, to avoid a separation bubble on the airfoil.

Measurement techniques employed in this investigation were; surface flow visualization, conventional pitot-static probe, the Preston tube, and the five hole probe. Surface flow visualization defined the boundaries of the vortex system and provided information on the direction and the magnitude of the wall shear stress. Local wall shear stress was measured using specially constructed Preston tubes. The characteristics of the Preston tube were also investigated. Total and static pressure distribution, including the flat plate surface, were made using a pitot-static

probe. The five hole probe was used, in the semi-nulling mode, to collect data on the three components of mean velocity. The measurements were carried out at eight axial stations downstream the trailing edge of the airfoil.

At every axial location downstream the trailing edge of the airfoil forming the corner with a flat plate, a single dominant horseshoe vortex was identified which persisted more than one chord length downstream. A smaller and weaker corner vortex was also identified. It dissipated and ceased to exist by a downstream axial location of approximately $0.2C$. There was no evidence of stress induced vortices in the region of this investigation. The secondary flow system redistributes the mean flow momentum and distorts total pressure profiles and contours. In planes parallel to the flat plate, total pressure profiles were found to be higher than the undisturbed two-dimensional boundary layer at that height. Surface static pressure was found to be at its maximum at the airfoil trailing edge location and it decreased in both the downstream and transverse directions. There was no significant static pressure variation in the spanwise direction.

To my parents

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NOMENCLATURE

C	Airfoil (blade) chord length
C_p	Static pressure coefficient
C_t	Differential pressure transducer coefficient
C_{pp}	Five hole probe pitch angle pressure coefficient
C_{PT}	Five hole probe total pressure coefficient
C_{PV}	Five hole probe velocity pressure coefficient
C_{TP}	Total pressure coefficient
D	Airfoil (blade) leading edge diameter
d	Preston tube outer diameter
H	Body height (Figure (2.3))
P_1	Pressure measured at the central hole of the five hole probe (Figure (3.12))
P_2, P_3	Pressures measured in the yaw plane of the five hole probe (Figure (3.12))
P_4, P_5	Pressures measured in the pitch plane of the five hole probe (Figure (3.12))
P_a	Reference static pressure (equal to ambient pressure)
P_p	Preston tube pressure
P_t	Total (stagnation) pressure
P_s	Static pressure
P_∞	Free stream total (stagnation) pressure
R_e	Reynolds number
s	Spacing between the bodies that generate the wake and the boundary layer (Figure (2.1))
V	The magnitude of the velocity vector at the five hole probe sensing elements
V_s	Resultant secondary flow velocity
V_x, V_y, V_z	The three orthogonal velocity components in the X-, Y- and Z- directions, respectively (Figure (3.12))
V_∞	Undisturbed free stream velocity
X, Y, Z	Distances in X-, Y- and Z- directions, respectively
X^*, Y^*	Non-dimensional parameters in the Preston tube calibration equations

Greek symbols

δ	Boundary layer thickness
θ	Angle in the pitch plane
φ	Angle in the yaw plane
Δ	Indicates a pressure difference
ν	Fluid kinematic viscosity
ρ	Fluid mass density
τ_w	Local wall shear stress
$\tau_{w\infty}$	Wall shear stress at one chord length upstream of the airfoil leading edge
τ_{wmin}	Local minimum wall shear stress (Figure (4.18))
τ_{wmax}	Local maximum wall shear stress (Figure (4.18))

CHAPTER I

INTRODUCTION

1.1 Problem Statement:

The flow in the region downstream of airfoils and struts mounted over other bodies and placed in an approaching stream with a non-uniform velocity is a complex three-dimensional, rotational, slender shear flow which occurs in many fluid dynamic engineering applications. This type of flow is generated when the boundary layer developed on one body encounters the wake of another body attached to the surface of the first body. The flow in such regions is regarded as an interacting flow, since at large distances from the corner each shear layer is unaffected by the other. Flow region behind the junction of wing, fin and tail planes with the aircraft fuselage, flow downstream of the inlet and the outlet supporting struts, guide vanes, rotor and stator of a turbomachine near the hub and tip regions, roots of fins in heat exchangers, boundary layer probe and measuring surface, submarine fins and fuselage, bridge pier in river bed, air flow downwind of tall buildings and trees, are few examples of the type of flow under investigation. Actually, this problem is common to any geometry where the leading edges of the joined bodies are offset. A prevailing feature observed experimentally for such configurations is the occurrence of three-dimensional separation of the boundary layer, upstream of the wing leading edge, with a curved separation lines present on the body upstream of the wing.

When a two-dimensional boundary layer (vortex sheet) developed on a body approaches a bluff obstacle (wing) protruding from the first body (Figure 1.1), it will experience an adverse pressure gradient, causing the boundary layer to be skewed (deflected), giving rise to streamwise vorticity just upstream the leading edge of the wing. The boundary layer will separate in the vicinity of the leading edge and the separation point is called the "saddle point" (nodal or standoff point of separation). This separated (now three-dimensional) boundary layer will roll up forming a horseshoe shaped vortex that wraps around the leading edge. The flow dominated by the horseshoe vortex is known as "secondary flow of the first kind" or "skew induced secondary flow". Within short distance downstream of the wing leading edge, a corner vortex is formed in the region between the horseshoe vortex and the corner formed by the two bodies [1]. The rotational direction of the above mentioned corner vortex is opposite to that of the horseshoe vortex. The presence of a corner vortex was also reported by Dickenson [2] and Gorski et al. [3]. Hazarika and Raj [1] also observed the formation of a pair of counter-rotating vortices between the corner vortex and the corner formed by the two bodies downstream of the maximum thickness section of the wing. These counter-rotating vortices were noted to be produced by the action of anisotropic and inhomogeneous turbulence, i.e., Reynolds stress gradients in the plane normal to the primary (streamwise) flow direction [1]. They proposed that one of these vortices (next to the wing surface) will merge with the corner vortex (Figure 1.1). The size of the stress induced vortex is approximately half the size of the corner vortex [1].

Photographs of oil film surface flow visualization presented in [1] showed that the separation region generated ahead of the leading edge persists and influences the regions downstream of the trailing edge. A similar conclusion is arrived at from the

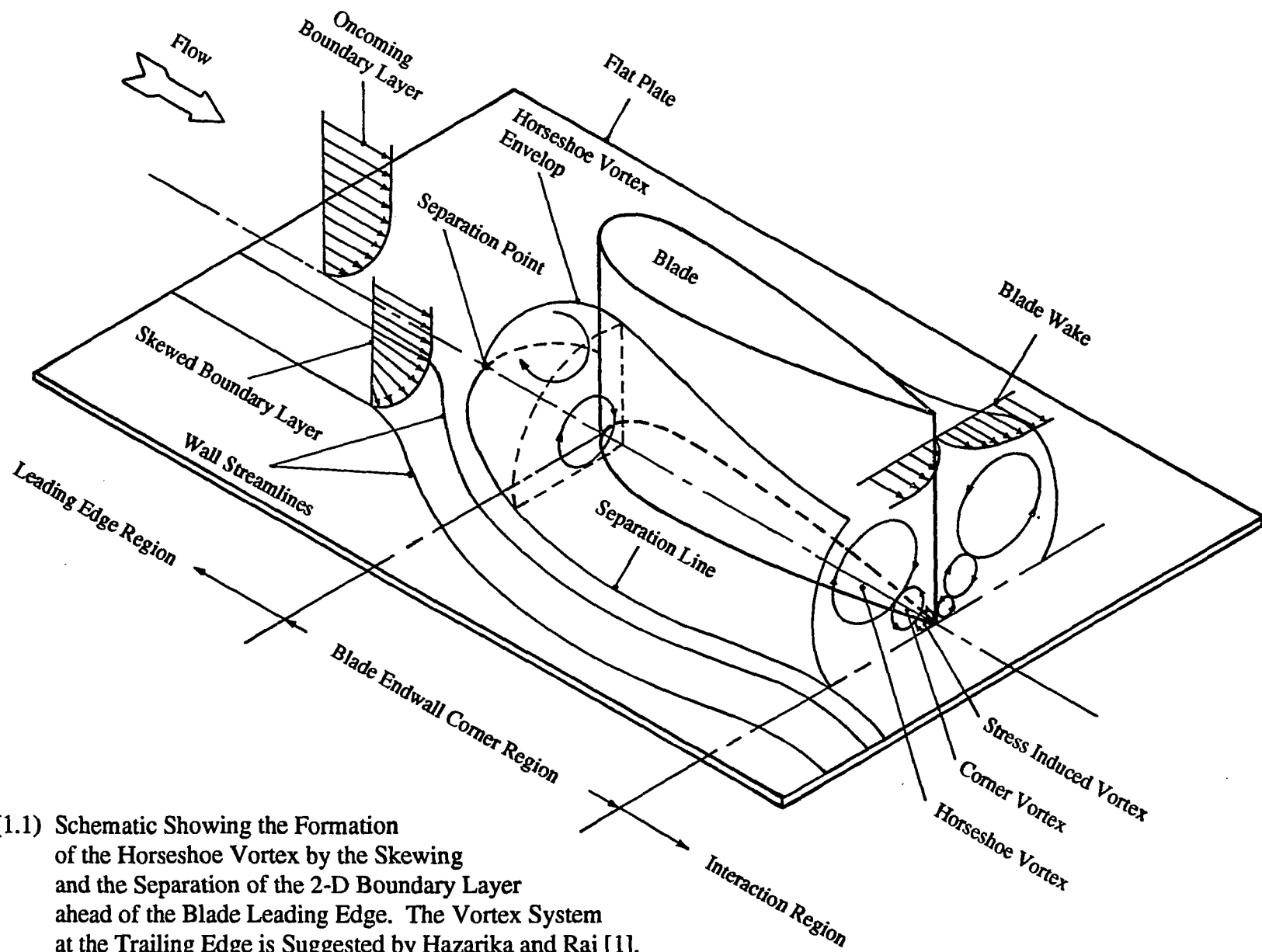


Figure (1.1) Schematic Showing the Formation of the Horseshoe Vortex by the Skewing and the Separation of the 2-D Boundary Layer ahead of the Blade Leading Edge. The Vortex System at the Trailing Edge is Suggested by Hazarika and Raj [1].

surface oil flow visualization made by Sepri [4] and Chu and Young [5]. These authors used NACA 0012 wing section connected with flat-plate and indicated that the horseshoe vortex continues downstream of the trailing edge of the wing. Also, the measurements made by Binder and Romey [6], Love [7], Armilli [8], Naranjit [9] and Barber [10], downstream of various corner configurations, implied that some secondary flow exists downstream of the trailing edge. It follows that the complex vortex structure generated ahead and after the leading edge will shed down and influence not only the endwall corner region but also the wake generated at the trailing edge of the wing.

In aircraft aerodynamics, one source of drag which is often appreciable, but not well understood, is fuselage-wing aerodynamic interference. This interaction may result in a drag force that is far different from the sum of drag forces sensed by the fuselage and the wing individually. Since the wake of an isolated airfoil has different characteristics as compared to the boundary layer/wake interaction region, the analytical techniques and results of measurements of isolated airfoil wake can not be used to predict the flow regions discussed above. Moreover, secondary flow at a wing-fuselage intersection may create a disturbance in the boundary layer on the fuselage downstream of the wing and may affect the performance of the propellers when installed in these regions (e.g., submarines and turboprop aircraft).

Secondary flows in the endwall corner regions have been studied experimentally as well as theoretically over the last decade, for different blade geometries. These studies have resulted in a fairly detailed description of the complex flow pattern. An extensive literature review is given by Shabaka [11], Hazarika and Raj [1] and Bhargava and Raj [12]. In the study of "Microclimate", i.e., the wind environment at ground level, some theoretical and experimental work have been conducted to describe the wakes behind two-dimensional surface obstacles (e.g.

Counihan et al., [13] , Olivari [14] and Marumo et al., [15]). These studies cannot be applied to many practical situations as the height of these obstacles is small compared to the thickness of the boundary layer flowing over it, i.e., they are perturbations within the boundary layer.

From the literature survey (chapter II) and from the above observations, it is apparent that very little work has been done to evaluate the nature of this interacting region, and a more detailed study is needed to understand this type of flow which is crucial to improved design, efficiency and performance of turbomachinery. The optimization of the turbine flow requires detailed knowledge of this interacting flow impinging on the downstream rows of airfoils. Compressor inlet guide vanes, which are often used for improvement in the stage efficiency and the operating range of axial flow compressors, generate secondary flow by the turning of the incoming wall boundary layer. The incoming flow to the succeeding row (i.e., rotor) would be significantly altered. The commencement of cavitation, which is a major factor in erosion of blades in turbomachinery utilizing a liquid medium, depends on the presence of the vortices in the interaction region. If knowledge of the flow characteristics in these regions is improved, it will enable a more accurate prediction of the optimum size and location of fillets at the endwall juncture to achieve an overall improvement in the flow field. Also, it may provide insights into the processes that are controlling performances in a "Horseshoe Vortex Combustor" or Circumferentially Stirred Combustor (e.g. Turan et al., [16]). Thus from the above examples, the study of the characteristics of the flow in the downstream interaction region is imperative.

The model geometry discussed in chapter III would seem more closely linked with the flow downstream of the junctions of most practical engineering applications than the one associated with flow downstream of a streamwise corner (Figure 1.2).

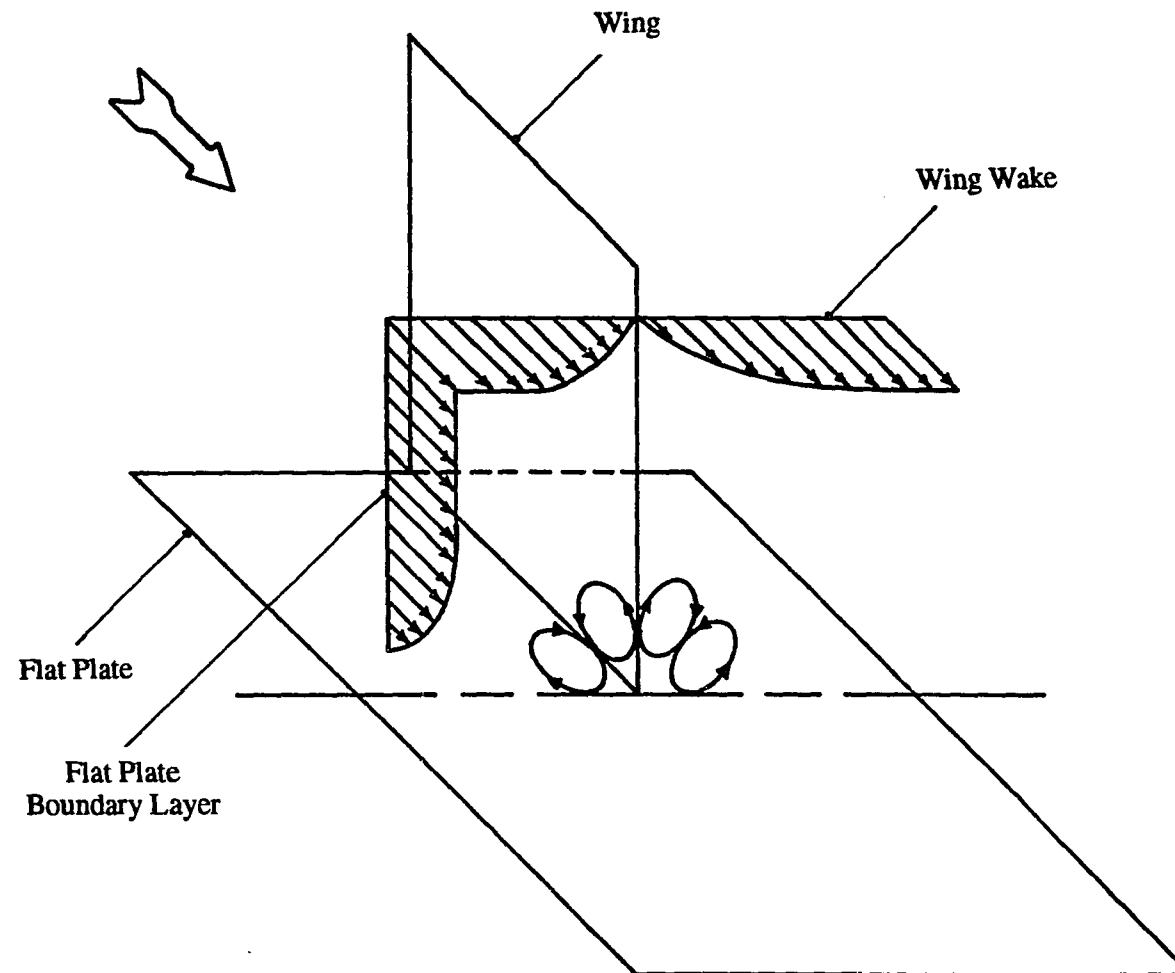


Figure (1.2) Vortex System of a Turbulent Corner Flow.

The flow downstream of a streamwise corner also has received some attention recently (e.g. Rahai [17] and Nakayama and Rahai [18]).

1.2 Objectives of the Present Investigation:

The present program of investigation of the interaction region formed by attaching an airfoil with its span normal to a flat plate has been taken up with the following objectives:

- a) To conduct flow visualization study to establish different regions of interest and to understand the overall flow structure present in the interaction region. Also, to obtain the direction of the local wall shear stress.
- b) To measure the local wall shear stress in the interaction region.
- c) To survey the interaction region for the total and static pressure variations, including wall static pressure, and examine pressure profiles to detect flow distortions due to the presence of vortices.
- d) To measure the three components of mean velocity in the interaction region and to examine the mutual effect between the secondary flow and the wing wake.

1.3 Method and Means of Investigation:

The investigation was carried out in a low turbulence, low speed, open circuit wind tunnel. The model used by Hazarika and Raj [1] with extended length of the flat plate was found to be suitable for studying the interaction region. The model gives large enough incoming boundary layer thickness so that sufficient measurements can

be made to properly analyze the interaction phenomenon in the range of Reynolds numbers encountered in subsonic turbomachines.

Based on the available literature on flow visualization, the smoke flow visualization technique has been ruled out, since at high velocities turbulence is expected to smear the smoke too greatly to study the flow. Surface flow visualization technique was successful in obtaining permanent traces of the flow very near the surface. Two different techniques were used, namely, oil film and dot matrix surface flow visualizations.

Local wall shear stress measurements were conducted using specially constructed Preston tubes along with the calibration equations suggested by Patel [19]. Static pressure measurements, including local surface static pressure were measured using the smallest available static pressure probe of 1.5 mm sensing head diameter. Total pressure measurements were made using total pressure probe of 1.08 mm diameter.

The three components of mean velocity was measured using the smallest available five hole probe with 2.3 mm sensing head diameter. The five hole probe was calibrated in the range of Reynolds numbers expected in this investigation. Since the diameter of the probes is expected to be less than 10% of the incoming boundary layer thickness, the introduction of these probes in the interaction region was expected to have negligible effect on the flow measurements. The grid points for the flow measurements were based on total pressure survey. The outputs of the probes were read on digital voltmeters and recorded for calculations.

CHAPTER II

LITERATURE SURVEY

The existing literature on boundary layer and wake interaction can be discussed depending upon the relative geometrical positioning of the bodies that generate the boundary layer and the wake. Depending on the relative arrangement, the interacting flow can be two or three dimensional and with or without corner flow effect (secondary flow of the first or the second kind). The literature can be divided into three main groups;

- a) The boundary layer of one body is intercepted by the wake of another body and vice versa, (Figure 2.1).
- b) The boundary layers of the two bodies start at the same axial location, i.e., coincident leading edges, with one body shorter than the other and mounted normally to it. The wake of the short body interacts with the boundary layer of the long body, (Figure 2.2).
- c) The boundary layer developed on the surface of one body interacts with the wake of the other body mounted normally to the first one, i.e., offset leading edges, (Figure 2.3).

The fundamental difference between group (a) type of flow and that of groups (b) and (c) is the absence of the 3-D corner flow in group (a). In group (a) there is no corner formed by the two bodies. In group (b), where both of the turbulent boundary

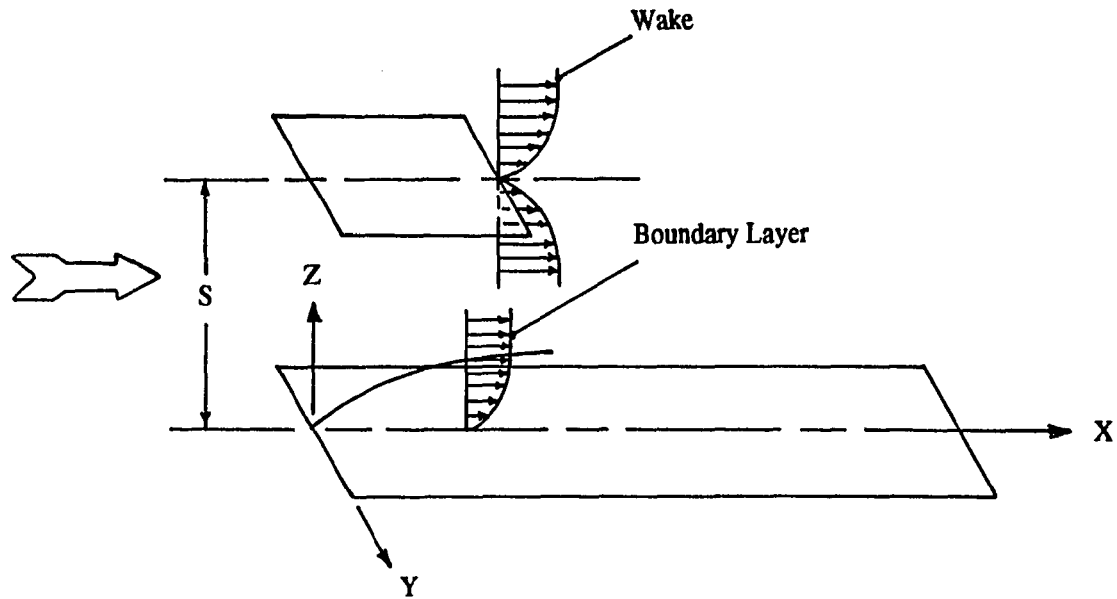


Figure (2.1-a) Boundary Layer of One Body is Interacted with the Wake of Another Body, (2-D Case).

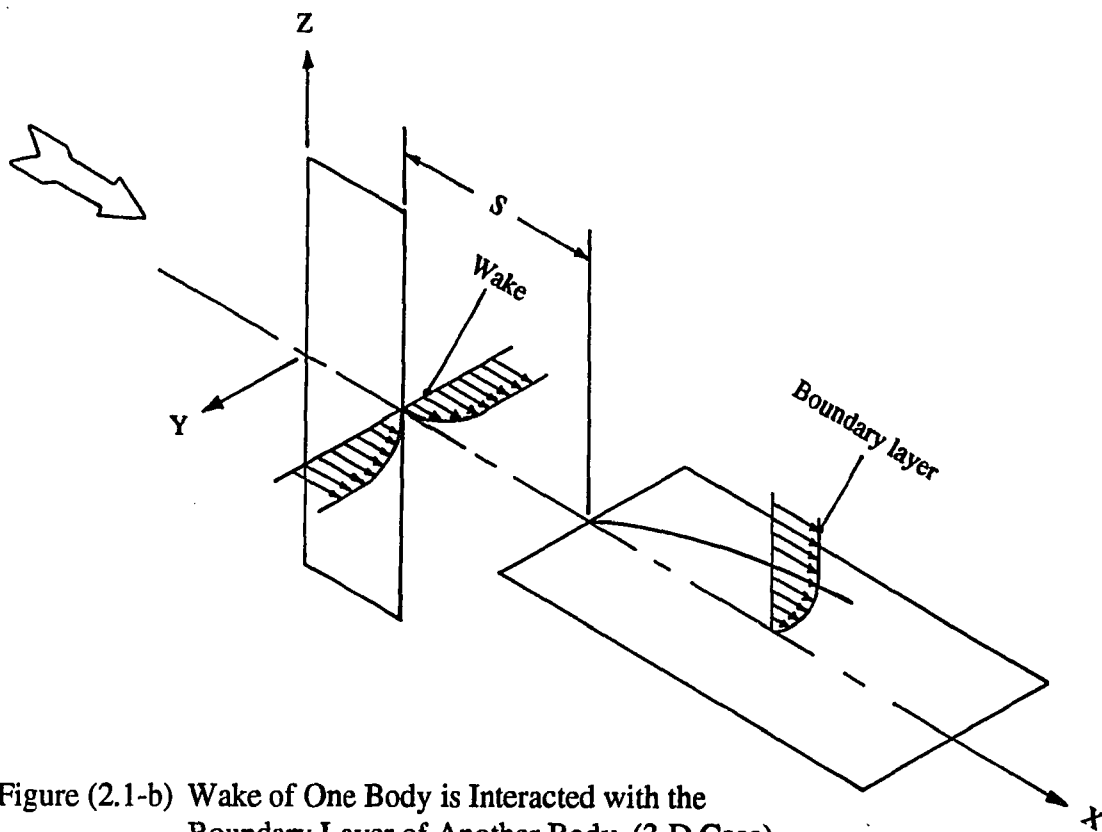


Figure (2.1-b) Wake of One Body is Interacted with the Boundary Layer of Another Body, (3-D Case).

Figure (2.1) Schematics of Boundary Layer and Wake Interaction in the Absence of Corner Flow.

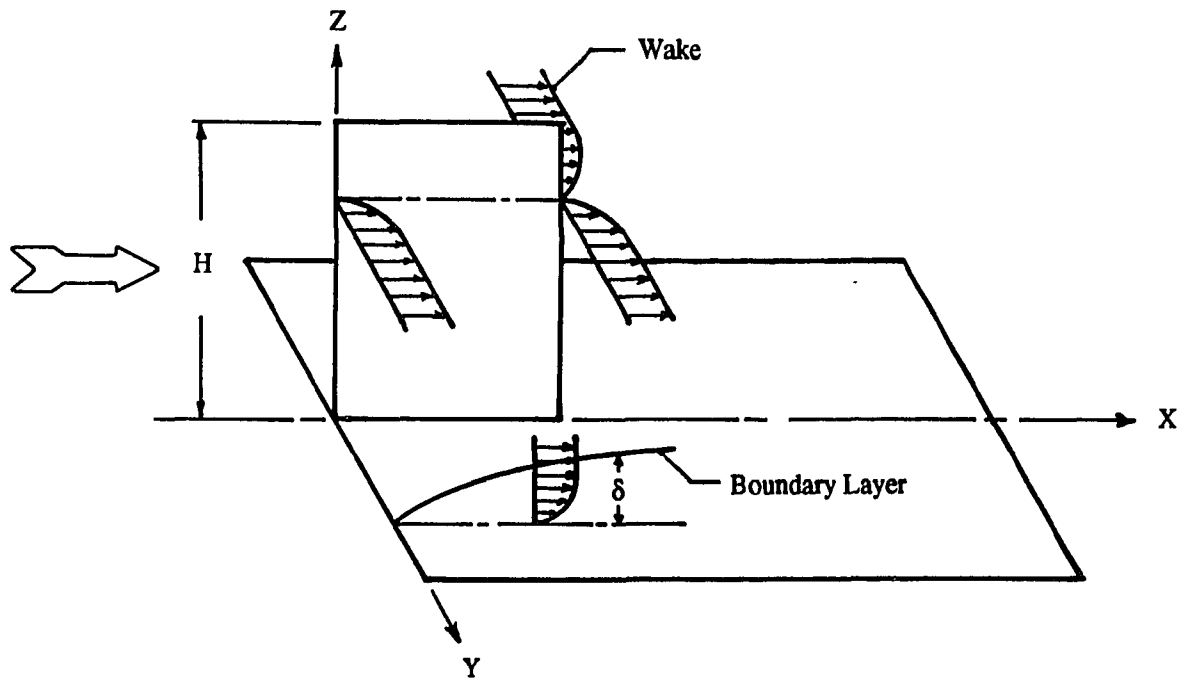


Figure (2.2) Coincident Leading Edges.

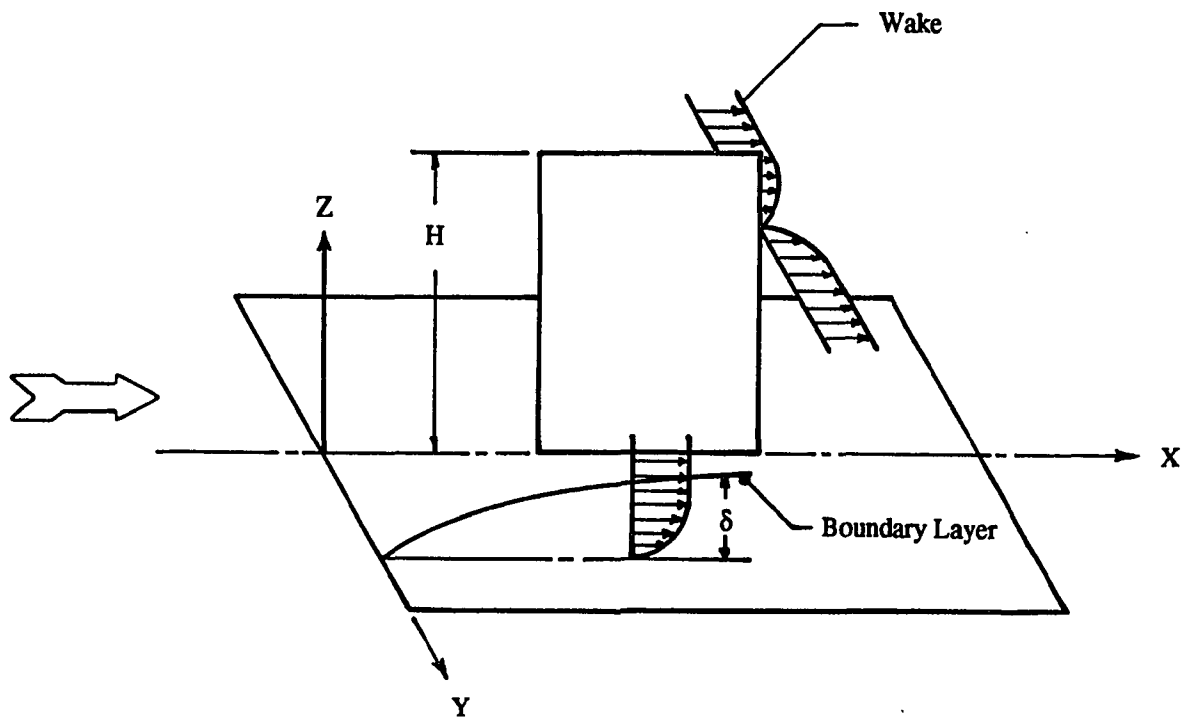


Figure (2.3) Offset Leading Edges.

layers start at the same axial location, stress induced secondary flow will dominate the corner region. A pair of counter rotating vortices across the corner bisector develops and will influence the flow downstream the trailing edge. While in group (c), a horseshoe vortex (skew induced secondary flow) has a dominant effect in the corner region and the region downstream the trailing edge.

The model geometry used in this investigation falls into group (c). Thus, the literature concerning group (c) type of flow will be discussed in details. However, for the sake of completeness, a brief literature review concerning groups (a) and (b) is also given, which is discussed in the following sections.

2.1 The Boundary Layer of One Body is Intercepted by the Wake of Another Body:

Gartshore [20] investigated the mixing of flat plate boundary layer with the wake of circular rod mounted above the flat plate. The rod axis was parallel to the flat plate and normal to the flow direction. A theoretical model for the mixing region was proposed by superimposing the linearized solutions for the wake and the boundary layer and using a suitably defined eddy viscosity to predict the mean velocity in the combined flow. This theoretical model is valid for the mixing of two-dimensional boundary layer and wake with small deficit. In other words, it is suitable for far wake mixing when the rod is farther off the flat plate and the mixing is delayed until the wake and the boundary layer have developed more completely.

Marumo et al. [15] and Tsiolakis et al. [21] investigated the effect of cylinder wake on the behavior of flat plate turbulent boundary layer. The cylindrical bar was placed above the flat plate with its axis parallel to it and at different distances (both inside and outside the boundary layer). The boundary layer exhibited different

behavior according to the position of the bar. Reduction in plate skin friction coefficient due to the presence of the cylinder cannot be delineated from Marumo et al. [15] results. Kreplin et al. [22] carried out mean velocity and Reynolds stress measurements in two merging flat plates, two dimensional wakes and flows where these double wakes interact with a flat plate boundary layer. These measurements are an extension of Nakayama et al. [23] measurements of the merging of a flat plate wake and boundary layer, both at zero pressure gradient, with all cases having the same order length scale ratios. The mean velocities in the initial merging regions were a superposition of the defect velocities of the isolated wakes in the case of the double wakes. However, a superposition could not be verified for double-wake/boundary layer as well as wake-boundary layer merging of Nakayama et al. [23]. For the double-wakes/boundary layer cases, the turbulent energy production is strongly reduced in the regions of small mean velocity gradients. The turbulent kinetic energy is significantly decreased in the interaction region of the boundary layer with the double-wakes. Recently, Kreplin et al. [24] suggested that eddy-viscosity model works well in the interaction region of this type.

Pal [25] studied experimentally as well as analytically the interaction of the wake, developed from the trailing edge of a thin and smooth flat plate, with the boundary layer developed on an artificially roughened flat plate placed downstream at a distance and at plane normal to the plane of the first plate, Figure (2.1.b). The study included the effect of freestream turbulence and spacing between the two plates on the interacting flow characteristics. This phenomena can be better regarded as the development of boundary layer under non-uniform free stream velocity rather than a wake /boundary layer interaction phenomena.

Ziada and Rockwell [26] investigated the interaction mechanism of a mixing layer flow, formed by the merging of high and low speed streams at the trailing edge of

a thin plate, impinging upon the leading edge of a wedge. Interaction mechanism near the leading edge is strongly influenced by the position of the leading edge relative to the incident vortex. The vortex shed from the leading edge has a vorticity direction opposite to that of the incident vortex. Kaykayoglu and Rockwell [27] extended this work by investigating the instantaneous pressure field on the surface of the wedge and its relation to the mechanism of vortex-edge interaction. Maximum pressure amplitude always occurs at the tip of the wedge, irrespective of the relative location of the leading edge to the incident vortex.

Recently, aerodynamicists showed increasing interest in reducing turbulent drag using "Large-eddy breakup devices", "LEBUS", or flow manipulators; for example Dowling [28], Sahlin et al. [29] Browne [30], and Vukoslavcevic et al. [31]. The device consists of short and thin airfoils, arranged in tandem, placed inside a longer-body turbulent boundary layer. The vorticity, shed from the trailing edge of such airfoils, is of opposite sign to the incidence vorticity and it will cancel its effect and reduce the velocity fluctuations in the vicinity of the body. This effect seems to persist far downstream the airfoils. The total drag on both the body and the airfoil is less than that on the body without the airfoil [28, 31]. However, Sahlin et al. [29] reported no drag reduction in their experimental investigation. One can conclude that more investigation is needed to obtain the optimum size and location of such devices to accomplish the desired reduction in drag.

2.2 Boundary Layers Developed on the Surface of the Two Bodies Start at the Same Axial Location with One Body Shorter Than the Other:

Two published studies on this type of flow, carried by Rahai [17] and extended by Nakayama and Rahai [18]. Rahai obtained mean velocity measurements in the

interaction region between a turbulent boundary layer developed on a flat plate and the wake of a second flat plate, with a sharp trailing edge, attached perpendicular to the first one at zero angle of attack and with aligned and rounded leading edges. The general shape of isovel contours showed the three dimensionality of the flow with no appreciable skew induced flow, which is consistent with the 90° corner flow data of Mojola and Young [32]. The wing wake grew much faster than the flat plate boundary layer and the real interaction region is confined to a small region directly downstream of the wing near the body and the rest of the flow appeared to be an extension of the upstream corner flow. However, this interaction region expands faster in the transverse direction than in the spanwise direction. Pressure is maximum at the trailing edge corner and decreases in both downstream and transverse directions. This region with induced pressure gradients extends about two corner flow thickness downstream. At distances of about ten corner flow thickness downstream of the trailing edge, the corner flow characteristics seem to have died out. Nakayama and Rahai [18] extended the work by Rahai [17] by measuring mean streamwise and normal components of velocity and five Reynolds stress components using a cross-wire probe. They confirmed Rahai's conclusion [17] that the pressure distribution on the body was uniform and within a scatter of 0.5% of the free stream dynamics pressure except in the area of 5 cm from the root of the wing trailing edge. Small regions of negative eddy viscosity with positive mean velocity gradient existed close to the wing trailing edge in which eddy viscosity model will fail. Also calculated, skin friction coefficient by Clauser's method of fitting the logarithmic portion of mean velocity profiles, skin friction coefficient values appeared to be always smaller inside the wake-influence region than in the asymptotic boundary layer away from the corner.

2.3 Boundary Layer Developed on the Surface of One Body is Interacted with the Wake of the Other Body Mounted Normally to the First One:

This type of flow is dominated by skew induced secondary flow; secondary flow of the first kind. The literature available on this type of flow can be divided into two groups;

2.3.1 Body Height is Shorter Than the Incident Boundary Layer:

In recent years, the study of Microclimate; the wind environment at ground level, became a matter of concern amongst architects and urbanists especially airflow downwind tall buildings. The buildings can be modeled as perturbation within the atmospheric boundary layer. With this in mind, Counihan et al. [13] investigated analytically and experimentally the characteristics of the wake behind two dimensional surface obstacle (rectangular bar) placed within large turbulent boundary layer. Eddy viscosity model adequately describes the distribution of mean velocity and was in a good agreement with the experimental results. They pointed out that the theory needs further development as it does not adequately describe turbulence quantities and is primitive in nature. Hunt et al. [33] discussed the flow around various blunt bodies (e.g. cuboids and bell shaped humps) attached to a flat plate. This study recommended the application of kinematical principles, as an aid, in constructing flow patterns from the interpretation of surface flow visualization. Olivari [14] studied the effect of wind environment at ground level in the vicinity of buildings. The study included flow visualizations, total and static pressure around building-like prototype. He recommended that wind tunnel tests should be performed on models at the earliest stages of planning the layout of a building or group of structures.

2.3.2 Body Height is Greater Than the Incident Boundary Layer:

The survey in this section will be concentrated on those studies with secondary flow around a single perturbation. Detailed literature reviews on the secondary flows in turbomachines were given by Lakshminarayana and Horlock [34], Dunham [35], Shabaka [11], Langston [36], Sieverding [37], Hazarika and Raj [1], Bhargava and Raj [12] and the recent studies by Chung et al. [38] and Tang et al. [39].

The characteristics of the interaction region are expected to be controlled by the nature of flow phenomena upstream of the interaction region. Therefore, a brief literature survey of the flow upstream the interaction region similar to the one that would have existed for the present investigation is also given here in the following paragraphs.

The investigation of the flow upstream the interaction region may be divided into those focused on the skewing and the separation of the boundary layer ahead of the leading edge and those focused on the secondary flow in the junction.

Regarding the behavior of the three dimensional boundary layer ahead of an object, Hornung and Joubert [40] investigated experimentally the mean velocity distribution in a low-speed three-dimensional turbulent boundary layer. The secondary flow was produced by the pressure field introduced by a semi-circular leading edge wedge placed normal to the flat plate. They concluded that the wall-wake model proposed by Coles [41] for the velocity profile in the three dimensional boundary layer does not apply to flow of this kind and the triangular model proposed by Johnston [42, 43] was confirmed but with limitations, i.e., it is not universal. This triangular model establishes a relationship between the cross-flow and main-flow components of the local velocity vector within the three dimensional boundary layer. Francis and Pierce [44] and Klinksiek and Pierce [45] reported investigations of the skewed three

dimensional, turbulent, incompressible boundary layer in curved and recurved channels.

Stanbrook [46] reported results of experiments on various wing-body junctions which included swept and unswept wings both with rounded and sharp leading edges. The study included total pressure measurements and oil film surface flow visualization at both subsonic and supersonic speeds. The streamlines indicated that a vortex system wrapped around the leading edge, whose formations was attributed to the separation and rolling up of the boundary layer ahead of the leading edge, was present for rounded leading edge wings, swept and unswept, with or without incidence, at both subsonic and supersonic speeds. This vortex system was not observed for wings with sharp leading edges at zero incidence [11, 47, 48]. From tests on delta wing, he concluded that the boundary layer thickness has little effect on the strength and position of the vortex system. While on the other hand, a reduction in Reynolds number will cause the vortex system to be weaker and closer to the wing. The Reynolds number effect was confirmed by Schwind [47] flow visualization studies. Schwind [47] showed the formation of a horse shoe vortex ahead of 60° included-angle wedge using low Reynolds number, laminar flow smoke visualization studies. Different flow geometries (called flow regimes) were identified as the Reynolds number was increased and one or two separation lines may be generated ahead of the leading edge. He confirmed Stanbrook [46] observation that the distance from the wedge leading edge and the separation line increases with Reynolds number.

East and Hoxby [49,50], Prahlad [51], and Hsing and Teng [52] conducted experimental investigation on the flow ahead and around the leading edge of a wing-like object mounted normal to a flat plate. The wing had a circular arc nose followed by a short constant thickness portion, after which thickness was reduced linearly and symmetrically to a sharp trailing edge. With the exception of the work conducted by

Prahlad [51], the measurements in all of these cases were basically carried out ahead and around the leading edge. Whereas Prahlad [51], included measurements in the endwall corner region. East and Hoxby [49, 50] and Prahlad [51] found that agreement with Johnston triangular model was obtained only ahead of the free-stream line of inflection points and farthest from the obstacle. The secondary flow velocity profiles downstream of an inflection point can exhibit complex shapes which make this model invalid. Mean velocity profiles measurements by Hsing and Teng [52] were reported to follow wall-wake law in the attached flow region. Whilst, the constants in Coles velocity law were no longer kept constant, the constants and the mixing length varied with pressure gradient and the curvature of the streamline.

Peake, et al., [53] gave a comprehensive review of the three dimensional flow separation in external aerodynamics. They discussed the extent of the separation region ahead of various wing/body combinations as inferred from flow visualization studies.

Kaufman et al. [54] investigated the separated flow ahead of a constant thickness fin placed normal to a flat plate at supersonic speeds. They confirmed the presence of the horseshoe vortex and found an increased heat transfer rate in the separated region which exceeded ten times the undisturbed surface values. Turbulent boundary-layer separation occurs approximately two diameters ahead of the fin. The separation is insensitive to Mach number, Reynolds number and the incoming boundary layer thickness for a very wide range of Mach numbers, as long as the fin is larger than the undisturbed boundary-layer thickness [54].

Han et al. [55] studied the heat transfer from a heated cylinder with and without endwall effect to assess the effect of the horseshoe vortex on heat transfer rate. The experimental results indicated an increase in the heat transfer from the

cylinder near the endwall (i.e., when the horseshoe vortex was present). Larger horseshoe vortex does not necessarily cause higher circumferential heat transfer in the endwall region. The results also showed peaks of heat transfer coefficients on the stagnation line of the cylinder. They suggested further investigations to understand the flow in the separated region and to explain the circumferential variation of heat transfer in the endwall region.

Blair [56] reported surface flow visualization and surface heat transfer distribution around the leading edge of a rectangular block. He reported a complex heat transfer pattern with areas of up to 100% increase in Stanton number, as compared to its far upstream (i.e., 2-D) value.

In his literature review, Shabaka [11] reported studies done by Sepri [4], Chu and Young [5] and Young [57] in the juncture region formed by NACA 0012 wing section and a flat plate with semi-circular leading edge. Smoke and surface oil flow visualization confirmed Schwind [47] observations that the formation of one or more vortices depend on the flow Reynolds number. The distance between the horseshoe vortex core and the wing surface increases slightly as it proceeds downstream the leading edge. It is also indicated that the vortices complex structure and separation lines of the leading edge not only influence the corner region but also the region downstream the trailing edge. The same corner features were observed at incidence but they were not symmetrical.

Shabaka [11] investigated the flow in idealized wing/body junction in detail. The corner was formed by the wind tunnel test section floor and a non-lifting constant thickness wing with half elliptic leading edge. The investigation included surface oil film flow visualization, static pressure and shear stress measurements on both surfaces and the mean and turbulent velocity distribution in the corner region. He confirmed

the existence of a strong horseshoe vortex (skew induced secondary flow) in the junction and there was no evidence of the secondary flow of the second kind (stress induced secondary flow). However, due to the diffusive action of Reynolds stresses, the strength of the horseshoe vortex reduced rapidly in the downstream direction. Regions of negative shear stress existed in the corner region. He concluded that eddy viscosity and the mixing length hypothesis are not suitable for such flows where asymmetrical boundary layers interact.

Railly and Sharma [58] obtained a second momentum integral equation, in the annulus wall boundary layer, from the momentum integral equation presented by Mellor and Wood [59] which contains terms of "apparent stresses". The measurements in the single stage slotted-blade (to obtain high blade loadings) axial compressor showed that the theory cannot account for tip clearance. Murray [60] constructed a general curvilinear coordinate transformation involving complex analytic functions, suitable for numerical solutions applied for wing-body combination. Numerical solutions of Navier-Stokes equations were accomplished by Shang [61] for flow fields around a wing fuselage configuration at 6.2 Mach number. The numerical simulations were successful for the flow without incidence and at ten degree angle of attack. Gorski et al. [3] extended the use of the "space marching" technique to develop a solution of the Navier-Stokes equations in the endwall corner region. Secondary flow velocities were computed using the K- ϵ and the algebraic eddy viscosity models with Shabaka's [11] data used as the initial conditions. The K- ϵ model did not work well in regions with negative eddy viscosity reported by Shabaka. They deduced the formation of a corner eddy and characterized it as a secondary flow of the second kind. This corner eddy was not reported by Shabaka [11]. However, Hazarika and Raj [1] did report the existence of such eddy.

Oguz [48], Kubendran [62] and Kubendran et al. [63] investigated the flow field in the endwall corner, formed by a flat plate and a constant thickness wing with a semi-elliptic leading edge. Oguz [48] investigation was for two values of the flat plate boundary layer thickness. A 30 percent reduction in the incoming boundary layer thickness causes the effective core of the horseshoe vortex to move 20 percent closer to the wing surface. Furthermore, the corner effect was not felt (i.e., two-dimensional flow) beyond lateral distances of four undisturbed local boundary layer thickness. Kubendran [62] and Kubendran et al. [63] concluded that the vortex strength and the streamwise location of its core are affected by the slenderness ratio of the wing leading edge, which in turn, is a major factor determining the flow field around the juncture and the flow field downstream the leading edge. Regions of negative shear stress were also reported. Kubendran et al. [63] supported Dickinson [2] observation that the horseshoe vortex at its origin is confined to a small region above the flat plate surface.

Thompson [64] reported analytical and experimental study of wing-body aerodynamic interference. Potential flow numerical solution was found to be useful in designing wing-body configurations with pressure gradients reduction, thus reducing the tendency of boundary layer separation. Manor [65] performed experimental studies to determine the effects of large angles of pitch and yaw on the performance and stability of double-delta wing, cylindrical body and wing/body configurations at subsonic speeds. Tests included six component force measurements, wake total pressure surveys and fluorescent oil flow visualization to aid in the analysis of wing/body vortex interactions.

Mehta [66] investigated experimentally the effect of the wing leading edge shape on the flow in a wing/body junction. For a given entry shear, the size and strength of the horseshoe vortex increases with nose bluntness. The circulation in the

horseshoe vortex can be reduced by about 40% if the leading edge shape is changed from super-ellipse (blunt) to a wedge-ellipse.

Numerous studies have been recently reported on the horseshoe vortex system formation upstream the leading edge of a bluff body mounted on flat plate [67-84]. Depending on the size and shape of the body, the boundary layer thickness, whether the boundary layer is laminar or turbulent, and the Reynolds number of the oncoming boundary layer, the number of vortices reported in these studied varied from one to six. For a laminar oncoming boundary layer, Baker [67] observed that the number of vortices formed upstream of a cylinder increased (2, 4, or 6) with the increase of Reynolds numbers. For a turbulent boundary layer, $4000 < Re_D < 9000$, Baker [68] reported the existence of only four vortices. Moore and Forlini [69] oil surface flow visualization on the junction formed by Rankine half-body (rounded leading edge) and a flat plate suggested a flow pattern similar to the one given by Baker [68]. Eckerle and Langston [70] carried out surface flow visualization, in the endwall corner region around a cylinder showed the existence of a single endwall saddle point and a single endwall attachment point. Their five hole probe measurements confirmed the formation of a single main vortex off the plane of symmetry.

Sheiman and Kubendran [71] investigated juncture flow with and without streamwise pressure gradients. The wing had a sharp leading edge and two surfaces; a curved-surface and a flat-surface. This wing was attached to a flat plate. Their LDV measurements reported no evidence of a horseshoe vortex on either side and attributed the secondary flow, in the curved-surface side of the juncture, to the lateral skewing of the boundary layer. Juncture fillet effect was also studied on the flat-surface side of the wing. They reported that the use of fillet can improve the flow near the trailing edge of the wing by precluding regions of low momentum fluid. Hasan et al., [72]

reported detailed results on wall pressure fluctuations upstream and along a wing (semi-elliptical nose, parallel midbody and an NACA 0020 tail profile) - body junction.

Hazarika and Raj [1] carried out experimental study in the endwall corner-region with a blade similar to the one used in this investigation. They reported the formation of a horseshoe vortex with a single separation point and with a size of the order of the incident boundary layer thickness. Within a short distance downstream the leading edge, a corner vortex was detected between the horseshoe vortex and the two surfaces forming the corner with vorticity direction opposite to that of the horseshoe vortex. A pair of contra-rotating vortices is formed downstream the blade maximum thickness, between the corner vortex and the corner formed by the two surfaces, due to the action of inhomogeneous and anisotropic turbulence. They suggested that the vortex next to the airfoil surface, which has the same vorticity direction as the corner vortex, merges with the corner vortex. This vortex system is depicted schematically in Figure (1.1). Regions of negative shear stress were reported in the corner.

Lai and Makomaski [73] carried out numerical analysis of the flow upstream of a rectangular body placed in a two-dimensional turbulent boundary layer. In contrast to the four-vortex model suggested by Hunt et al. [33] and Baker [68], only one primary vortex was observed in the plane of symmetry. At lower Reynolds numbers, in addition to the primary vortex, they reported the existence of a counter-rotating corner vortex.

Menna and Pierce [74], Pierce and Harsh [75] and Pierce and Tree [76] documented detailed measurements of the flow upstream, in the separated region and around the corner formed by a streamlined cylinder and a flat plate. Pierce [77] summarized the above studies and offered these data as "a standard test case" in ascertaining the effectiveness of turbulent computational models. While, Pierce and

Harsh [75] five-hole probe measurements showed only one dominant junction vortex. They interpreted the computed vorticity field at the trailing edge location as indicative of the existence of a pair of smaller counter rotating vortices. On the other hand, Pierce and Tree [76] LDV measurement on the plane of symmetry, also reported a single dominant vortex with one "clear" separation line, but their computed vorticity field postulated the existence of only one very small counter-rotating corner vortex.

Devenport and Simpson [78] investigated the flow in the plane of symmetry upstream of a wing-body junction. The 1.5:1 elliptical nose wing had a NACA 0020 tail joined at the maximum thickness. Their three-component LDV measurements exemplified the presence of bimodal probability density functions of velocity fluctuations (in U and V). They proposed a mechanism that describes the formation of the intermittent reversed flow "jet" observed by Falco [79]. The instantaneous frame of the smoke flow visualization, presented by Falco [79] of a turbulent boundary layer approaching an upright cylinder, showed the formation of an "intermittent reversed jet" in the separated region. Recently, Devenport and Simpson [80] extended this study and presented a detailed three-component laser doppler velocimeter of mean velocity and turbulence quantities in six planes surrounding the wing-body junction. These planes included: the plane of symmetry, four in the junction, and a plane at $0.2T$ (T =maximum blade thickness) downstream of the trailing edge. The prediction of turbulence shear stress vectors from the measured mean velocity vectors or turbulence kinetic energy was tested using six turbulence closure models. They concluded that the Cebeci-Smith and the algebraic stress models were among the best and that the $K-\epsilon$ model is not suitable for this kind of flow.

Eckerle and Awad [81] investigated the effect of free stream momentum and boundary layer thickness on the secondary flow formation around an upright wall mounted cylinder. For a high-velocity incoming boundary layer, a vortex was not

formed in the plane of symmetry; rather a back (reverse) flow which was carried out around the cylinder. For a low-velocity incoming boundary layer the reverse flow had little momentum, causing it to roll up into a vortex in the plane of symmetry.

Recently, Hung et al., [82] carried out a numerical study of a steady laminar flow upstream of an upright cylinder mounted on a flat plate for a range of Reynolds numbers. Their study warned of the construction of external flow (structure) topology based on the interpretation of the experimental surface oil-flow visualization only. i.e., For the same oil-streakline patterns, depending on flow parameters, two different flow structures can be conceived; one of flow separation and another of flow attachment. This numerical study was extended by Chen and Hung [83] for laminar/turbulent and subsonic/supersonic flow around a cylinder. This extended study supported Schwind [47] and Baker [67] observations that the number of vortices increases with the increase of Reynolds number. However, the number of vortices generated ahead of the cylinder increased from one up to only three with the increase of Reynolds number.

Agui and Andreopoulos recent study [84] concentrated on the instantaneous flow structure upstream of an upright cylinder mounted to a wind tunnel floor. Their laser-sheet smoke flow visualization showed the presence of unsteady horseshoe vortex that "wanders" in space. This wandering prompts an eruption of wall fluid in a manner similar to the intermittent "jet" observed by Falco [79]. This intermittent eruption causes the formation of a counter-rotating or "mushroom" vortices upstream of the horseshoe vortex. In addition to these vortices, they reported the sporadic existence of another vortex upstream of the mushroom vortices.

Previous studies that are more pertinent to the present investigation are discussed in the following paragraphs.

Amongst the first to study the effect of the leading edge shape on the flow downstream was Hawthorne [85]. He examined theoretically the inviscid incompressible flow around struts with the approaching velocity varying in the spanwise direction. The slender body theory was invalidated when applied to the flow around a cylinder, a non-slender body. Hawthorne showed in his analysis that the shape of the strut profile at the nose influences the size of the disturbances appreciably. The disturbances caused by the presence of the strut are much less with a wedge-shaped than the rounded nose and are least with a bicusped nose. He confirmed the analytical solution by experimental results of the erosion which occurred in the sandy bottom of a water channel in the presence of perpendicular struts of elliptical and bicusped cross-sections. The bicusped profiles drag causing only one-tenth the erosion of elliptical profile, although they have the same finess ratio (same maximum thickness and chord length). Thus, this experiment confirmed the prediction that profiles can be designed to minimize secondary flows although their corresponding drags are small and difficult to measure accurately. He further notes that "The most important effects of the secondary flow at a wing-fuselage intersection may, therefore, not necessarily be to increase the drag on the wing but to cause a harmful disturbance in the boundary layer on the fuselage downstream of the wing, to sweep and collect the boundary layer on the wing into regions of locally stalled flow, and to change the lift on the wing,".

Armilli [8] investigated experimentally the interaction of a turbulent flat plate boundary with the wake of circular cylinder mounted normal to the flat plate. From mean-velocity measurements he indicated that the flow beyond 100 diameters downstream is similar to a two-dimensional wake, while between 20-100 diameters, it is not similar due to the presence of velocities higher than the uniform velocity that exists far from the plane of symmetry. He postulated that direct comparison of results,

from a similar geometry by Love [7], cannot be made as Love's measurements were made very close behind the cylinder and the pressure gradient in the undisturbed flow was non zero. Both Arimilli [8] and Love [7] results suggest that the properties of the wake imbedded in a boundary layer are different from the plane two-dimensional wake.

Chu and Young [5] used 45° chamfered-mouth tube, covered with an outer tube to eliminate the effect of mean velocity gradients, to measure total pressure profiles at the trailing edge region and the wing wake of the same model used by Sepri [4], [11]. At zero angle of incidence lateral pressure profiles showed two regions of low velocity, one on each side of the plane of symmetry, which is the extension of the horseshoe vortex, in addition to a central low velocity which corresponds to wing wake center. At 9.5° angle of incidence, the horseshoe vortex leg from the suction side was strengthened and remained close to the continuation of the wake, while the one from the pressure side was weakened and moved away from the wake. The centerline of the wing wake remained unchanged throughout the plate boundary layer [11].

Naranjit [9] carried out total pressure measurements in the wing wake and body layer, both immediately behind the wing trailing edge and just behind the fuselage tail of a "simple" wing-body combination [11]. At 9° angle of incidence, just behind the wing trailing edge, Naranjit predicted that wing wake merged without any distortion with the fuselage boundary layer. Behind the body trailing edge the fuselage boundary layer and the wing wake did not merge smoothly due to the downwash distribution which kept the main part of the wing wake above the fuselage boundary layer [11].

Barber [10] examined the intersection losses of a number of 65 series symmetric struts with strut thickness-to-chord ratios of 20% and 30% over an incidence range of 0-10 degree. Total pressure traverses across the strut wake were made at a plane located 0.4 of a chord downstream the strut trailing edge with inlet Mach number of 0.25. He identified the horseshoe vortex produced near the leading edge and the secondary vorticity induced by the curvature of the incident shearing flow about the strut, from surface oil flow visualization and total pressure profiles. The total pressure profiles were reduced to the undisturbed two dimensional wake profiles by 0.14 of a chord length above the flat plate. Total pressure profiles indicated that the effect of incidence is simply a distortion of symmetry and little loss dependence on the incidence angle and the primary effect of thickness appears to be an increase in the vorticity-induced distortion. To assess the effect of the incident boundary layer thickness on the endwall intersection losses, the test was conducted for two flat plate lengths which were labeled "long" and "short" plate. A thick boundary layer resulted in small intersection losses, weaker loss dependence on flow incidence and larger horseshoe vortex than a thin incident boundary layer.

Total pressure survey carried out by Binder and Romey [6] showed that the mixing of the wake behind a turbine stator (without rotor) results in an increase in the mixing losses, and that the secondary flow persists far downstream up to distances of two times the chord length of the stator blade. The survey did not include the region close to the trailing edge (less than 0.5 chord length) in which the horseshoe vortex and the secondary flows of the first and second kind may have important influence. Moreover, their model has an additional variable, that is the effect of the passage vortex (formed at the leading edge) and the secondary flow inside the wall boundary layer due to the pressure gradient between the suction and the pressure surfaces in a cascade of airfoils, which may not exist in modeling other flow geometries.

Rood and Keller [86] carried out two-point space time velocity measurements, one on each side of the plane of symmetry, in the region downstream of uncambered symmetrical wing at zero angle of incidence extending vertically from a flat plate and through its boundary layer, to determine a temporal variation with space correlation. The results suggest the existence of turbulent large scale, time dependent organized flow structures, which are associated with the horseshoe vortex. They maintain that the vertical extent of these coherent structures is about 20 percent of the boundary layer thickness. This work was extended by Rood [87] for symmetrical wings, having different ratios of nose radius to boundary layer thickness at incidence, by measuring two-point spectral velocity as well as the mean velocity. The plane of measurements was 2.5 chord lengths downstream from the wing trailing edge. His results indicated that the intensified leg of the horseshoe vortex (by stretching), on the low pressure side of the wing, dominates the downstream flow. The wake deficit in the lateral profiles of the mean velocity is not always coincident with the division between the flows from the two sides of the wing, and the data exhibited a trough (deficit) line skewed toward the low pressure side as the wall is approached. A possible parameter that governs the amount of skew is the ratio of the wing nose radius to the plate boundary layer thickness. Rood [87] maintains that Kutta-Joukowski condition (used in flow calculations to fix circulation about a lifting body) may not be satisfied inside the interaction region (three dimensional) even though it is satisfied outside the boundary layer (two dimensional).

To assess the effect of wing tail shape on the flow downstream of the trailing edge of a wing-plate junction, Rood and Anthony [88] reported results on mean velocities and two-point velocity correlations. Measurements were made at one chord length downstream of the trailing edge. Two tail profiles, a circular arc and an NACA 0020, were attached to 1.5:1.0 elliptic nose wing. The wing was at 5° degree angle of

incidence. The circular arc profile, showed more interacting and more effective transport of boundary layer momentum than the NACA 0020 profile.

Kubendran and Harvey [89] made total pressure measurements in the wake regions of a simulated wing-fuselage juncture, in a plane one chord length downstream the trailing edge of NACA 0012 cross-section, with and without leading edge fillets, in order to assess the effect of the leading-edge modifications on the flow field around the juncture. They concluded that at moderate angles of attack, there is some reduction in the drag and evidence of flow improvement in the juncture wake region with the use of leading-edge fillets. Flow characteristics started to deteriorate at some point as the fillet size was increased and an optimum fillet size may be required to achieve an overall improvement in the flow field. Recently, Devenport et al. [90] studied the effect of fillet on the flow in a wing-body junction. The fillet was of constant thickness wrapped around the entire junction. Two oncoming turbulent boundary layer thickness and three angles of attack were investigated. Aside from drag reduction, which could not be deduced from their results, they concluded that the use of fillet will not enhance the flow nor eliminate upstream separation. Rather, it increased the effective radius of the horseshoe vortex.

McMahon et al. [91] studied the juncture flow around a modified NACA 0020 airfoil (1.5:1 elliptical leading edge). Their hot wire measurements covered five planes; one upstream the leading edge, one downstream the airfoil maximum thickness, one at the trailing edge location, one at one chord length downstream the trailing edge and one at two chord lengths. They suggested, without conclusive results (evidence), the presence of a counter-rotating vortex in a plane just upstream of the trailing edge. The existence of this counter-rotating vortex was also suggested by Hazarika [1] and Dickinson [2]. This study reported limited hot wire measurements downstream the trailing edge (i.e. only in two planes, namely, at $X/C = 1$ and $X/C = 2$). Nevertheless,

it showed that the horseshoe vortex will persist even at axial location of 2 chord lengths downstream the trailing edge. This study also reported that static pressure variation throughout the flow was small and the secondary flow has negligible effect on static pressure field.

Ozcan and Olgmen [92] postulated a quasisteady streamline pattern of a "tornado" vortex downstream of a symmetrical unswept wing, at a 30° angle of attack, attached to a flat plate. Chang and Gessner [93] examined the juncture flow of a confined strut-endwall configuration. The NACA 0012 airfoil was bounded by parallel endwalls and parallel sidewalls. The reported secondary flow vectors, at two planes downstream of the trailing edge (at 0.8 and 2.3 chord length), showed the existence of a secondary vortex along with the horseshoe vortex which is of equal strength. The generation of this secondary vortex was attributed to the transverse pressure gradient caused by the presence of the side walls, which simulates another strut.

From the foregoing review, it is clear that most of the work done concentrated on regions ahead and around the juncture. The flow in these regions is fairly well understood for a particular configuration and oncoming flow parameters. To the author's knowledge there is no study, that fully describes the flow characteristics in the interaction region. Most of the reported work has been carried out at a distance of more than one chord length downstream of the trailing edge and not near the trailing edge. The purpose of this investigation is to collect experimental data that will give an understanding of the flow properties in this region.

CHAPTER III

EXPERIMENTAL EQUIPMENT, INSTRUMENTATION AND METHOD OF MEASUREMENT

In this chapter, the details of the experimental equipment, arrangements and methods of measurement deployed in the present experimental investigation on the development of the flow downstream the corner formed by an airfoil and a flat plate are given. The study included flow visualization, measurement of local wall shear stress, local surface static pressure, static pressure survey, total pressure survey and the three components of mean velocity.

3.1 Equipment and Instrumentation:

3.1.1 Wind Tunnel:

A low speed, subsonic, open circuit type wind tunnel, designed and constructed at the Turbomachinery Laboratory of the City College of the City University of New York, was used for the experiments. A centrifugal blower fitted with variable inlet guide vanes, driven by a 25 hp, 1770 rpm A.C. motor generates the flow in the wind tunnel. Air velocity at the 46 cm x 46 cm test section can be varied from about 5 m/s to 35 m/s by adjusting the inlet guide vanes setting. Non-uniformity of mean velocity profile, in the test section over 36 cm x 36 cm core was within $\pm 0.5\%$ at the maximum flow rate. The turbulence intensity in the test section was found to vary from 0.6% at a velocity of 6 m/s to 0.1% at a velocity of 20 m/s and higher. Inlet cloth filter was used to filter out the dust from the air passing through the wind

tunnel. The top and the bottom walls of the test section were removed from 10 cm ahead of the airfoil leading edge to facilitate probe traverses.

3.1.2 Test Model:

The flat plate was constructed from 2024T4 bare aluminum plate with dimensions of 116.3 cm x 46 cm x 1.9 cm (Figure (3.1)). The leading edge of the plate was made from a semi-circle aluminum bar of 1.9 cm diameter. The initial 7.0 cm of the leading edge of the flat plate was made rough by gluing a strip of sand paper with 0.2 cm grit size and 15 grits/cm² to promote a thicker turbulent boundary layer.

Based on a compromise of obtaining a thick boundary layer on the airfoil and its stall characteristics, NACA 65-015 base profile was chosen for the airfoil. The chord length of the airfoil was 25.4 cm. The leading edge and the trailing edge diameters of the airfoil were of 1.27 cm and 0.25 cm, respectively (Figure (3.2)). The airfoil was made in two sections with 21.9 cm span each and a 0.95 cm hole in the spanwise direction at the position of maximum thickness 'T' ($T = 3.81$ cm) on the chord line. A threaded rod passing through the holes in the flat plate and the airfoils, one on each side of the flat plate, holds the entire assembly together. The distance between the leading edges of the flat plate and the airfoil was 62 cm.

3.1.3 Traverse Mechanism and Rotary Devices:

3.1.3.1 Traverse Mechanism: The traverse mechanism was constructed with two compound slides held perpendicular to each other with the help of an angle plate (Figure (3.3)). The assembly was bolted to a table anchored to the floor. A device to

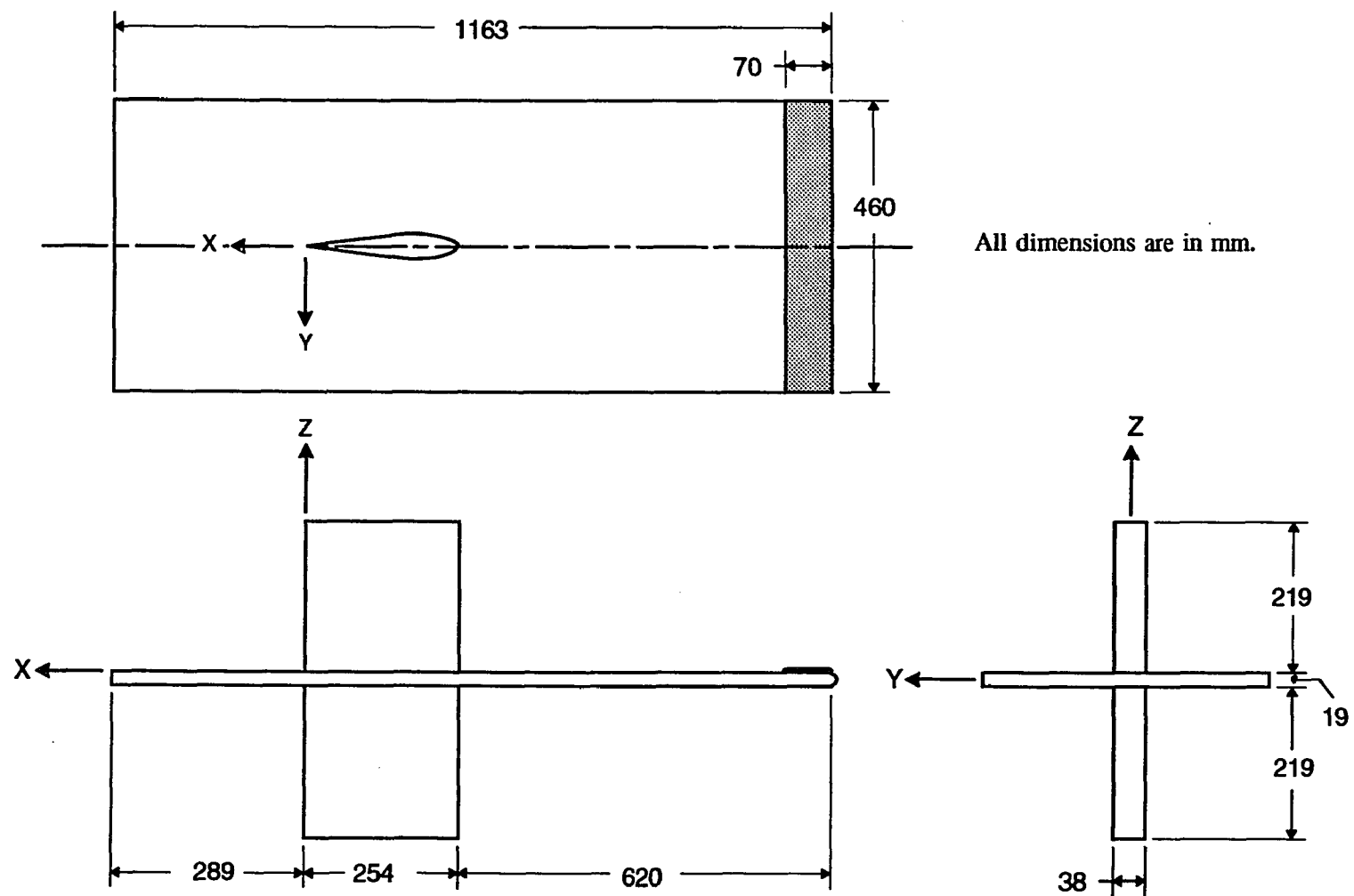
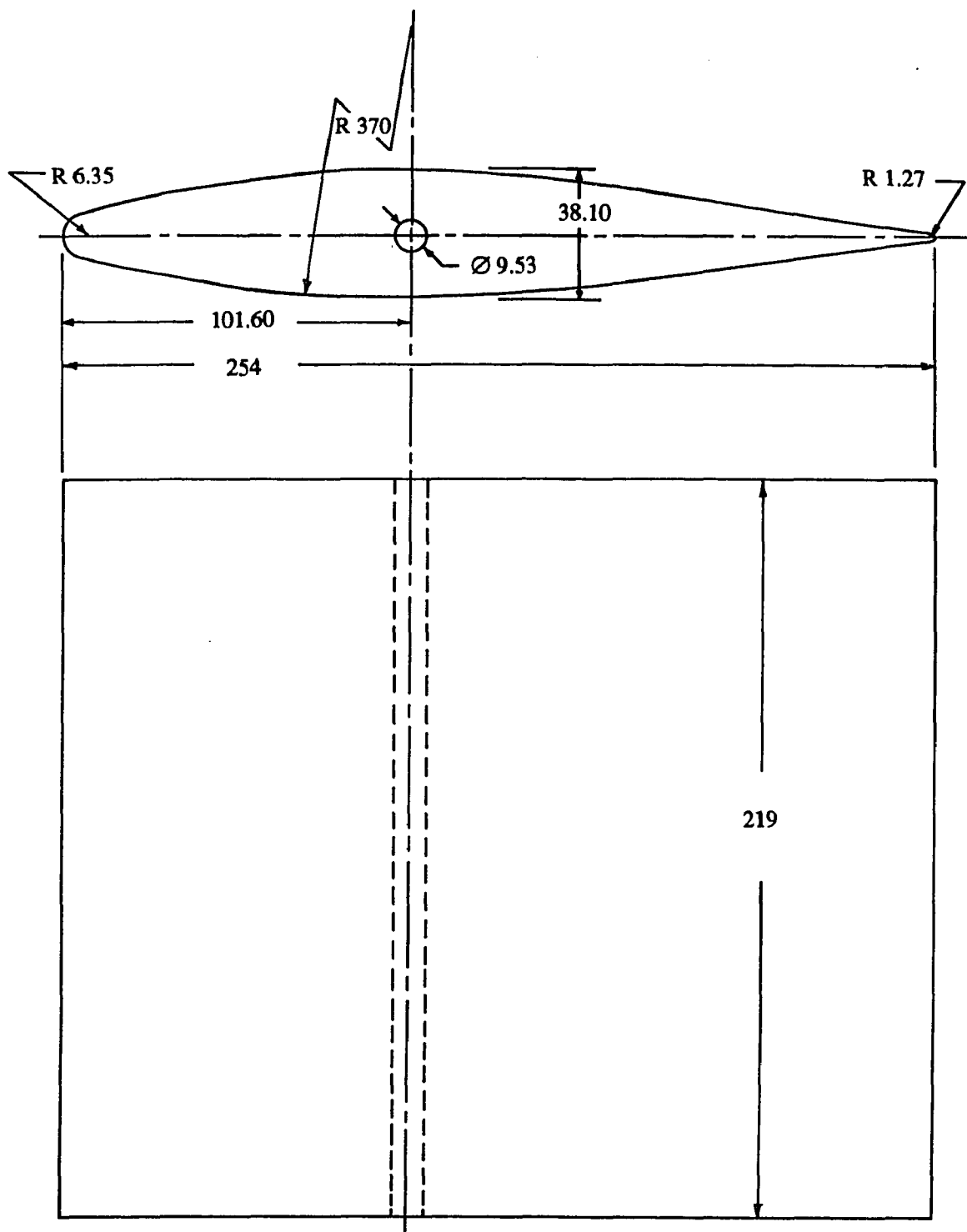


Figure (3.1) Schematic of the Test Model.



All dimensions are in mm.

Figure (3.2) Schematic of the Test Model Airfoil.

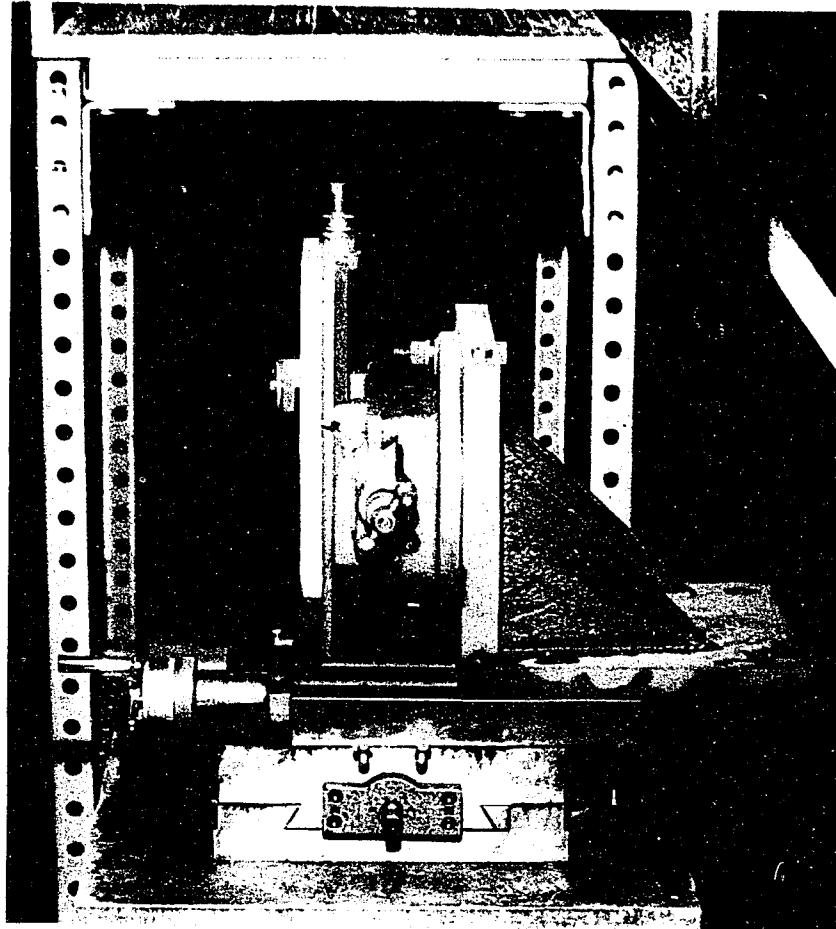


Figure (3.3) Photograph of the Traverse Mechanism.

rotate the probe about its own axis was bolted to the connecting bar. The connecting bar was clamped to the compound slide with its axis perpendicular to the axis of the test section.

The least count of the compound slides is 0.0254 mm. The total traverse distance parallel to the axis of the test section was 33 cm. The traverse was 20 cm perpendicular to the axis in the horizontal and vertical directions. Traverse distance of the probe in all the three directions can be extended by 5 cm by changing the clamping position of the connecting bar on the vertical surface of the compound slide. Subsequent extension in traverse distances was possible by changing the position of the angle plate on the horizontal compound slide.

3.1.3.2 Rotary Devices: Two rotary devices were used in the investigation; the first rotary device, used for rotating the Preston tube about its own axis, was machined from brass stock in two principle components. The base which was bolted to the connecting bar has a conical hole in which it carried the rotating part in a shape of a frustum of a cone with a hole through its axis and a chuck to hold the guide tube. Three adjustable spring loaded steel balls were provided to apply sufficient pressure between the base and the rotating part to hold them together snugly without impeding rotary motion. Two supporting rods, bolted to the base, held a guide block which had a hole aligned with the axis of the guide tube holder chuck. The protractor mounted on the rotating part and the vernier mounted on the base allowed the change in the angle to be measured to the least count of 0.2 degrees. The probe and guide tube assembly were held by the guide tube holder chuck and the axis of rotation made an angle of 90° with the axis of the test section.

The second rotary device was used for rotating the five hole directional probe about its axis. It consisted of three components; the rotary vernier, the linear traverse bar and the mounting base (Figure (3.4)). The rotary vernier was connected to a slide through which the linear traverse bar slides and its movement was resisted by friction of flat springs sliding on the bar, adjustable by a thumb screw which could be tightened to lock the probe at any location. The linear traverse bar was connected at one end to the mounting base via two 6.35 mm (0.25 inch) bolts.

This assembly held the five hole probe rotation firmly in two places, one to the rotary vernier by a two-piece 6.35 mm (0.25 inch) collet to lock the probe firmly in position without danger of denting the stem. The second through the mounting base via flat mounting pressure seal, which prevented the probe from vibrating when placed in the flow and ensured an aligned rotation by the rotating vernier. Angle of rotation of the probe, used for measuring flow direction (yaw angle), was measured in movements of 0.2 degree over a full 360 degrees on a protractor graduated in 2 degrees divisions. Both vernier and protractor were friction loaded with adjustable springs and balls for easy and accurate rotation. The adjusting spring screw could be fitted by a thumb screw or an Allen wrench to lock the probe at any angle. The vernier was fully adjustable, permitting the zero angle point to be set at any time without loosening the probe in the collet and losing the distance setting (Figure (3.4)).

This unit was clamped to the vertical compound slide of the traverse mechanism described in Section (3.1.3.1), which ensured a right angle between the probe axis of rotation and the axis of the test section.

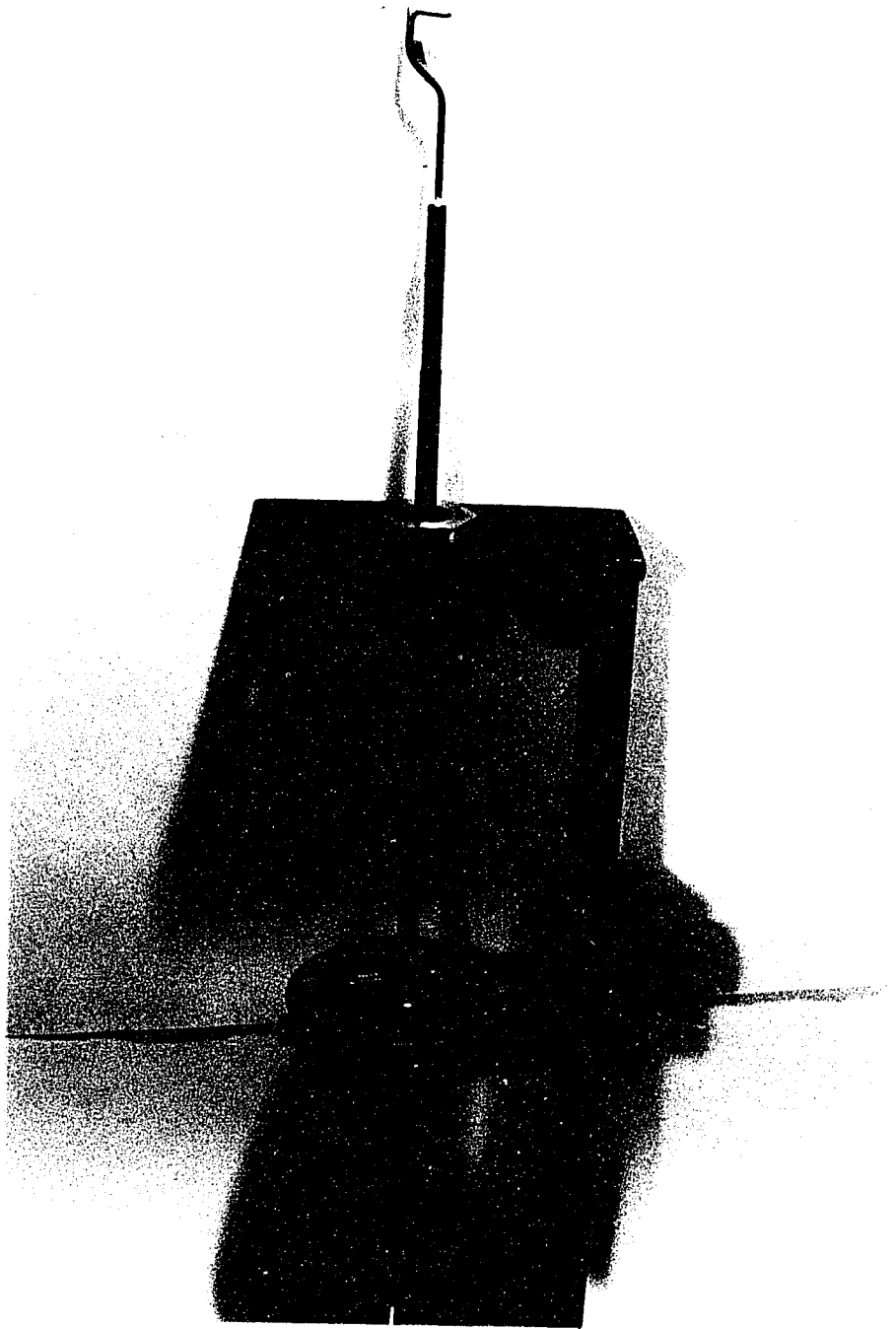


Figure (3.4) Photograph of the Second Rotary Device with the Five Hole Probe.

3.1.4 Grids:

A combination of two grids was used to create free stream turbulence level of 1.5% at 107 cm downstream of the grids (location of the leading edge of the airfoil) to avoid separation bubble on the airfoil (which appeared in the absence of the grids) and to increase the boundary layer thickness on the surface of the flat plate. The grids selected for this purpose were 1 mesh 2 mm diameter galvanized iron wire mesh followed by a 4 mesh 0.6 mm diameter galvanized iron wire mesh (Figure (3.5)). The second mesh had a 45° inclination to the first mesh. The grids were placed 45 cm ahead of the leading edge of the flat plate. The non-uniformity in the mean free stream introduced by the grids was less than 0.5% at a test section velocity of 27.0 m/s.

3.1.5 Probes and Instrumentation:

The probes used in these investigations were: static pressure probe for measuring surface static pressure, pitot-static pressure probe to measure ($p_t - p_s$) in the calibration of the five hole probe, three sizes of Preston tube for measuring near surface total pressure, Kiel probe for reference total pressure, total pressure probe of 1.08 mm diameter for total pressure measurements, five hole three dimensional directional probe for measuring yaw and pitch angles of the flow, total and static pressure (used in calculating the three components of mean velocity).

Both the static pressure probe and pitot-static pressure probe, made by United Sensor Corporation, were of 1.59 mm sensing head diameter (d), 14 d long head type. Three sizes of Preston tubes were constructed at the Turbomachinery Laboratory, made from stainless steel hypodermic needle tubing supplied by Unimetrics Corporation, and have the dimensions given in Table (3.1). Figure (3.6) shows a schematic of the Preston tubes employed in the experiment.

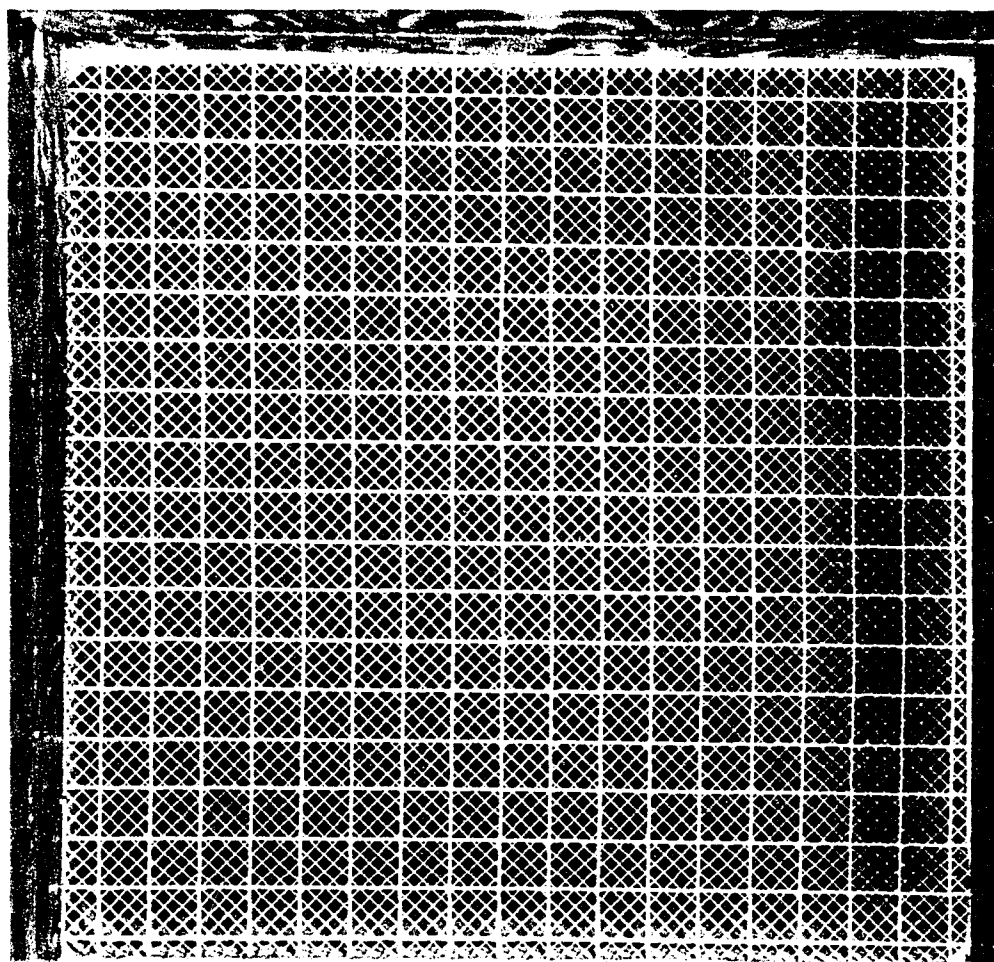
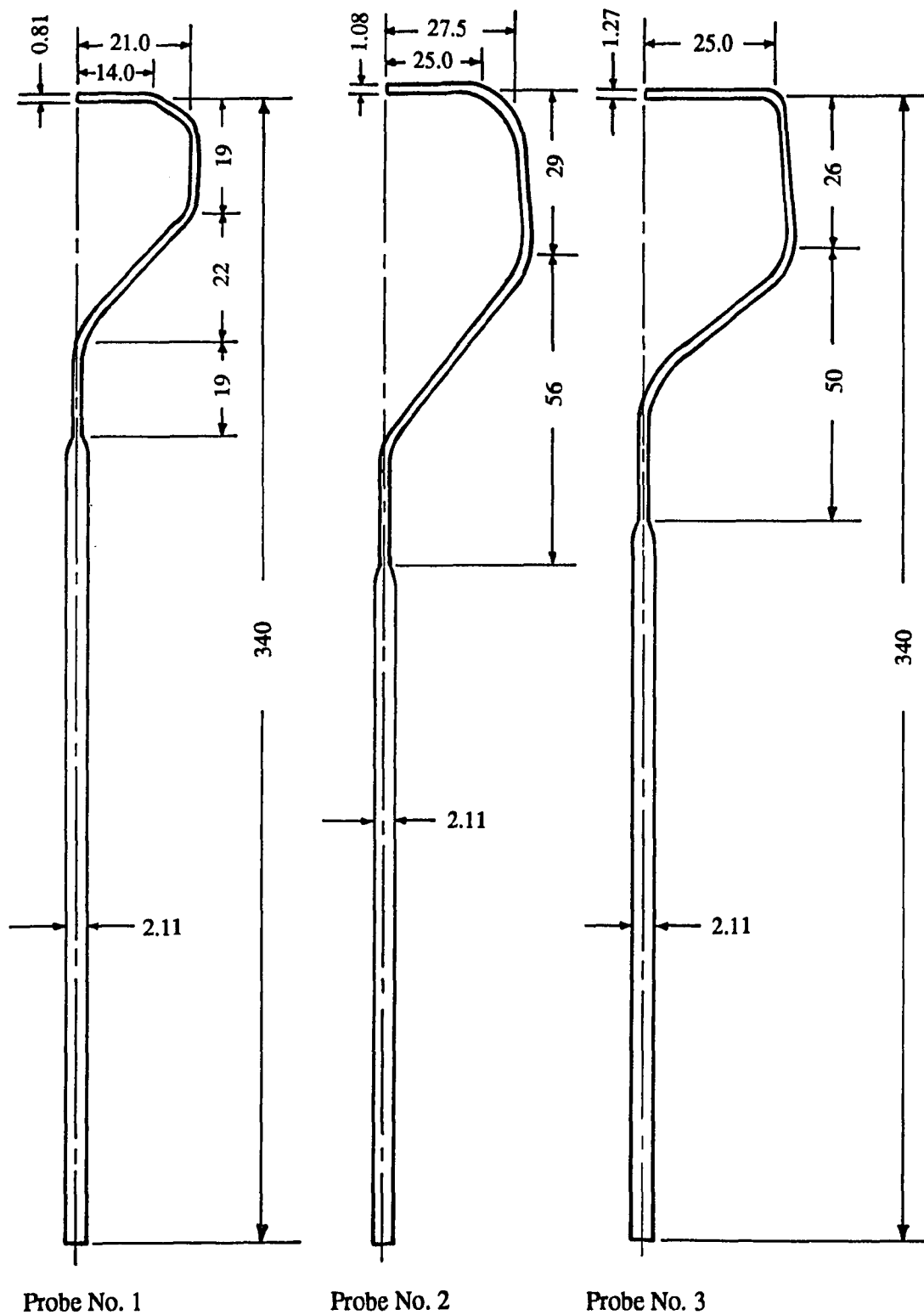


Figure (3.5) Photograph of the Turbulence Generating Grids.



All dimensions are in mm.

Figure (3.6) Schematic of the Preston Tubes.

Probe No.	size	outer diam. (mm)	inner diam. (mm)	outer/inner diam.
1	Small Preston tube	0.8128	0.4953	0.6094
2	Medium Preston tube	1.0795	0.6858	0.6353
3	Large Preston tube	1.2700	0.8380	0.6600

Table (3.1) Preston tube details

The geometrically similar circular Preston tubes were chosen to have as nearly as possible a ratio of internal to external diameter of 0.6, to be within the size range of Preston tubes used by Patel [19] in obtaining the calibration equations for measuring the local wall shear stress, as described in Section (3.2.2). Also the available sizes of hypodermic needles have approximately this ratio. Finally these types of tubes are robust and any thinner tubes will be liable to damage and difficult to construct. The tubes were bent to the shapes shown in Figure (3.6) so that the tip of the probe could be rotated concentrically about its axis of rotation (stem). This was important when aligning the probe tip with the flow direction to avoid displacing the tip away from the desired measuring point. Because in a flow situation where wall shear stress changes appreciably over a small distance, the Preston tube tip displacement can give an error in the measured value of the wall shear stress. This was checked several times and after each station measurement by rotating the probe tip (using the first rotary device) around a fixed needle to ensure concentricity. These tubes were soldered to an increasing diameter tubes such that the probes could be mounted in the available fixture. The medium Preston tube (No.2) was used in the measurements of the near

wall total pressure (used in the calculations of the local wall shear stress). The other two sizes (No.1 and No. 3) were used to corroborate the values of the local wall shear stress at two stations (Station #3 and Station #5).

A miniature sensing head Kiel probe of 1.6 mm diameter, supplied by United Sensor (model KAA-12), was used to measure the free stream total pressure for reference, which was maintained nearly constant during the experiment.

A miniature five hole probe, supplied by United Sensor, was type DC-095-250-13-125, the smallest size available commercially, with a conical sensing head of 2.3622 mm diameter and the dimensions shown in Figure (3.7). The probe is made of corrosion resistant non-magnetic stainless steel silver brazed.

The pressure measuring probes discussed above were connected to a Validyne made pre-calibrated differential pressure transducer with a range of ± 0.5 psid (model DP15) and CD15 carrier demodulator. The outputs were read from an integrating digital voltmeter (DISA 55D31). The pressure transducers were calibrated using model 1125 calibrator supplied by Thermo Systems Inc. against a micrometer with 0.0254 mm of water pressure resolution.

3.2 Methods of Measurement:

All testing for this study was conducted at a free stream velocity of 27.3 m/s at the location of the airfoil leading edge which corresponds to a Reynolds number, based on the length of the flat plate upstream of the leading edge of the airfoil (620 mm), of 1×10^6 and a Reynolds number, based on the airfoil chord length, of 4×10^5 . Boundary layer equilibrium on the flat plate at this location had been verified by earlier investigation [1]. On the flat plate, at the location of the airfoil leading edge, the

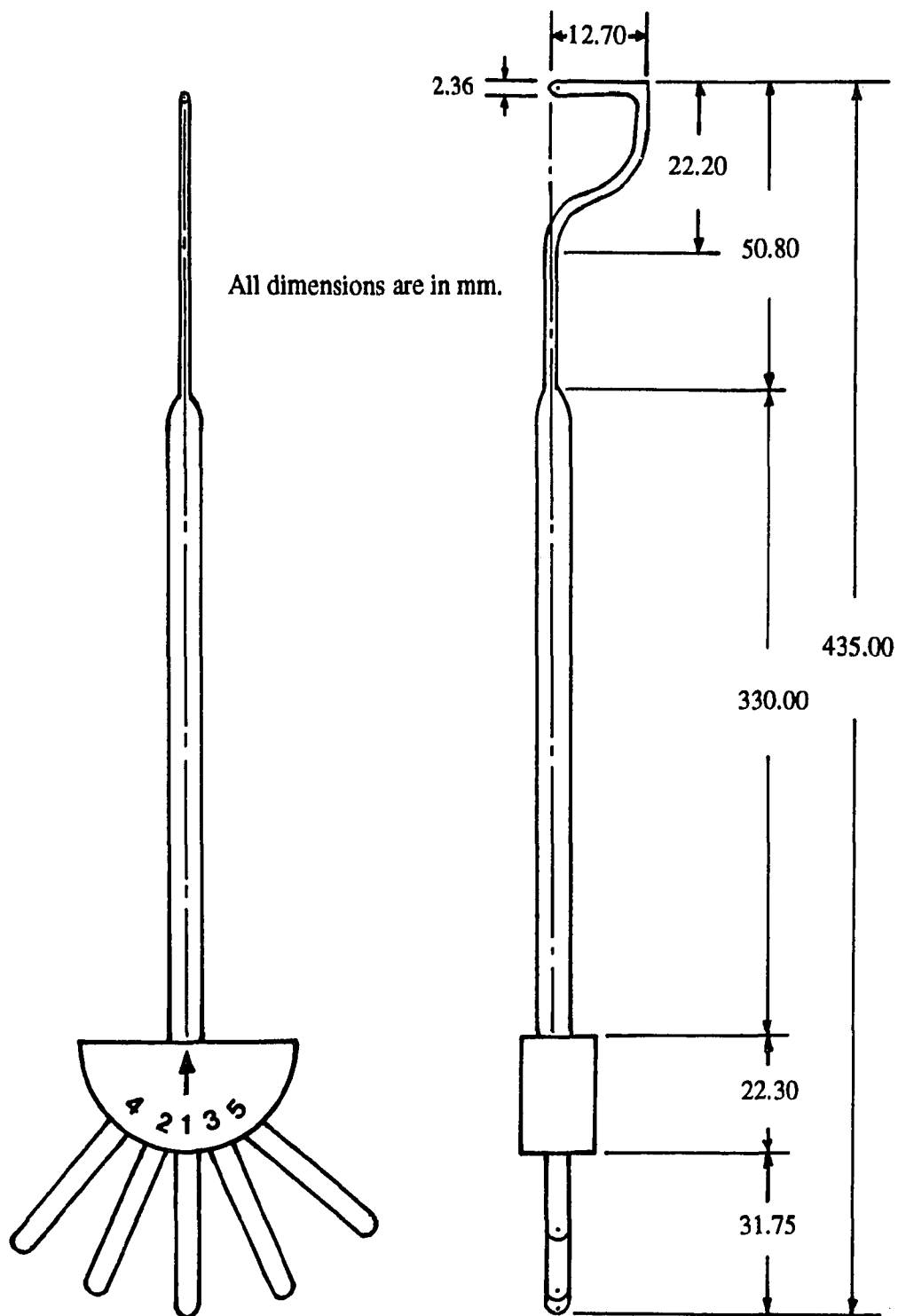


Figure (3.7) Schematic of the Five Hole Probe.

boundary layer thickness was 29 mm, momentum thickness was 2.56 mm and Reynolds number was 4.36×10^3 , based on momentum thickness [1].

3.2.1 Flow Visualization:

Flow visualization was performed first in order to obtain a qualitative picture of the interaction phenomenon that aided in selecting the location of discrete quantitative measurement.

Several flow visualization techniques were available in the literature e.g., laser holography, holographic interferometry, infrared thermography, smoke visualization and surface flow visualization [94]. Surface flow visualization was a good choice for this investigation as it provides permanent traces that show surface streamline direction and shape in relatively short period of time. It was economical, easy to perform and did not require complicated instrumentation. Surface streamline or "limiting streamline" was the streamline infinitely close to the surface, so it has infinitely small velocity but a definite direction.

Two techniques were used for flow visualization on the flat plate surface which included flow upstream the airfoil, the endwall corner region and the region downstream the trailing edge. Namely, surface oil film flow visualization and dot-matrix surface streamline flow visualization. The first technique provided a qualitative picture of the flow and the strength of the vortices in the corner and their extent downstream the trailing edge. The second technique provided the angles of the wall shear stress vector and helped to identify the different flow regions (i.e., separated region, wake region, interaction region and the undisturbed region). Flow velocity in the test section was set at 27.3 m/s by adjusting the inlet guide vanes, before conducting the flow visualization.

3.2.1.1 Surface Oil Film Flow Visualization: The oil mixture was formed from Kerosin (or #2 Diesel oil) pigmented with carbon lampblack (or Titanium Dioxide, TiO_2) with Oleic Acid as a dispersing agent. Heated magnetic stirrer was used to obtain a uniform mixture. Several trials were made to determine the ratio of the pigment to the oil in order to obtain well defined pattern on the flat plate surface for the present flow speeds.

The mixture was economically and evenly brushed on to the flat plate surface to avoid accumulation of mixture in regions of low velocity. Regions upstream the leading edge of the airfoil (about 80 mm), the endwall corner as well as downstream the trailing edge were covered with this mixture. The wind tunnel blower was started with the pre-set free stream velocity of 27.3 m/s. From 10 minutes of running time (using # 2 Diesel) to 30 minutes (using kerosin) were needed to obtain permanent patterns and all excess mixture was blown away. Video film was recorded (with camera viewing from the top) and photographs were taken (from the right side at 45° , looking downstream) during and after the test.

Once a permanent pattern was obtained, the wind tunnel was turned off. Tracing paper was carefully applied at the flat plate surface to replicate the patterns. It was carefully removed and an exact full size pattern was available. Photographs obtained and the discussion of results are presented in Chapter IV.

3.2.1.2 Dot Matrix Surface Streamline Flow Visualization: This technique is similar to the one developed by Langston and Boyle [95] and used by Hazarika and Raj [1]. This method provided a permanent record of ink traces that shows the shape and direction of the limiting streamlines. A dot of tracer (ink or dye) was applied to the surface of the region to be studied. The dotted area was then covered with continuous

film of liquid (solvent). After the tracer starts to dissolve, the air was allowed to flow over the region of interest, the continuous film was then acted upon by the local wall shear stress and flow in their direction. Thus, the tracer left streaks of paint conformable to the action of the wall shear forces. The dot acted as a stationary source of tracer.

Langston and Boyle [95] made the dots with a felt-tipped pen containing blue ink on a polyester drafting sheet that covers the region of interest. The dotted area of the drafting film was covered with continuous film by spraying Oil of Wintergreen (Synthetic Methylsalicylate) using aerosol sprayer.

Depending on the type of flow to be visualized, one dot of tracer can be applied and sprayed onto and traced after the first application, i.e., several flow runs, or several dots are applied at one time (matrix) and sprayed onto, i.e., one flow run.

For the smooth aluminum plate, a combination of Gloss Polyurethane Oil Enamel (as a tracer) and Mineral Spirits (as a paint thinner) was successful for well defined streaks. Different colors of the Polyurethane Oil Enamel of the same trade mark possessed different characteristics, e.g., dry or diffuse at a different time rate. Tracers used for this investigation were: Gloss Polyurethane Oil Enamel ("Red Devil" Trade mark) for blue color and Polyurethane Fortified Oil Gloss Enamel ("Enterprise" Trade mark) for red color. Mineral Spirits ("Nankee" Trade mark) as paint thinner. Hypodermic needles of 1.1 mm outside diameter connected to a plastic tube (as a reservoir), shown in Figure (3.8), were used in applying the tracer dots. These hypodermic needles gave a dot size of approximately 1 mm. Before starting the visualization tests on the flat plate surface, flow visualization on the airfoil surface was made to be certain that a separation bubble did not exist. The wind tunnel test section was turned 90° about its axis. At this position, airfoil span was horizontal. Dots of

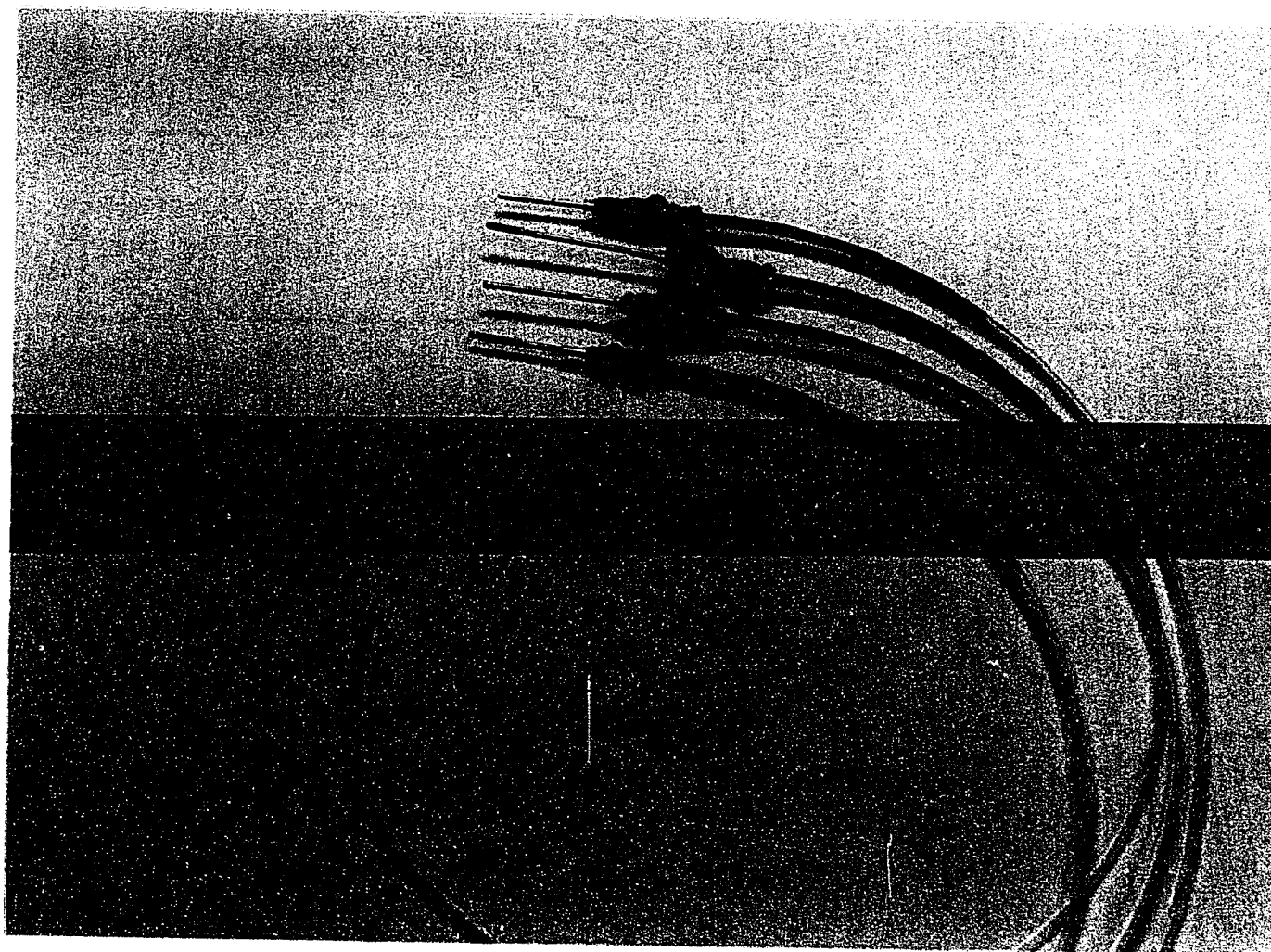


Figure (3.8) Photograph of the Hypodermic Needles Used in Applying the Tracer Dots.

red paint were applied at random covering the whole surface. Then, it was sprayed with paint thinner using a small hand held aerosol sprayer to give a thin and continuous film. Discontinuous film may give erroneous patterns. Within few seconds the wind tunnel blower was turned on until the surface was dry. No separation bubble was observed on the airfoil surface.

Two runs of flow visualization using this technique were made. The first run included areas upstream the leading edge and the corner region to visualize the extent of the separated limiting streamlines and the secondary flows downstream the trailing edge. The second run included areas only downstream the trailing edge to measure the angles of the local wall shear stress vector. In the first run, 51 mm upstream the leading edge of the airfoil, dots of blue paint were applied at 5 mm intervals in a row up to 25.4 mm on either side of the center plane. While the paint was wet, paint thinner was sprayed on the surface with the aerosol sprayer. Within a few seconds the paint started to dissolve and diffuse into the thinner. The wind tunnel blower was turned on immediately with the pre-set test velocity. Within a few minutes the thinner had been swept downstream or evaporated, leaving streaks of paint. Then, the wind tunnel was turned off. Another row of dots, 25.4 mm downstream of the first row, were applied on the streaks left from the first application and the process was repeated until well defined streaks were obtained up to one chord length downstream the trailing edge. The above streaks were in the region outside the separation line formed by the separated oncoming boundary layer, i.e., outside the region where the horseshoe vortex is dominant. Dots of red ink were applied at the flat plate surface, 25.4 mm upstream the trailing edge, between the separated limiting streamlines and the corner and the process was repeated downstream. These red streaks tended to move away from the centerplane (probably due to the presence of the wake). Dots of blue ink were applied to the wake region and the process was repeated. The model was

removed from the test section and photographs were taken. Photographs and discussion of the results are presented in Chapter IV.

In the second run, a dot matrix of paint was applied covering the region one chord length downstream the trailing edge and extends 60 mm on each side of the centerplane. Dots location was at random. The dotted area was sprayed with a paint thinner. After a few seconds, the wind tunnel blower was turned on until the surface was dry, and then, it was turned off. The streaks pattern was photographed and photocopied to get full size pattern of the streamlines.

3.2.2 Local wall shear stress:

The local wall shear stress was measured using the Preston tube, which is simply a rounded total pressure probe resting on the plate surface. While various techniques to measure the local wall shear stress exist in the literature, summarized by Brown and Joubert [96], Winter [97] and Bhargava and Raj. [12], Preston's method was employed in this investigation for its simplicity in construction and convenience of use. This method had shown consistent value of skin friction for two-dimensional as well as three-dimensional flows if the Preston tube was aligned with the local flow direction [98]. For example, Prahlad [51] used this method in three-dimensional turbulent boundary layer, Bragg [99] and Mojola and Young [32] utilized the Preston tube to measure the local wall skin friction in a streamwise corner, Shabaka [11] used the above method in a simplified wing body junction and Bhargava and Raj [12] used Preston tube No.1 (Figure (3.6)) to measure local wall shear stress in the endwall corner region as well as ahead of the leading edge of the same airfoil under this investigation. This method which was developed by Preston [100] was based on wall similarity, i.e., the law of the wall. Although several calibration equations for the use

of the Preston tube have been proposed, summarized by McAllister et al. [101] and Bhargava and Raj [12], Patel's calibration equation [19] has been widely used for its wider range and agreement with other techniques.

Patel's calibration equations are as follows;

$$0 < X^* < 2.9 \quad X^* = 0.5 X^* + 0.037 \quad 3.1$$

$$2.9 \leq X^* < 5.6 \quad Y^* = 0.8287 - 0.1381 X^* + 0.1437 X^{*2} - 0.0060 X^{*3} \quad 3.2$$

$$5.6 \leq X^* < 7.6 \quad X^* = Y^* + 2 \log_{10} (1.95 Y^* + 4.10) \quad 3.3$$

where

$$X^* = \log_{10} (\Delta P d^2 / 4 \rho v^2) \quad 3.4$$

$$Y^* = \log_{10} (\tau_w d^2 / 4 \rho v^2) \quad 3.5$$

$$\Delta P = p_p - p_s$$

p_p = Preston tube pressure

p_s = local wall static pressure

d = Preston tube outer diameter

ρ = density of air

v = kinematic viscosity of air

τ_w = local wall shear stress

It can be noted from the above equations that in order to calculate the local wall shear stress, the value of X^* should be calculated first from equation (3.4) in order to select a suitable calibration equation. This required the measurement of ΔP , which is the difference between the Preston tube pressure and the local wall static

pressure, p is calculated from the equation of state, u was read from the thermodynamic tables. Once the value of X^* was known, an appropriate calibration equation is chosen to calculate the value of Y^* and the local wall shear stress τ_w was calculated from equation (3.5).

Surface static pressure was commonly measured by wall pressure taps. However, due to the large number of measuring points needed at some stations, it was not practical to use wall pressure taps. Instead, a conventional static pressure probe (described in Section (3.1.5)) was used, resting on the surface, to measure the surface static pressure. Sproston and Goksel [102] calibrated a static pressure probe resting on the surface against the conventional wall pressure taps (static holes) in zero as well as non-zero pressure gradient flows. The agreement between the two methods was within 2 percent of the free stream dynamic head. More recently, Bhargava and Raj [103] investigated the surface static tube accuracy in three-dimensional turbulent flows and showed that the surface static tube can be used with maximum error of less than 7% of free stream dynamic head compared to the conventional technique.

Based on a compromise of the Preston tube size and its rigidity, Preston tube No.2 was employed in this investigation. The other two tubes (No.1 and No.3) were used to corroborate the results of tube No.2 at two stations. The Preston tube was installed in the first rotary device (described in Section 3.1.3.2) and this device was clamped to the traverse mechanism (described in Section 3.1.3.1). The contact of the Preston tube tip with the flat plate was ensured visually in addition to the simple electrical circuit shown in Figure (3.9). The voltmeter non-zero reading indicated the contact as the open circuit became a closed one. The resting of the Preston tube on the surface is crucial in the computation of the local wall shear stress as the measured total pressure should be as close as possible to the true local total surface pressure and not at a distance from the plate. This step was followed with the static pressure probe

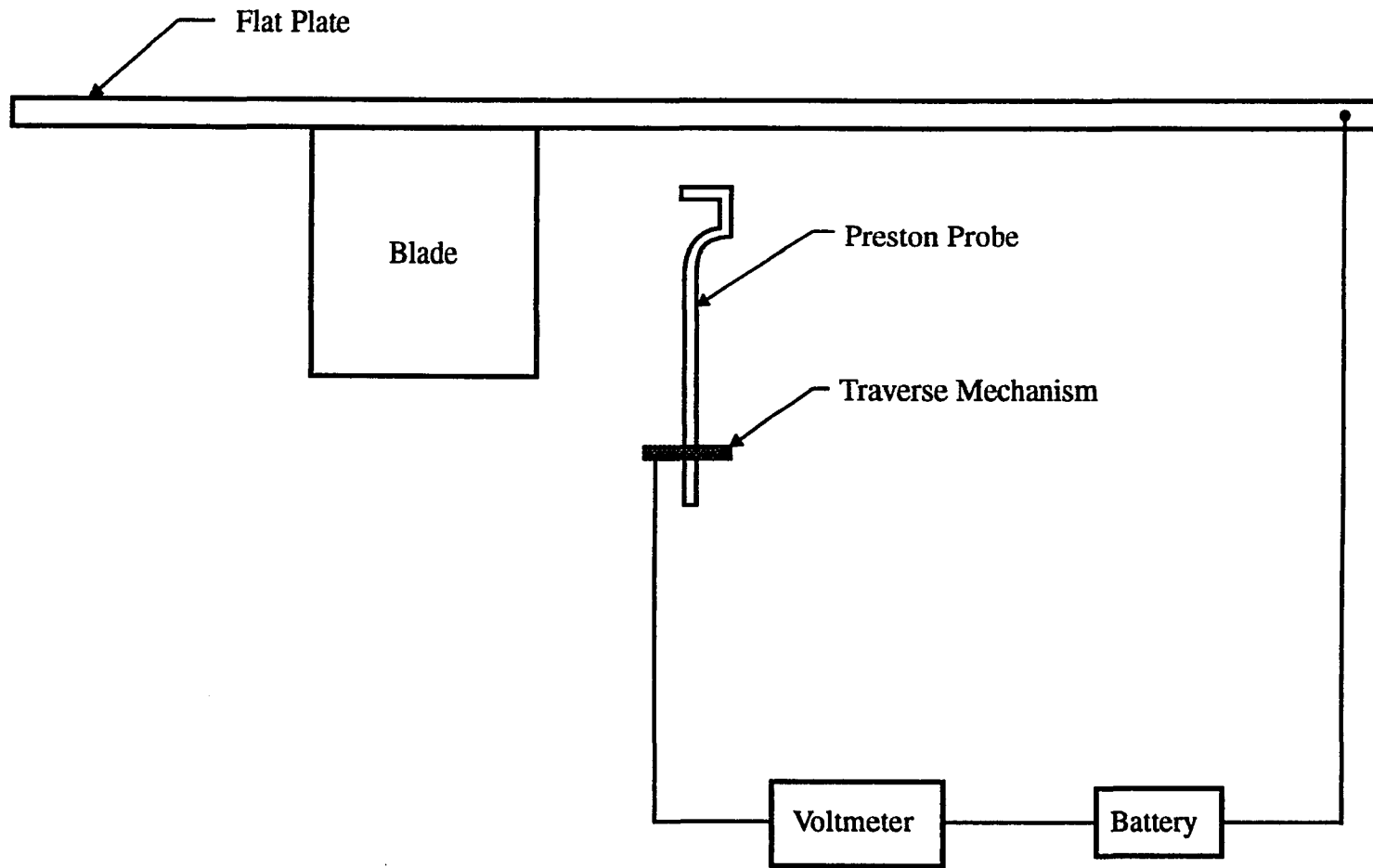


Figure (3.9) Simple Electrical Circuit Used to Ensure the Contact of the Preston Probe with the Flat Plate Surface

as well. Once the contact was ensured, the Preston tube was rotated about its axis using the first rotary device until the maximum value of pressure was achieved. This maximum value of pressure was recorded and used in the computation of τ_w . This procedure was adopted at all measurement points on the flat plate. The angular position of the Preston tube was not recorded as the pressure readings from three sizes of Preston tube were not sensitive to ± 6 degrees of angular rotation (Section 4.3.2), and the local wall shear stress direction was measured from the surface streamlines obtained from flow visualization (Section 3.2.1.2). Surface static pressure and Preston tube traverses were carried out on the flat plate at 8 axial locations downstream the trailing edge of the blade (airfoil), namely, Stations #0 through Station #7. The location of these stations is shown in Figure (3.10) and are tabulated in Table (3.1). Care was taken to ensure the contact of both probes with the flat plate at the same physical measuring point.

It should be mentioned here that all measurement points were taken on the right-hand side (looking downstream) of the geometrical plane of symmetry (will be referred to as the centerplane), i.e., positive Y. This was based on the assumption that the flow is symmetrical on both sides of the geometrical plane of symmetry. Check for symmetry was carried out on both sides of the plane at Stations #5 and #6 using Preston tube No.2. Asymmetry existed in the values of wall shear stress (Section 4.3.3).

The Kiel probe described in Section (3.15), was placed 50.8 mm (2 in.) ahead of the airfoil leading edge and 76.2 mm (3 in.) above the flat plate surface. At this position the Kiel probe was outside the boundary layer developed on the flat plate and outside the separation region ahead of the airfoil leading edge. At this location, the static pressure is equal to the atmospheric pressure. This Kiel probe was used to measure the reference total pressure (P_∞) and was kept nearly constant during the

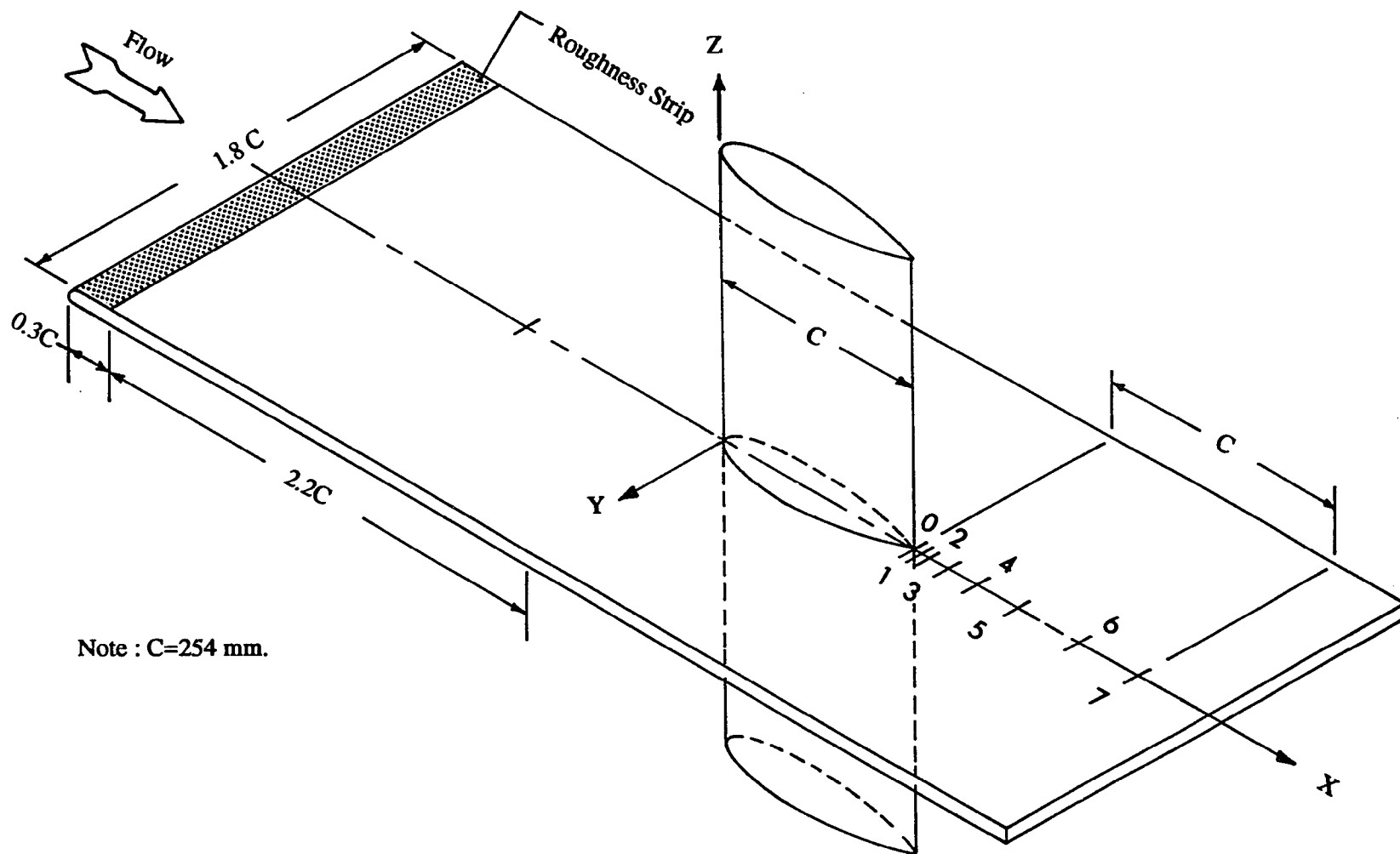


Figure (3.10) Schematic of the Test Model and the Measuring Stations.

Station Number	X mm	X in.	X/C	X/D
0	0.00	0.00	0.000	0.0
1	6.35	0.25	0.025	0.5
2	12.70	0.50	0.050	1.0
3	25.40	1.00	0.100	2.0
4	50.80	2.00	0.200	4.0
5	101.60	4.00	0.400	8.0
6	177.80	7.00	0.700	14.0
7	254.00	10.00	1.000	20.0

Table (3.2) Location of the Axial Stations

experimentation by adjusting the wind tunnel inlet guide vanes to give a corresponding nominal free stream velocity of 27.3 m/s. Also, the atmospheric pressure and temperature were recorded several times during the experimentation and the corresponding values were used in computing the local wall shear stress using a FORTRAN program employed by Bhargava and Raj [12].

3.2.3 Five Hole Probe:

The five hole probe was used to measure total and static pressures distribution, yaw and pitch angles and the three components of mean velocity at the six axial stations shown in Figure (3.10). Calibration apparatus and procedure are described in Appendix (A).

3.2.3.1. The Application of the Five Hole Probe: The connection arrangement in applying the five hole probe is shown schematically in Figure (3.11). Care was taken when clamping the manual rotary device, that holds the five hole probe, and the vertical compound of the traverse mechanism. The position of the rotary device with respect to the traverse mechanism was such that the value of C_{pp} [$C_{pp} = (P_4 - P_5)/(P_1 - P_2)$] read from the digital voltmeters is equal to the reading of C_{pp} that corresponds to zero pitch angle on the pitch angle pressure coefficient (C_{pp}) versus pitch angle curve (Appendix A). When the five hole probe is used in the semi-nulling mode of operation, it provided three differential pressure measurements via the differential pressure transducers. If the probe is rotated about the stem by the manual traverse unit until $P_2 - P_3 = 0$ as a read out on the differential pressure transducer (i.e., aligning the probe with the stream's yaw angle), then the yaw angle of the flow could be read on the rotary vernier. The differential pressures ($P_4 - P_5$), ($P_1 - P_2$) and ($P_1 - P_a$)

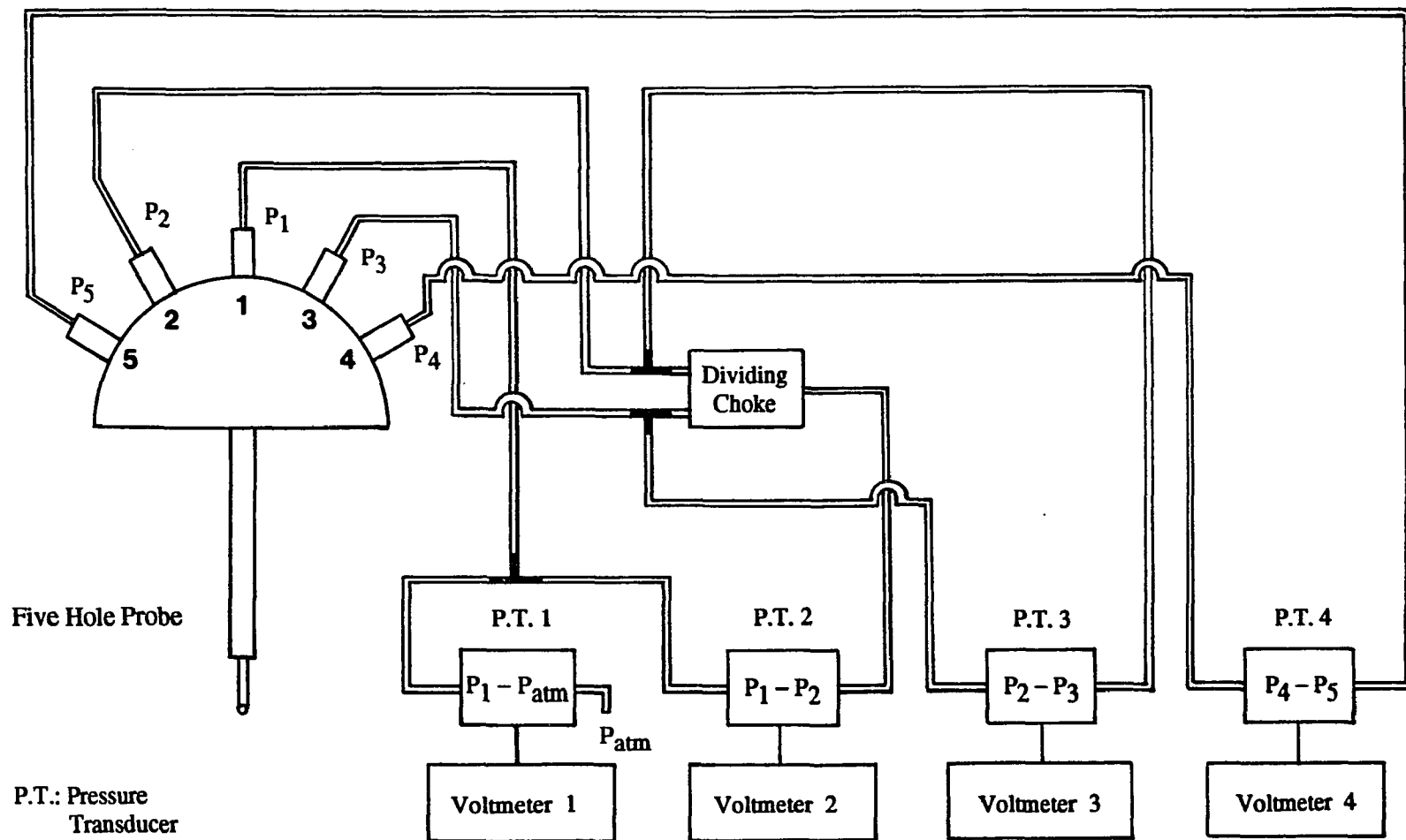


Figure (3.11) Schematic of Transducers Connection Used in the Application of the Five Hole Probe.

would then be observed. The pitch angle pressure coefficient (C_{pp}) would then be calculated (after multiplying each pressure differential with its corresponding differential pressure transducer coefficient). From the calibration spline curve fitting equation of C_{pp} vs. pitch angle, pitch angle θ was determined. The pitch angle was used as an input in the equation of the curve fitting C_{pv} vs. θ graph to find C_{pv} [$C_{pv} = (P_t - P_s)/(P_1 - P_2)$], and hence from the observed $(P_1 - P_2)$, $(P_t - P_s)$ was calculated. Similarly, θ was used in the equation of the curve fitting of C_{pt} vs. θ graph to determine C_{pt} [$C_{pt} = (P_1 - P_t)/(P_t - P_s)$]. With $(P_t - P_s)$ found from the previous step, $(P_1 - P_t)$ was determined. Another reading, $(P_1 - P_a)$, was observed across differential pressure transducer #1. The results are summarized in order as follows:

1. Yaw angle , ϕ
2. Pitch angle, θ
3. $(P_t - P_s)$
4. $(P_1 - P_t)$
5. $(P_1 - P_a)$

Bernoulli's equation was used to calculate the magnitude of the local total velocity vector V ,

$$V = \{ 2 (P_t - P_s) / \rho_a \}^{0.5}$$

where ρ_a is the air density at the time of taking the measurement at that point.

The three components of mean velocity can be calculated as follows;

$$V_x = V \cos(\theta) \cos(\phi)$$

$$V_y = V \cos(\theta) \sin(\phi)$$

$$V_z = V \sin(\theta)$$

Velocity vector resolution is illustrated in Figure (3.12).

The local total pressure, $(P_t - P_a)$, is calculated from:

$$(P_t - P_a) = (P_1 - P_a) - (P_1 - P_t)$$

and the local static pressure, $(P_s - P_a)$, is calculated from:

$$(P_s - P_a) = (P_t - P_a) - (P_t - P_s)$$

Thus, local total and static pressures, local yaw and pitch angles and the three components of mean velocity can be calculated for all the measuring points. A FORTRAN program is presented in Appendix (B).

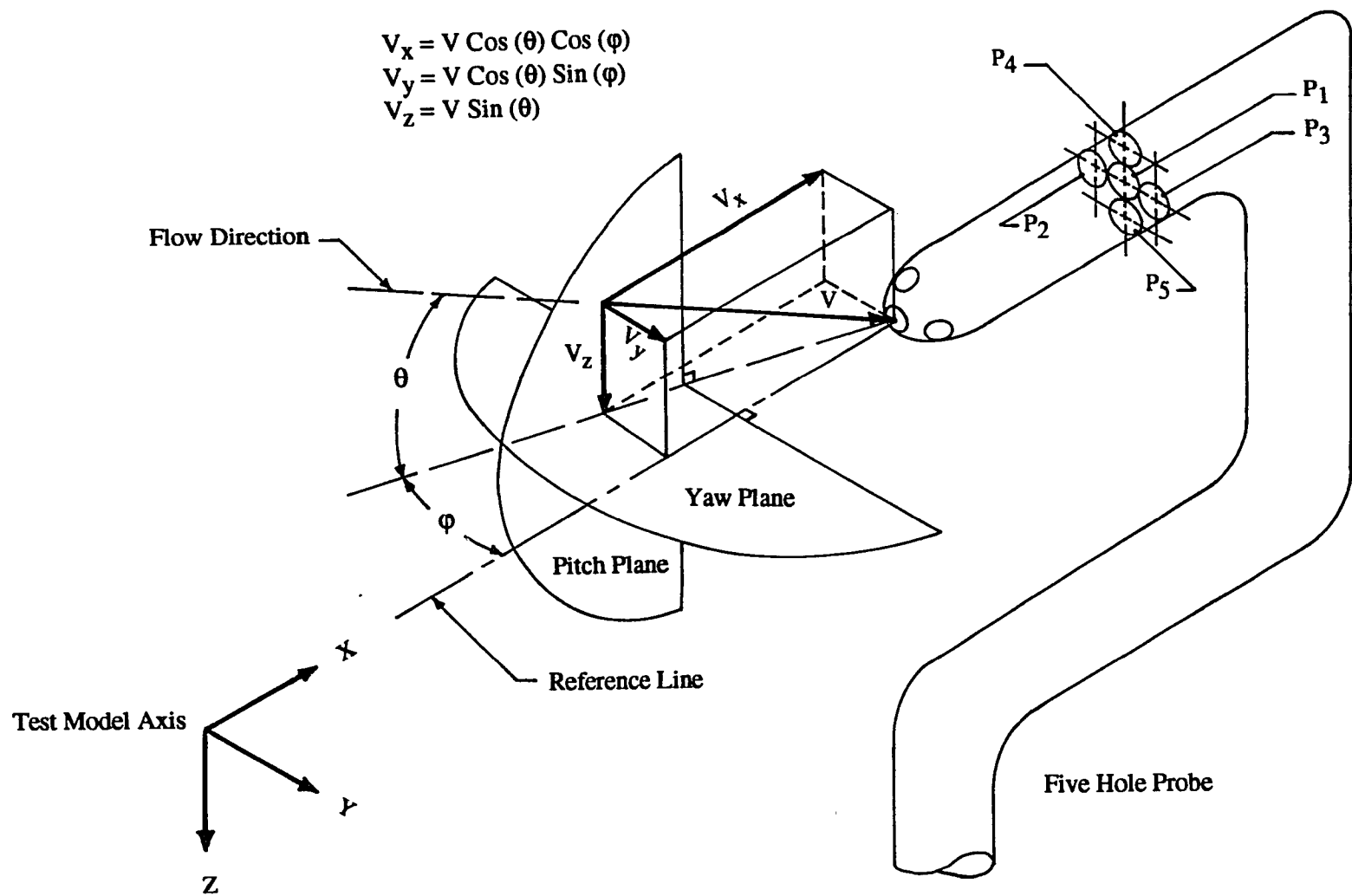


Figure (3.12) Velocity Vector Resolution of the Five Hole Probe.

CHAPTER IV

RESULTS AND DISCUSSION

In this chapter, the results of experimental measurements downstream of the corner, formed by a NACA 65-015 base profile airfoil (blade) and a flat plate, are presented and discussed. This includes the experimental results of flow visualization, local surface static pressure, local wall shear stress, static pressure survey, total pressure survey and secondary flow velocity vectors. The calibration procedure of the five hole probe is discussed and presented in Appendix A. The FORTRAN program used in processing the five hole probe measurements is presented in Appendix B.

4.1 Flow Visualization:

An overall photograph of the flow pattern on the surface of the flat plate after the blade has been removed is shown in Figure (4.1). A close-up photograph of the flow pattern ahead of the blade leading edge (after the blade has been removed) is shown in Figure (4.2). Other close-up photographs of the flow pattern downstream the trailing edge of the blade are shown in Figures (4.3) and (4.4). In these photographs, the dot traces are actually wall streaklines, or in other words, the curve composed of all fluid particles which have passed over the dye dot. The wall streamlines (limiting streamlines) and wall streaklines are identical for a steady flow. Langston and Boyle [95] showed that the wind tunnel start-up does not measurably affect the steady-state flow visualization final pattern.

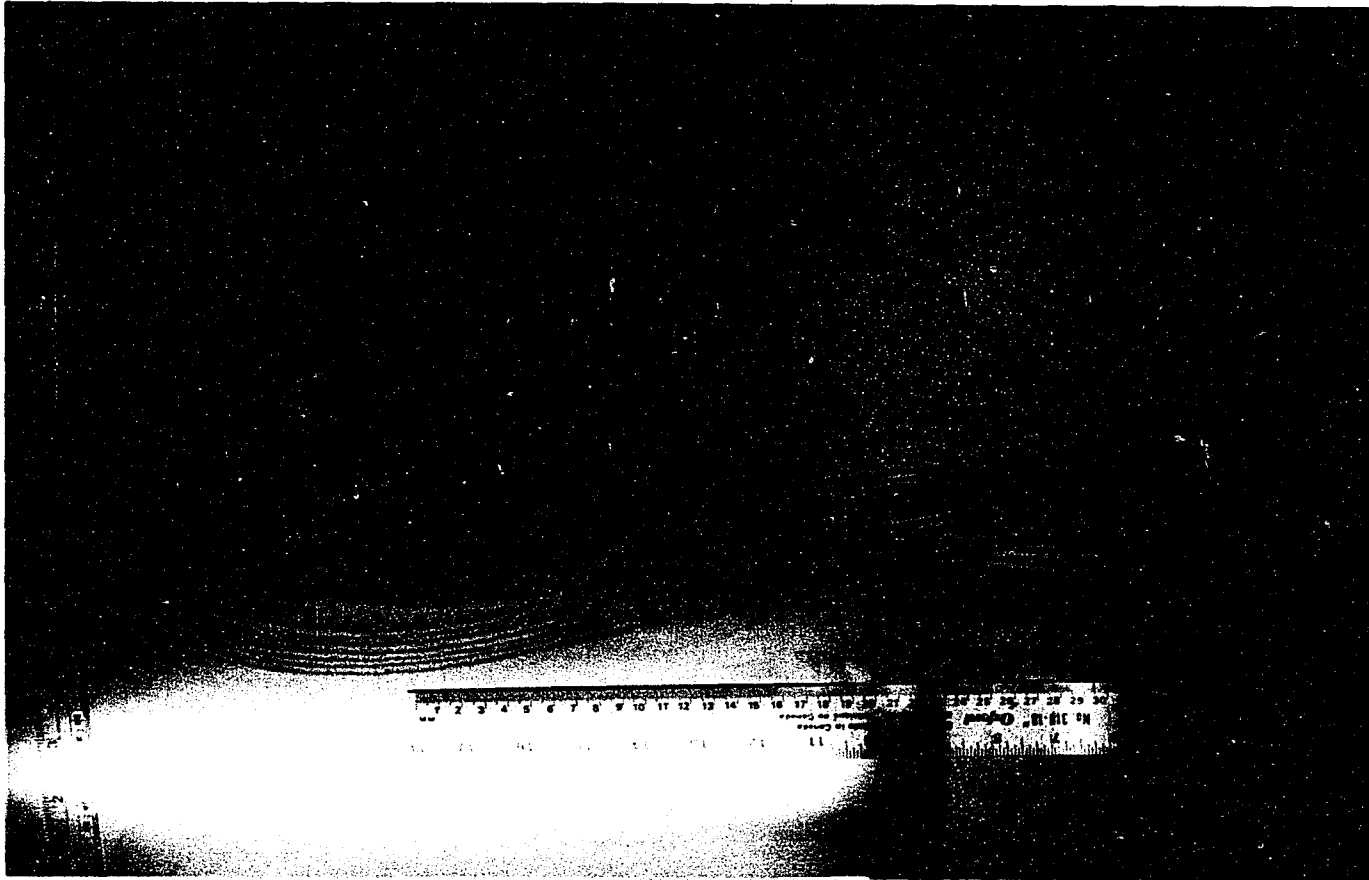


Figure (4.1) Photograph of the Surface Streamlines on the Flat Plate. Overall View. (Blade removed)

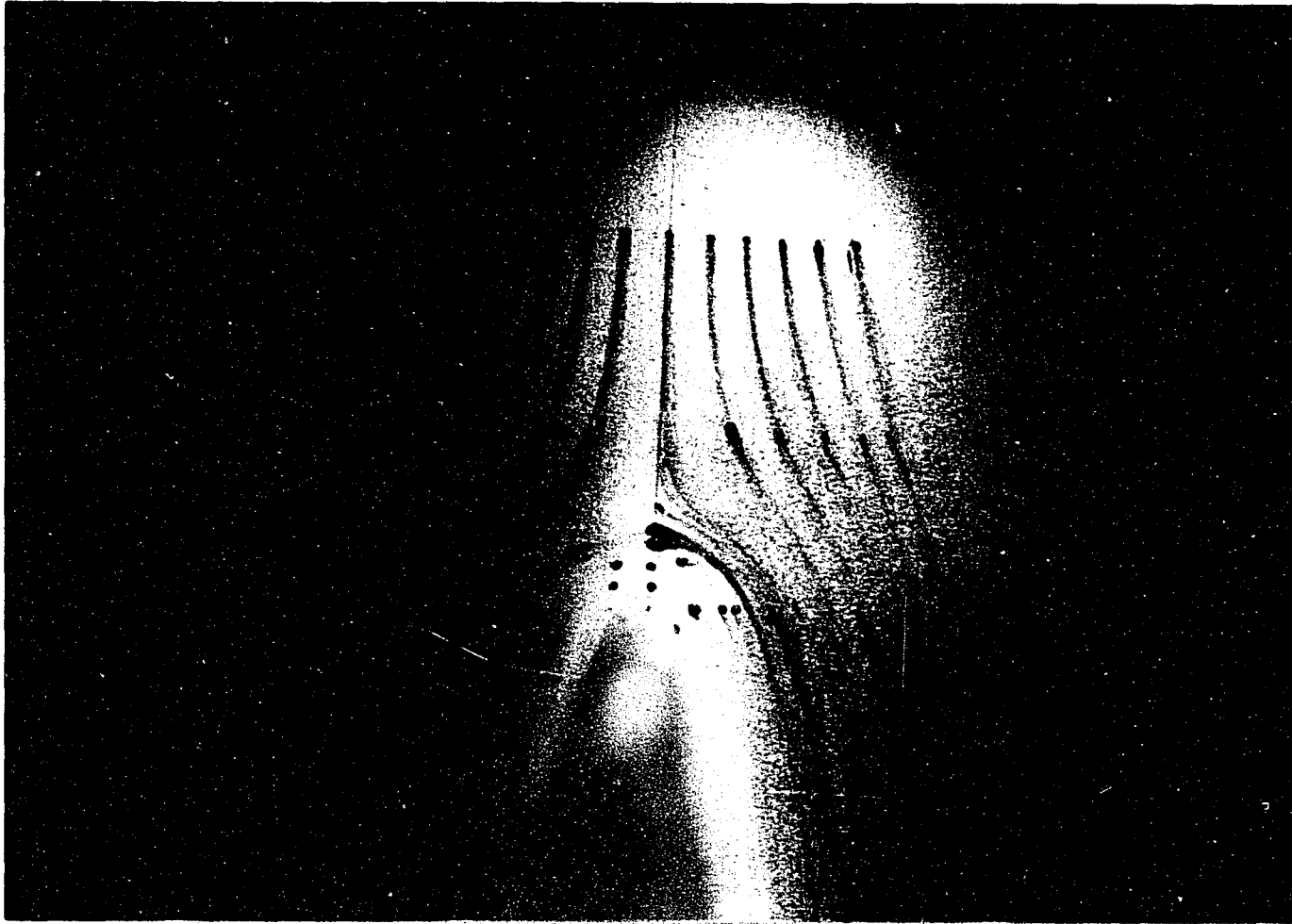


Figure (4.2) Photograph of the Surface Streamlines on the Flat Plate.
Close-up View at the Leading Edge Region. (Blade removed)

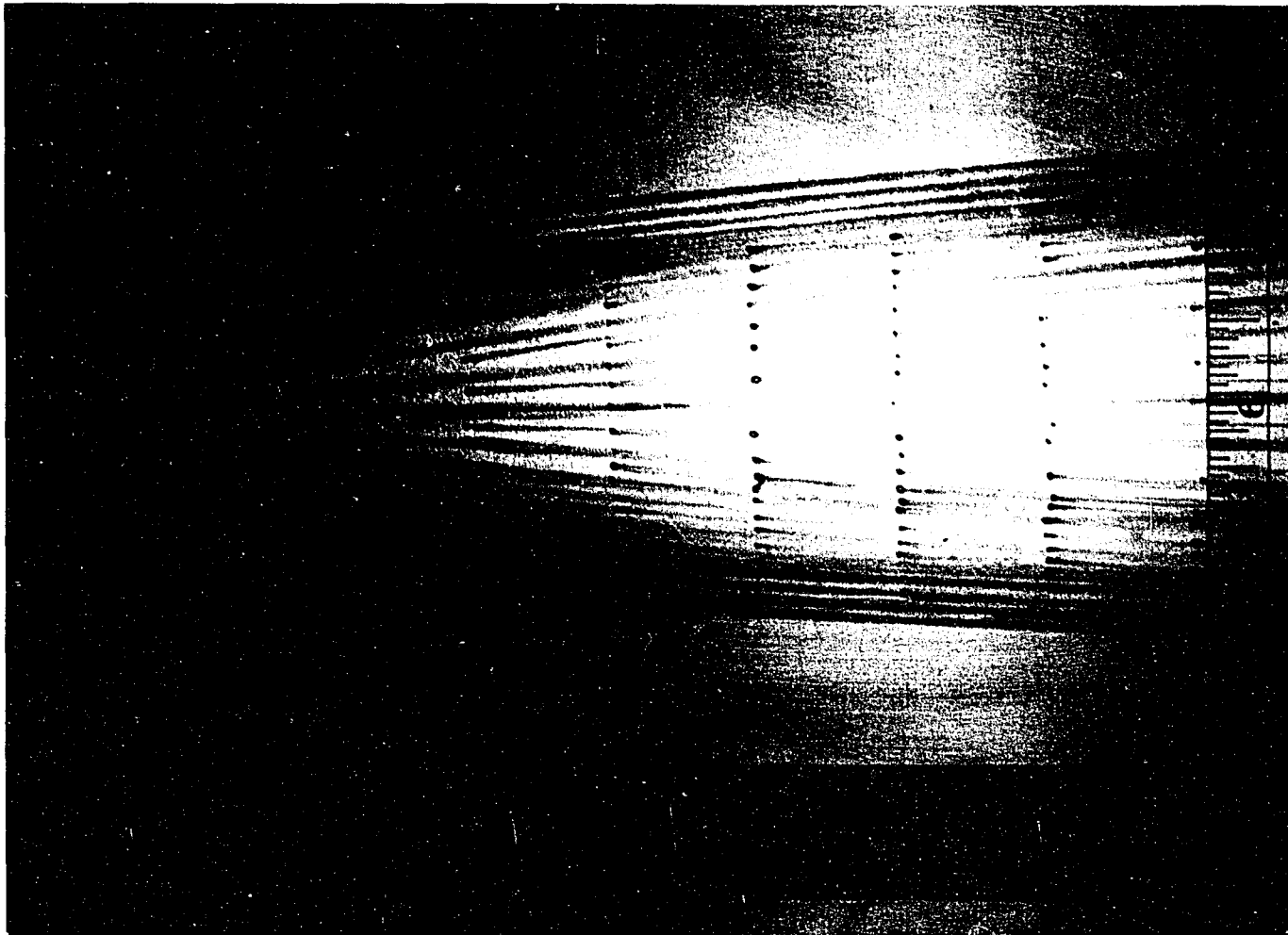


Figure (4.3) Photograph of the Surface Streamlines on the Flat Plate Downstream of the Trailing Edge Region.

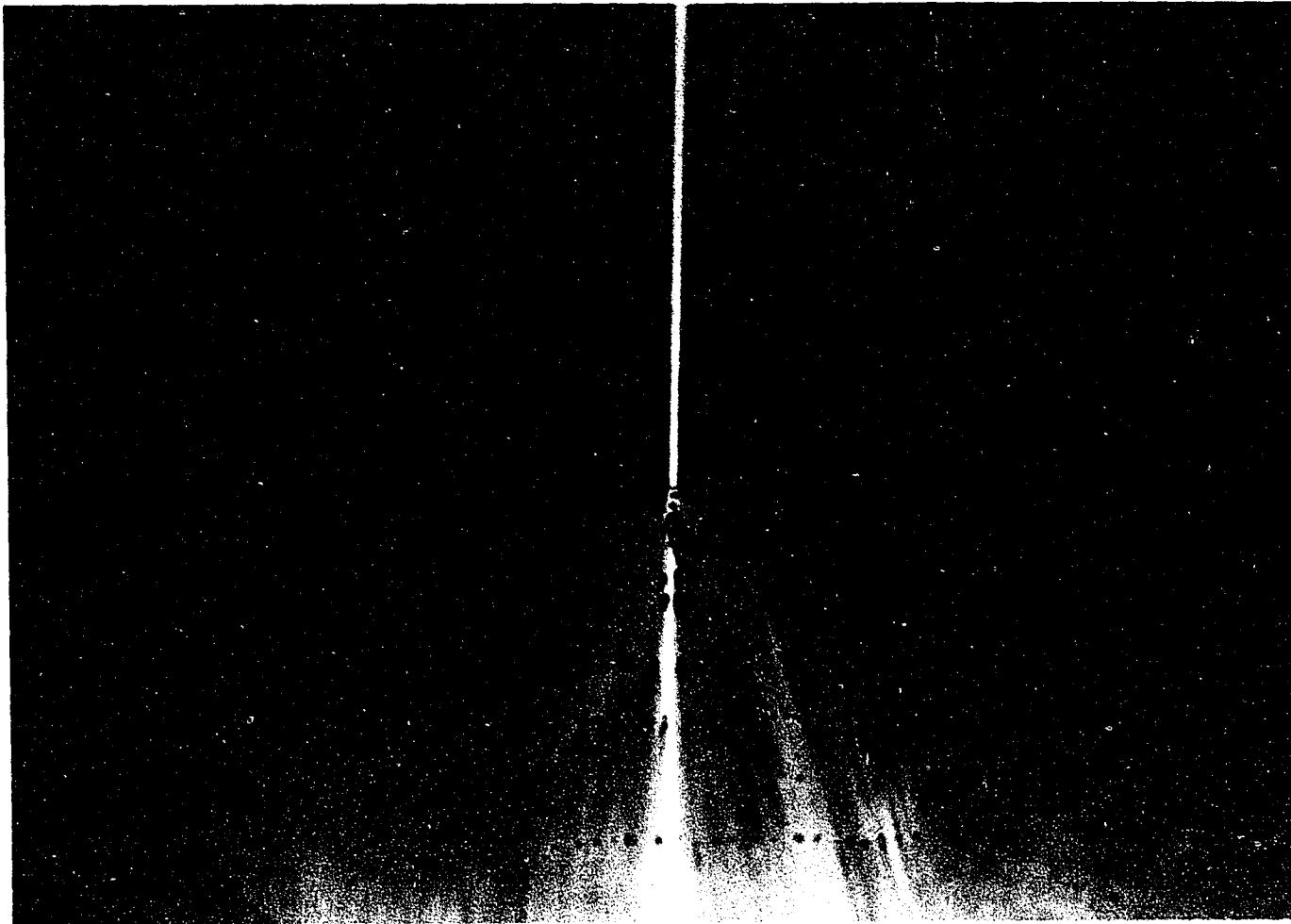


Figure (4.4) Photograph of the Surface Streamlines on the Flat Plate. Close-up View of the Trailing Edge Region.

The resulting flow pattern on the flat plate surface shows a typical horseshoe vortex that forms ahead of the blade leading edge with a stagnation point (saddle point) located at a distance of approximately 10 mm upstream of the blade leading edge. From this stagnation point two separation streamlines originate and wrap around the blade, one on each side. The perpendicular distance (distance in the Y-direction) between the separation streamline and the blade surface increases to a maximum of approximately 13 mm at the location of maximum blade thickness, then starts to decrease to a minimum of 7 mm at downstream axial location of 203 mm. These observations are identical to those observed by Hazarika and Raj [1] on the same airfoil. As the flow proceeds downstream, the perpendicular distance between the limiting separation streamline and the blade surface increases, reaching 12 mm at the location of the trailing edge.

Proceeding downstream the trailing edge, the distance between the limiting separation streamline and the centerplane continues to increase measuring 40 mm at axial location of one chord length downstream the blade trailing edge.

At the trailing edge location, two more limiting streamlines form, one on each side of the centerplane, which will be labeled as "wake limiting streamlines". A general flow diagram was constructed from flow visualization and is presented in Figure (4.5). The wake limiting streamlines seem to diverge the separation limiting streamlines away from the centerplane. The curvature imposed upon the separation streamlines is at its largest at the trailing edge location and diminishes as the flow proceeds downstream.

The displacement effect imposed by the blade wake limiting streamlines upon the oncoming upstream flow is caused by the induced pressure gradients in the downstream vicinity of the trailing edge (Section 4.2, Figure (4.10)). This effect

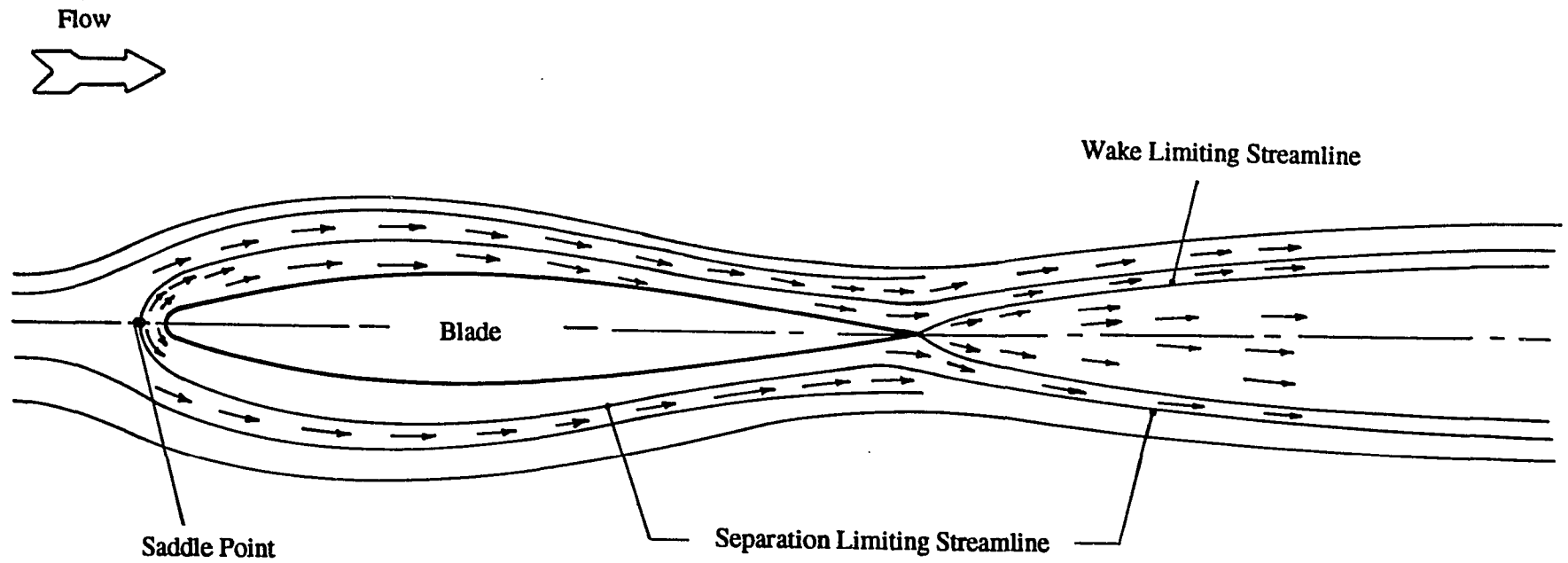


Figure (4.5) Flow Diagram of the Surface Streamlines obtained from Surface Flow Visualization.

ceases as the flow proceeds farther downstream of the blade trailing edge and the limiting separation streamlines become essentially parallel to the centerplane. The divergence of the streamlines can be attributed to the downwash contribution of the horseshoe vortex in this region.

Figure (4.6) shows a photograph of the resulting streamlines patterns obtained from the dot matrix flow visualization. The resulting pattern basically shows the same features observed in Figures (4.3) and (4.4).

Surface oil film flow visualization results gave a continuous pattern of these streaks at the vicinity of the leading edge. Figure (4.7) and (4.8) show the pattern of these streaks at the vicinity of the leading edge and the trailing edge, respectively. Figure (4.9) shows the tracing paper duplicate of the resulting streaks downstream the trailing edge.

It is clear that the flow pattern obtained from the surface oil flow visualization is in agreement with the results of the surface streamline flow visualization in showing the important features of the flow very near the surface.

The video tape recorded, while performing the surface oil film flow visualization, showed that there were two regions in which the oil film was swept away first. In other words, two regions where the flat plate surface beneath the oil film has been revealed earlier than the rest of the surface. The first region, in the endwall corner region, covers the area that extends from $X/C = 0.1$ to $X/C = 0.3$ downstream of the blade leading edge and between the cornerline and the limiting separation streamline. The second region, is the area just downstream the corner formed by the blade trailing edge and the flat plate surface and bounded by the wake limiting streamlines.



Figure (4.6) Photograph of the Surface Streamline Flow Visualization (Dot Matrix) on the Flat Plate.

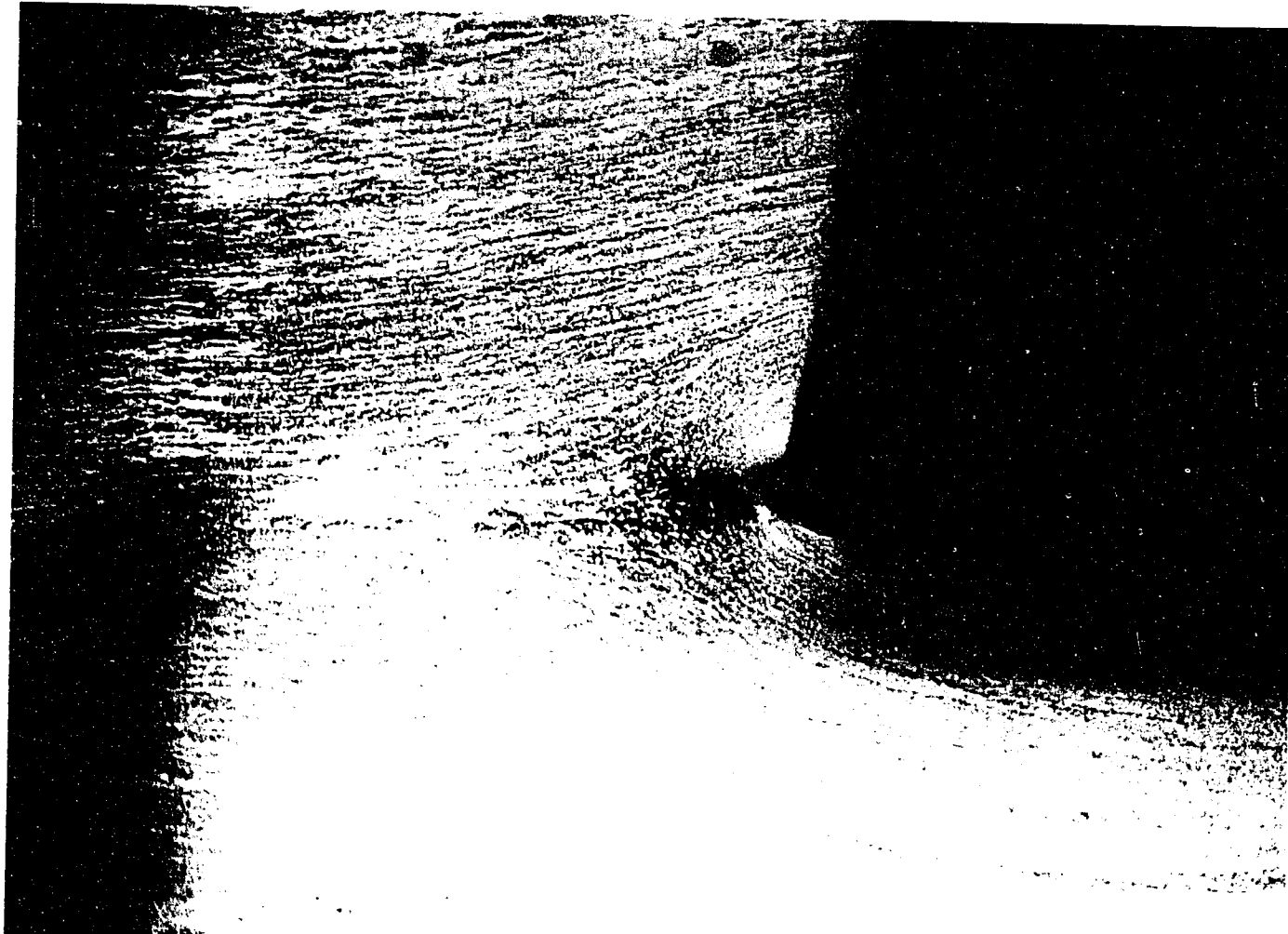


Figure (4.7) Oil Film Flow Visualization on the Flat Plate Surface in the Vicinity of the Leading Edge.

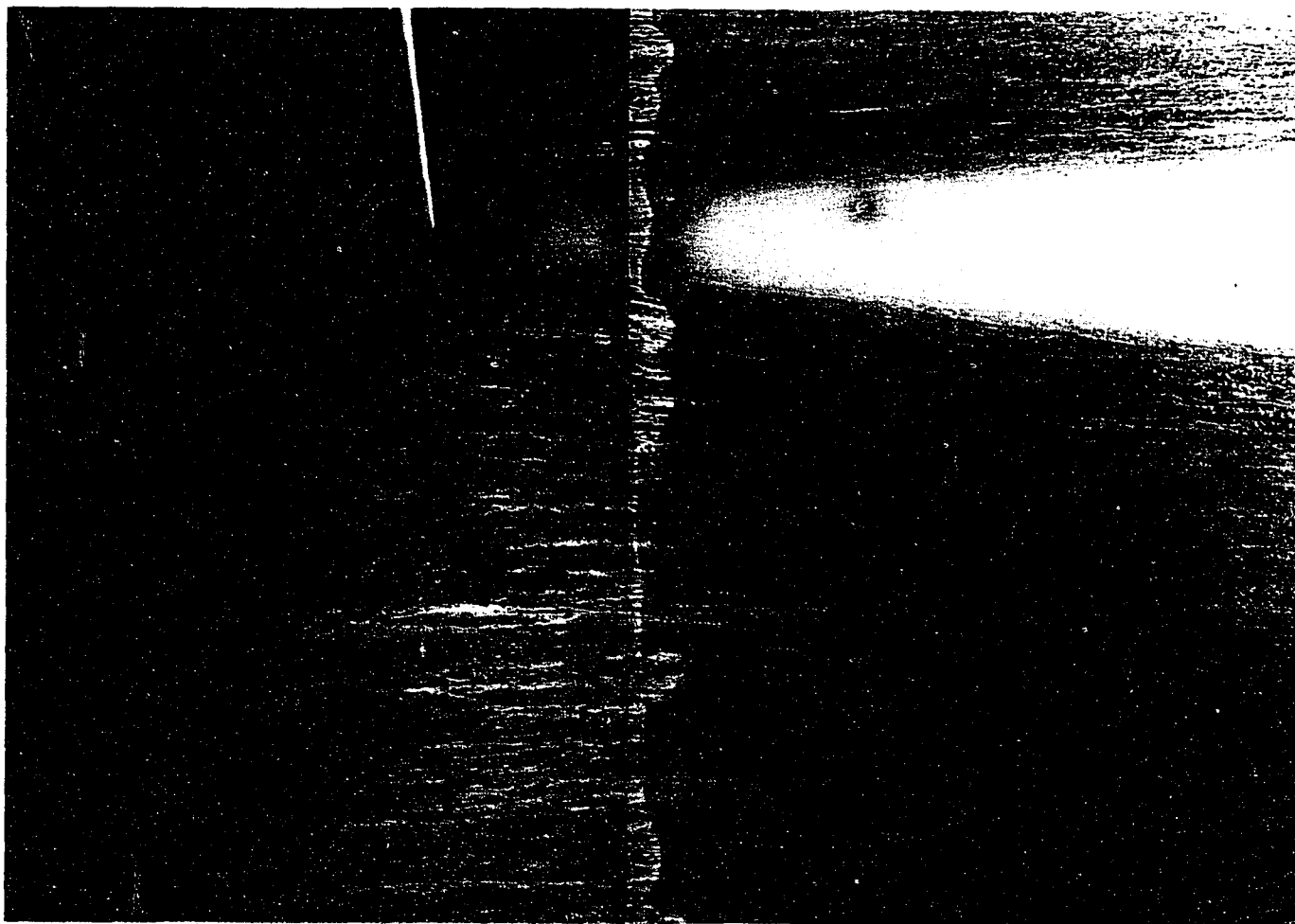


Figure (4.8) Oil Film Flow Visualization on the Flat Plate Surface in the Vicinity of the Trailing Edge.

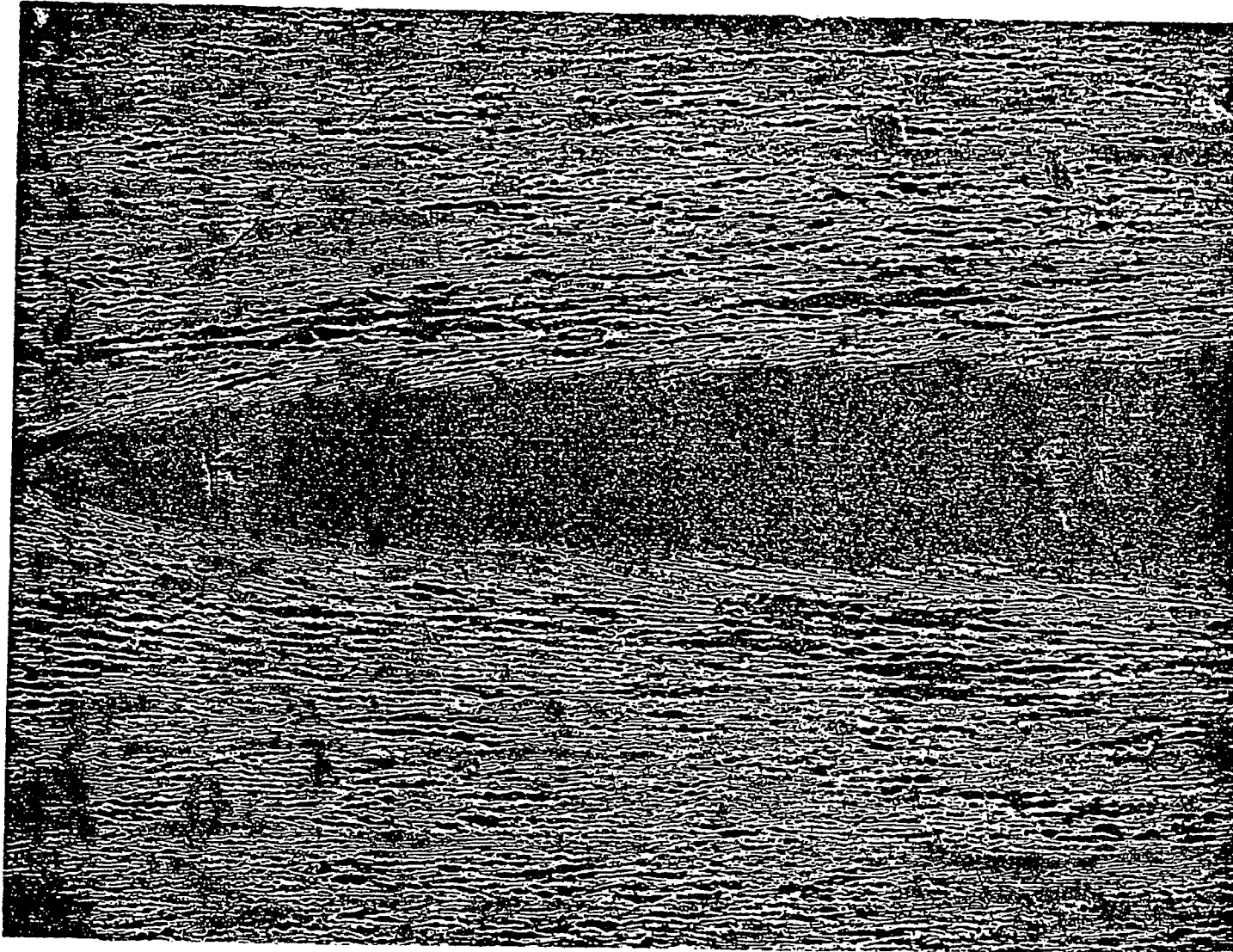


Figure (4.9) Duplicate of the Oil Film Flow Visualization on the Flat Plate Surface Downstream of the Trailing Edge.

The appearance of the first region can be explained in conjunction with the results of the local wall shear stress reported by Bhargava and Raj [12] (also by Abdulla et al. [104]), in the endwall corner region on the same airfoil. The high magnitude of wall shear stress in this region sweeps the oil film downstream with greater shearing action than the other regions (see Figure (4.17-a)). A similar observation was made by Pierce and Harsh [75] flow visualization in the corner between a streamlined cylinder and a flat plate. In the second region, i.e., downstream of the trailing edge, the wall shear stress levels are of low magnitude (see Figure (4.18)) and not of high magnitude as suggested by Devenport and Simpson [105]. The appearance of the surface in this case (i.e., in the second region) can be attributed to the induced pressure gradients at this region (section 4.2, Figures (4.10) and (4.11)) which displace the oncoming upstream oil film away from this region and prevent it from being covered again.

4.2 Surface Static Pressure:

Flat plate surface static pressure is represented by the static pressure coefficient, C_p , which is defined as;

$$C_p = (P_s - P_a) / (P_\infty - P_a)$$

where

P_s = Local wall static pressure

P_a = Atmospheric pressure

P_∞ = Free stream total pressure

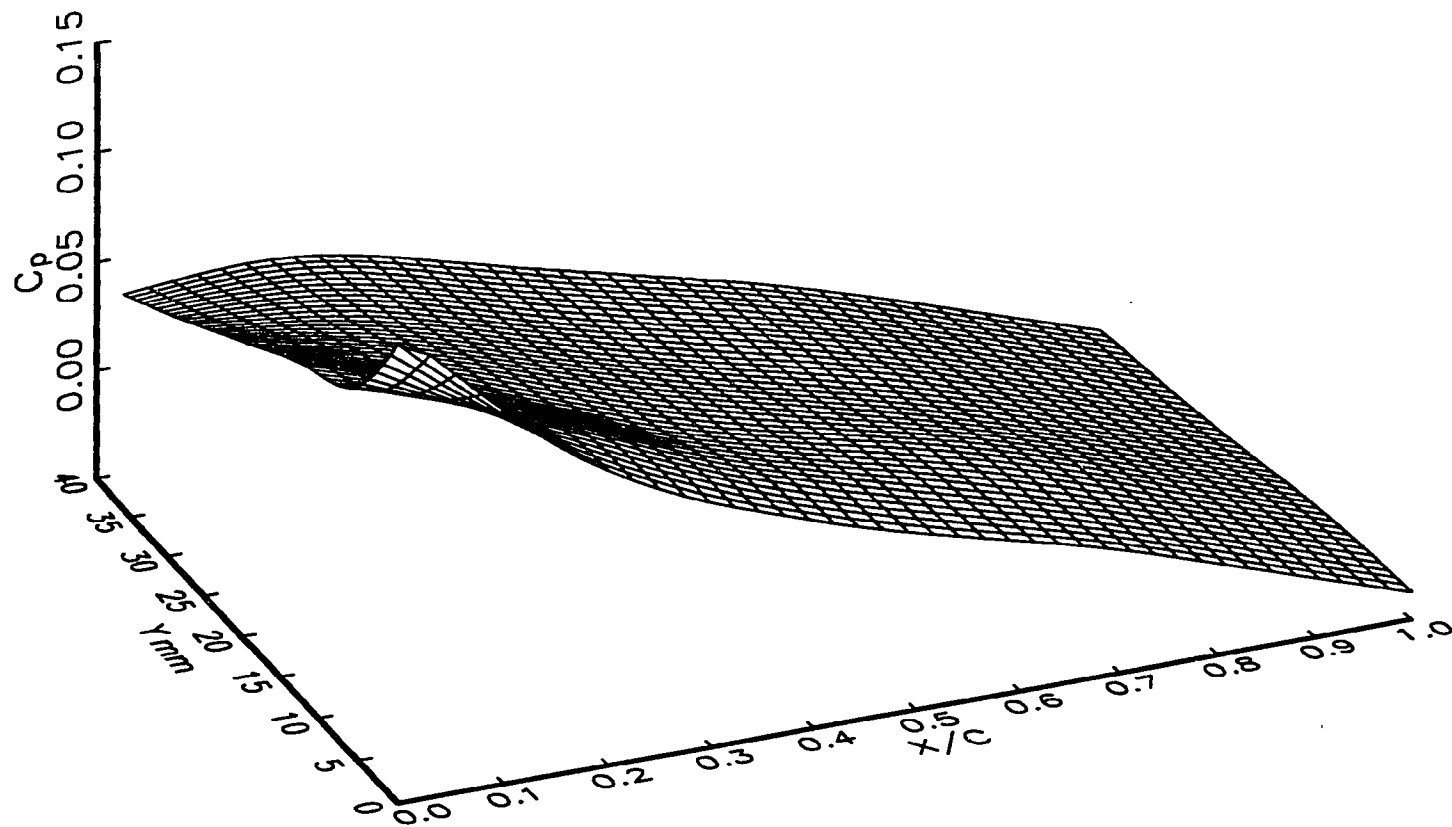


Figure (4.10) Surface Static Pressure Variation Downstream of the Trailing Edge (Isometric presentation).

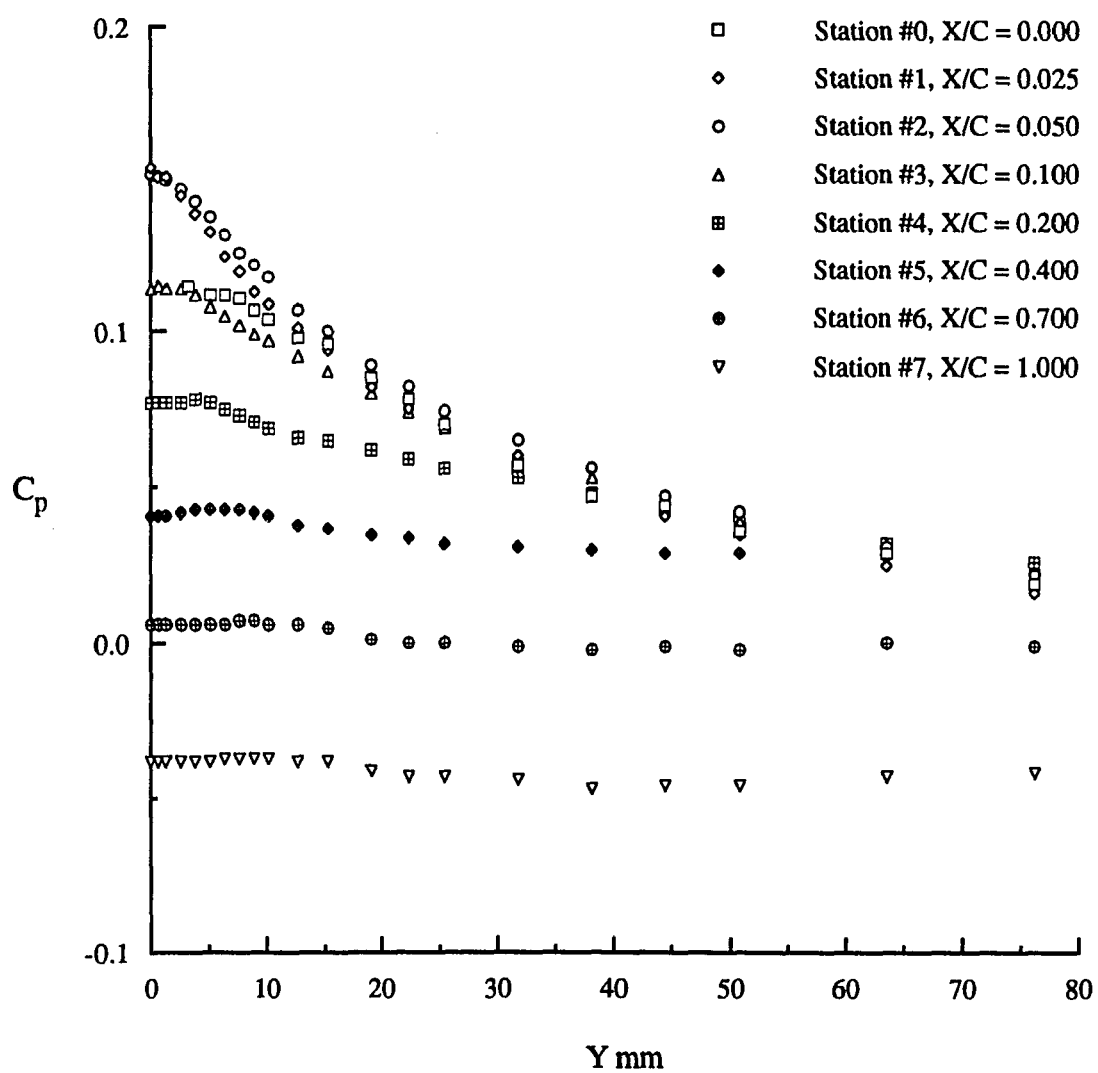


Figure (4.11) Surface Static Pressure Variation Downstream of the Trailing Edge.

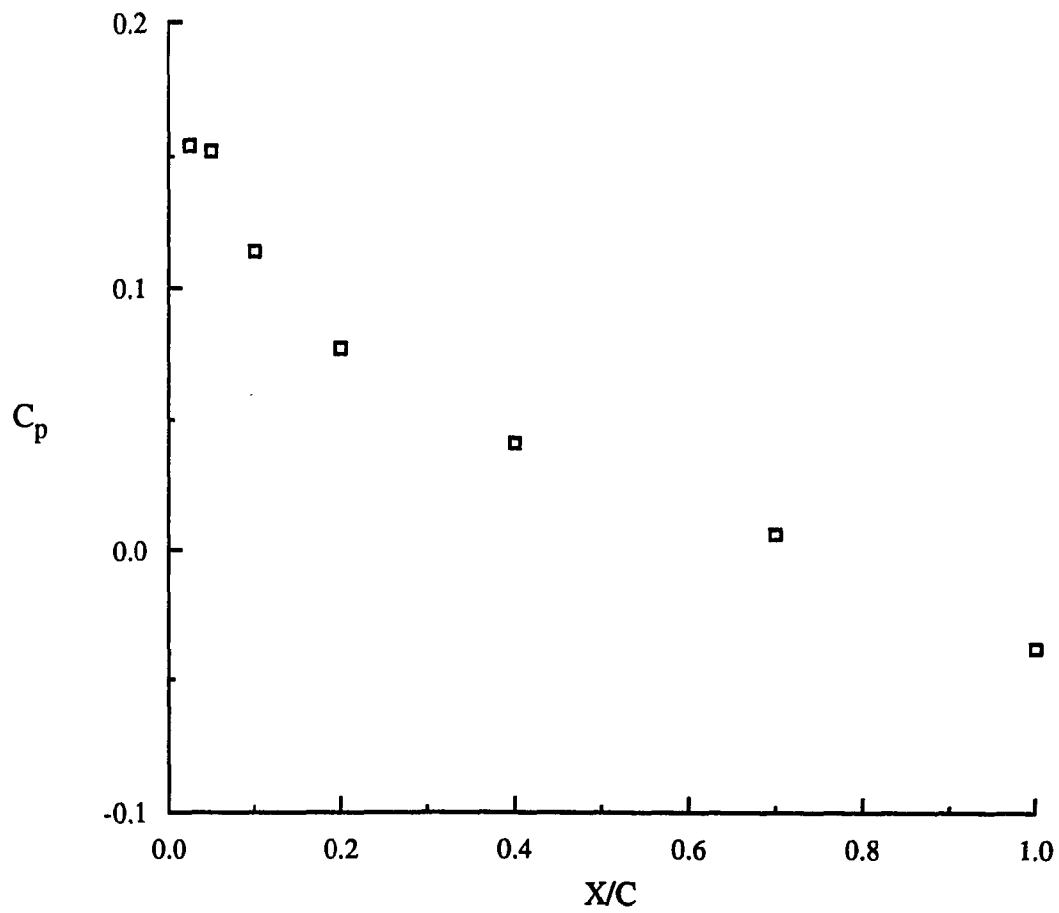


Figure (4. 12) Surface Static Pressure Distribution along the Centerplane (Y = 0 mm).

Surface static pressure distribution on the flat plate for all axial stations downstream the trailing edge is shown in Figures (4.10, 4.11 and 4.12). Again, the coordinate ($Y = 0$) represents the location of the centerplane (geometrical plane of symmetry). Figure (4.10) shows C_p as a function of x and Y , in an isometric form. Surface static pressure is maximum at the corner region formed by the trailing edge and the flat plate, as a result of the stagnation point there. It decreases in both the downstream (X -) and the transverse (Y -) directions away from the trailing edge. Figure (4.11) shows that the rate of decrease in C_p is higher at axial stations in the vicinity of the trailing edge and it reduces downstream. Flat plate surface static pressure distribution downstream the blade is similar to the static pressure distribution of the isolated airfoil wake reported by Munukutla [106], (Figure (96), p.198).

The presence of maximum surface static pressure, at the trailing edge corner formed by two flat plates with aligned leading edges has also been reported by Rahai [14] and Nakayama and Rahai [15]. It is consistent with the endwall corner region flow data of Hazarika and Raj [1] and Bhargava and Raj [12], on the same airfoil, at axial stations in the upstream vicinity of the trailing edge. This region of induced pressure gradient exists downstream of the blade trailing edge up to $X/C \cong 0.4$ (i.e., Station #5).

Surface static pressure variation along the centerplane is shown in Figure (4.12). The decrease in surface static pressure downstream of the blade trailing edge bears resemblance to the displacement thickness of airfoil wakes.

4.3 Preston Tube Results:

4.3.1. Check for Symmetry:

Surface total pressure measurements at Station #6 on both sides (i.e., $\pm Y$) of the geometrical plane of symmetry ($Y = 0$) using Preston tube No.2 are shown in Figure (4.13). Surface total pressure values were not symmetrical on the two sides, both in the magnitude and the location of its maximum and minimum values. This may have been caused by slight asymmetry in the construction of the airfoil (e.g., leading edge shape or location and size of the maximum thickness). In other words, any asymmetry in the construction of the airfoil, especially at the leading edge, will be magnified downstream. Another possible reason is the misalignment of the airfoil with the geometrical plane of symmetry, i.e., the airfoil is at an angle of incidence. Attempts were made to attain flow symmetry across the centerplane by placing two pitot-static probes, one on each side, at mirror image locations, and the blade alignment was adjusted to give equal pressure readings for both probes. However, absolute flow symmetry could not be achieved. Similar observations can also be made from the total pressure profiles of Barber [3], and McMahon et al. [26].

4.3.2 Angle of Sensitivity:

Figure (4.14) shows the local surface total pressure sensitivity to the angle of rotation about tube axis for the three sizes of Preston tube at a surface measuring point located at distance $X = 106.1$ mm, $Y = 106.1$ mm. (i.e., 2-D flow). Local surface total pressure was non-dimensionalized with respect to the free stream total pressure. It is clear that the three sizes of Preston tubes are insensitive to angles of rotation in the range of ± 6 degrees. Limiting streamlines obtained from surface flow

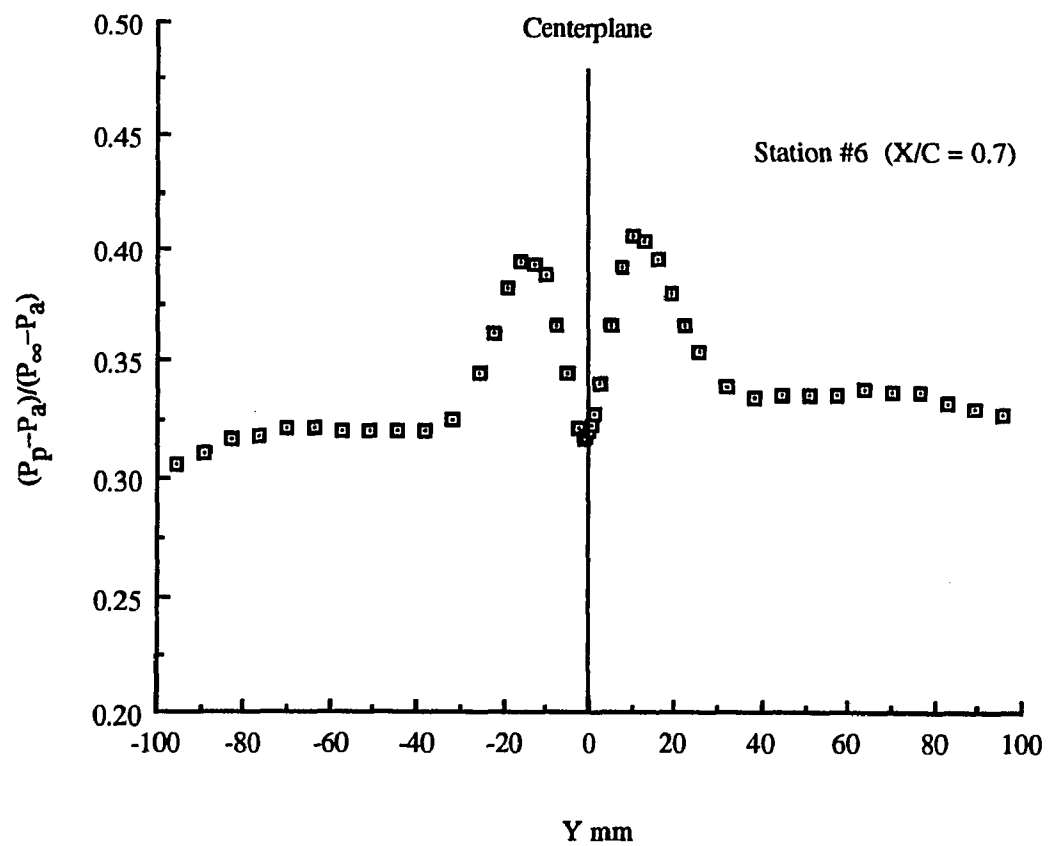


Figure (4.13) Flow Symmetry Across the Centerplane.

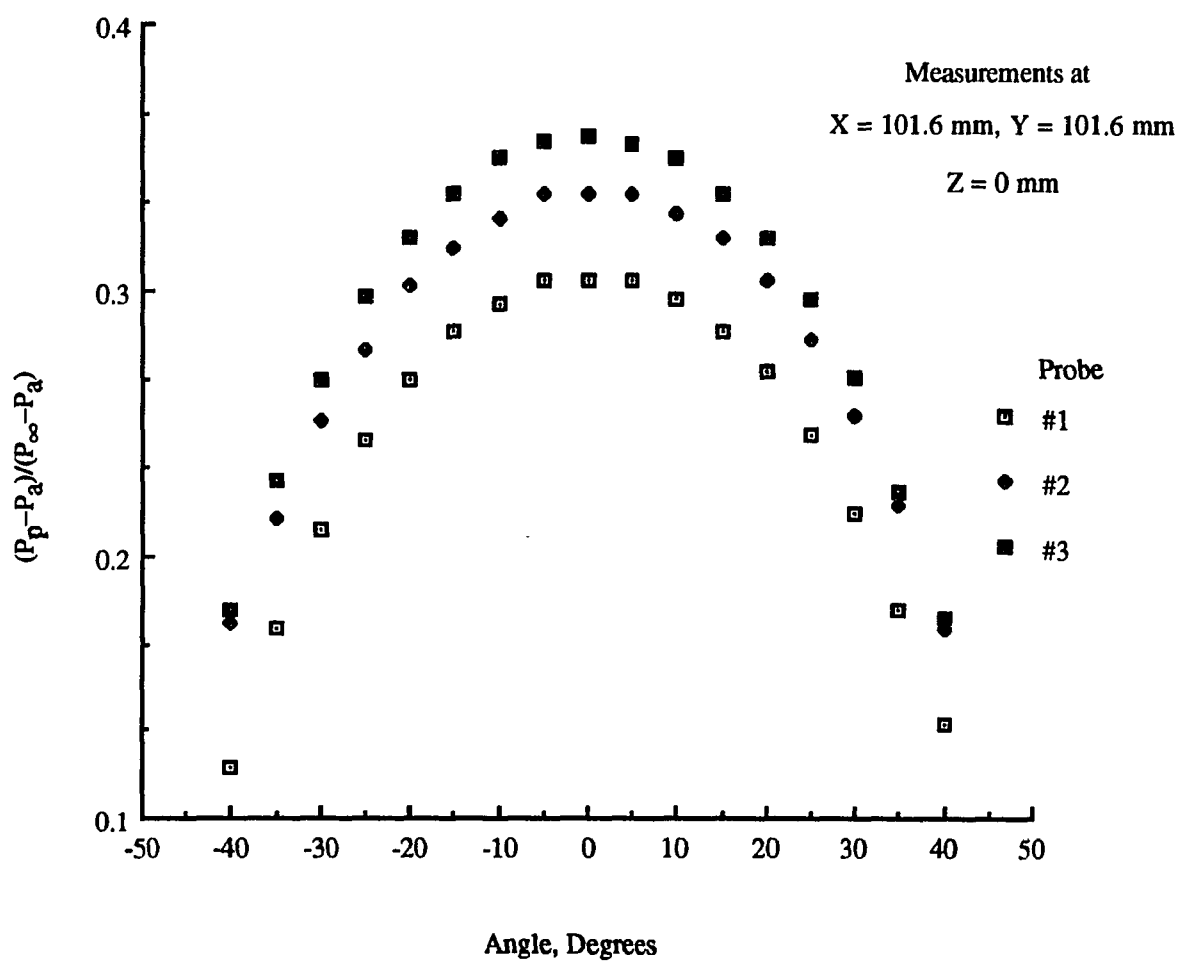


Figure (4.14) Angle Sensitivity of Preston Tubes .

visualization showed a maximum turning angle of about 16 degrees (with respect to the main flow direction). This suggests that the angles of direction of the local wall shear stress obtained from the Preston tube will not be appropriate. In three dimensional flow, the direction of the flow changes in planes normal and parallel to the surface and as the effective center of the Preston tube tip is at a finite distance from the surface, it will not record the true direction.

4.3.3 Preston Tube Size Effect:

Local wall shear stress distribution, measured by the three different sizes of Preston tubes, at two downstream axial stations, namely, Station #3 and Station #5 ($X/C = 0.1$, and $X/C = 0.4$) is shown in Figure (4.15). Values of the local wall shear stress for the three sizes compared very well and were within an error of less than 5%. It is clear that Patel's calibration equations adequately accounts (within the experimental error) for the tube size.

4.3.4 Local Wall Shear Stress:

Local wall shear stress downstream of the blade trailing edge is expected to be influenced by the flow behavior upstream of it. Therefore, it seems relevant to discuss the behavior of the wall shear stress in the regions upstream of the blade trailing edge. This includes two regions; the region upstream of the blade leading edge and the corner region. This is discussed in sub-section (4.3.4.1). It should be mentioned that these results were reported independently earlier by Bhargava and Raj [12] and also by Abdulla et al. [104]. Since the location of these results were upstream of the blade trailing edge, the streamwise distances (i.e., X/C) are in the negative, with $X/C = 0$ is at the blade trailing edge location.

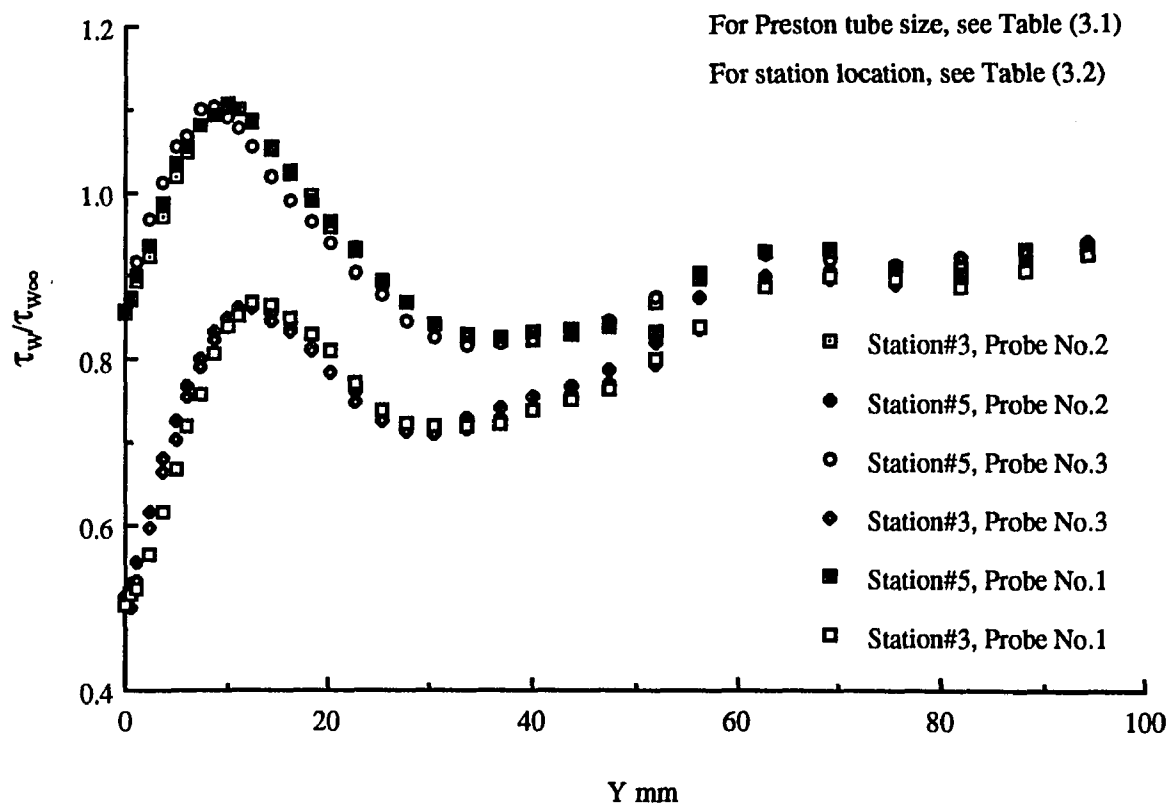


Figure (4.15) Comparison of Wall Shear Stress Values Measured by the Three Sizes of Preston Tubes.

All values of the local wall shear stress presented in sub-sections (4.3.4.1) and (4.3.4.2) were normalized by the far upstream value of the wall shear stress at $X/C = -2$ and $Y/C = 0$, i.e., one chord length. This presentation of the local wall shear stress instead of the usual skin friction coefficient is selected because the local free stream velocity varies at different axial locations. Moreover, in the corner region it is difficult to define the local free stream velocity because of the interacting nature of shear layers. The local wall shear stress at $X/C = -2$ and $Y/C = 0$ is approximately (within 3%) equal to the two-dimensional value at the same axial location.

4.3.4.1 Review of the Wall Shear Stress Behavior Upstream of the Blade

Trailing Edge: The non-dimensional wall shear stress upstream of the blade leading edge at 4 axial locations is shown in Figure (4.16). It is clearly observed that the wall shear stress decreases as the blade leading edge is approached. The maximum decrease ($\cong 40\%$) in the wall shear stress occurs near the stagnation streamline where the adverse pressure gradient is also maximum. The wall shear stress values are found to reach the two-dimensional value within a distance of $\cong 0.3C$ in Y-direction at different axial locations.

Wall shear stress variation on the flat plate surface, in the corner region, is shown in Figure (4.17-a). At different axial locations up to $X/C = -0.4$, the shear stress increases as the cornerline (defined as the line of intersection of the two surfaces forming the corner region) is approached and very close to the cornerline its value decreases. However, at two axial locations, $X/C = -0.2$ and $X/C = -0.01$, the wall shear stress value first decreases then increases as the cornerline is approached. In the vicinity of the cornerline, wall shear stress again decreases. It is quite evident that the maximum value of the wall shear stress and its distance from the cornerline changes at

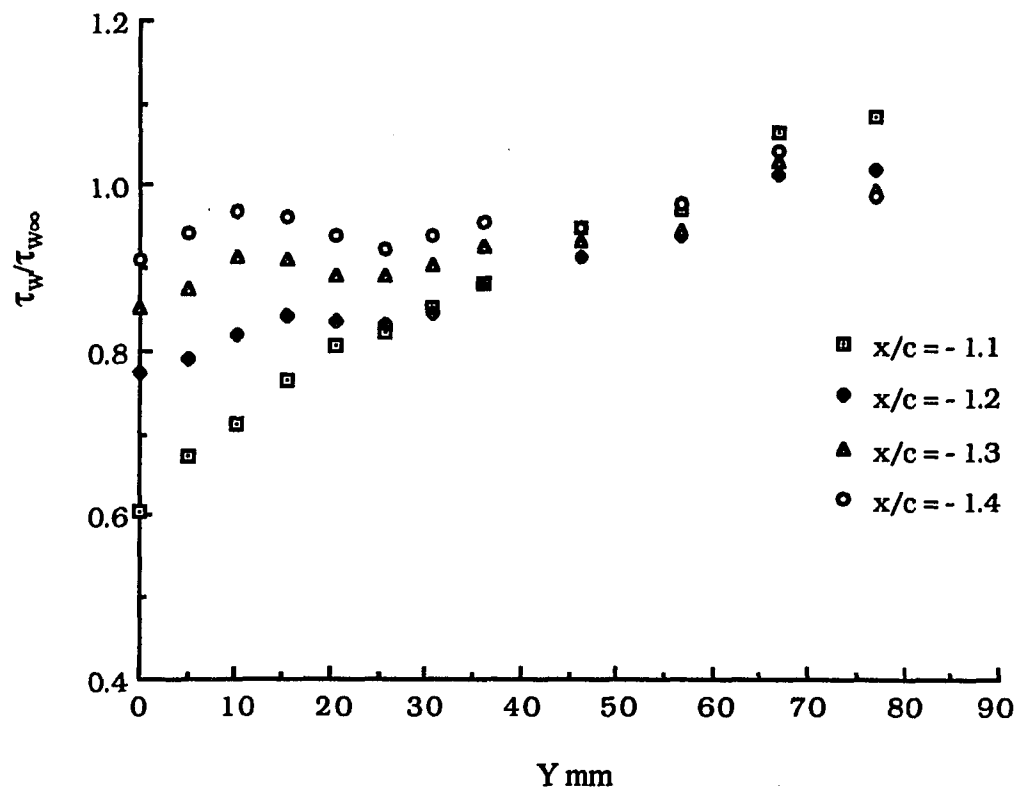


Figure (4.16) Wall Shear Stress Variation Upstream of the Blade Leading Edge [12].

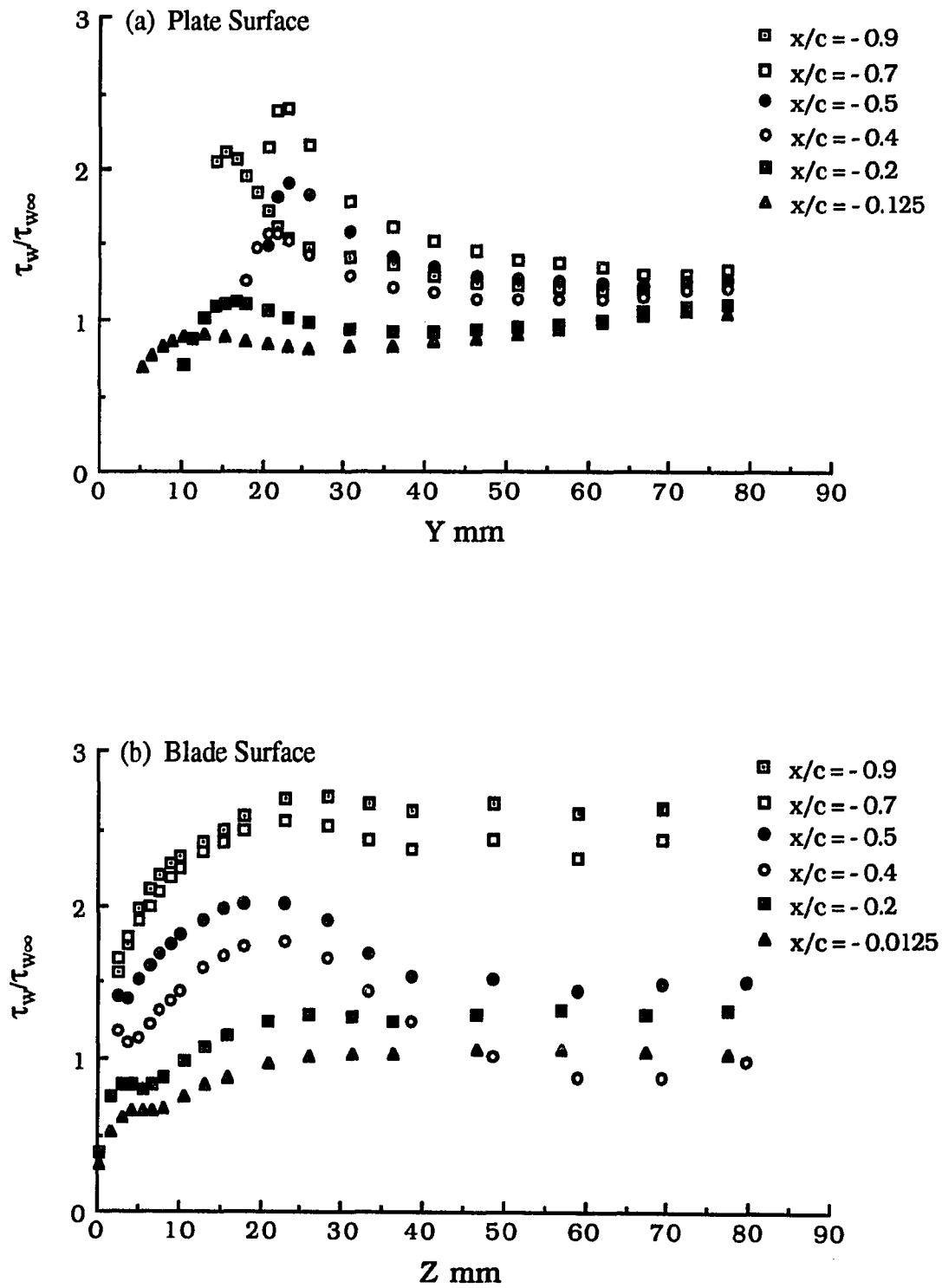


Figure (4.17) Wall Shear Stress Variation in the Blade Endwall Corner [12].

different axial locations. The maximum value of $\tau_w/\tau_{w\infty}$ occurs at $X/C = -0.7$ and it is almost 2.4 times the far upstream value of the wall shear stress.

On the blade surface (Figure (4.17-b)), at a given axial station, the local wall shear stress increases under the horseshoe vortex core and achieves a maximum value in the corner region and then alleviates at far distances and reaches an asymptotic value (the wall shear stress value at $Z/C \cong 0.3$ i.e., $Z \cong 76$ mm). Also at axial locations downstream of the maximum blade thickness location, the asymptotic values of the wall shear stress are less than these upstream. The maximum value of the wall shear stress and its location from the cornerline changes. However, the trend in variation of the location of maximum value at various axial locations is different in comparison to that observed on the flat plate surface in the corner region. It can be noted that at two axial locations ($X/C = -0.2$ and $X/C = -0.01$), Figure (4.17-b), a relatively small increase (or at least remains constant) in the values of wall shear stress as the cornerline is approached. This indicates the presence of vortical flow closer to the blade and can be due to the corner vortex suggested by Hazarika and Raj. [1].

In the corner region, at different axial locations, the maximum values of the wall shear stress are consistently higher on the blade surface compared to that on the flat plate surface of the test-model. This is caused by the thinner boundary layer on the blade surface, resulting in higher velocity gradients, compared to that on the flat plate at the same axial location. Asymptotic values (the wall shear stress value at $Y/C \cong 0.3$, i.e., $Y \cong 76$ mm) of the wall shear stress on the flat plate surface in the corner region are higher by approximately 25% than the far upstream value. However, on the blade surface preceding the maximum thickness of the blade, asymptotic values are still high but are lower than its values upstream of the blade maximum thickness. This is caused by the presence of blade curvature. The flow is expected to accelerate up to the location of maximum thickness giving rise to higher velocity gradients. In

the region downstream of the blade maximum thickness, the flow is expected to decelerate resulting in lower velocity gradients. On the flat plate surface the location of maximum value of the wall shear stress moves from 4 mm near the blade leading edge location to 11 mm at an axial location just upstream the trailing edge (i.e., $X/C = -0.01$). However, on the blade surface the location of the maximum value of the wall shear stress first moves closer to the cornerline and then moves away from it to a distance of $\cong 45$ mm at the axial location of $X/C = -0.01$).

From the aforementioned observations, it can be summarized that streamwise curvature thus introduced, will give rise to stretching or compressing of the horseshoe vortex depending on the type of curvature resulting in intensification or attenuation of vorticity. On the flat plate surface of the test-model, the maximum value of wall shear stress at different axial stations first increased up to the maximum blade thickness location. This increase can possibly be associated with the horseshoe vortex stretching because of the presence of acceleration in the flow. However, downstream of the blade maximum thickness location the flow decelerates, and the horseshoe vortex gets attenuated, and as a result the maximum value of the wall shear stress is observed to decrease. The variation in location of the maximum value of the wall shear stress on both the surfaces in the corner region suggests that the horseshoe vortex gets distorted in the downstream direction as it moves along the corner.

4.3.4.2 Local Wall Shear Stress Results Downstream of the Blade Trailing Edge: The variation of the normalized local wall shear stress on the flat plate surface downstream of the corner region is shown in Figure (4.18). A common trend is obvious at all stations; departing the centerplane ($Y/C = 0$), with local minimum wall shear stress (will be labeled τ_{wmin1}), wall shear stress rises sharply from τ_{wmin1} to a local maximum value of τ_{wmax1} . This is a region of downwashed wake, i.e., the horseshoe

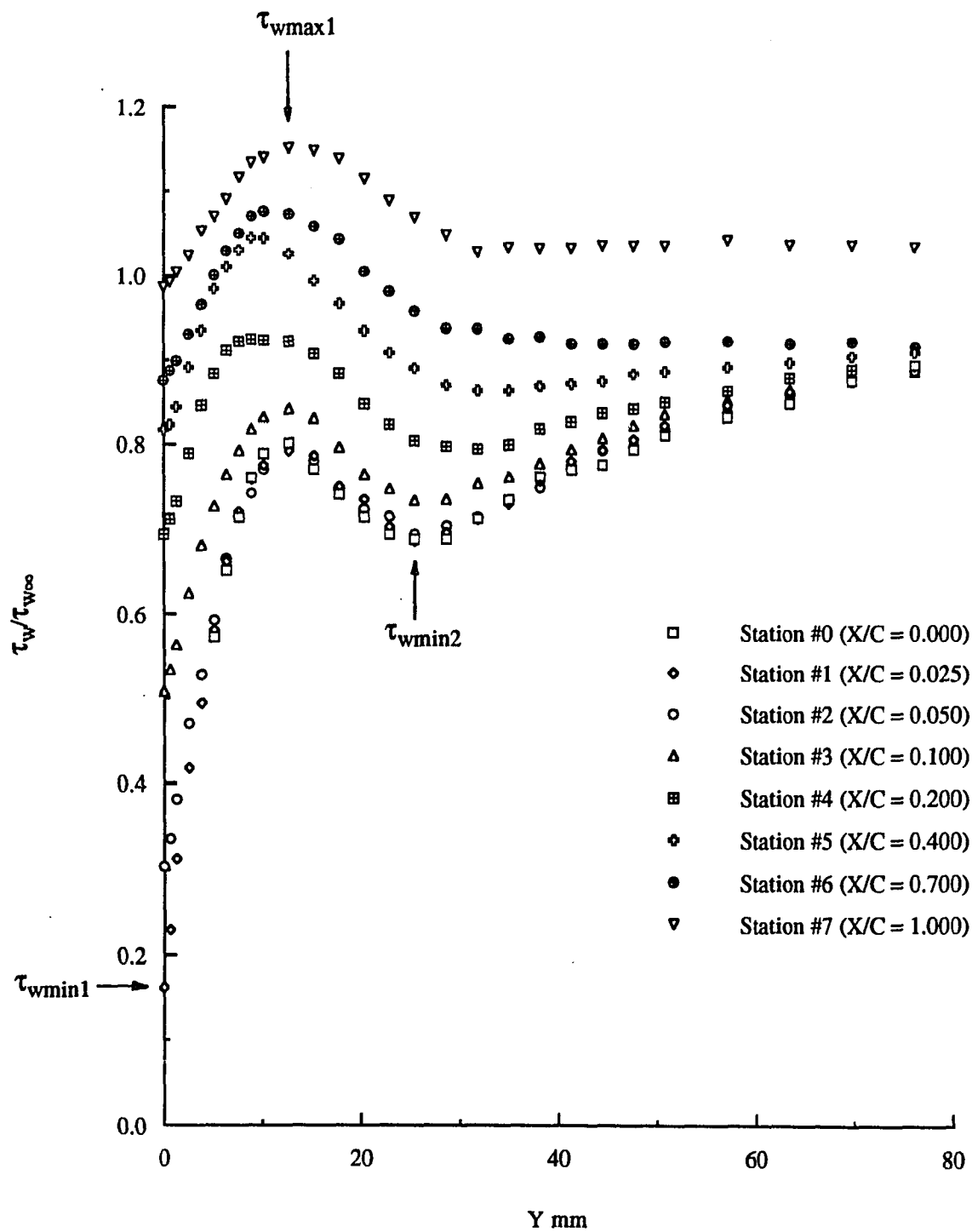


Figure (4.18) Wall Shear Stress Variation Downstream of the Blade Trailing Edge.

vortex is transporting the low momentum blade wake onto the flat plate surface. This rise is then followed by a sharp fall to another local minimum value of τ_{wmin2} (flow upwash) and finally, it rises to its local asymptotic value (wall shear stress far from the centerplane; i.e., two-dimensional value). It is clear from these trends that three dimensionality in the flow will persist far downstream of the blade trailing edge ($X/C = 1$, or one chord length). Proceeding downstream of the blade trailing edge, the rate of rise from τ_{wmin1} to τ_{wmax1} decreases while the rate of decrease from τ_{wmax1} to τ_{wmin2} essentially remains the same. The rate of increase from τ_{wmin2} to $\tau_{w\infty}$ decreases, as the flow proceeds downstream, and the increase ceases at an axial location of approximately $0.6C$ ($X/C = 0.6$). The monotonic increase in the values of τ_{wmin1} and τ_{wmax1} in the downstream direction is an indication of a recovery in the flow. Values of wall shear stress at the centerplane (τ_{wmin1}) are shown in Figure (4.19). It increases from a very small value of $(0.14 \tau_{w\infty})$ at Station #1 ($X/C = 0.025$) to the local asymptotic value at Station #7 ($X/C = 1$). This is an indication of the wake recovery, i.e., wake deficit is reduced.

4.4 Static and Total Pressure Variations Downstream of the Blade Trailing Edge:

Results of static and total pressure measurements at the eight axial stations downstream of the blade trailing edge are presented in this section.

4.4.1 Static Pressure Variation:

Static pressure variation is presented by the static pressure coefficient, C_p , which was defined in Section (4.2) and is shown in Figure (4.20) through Figure (4.43). Figure (4.20) to Figure (4.27) show C_p as a function of Y and Z in an isometric form. In Figure (4.28) to Figure (4.35), the variation of C_p parallel to the

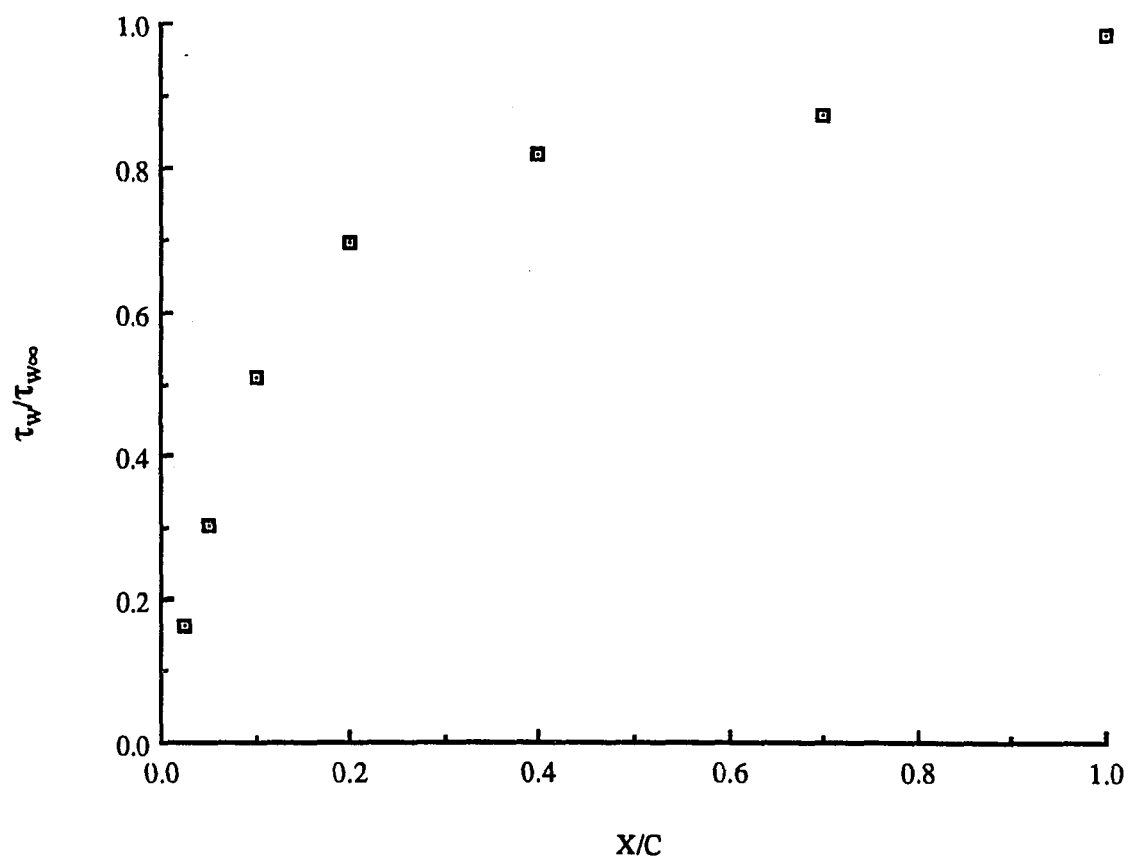


Figure (4.19) Wall Shear Stress Variation along the Centerplane ($Y = 0$ mm).

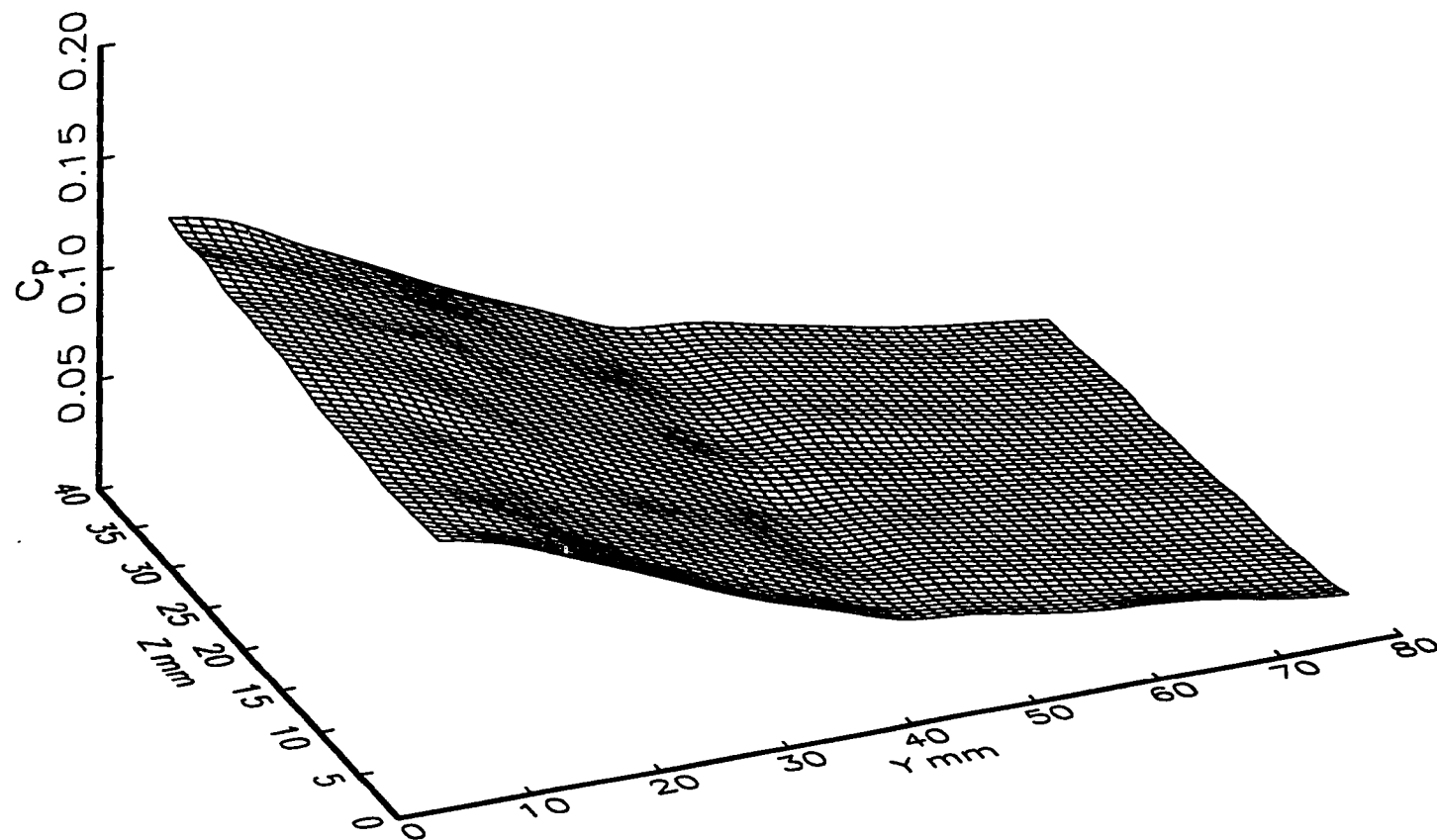


Figure (4.20) Static Pressure Variation at Station #0 ($X = 0.00$ mm, $X/C = 0.00$).

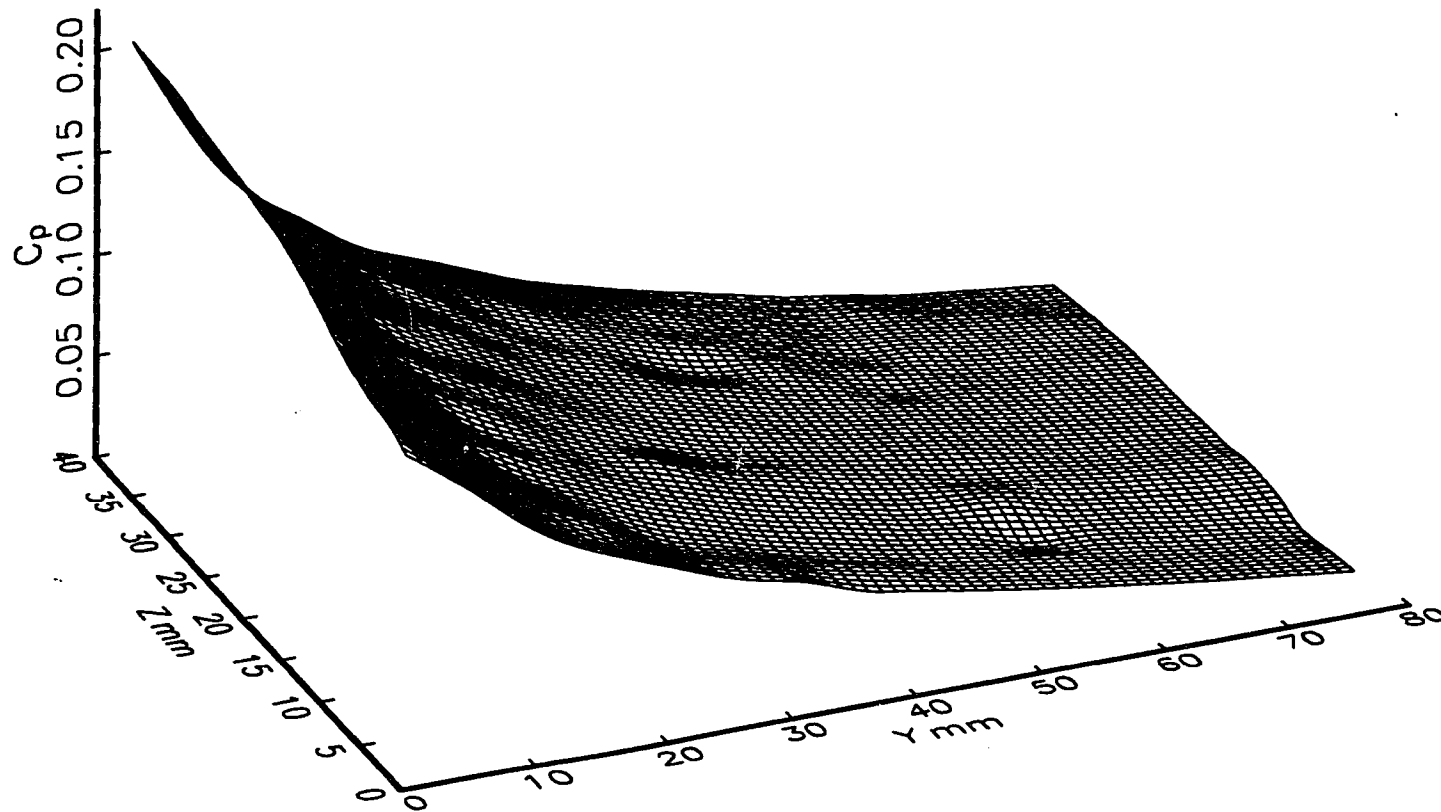


Figure (4.21) Static Pressure Variation at Station #1 ($X = 6.35$ mm, $X/C = 0.025$).

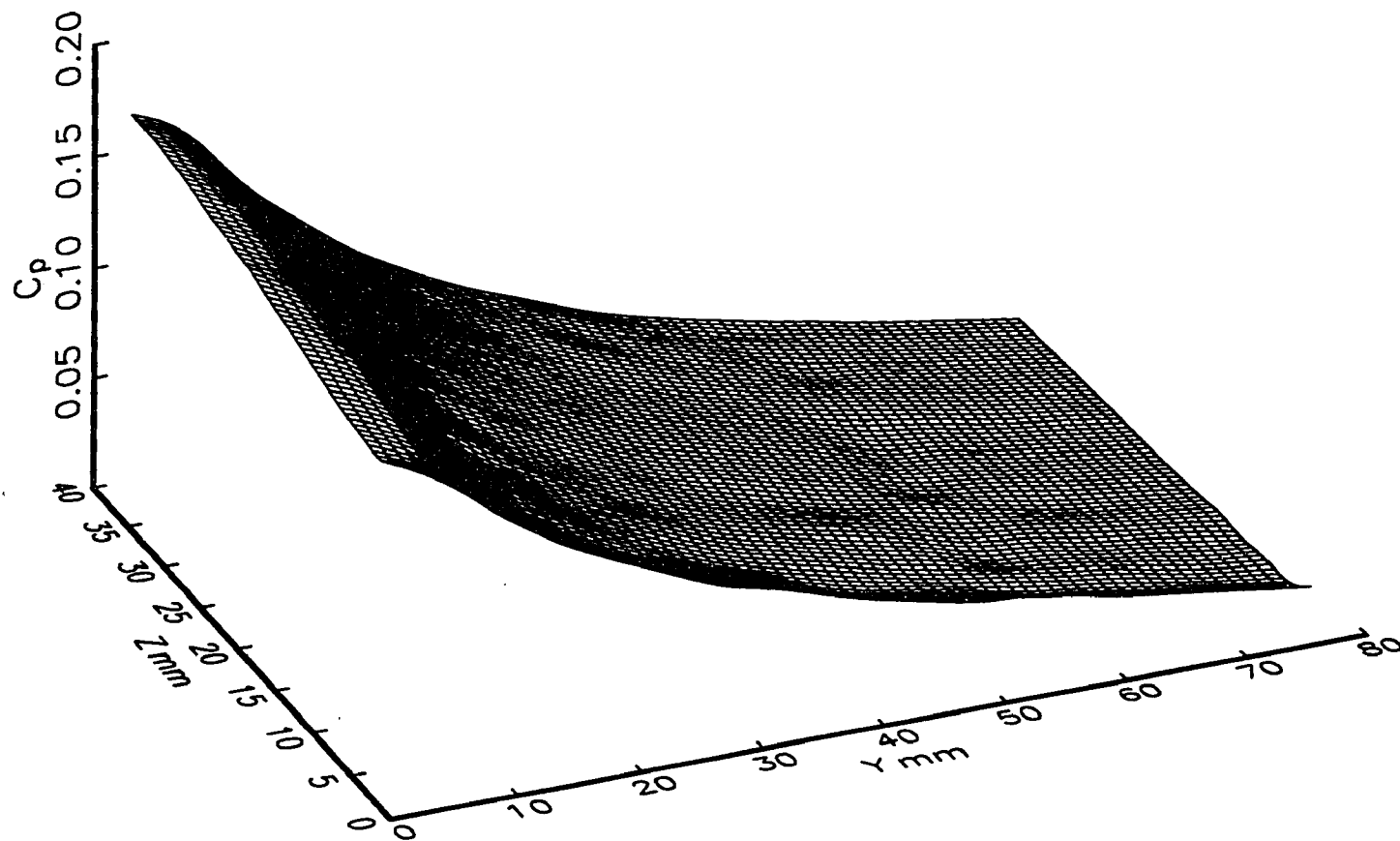


Figure (4.22) Static Pressure Variation at Station #2 ($X = 12.70$ mm, $X/C = 0.05$).

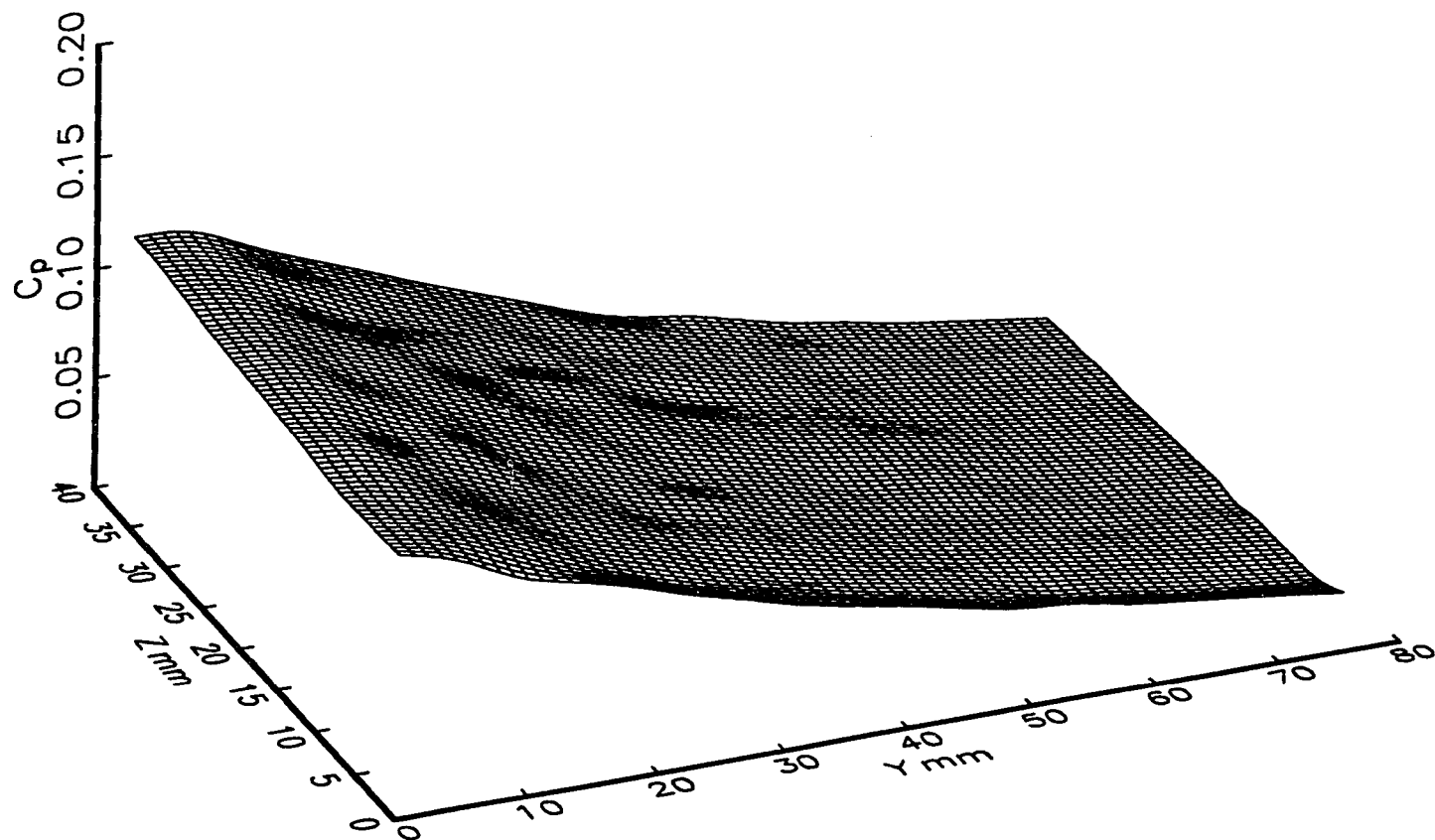


Figure (4.23) Static Pressure Variation at Station #3 ($X = 25.40$ mm, $X/C = 0.10$).

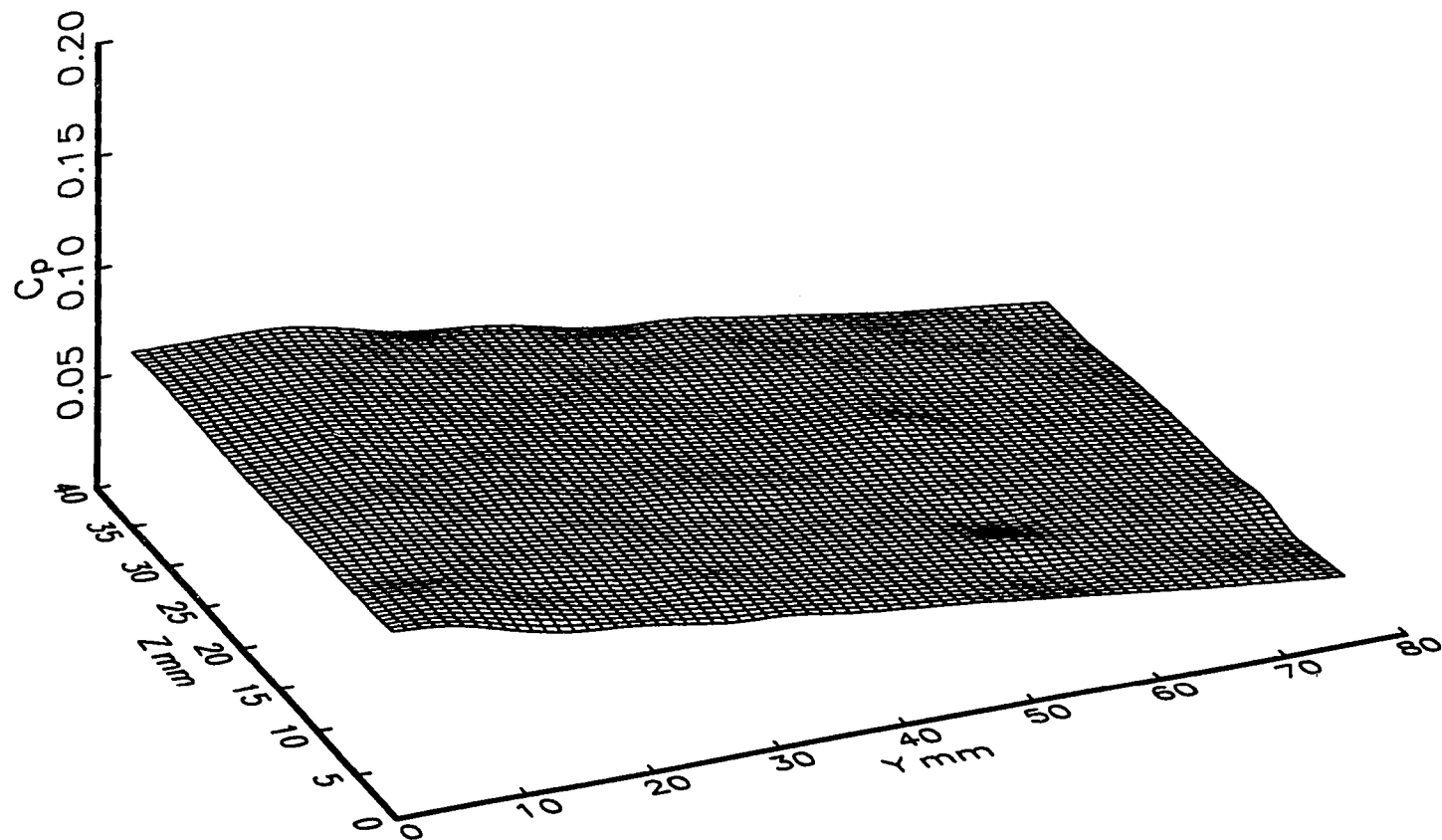


Figure (4.24) Static Pressure Variation at Station #4 ($X = 50.80$ mm, $X/C = 0.20$).

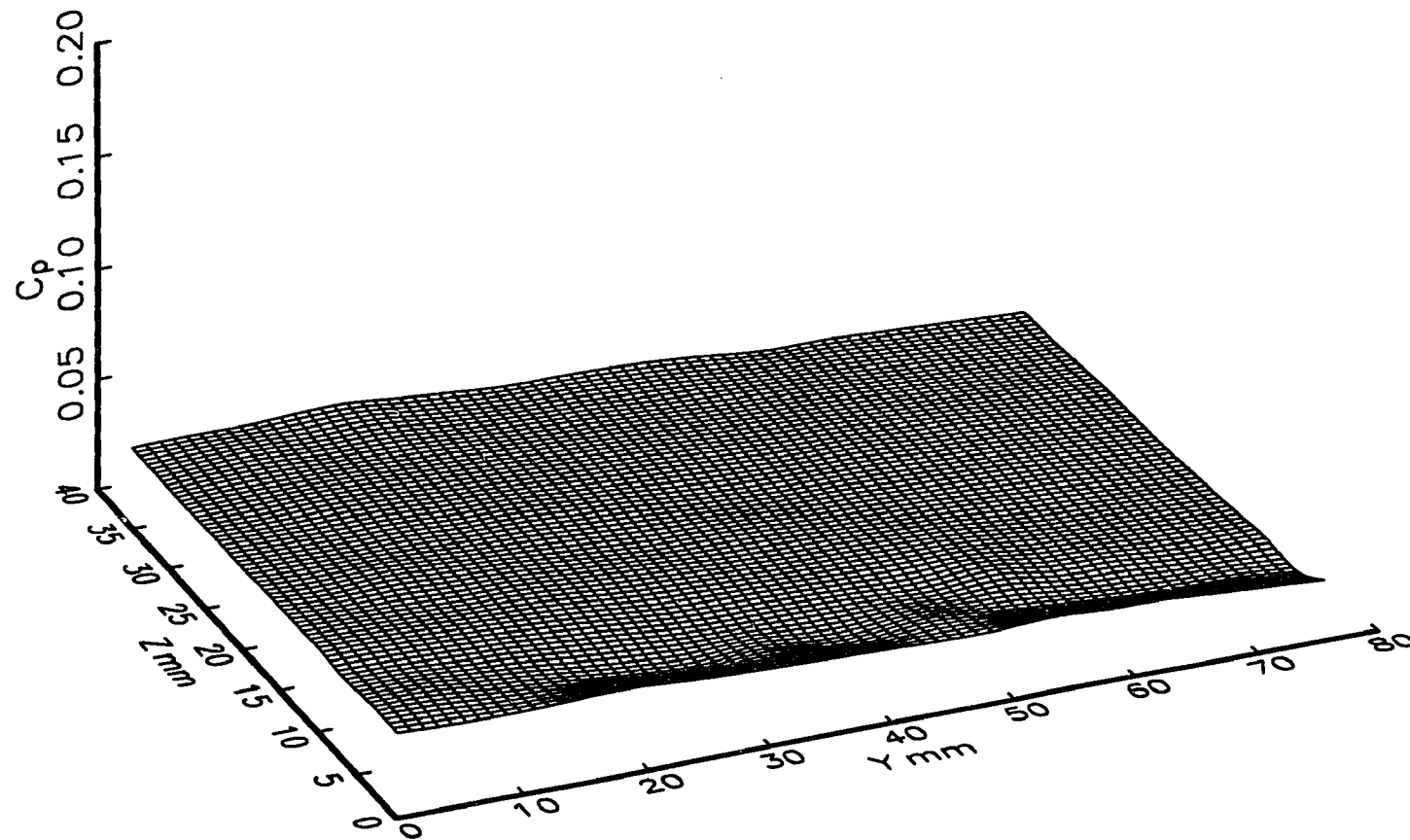


Figure (4.25) Static Pressure Variation at Station #5 ($X = 101.60$ mm, $X/C = 0.40$).

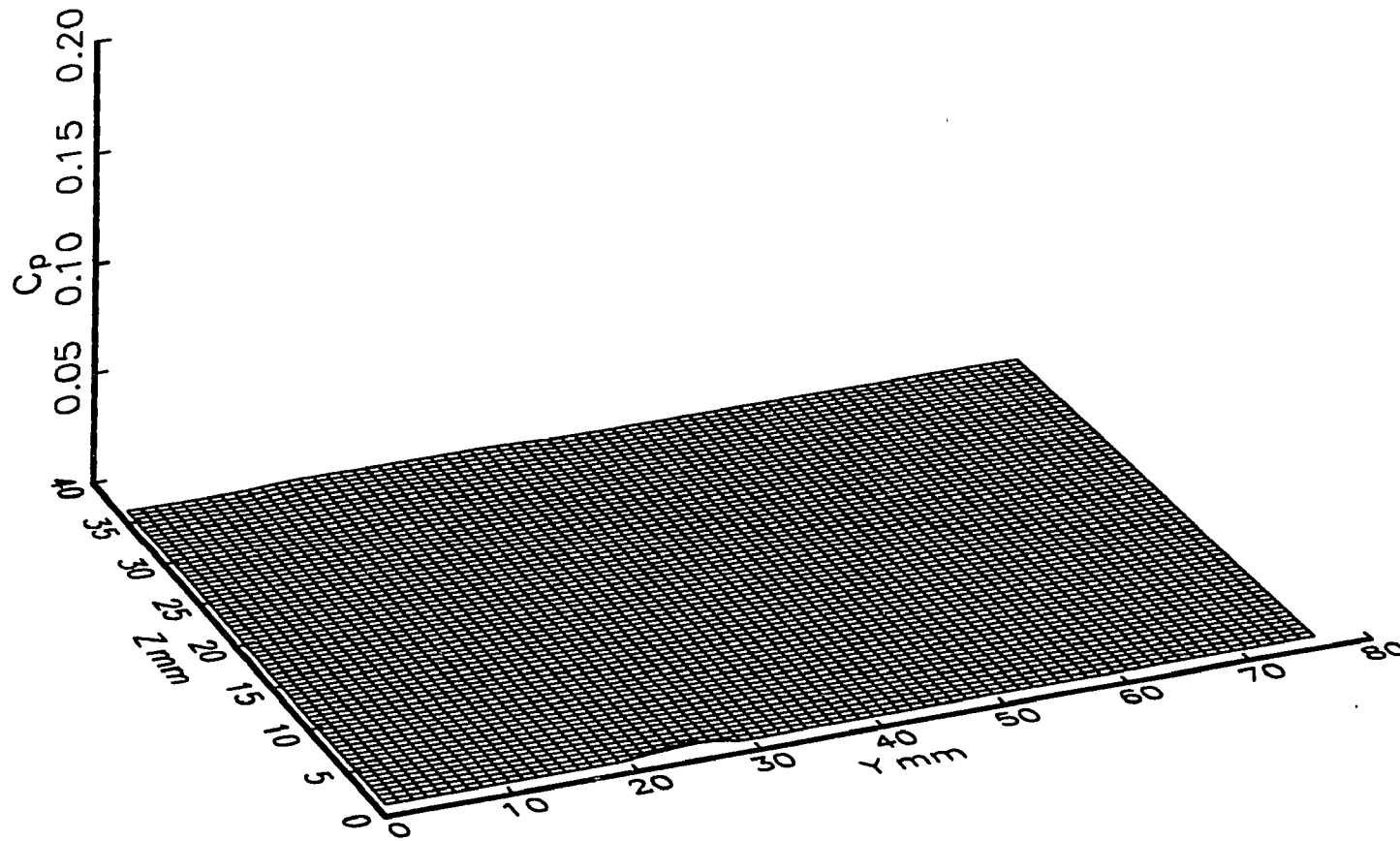


Figure (4.26) Static Pressure Variation at Station #6 ($X = 177.80$ mm, $X/C = 0.70$).

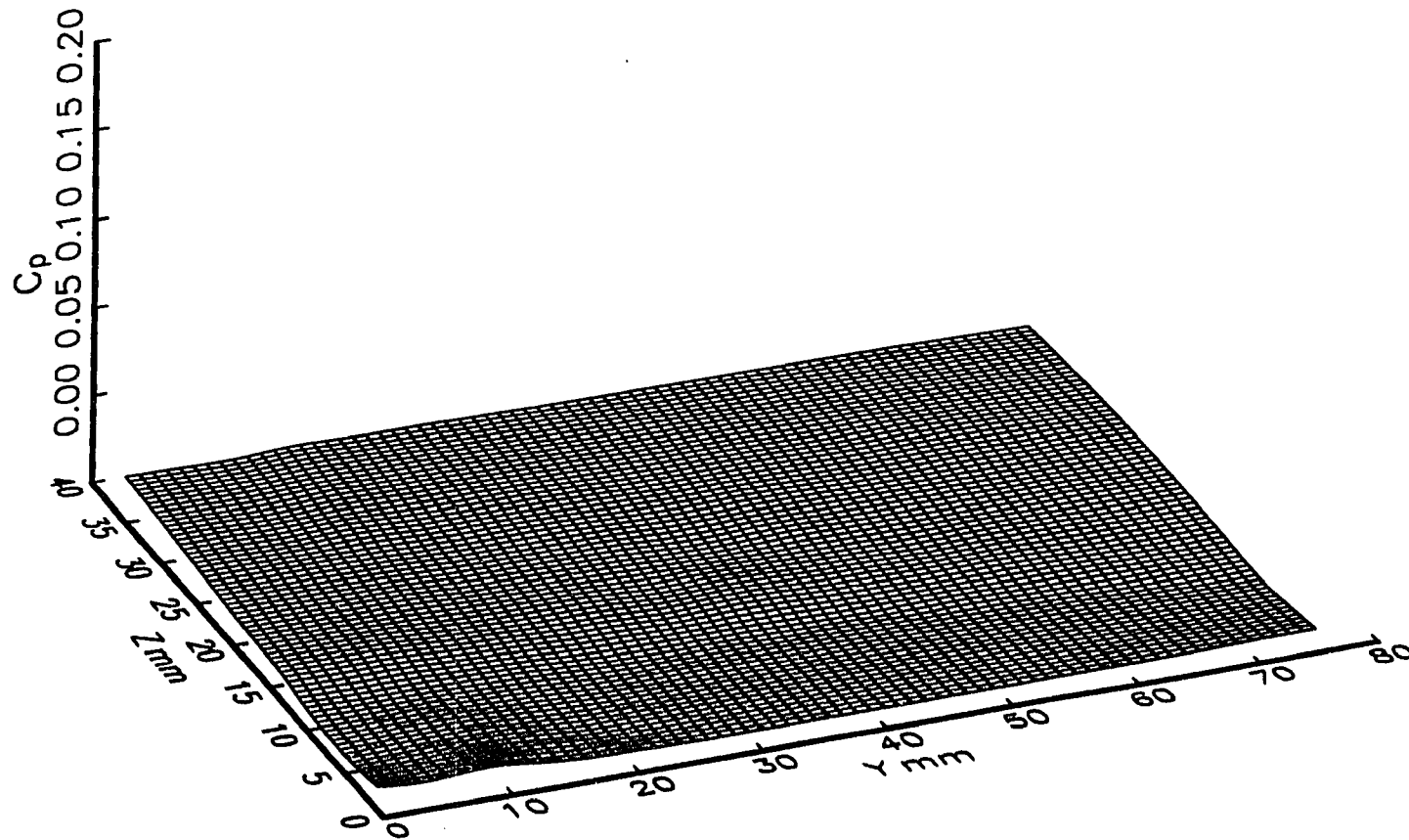
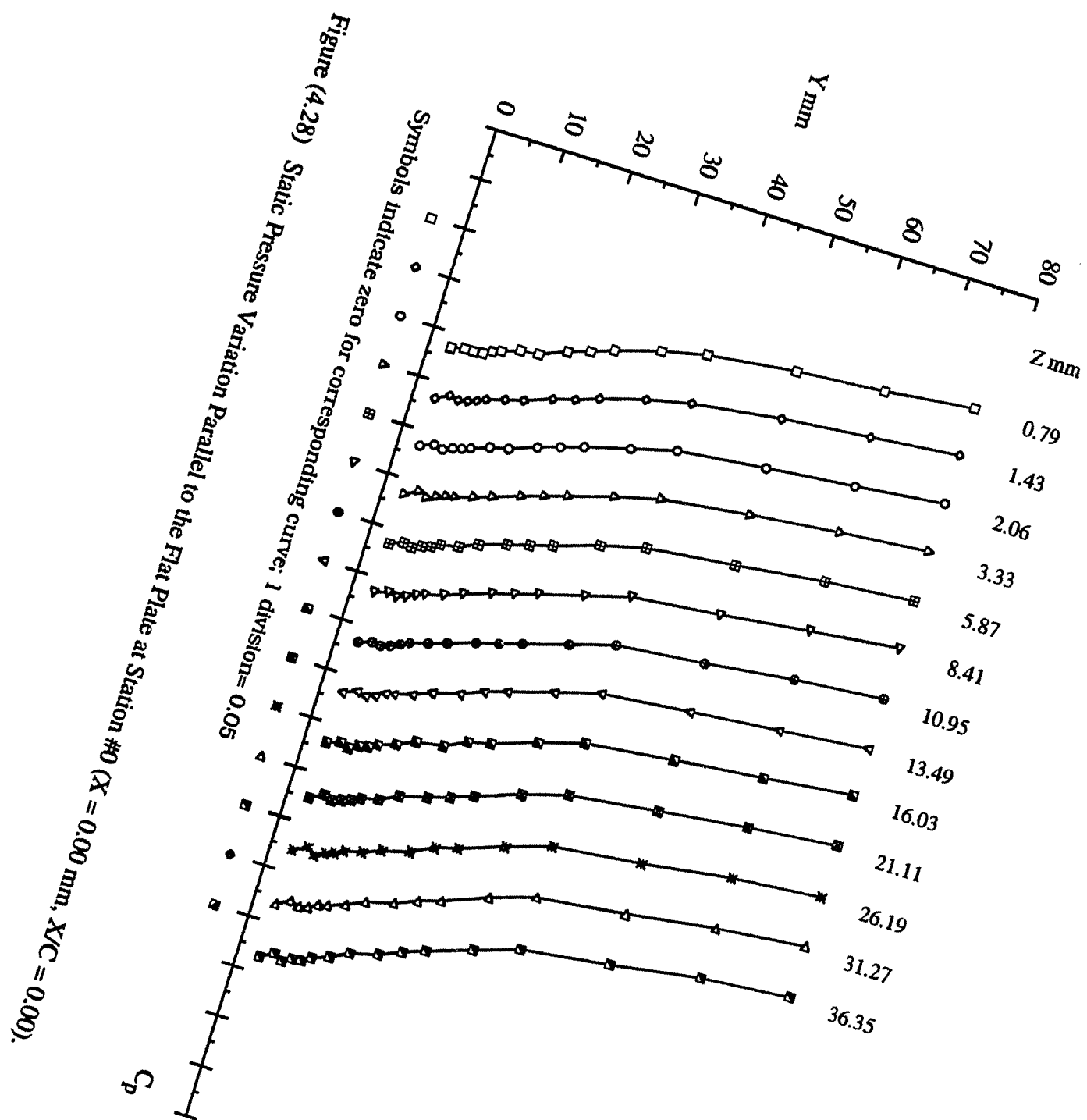


Figure (4.27) Static Pressure Variation at Station #7 ($X = 254.00$ mm, $X/C = 1.00$).



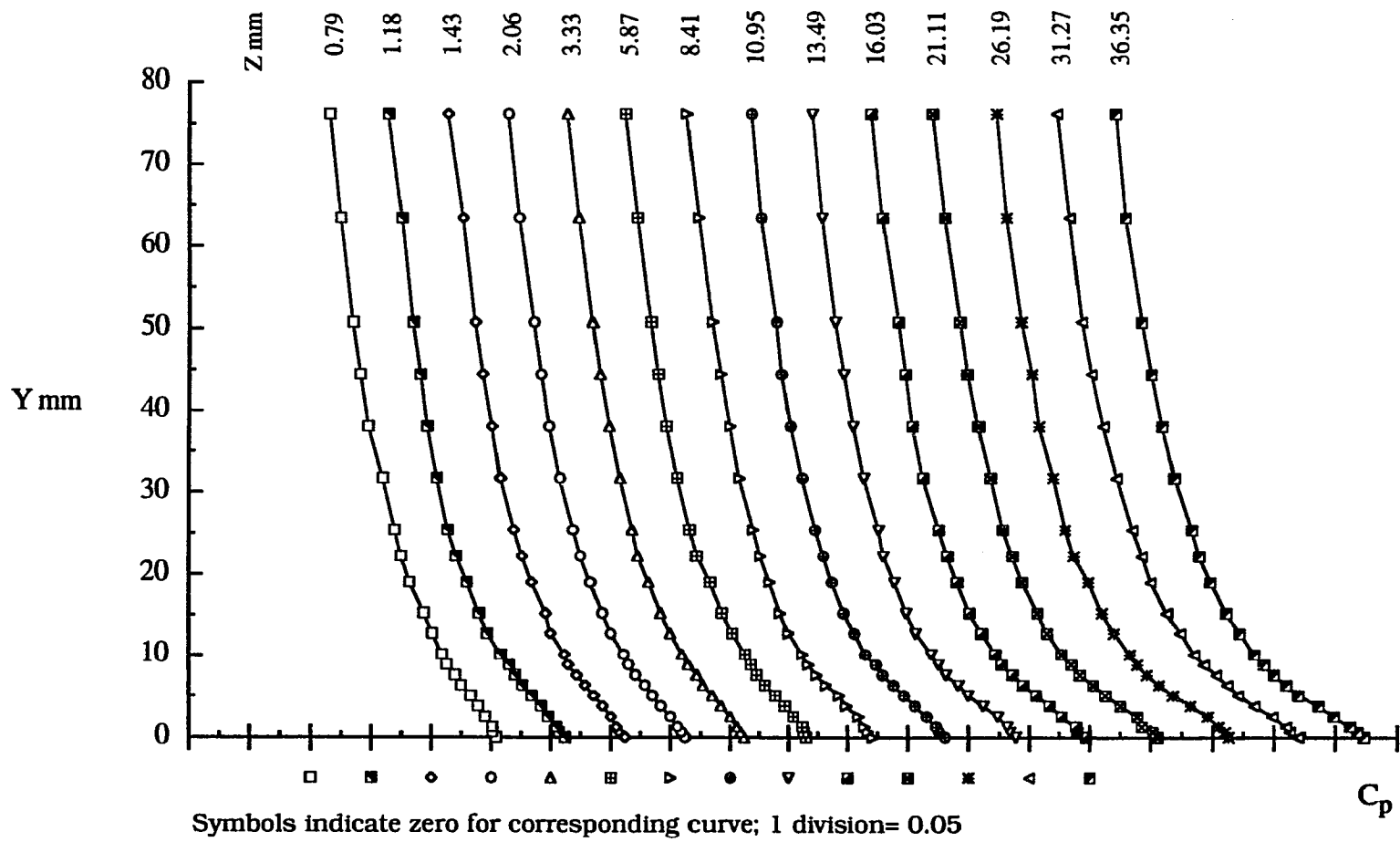


Figure (4.29) Static Pressure Variation Parallel to the Flat Plate at Station #1 (X = 6.35 mm, X/C = 0.025).

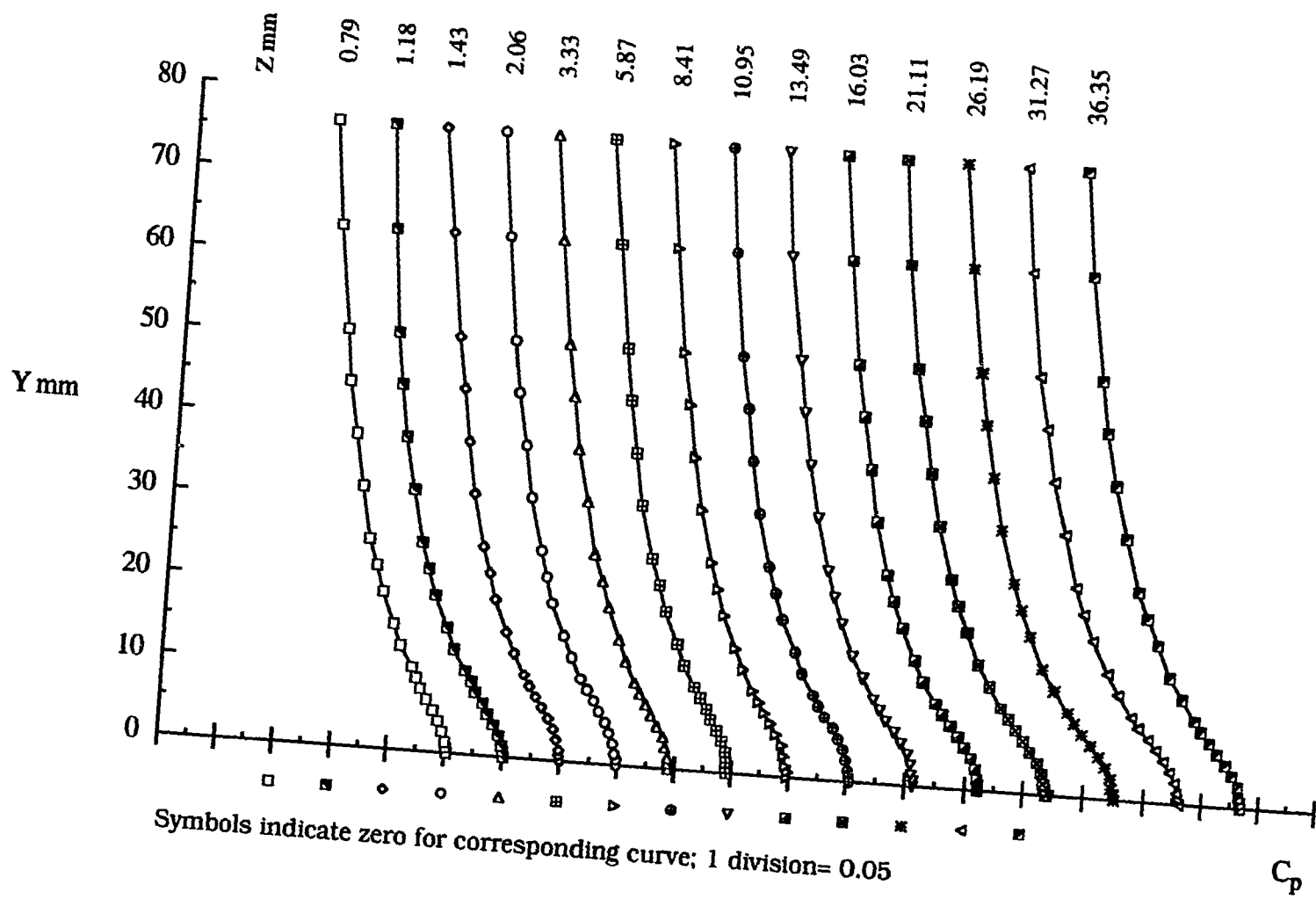


Figure (4.30) Static Pressure Variation Parallel to the Flat Plate at Station #2 ($X = 12.70$ mm, $X/C = 0.05$).

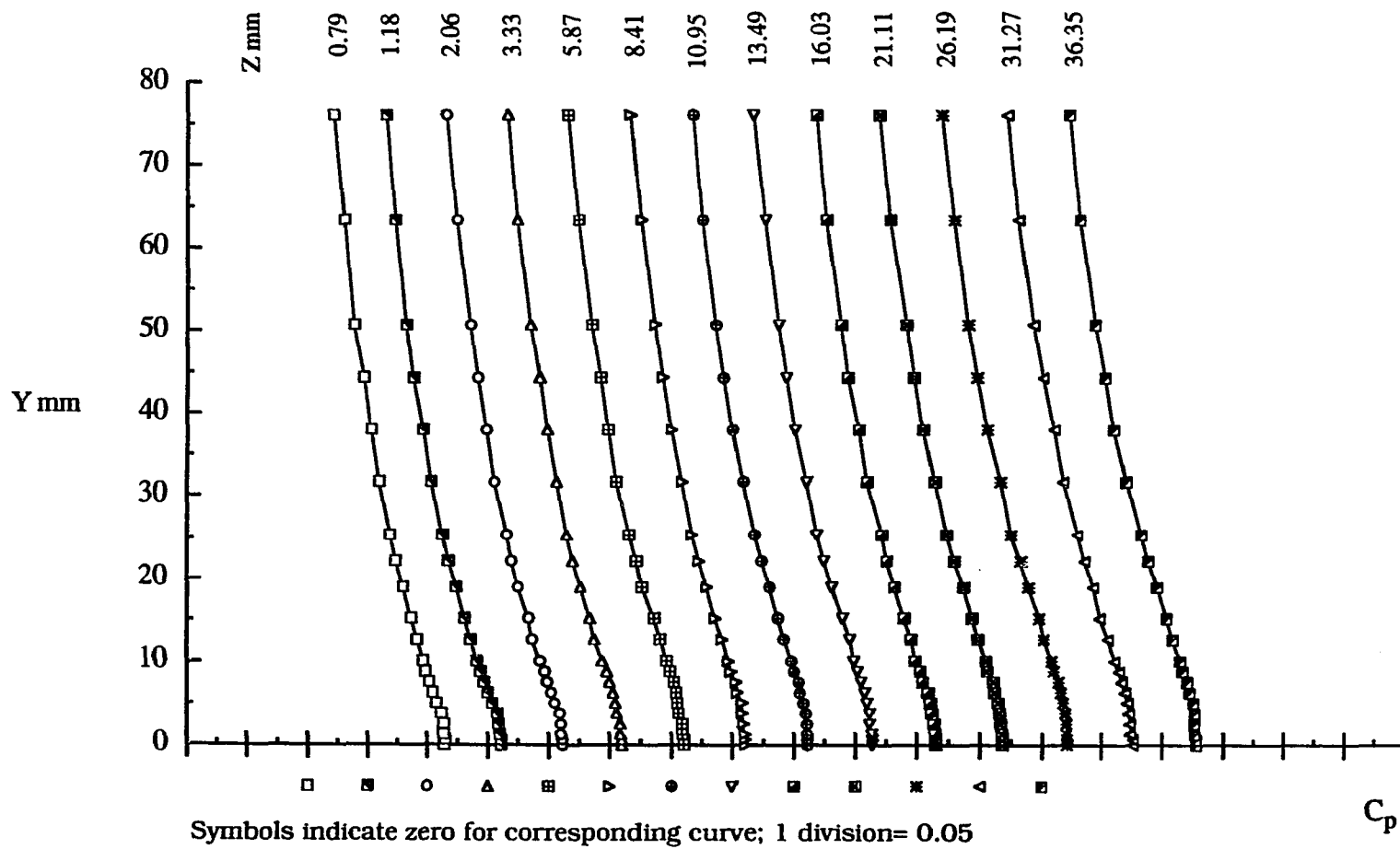


Figure (4.31) Static Pressure Variation Parallel to the Flat Plate at Station #3 ($X = 25.40$ mm, $X/C = 0.10$).

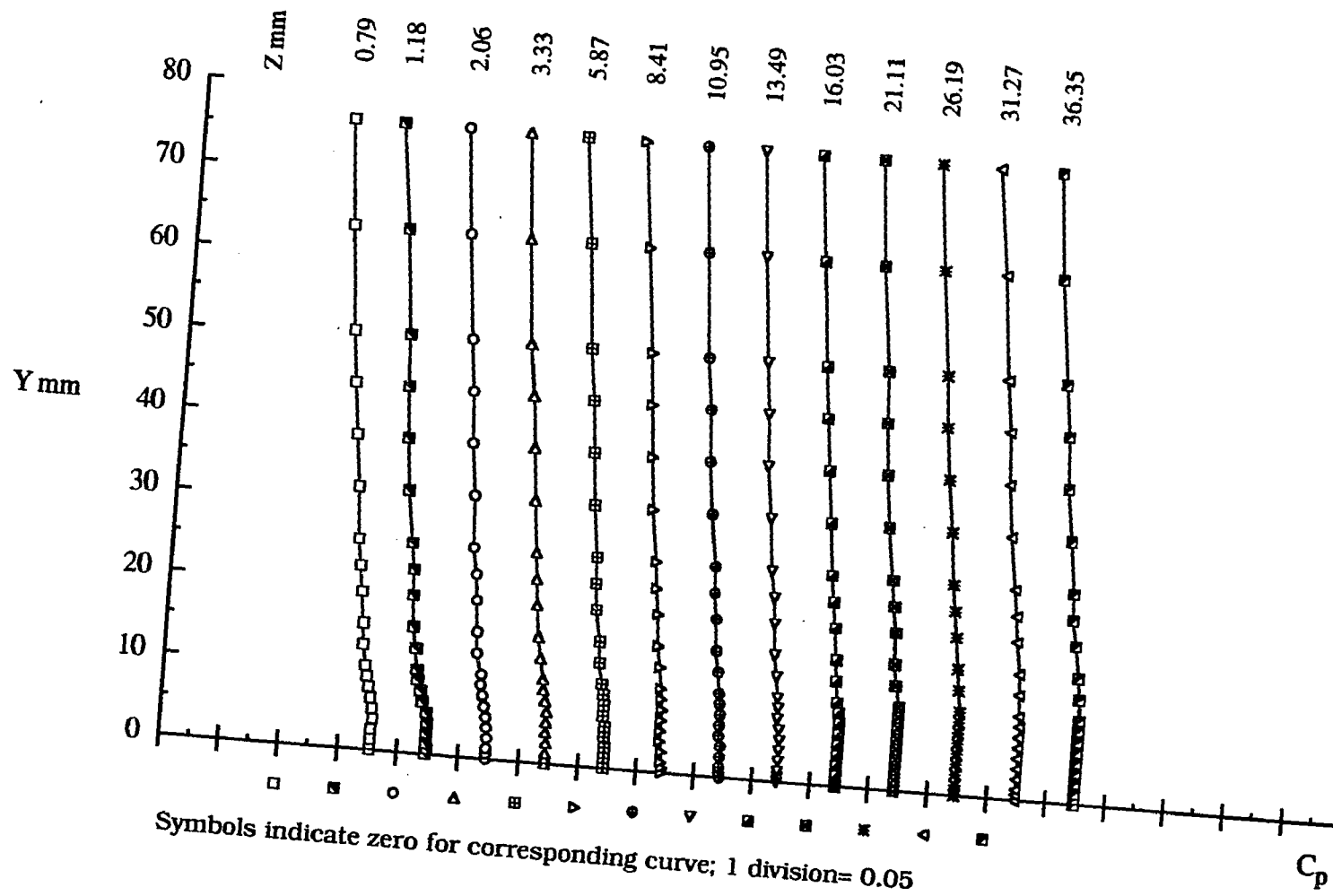


Figure (4.32) Static Pressure Variation Parallel to the Flat Plate at Station #4 ($X = 50.80$ mm, $X/C = 0.20$).

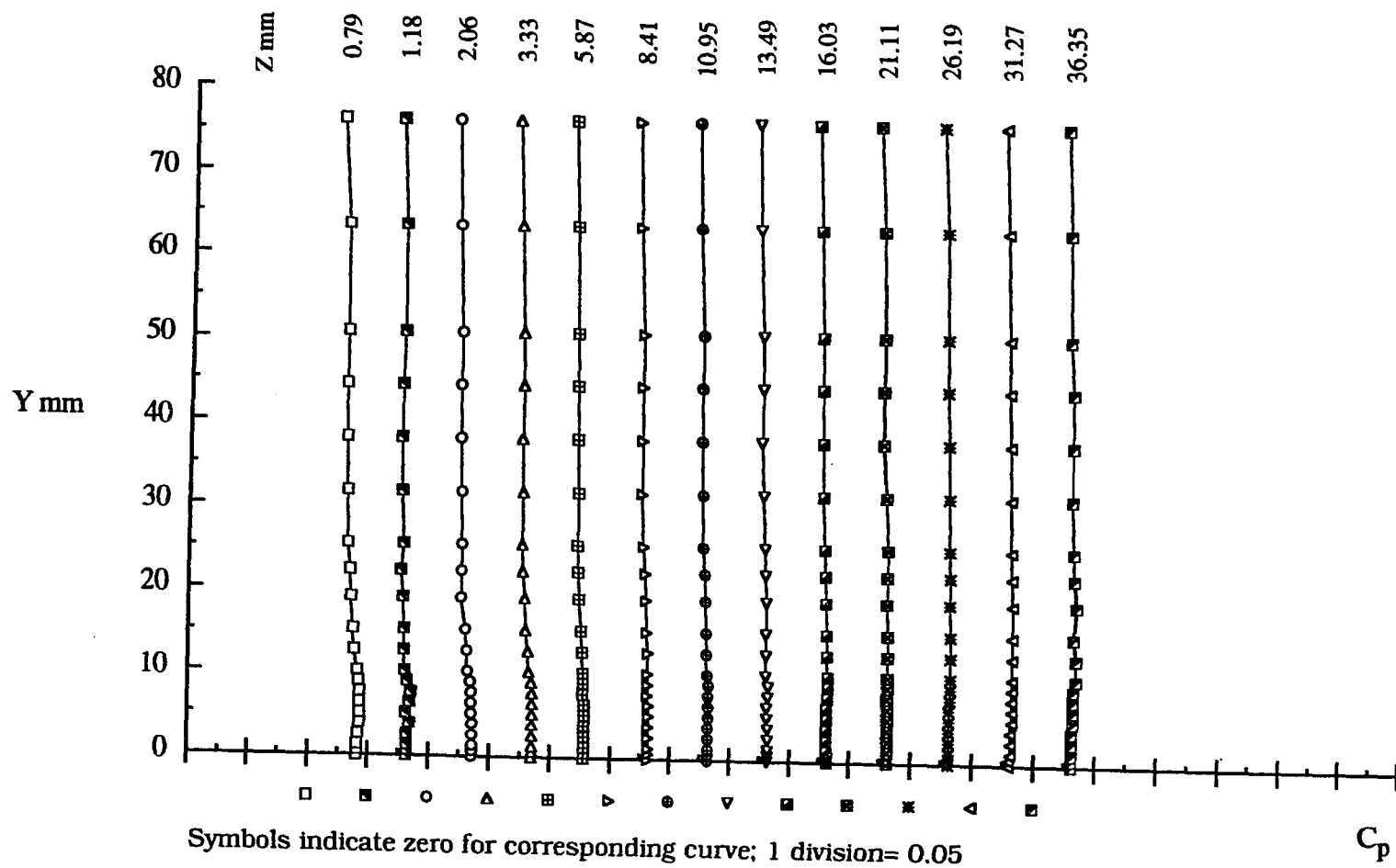


Figure (4.33) Static Pressure Variation Parallel to the Flat Plate at Station #5 ($X = 101.60$ mm, $X/C = 0.40$).

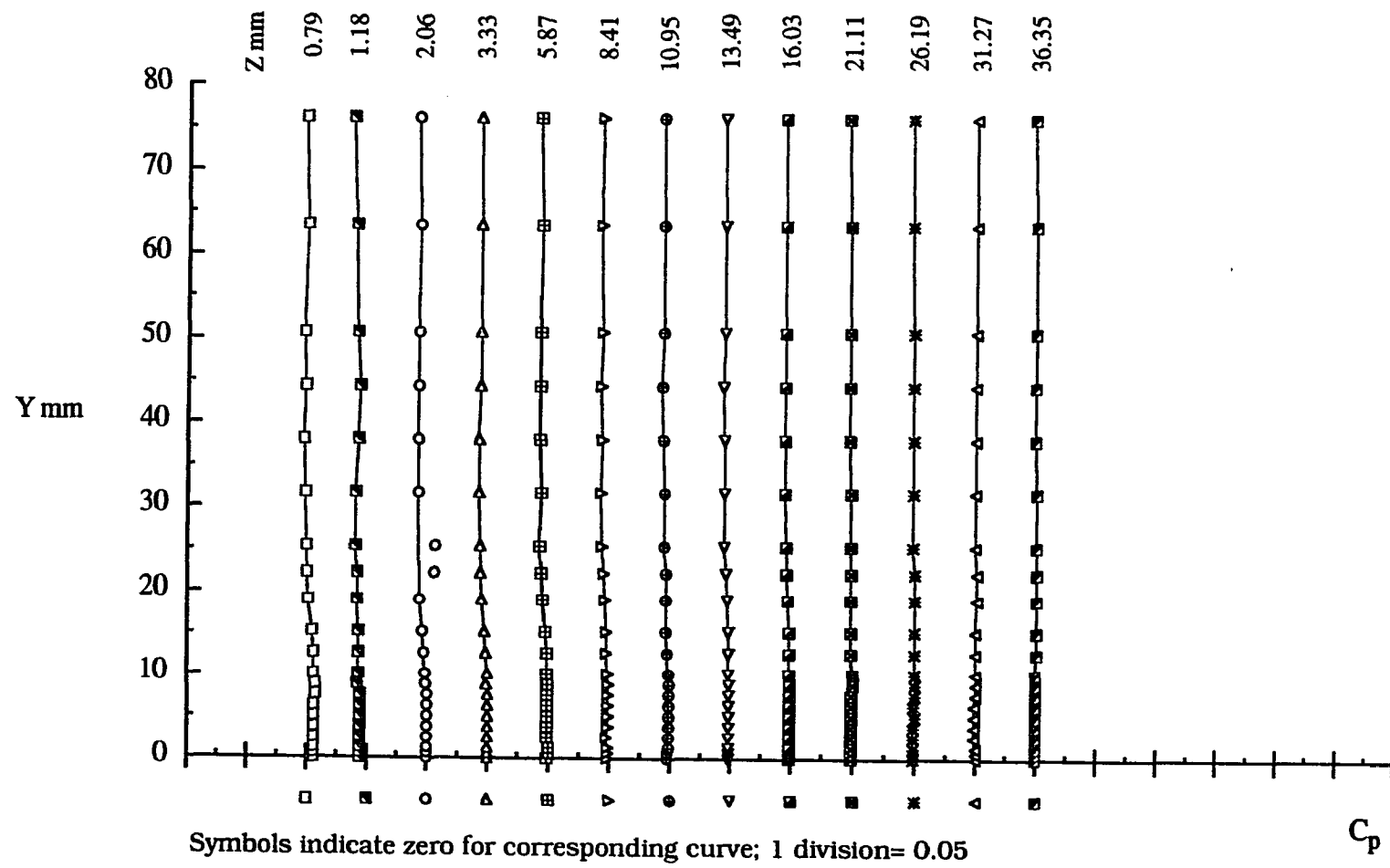


Figure (4.34) Static Pressure Variation Parallel to the Flat Plate at Station #6 ($X = 177.80$ mm, $X/C = 0.70$).

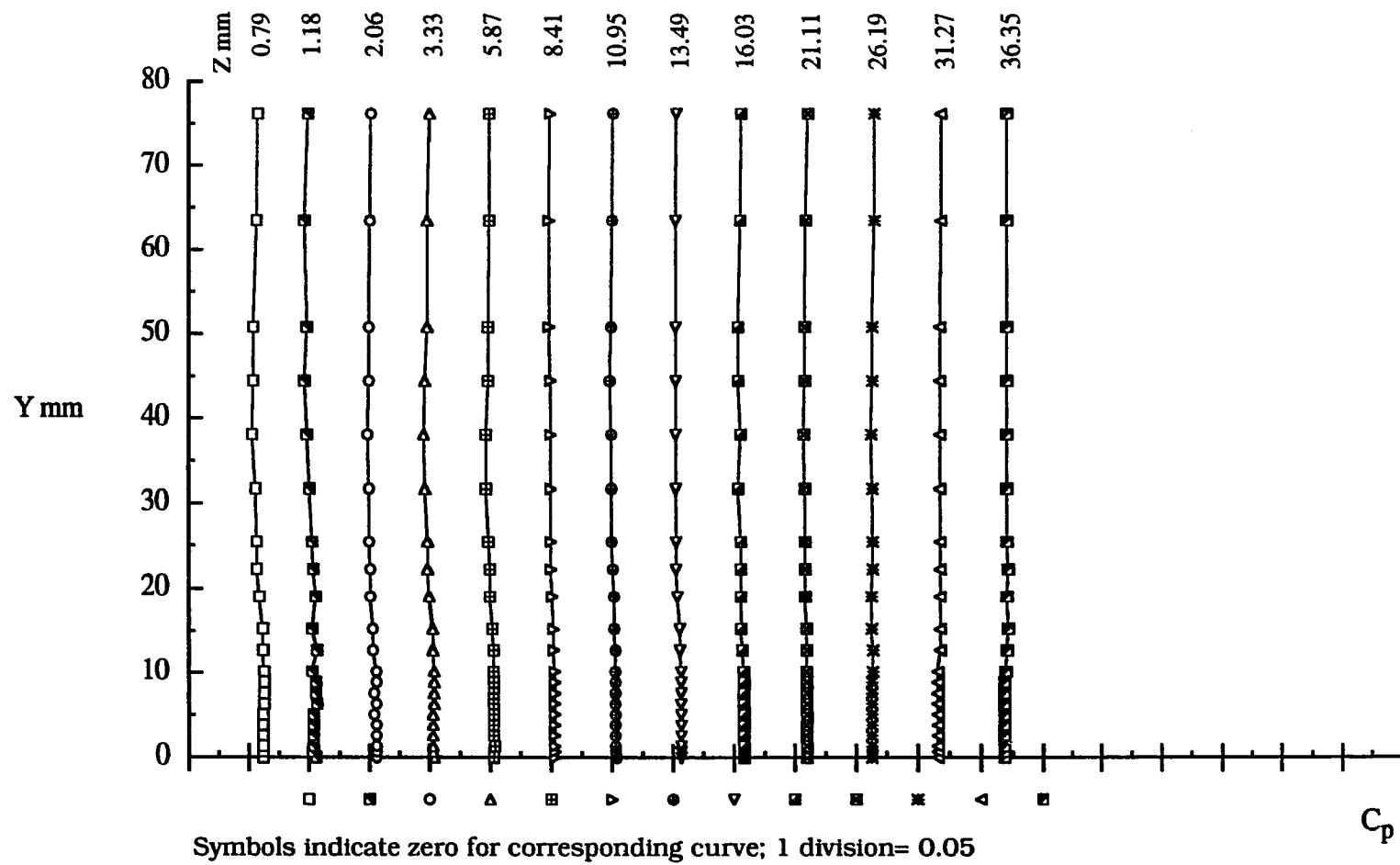


Figure (4.35) Static Pressure Variation Parallel to the Flat Plate at Station #7 ($X = 254.00$ mm, $X/C = 1.00$).

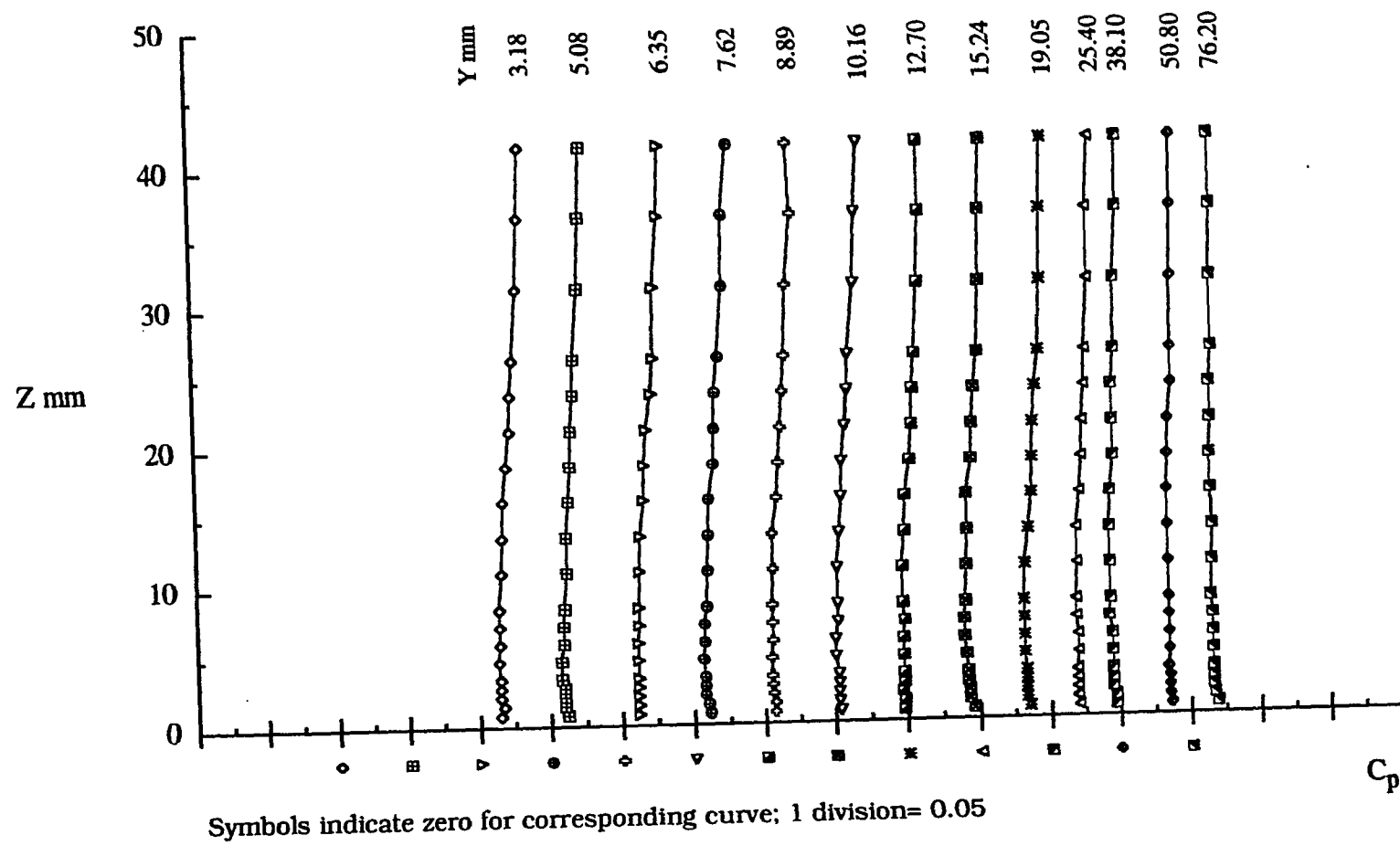


Figure (4.36) Static Pressure Variation Perpendicular to the Flat Plate at Station #0 ($X = 0.00$ mm, $X/C = 0.00$).

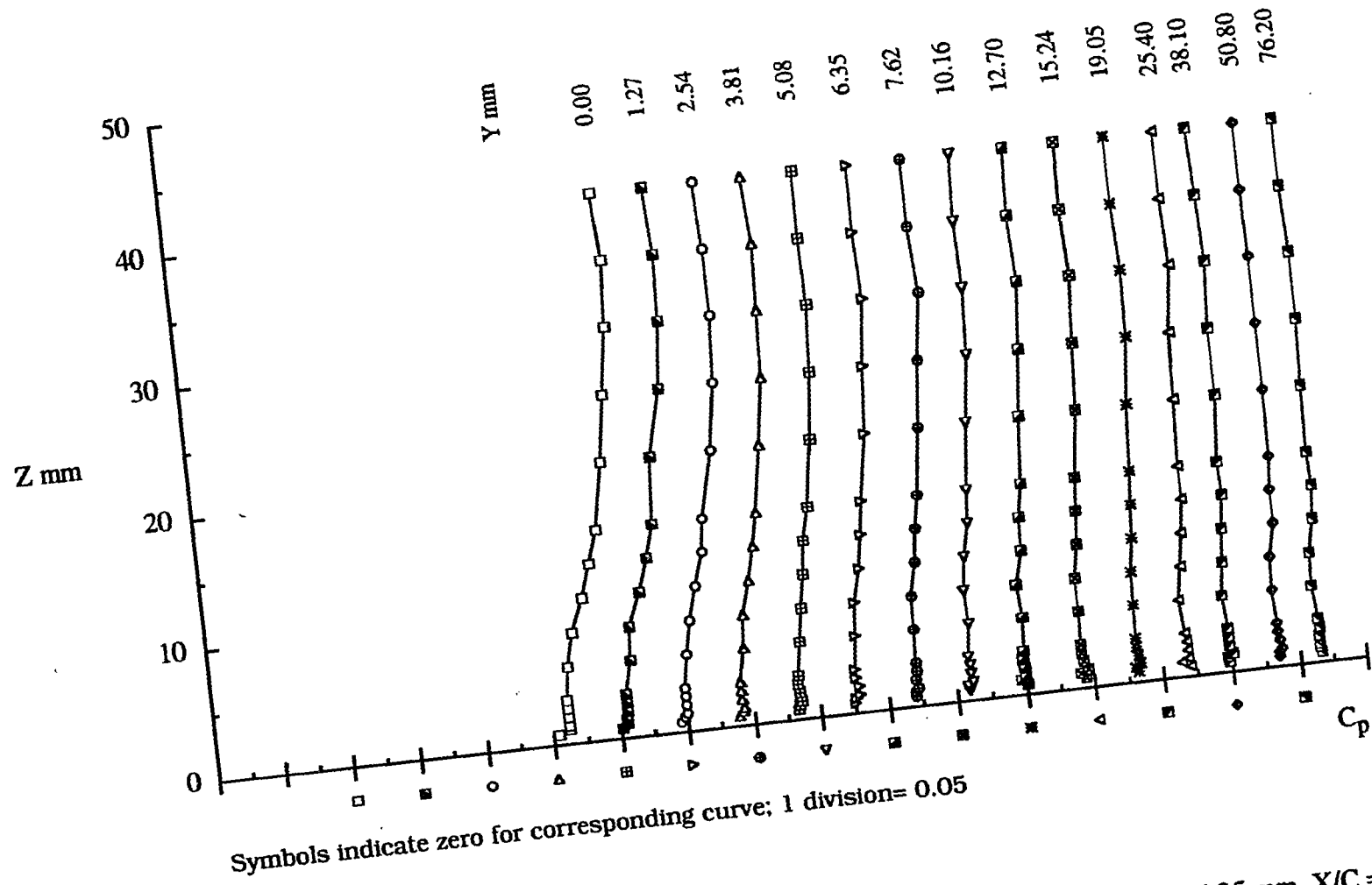


Figure (4.37) Static Pressure Variation Perpendicular to the Flat Plate at Station #1 ($X = 6.35$ mm, $X/C = 0.025$).

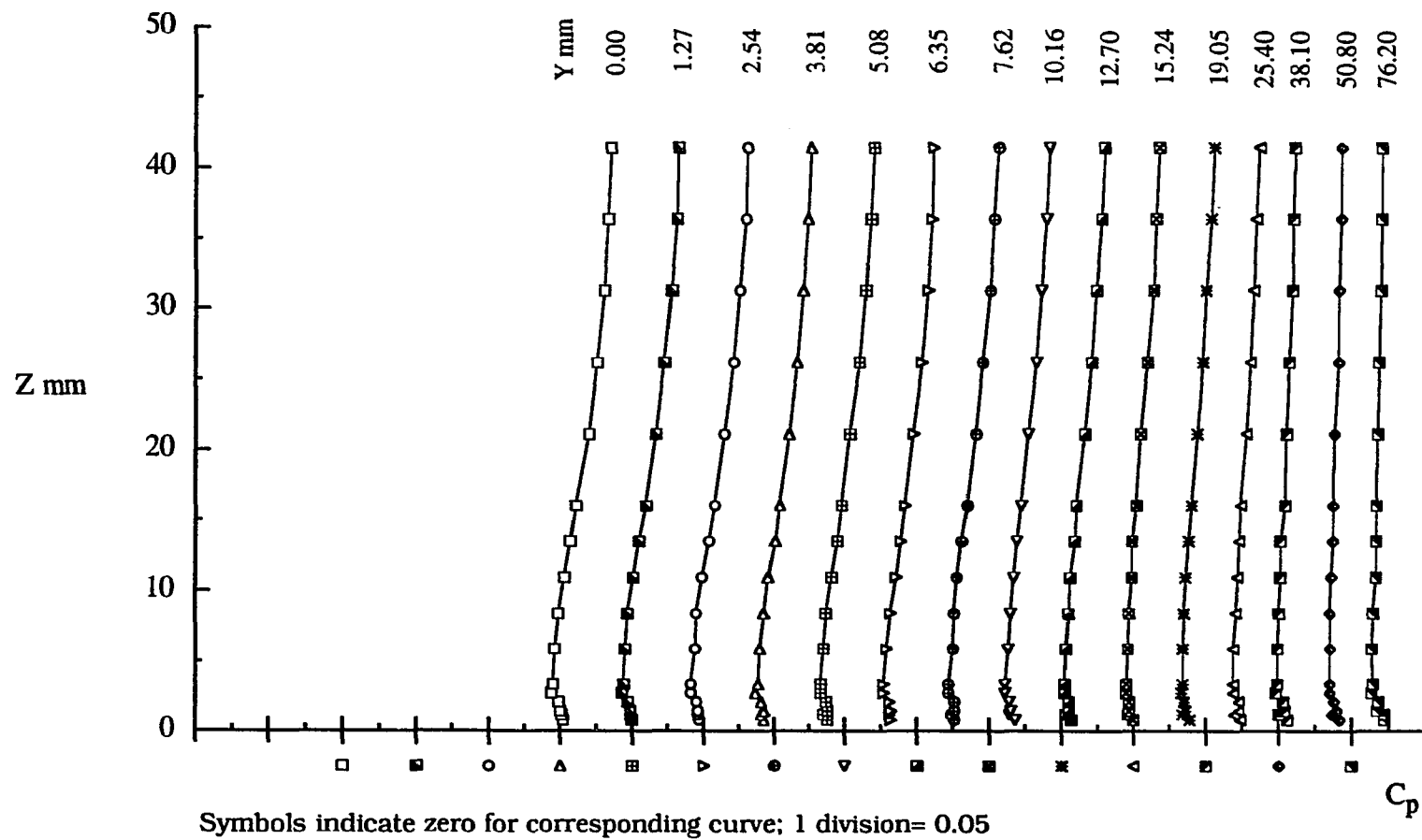


Figure (4.38) Static Pressure Variation Perpendicular to the Flat Plate at Station #2 ($X = 12.70$ mm, $X/C = 0.05$).

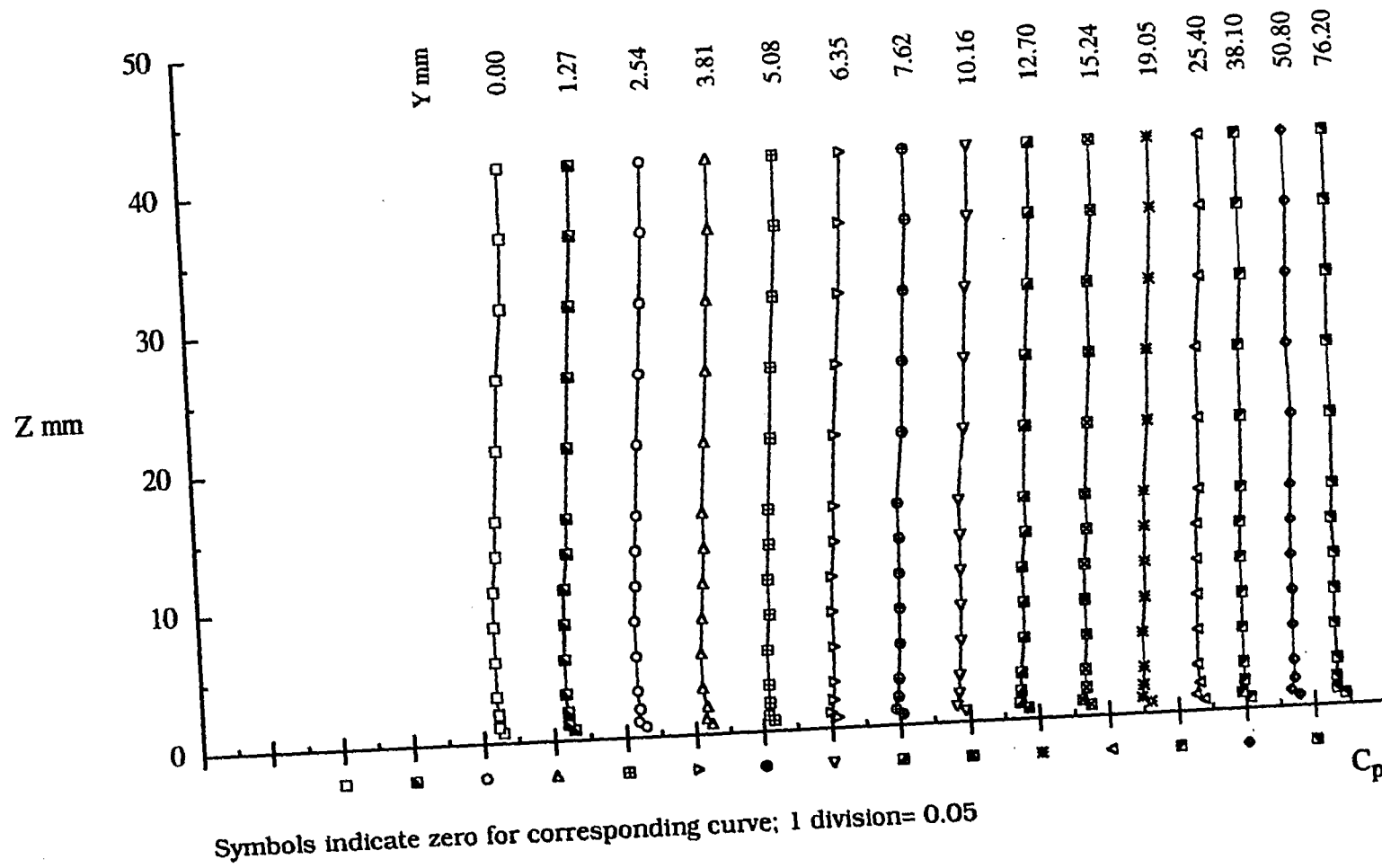


Figure (4.39) Static Pressure Variation Perpendicular to the Flat Plate at Station #3 ($X = 25.40$ mm, $X/C = 0.10$).

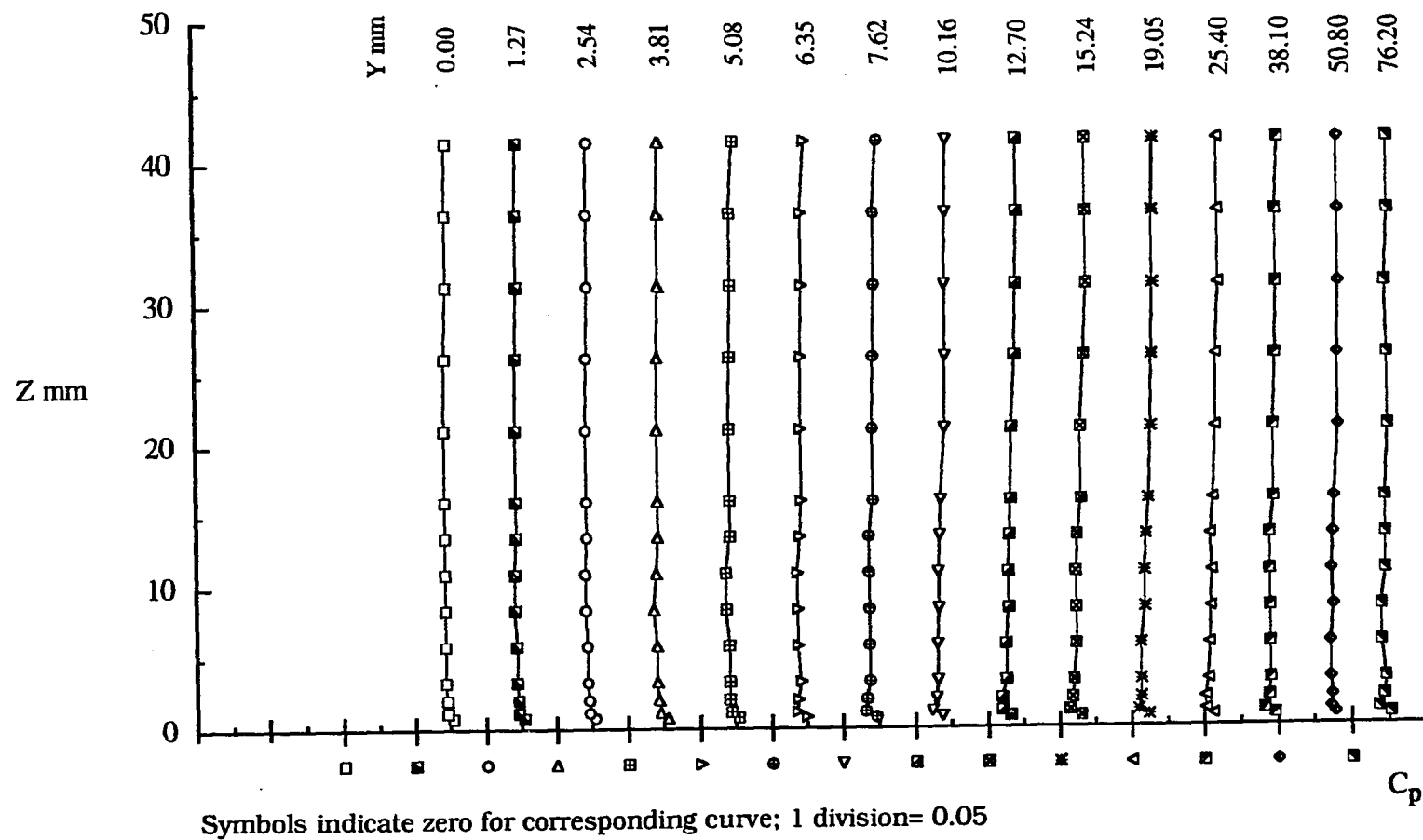


Figure (4.40) Static Pressure Variation Perpendicular to the Flat Plate at Station #4 ($X = 50.80$ mm, $X/C = 0.20$).

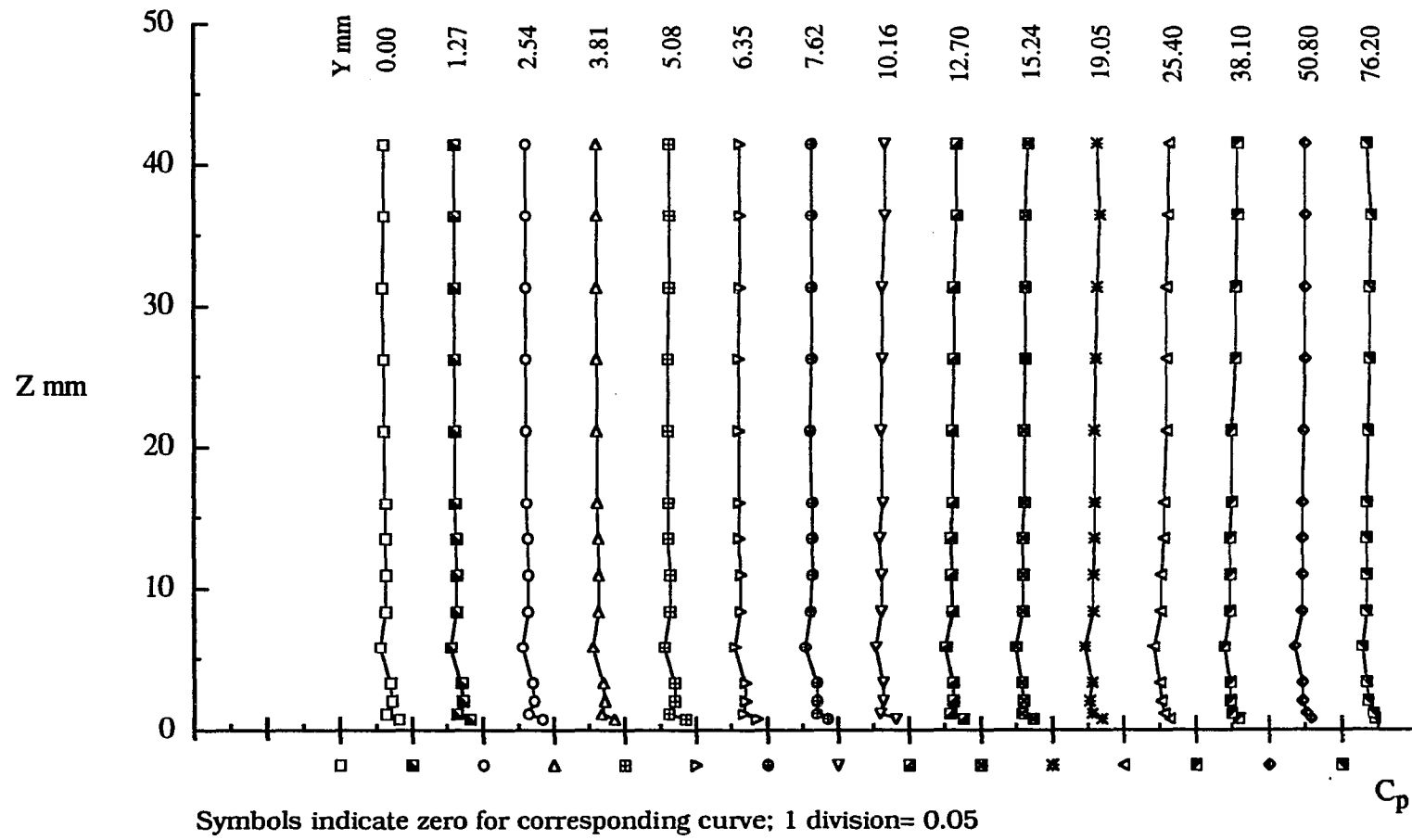


Figure (4.41) Static Pressure Variation Perpendicular to the Flat Plate at Station #5 ($X = 101.6$ mm, $X/C = 0.40$).

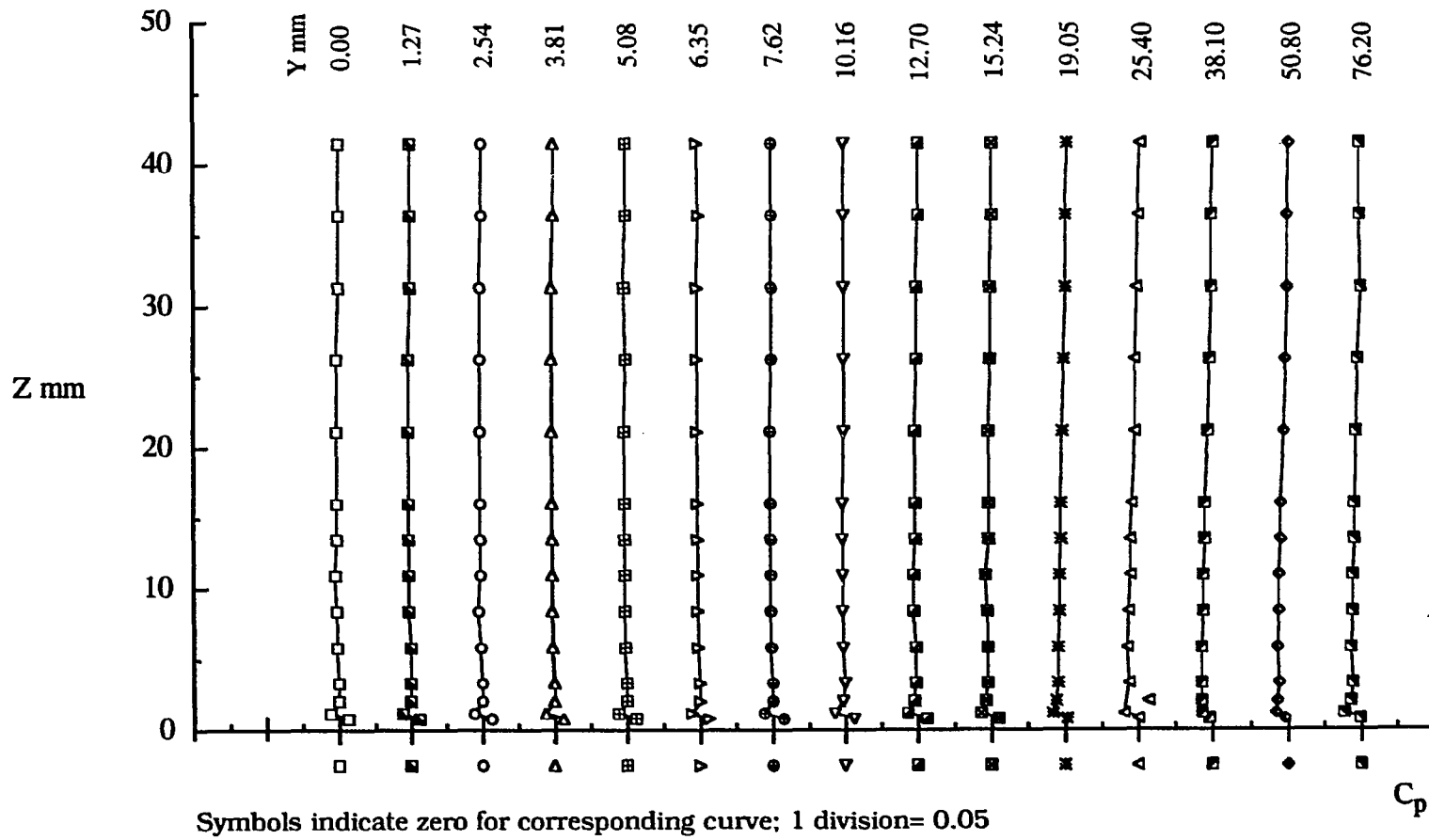


Figure (4.42) Static Pressure Variation Perpendicular to the Flat Plate at Station #6 ($X = 177.80$ mm, $X/C = 0.70$).

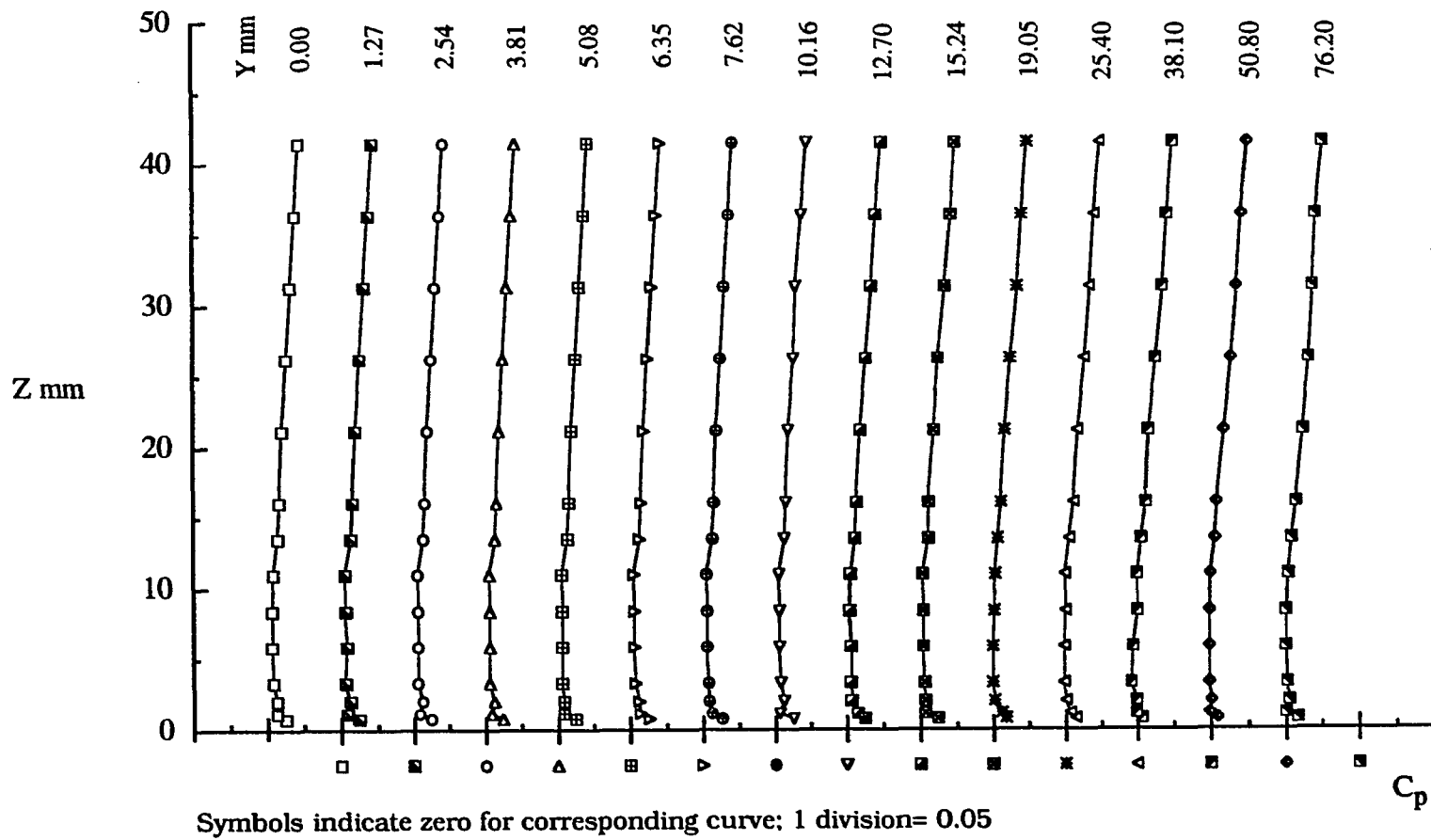


Figure (4.43) Static Pressure Variation Perpendicular to the Flat Plate at Station #7 ($X = 250.40$ mm, $X/C = 1.00$).

flat plate (i.e., variable Y) at different heights above the flat plate (i.e., constant Z) are presented. In Figure (4.36) to Figure (4.43), the variation of C_p perpendicular to the flat plate (i.e., variable Z) at different distances from the centerplane (i.e., constant Y) are presented.

At the trailing edge location (i.e., Station #0, $X/C = 0$), departing from the centerplane, static pressure is at its maximum value in the vicinity of the blade trailing edge and it decreases in the transverse (Y -) direction. There are no significant static pressure gradients in the spanwise (Z -) direction. In other words, the rate of decrease in C_p , in the transverse (Y -) direction, is almost constant for different heights above the flat plate. At this location the blade wake has not been felt yet.

In all of the axial stations downstream of the trailing edge (i.e., $X/C = 0.025$ to $X/C = 1$), static pressure coefficient is at its highest near the trailing edge and it decreases to the atmospheric conditions downstream. In the transverse (Y -) direction, at any height above the plate and for all of the axial stations, static pressure coefficient (C_p) decreases as one moves away from the centerplane. The rate of decrease in C_p is at its highest at axial locations in the vicinity of the blade trailing edge and it decreases downstream. In the spanwise (Z -) direction, for a given distance from the centerplane and at any axial station (except for Station #1 & Station #2), there is very little (if any) variation in C_p . This is consistent with the results reported by Hazarika and Raj [1] and McMahon et al. [91]. For Station #1 & Station #2, C_p increases slightly as one moves away from the flat plate. Even though this is a region of a downwashed wake, it was expected that the downwash effect of the horseshoe vortex to cease in regions far from its domain. This might suggest that, even at large distances ($Z/C \cong 1.4$) from the flat plate, the wing wake is not fully two dimensional (see section (4.5)). This rise in C_p is confined to the blade wake region ($Y/C = 0$ to $Y/C = 0.04$) and it ceases at downstream axial location of $X/C = 0.1$ (i.e., Station #3).

4.4.2 Total Pressure Variation:

The results of total pressure measurement are presented in this section in Figures (4.44) to (4.67). The total pressure coefficient C_{TP} is defined as:

$$C_{TP} = (p_o - p_a) / (p_\infty - p_a)$$

where

p_o = Local total (stagnation) pressure.

p_a = Atmospheric pressure.

p_∞ = Free stream total (stagnation) pressure.

Total pressure variation, parallel to the flat plate (i.e., variable Y) at different heights above it, at the eight axial stations downstream of the blade trailing edge are shown in Figure (4.44) to Figure (4.51). The plane of Station #0 ($X/C = 0$) was located just behind the trailing edge.

A general trend is clear at all stations; departing from the centerplane ($Y/C=0$), total pressure coefficient starts with a local minimum (which will be labeled as C_{TPmin1}). It sharply increases to a local maximum (C_{TPmax}), as it will be discussed later, this is a region of downwashed wake. Then values of C_{TP} decreases to a local minimum C_{TPmin2} (flow upwash) and again it gradually increases to its local asymptotic value at that height. This trend is very similar to that of the local wall shear stress. It can be noted that, at heights close to the flat plate and at all axial stations, the local maximum C_{TPmax} has values higher than that of the undisturbed boundary layer at that height. Their presence (i.e., C_{TPmax} values higher than that of the undisturbed boundary layer at that height) is confined to heights approximately half

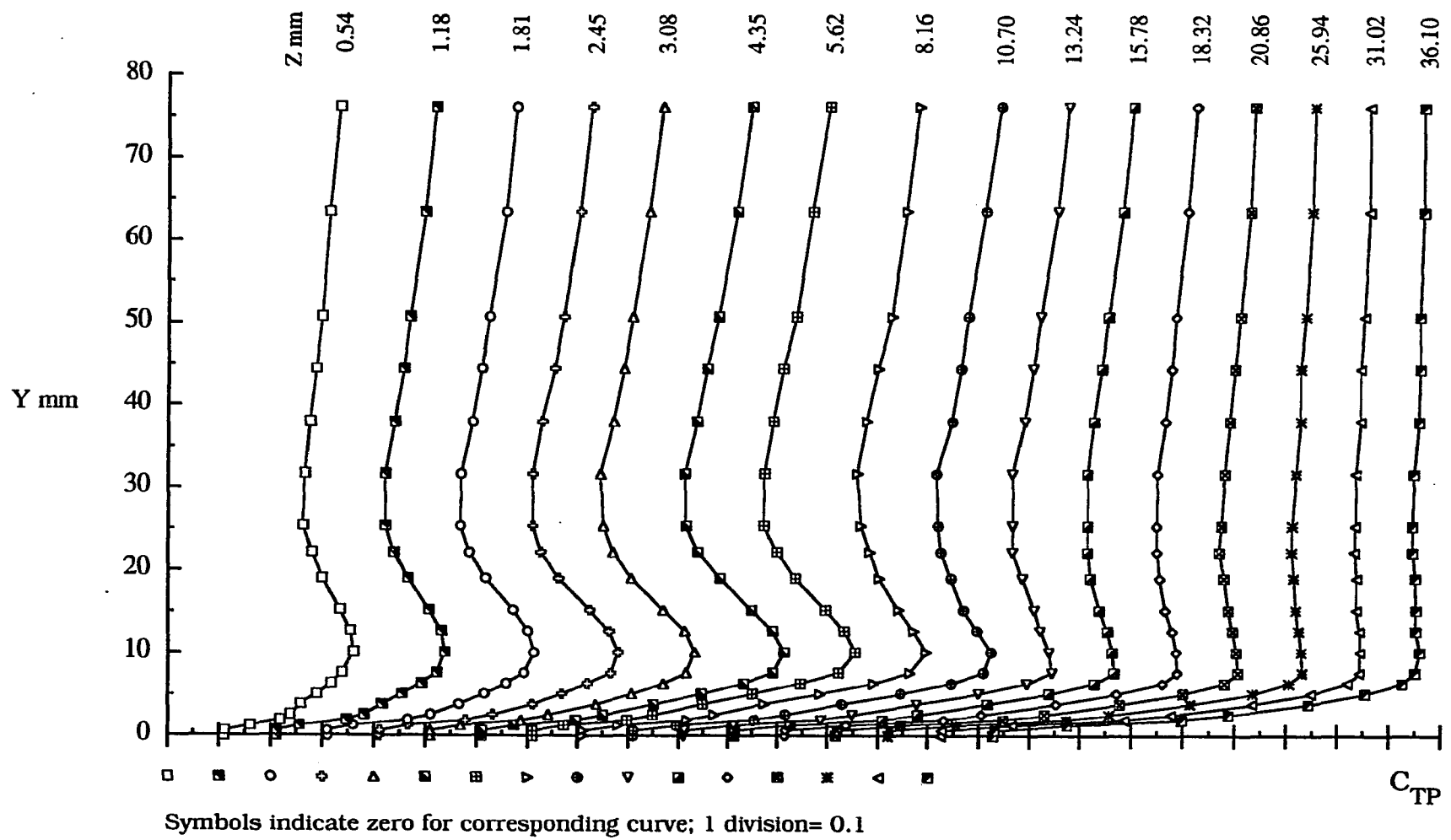


Figure (4.44) Total Pressure Variation Parallel to the Flat Plate at Station #0 ($X = 0.00$ mm, $X/C = 0.00$).

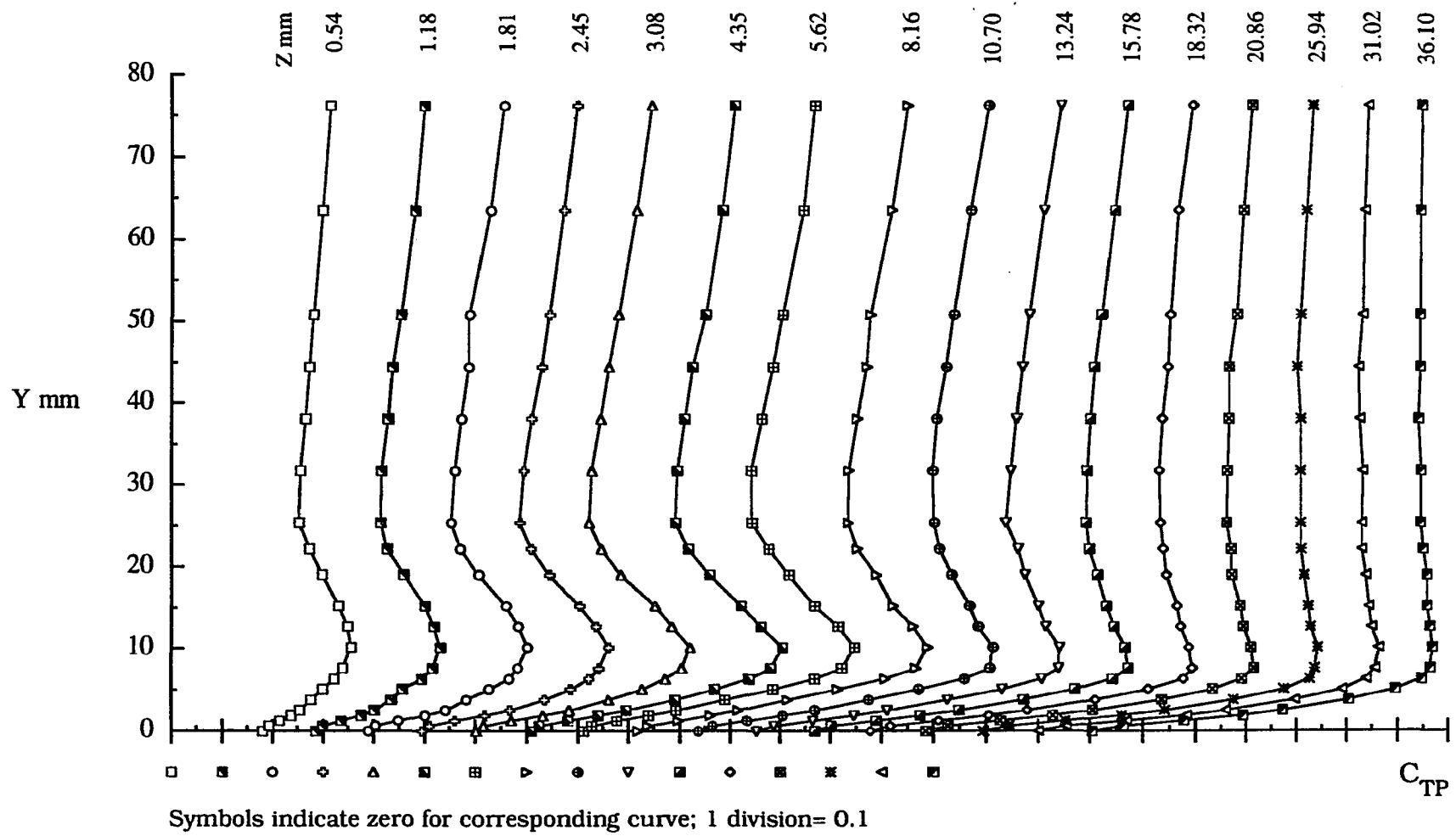


Figure (4.45) Total Pressure Variation Parallel to the Flat Plate at Station #1 ($X = 6.35$ mm, $X/C = 0.025$).

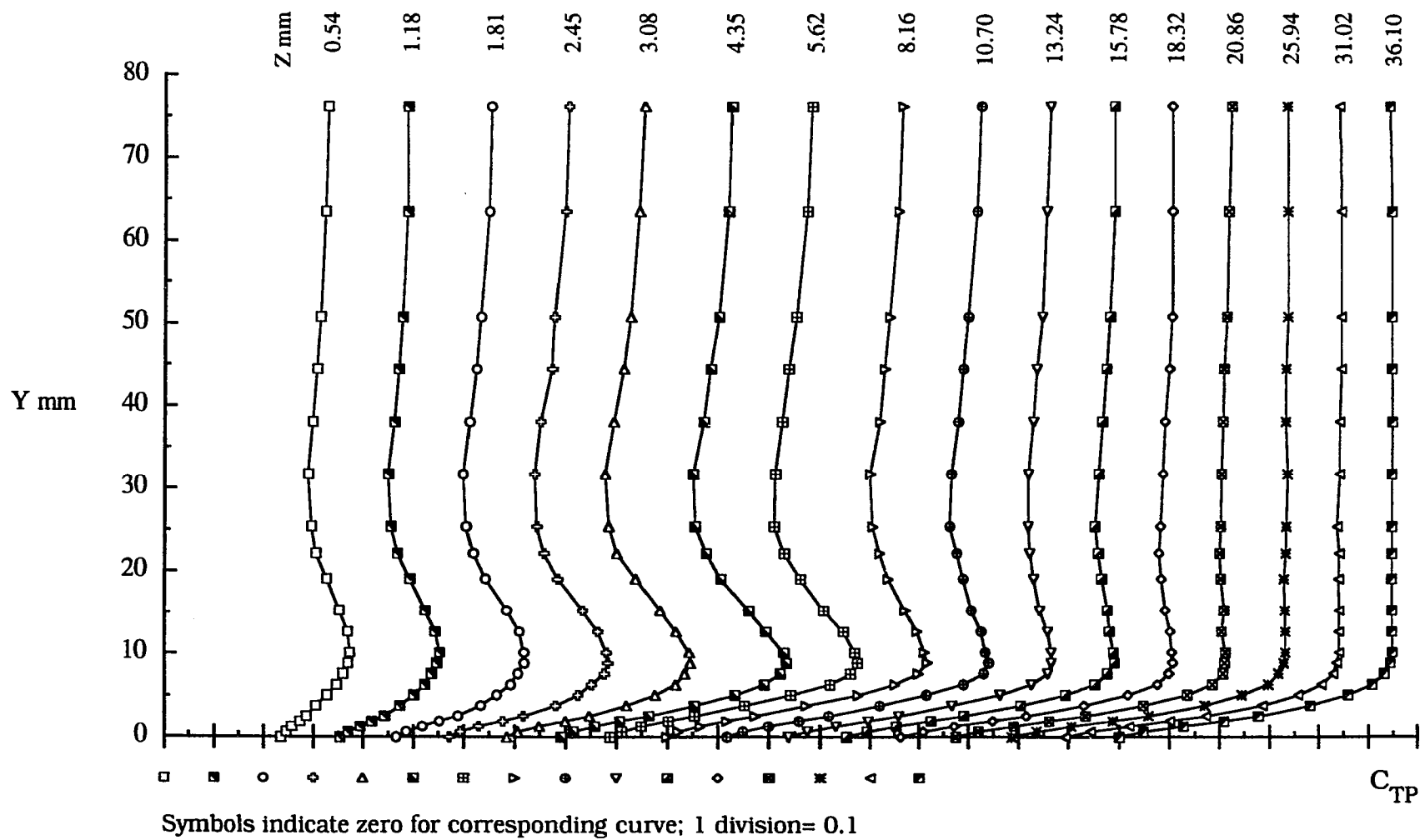


Figure (4.46) Total Pressure Variation Parallel to the Flat Plate at Station #2 ($X = 12.70$ mm, $X/C = 0.05$).

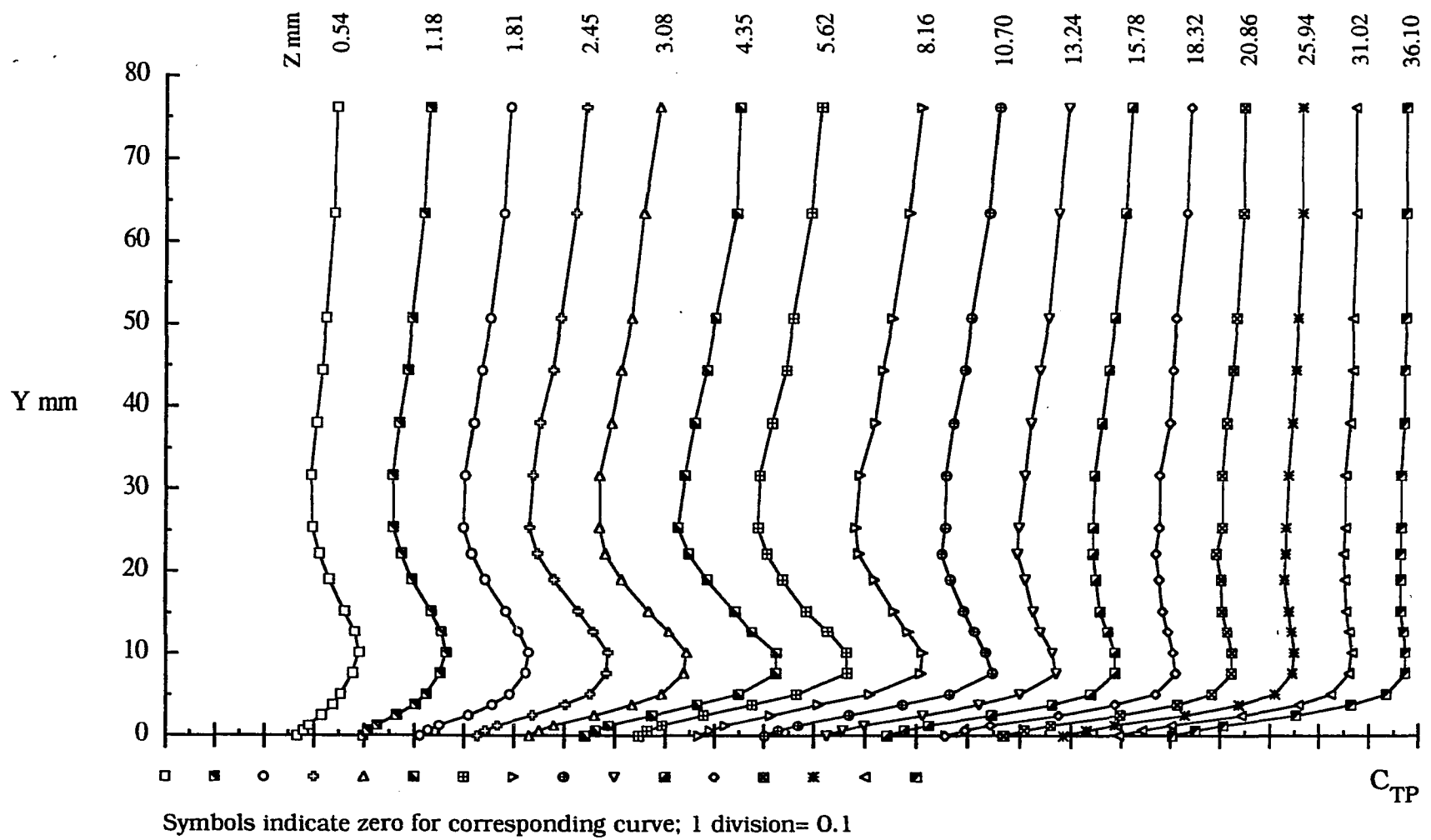


Figure (4.47) Total Pressure Variation Parallel to the Flat Plate at Station #3 ($X = 25.40$ mm, $X/C = 0.10$).

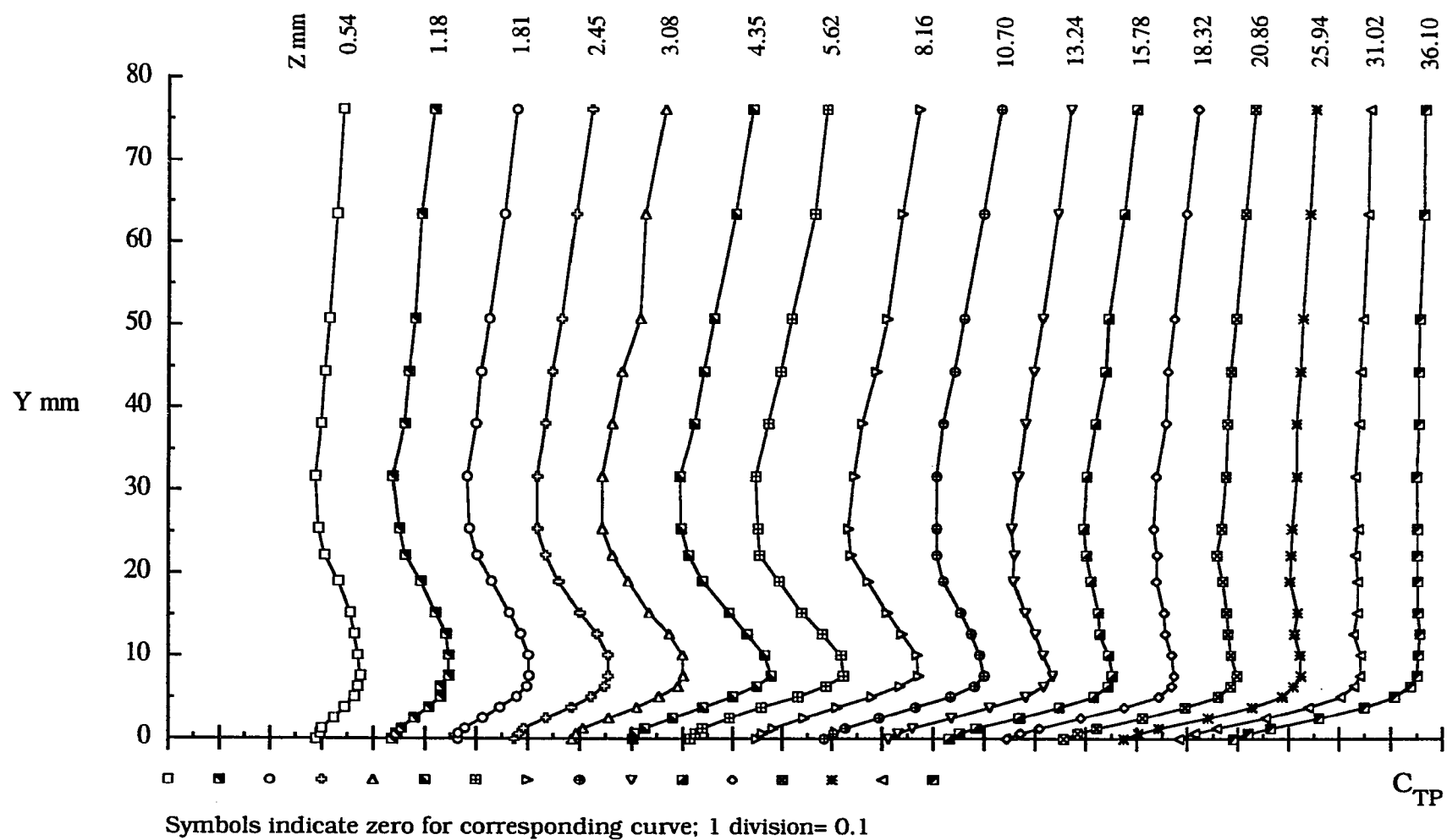


Figure (4.48) Total Pressure Variation Parallel to the Flat Plate at Station #4 (X = 50.80 mm, X/C = 0.20).

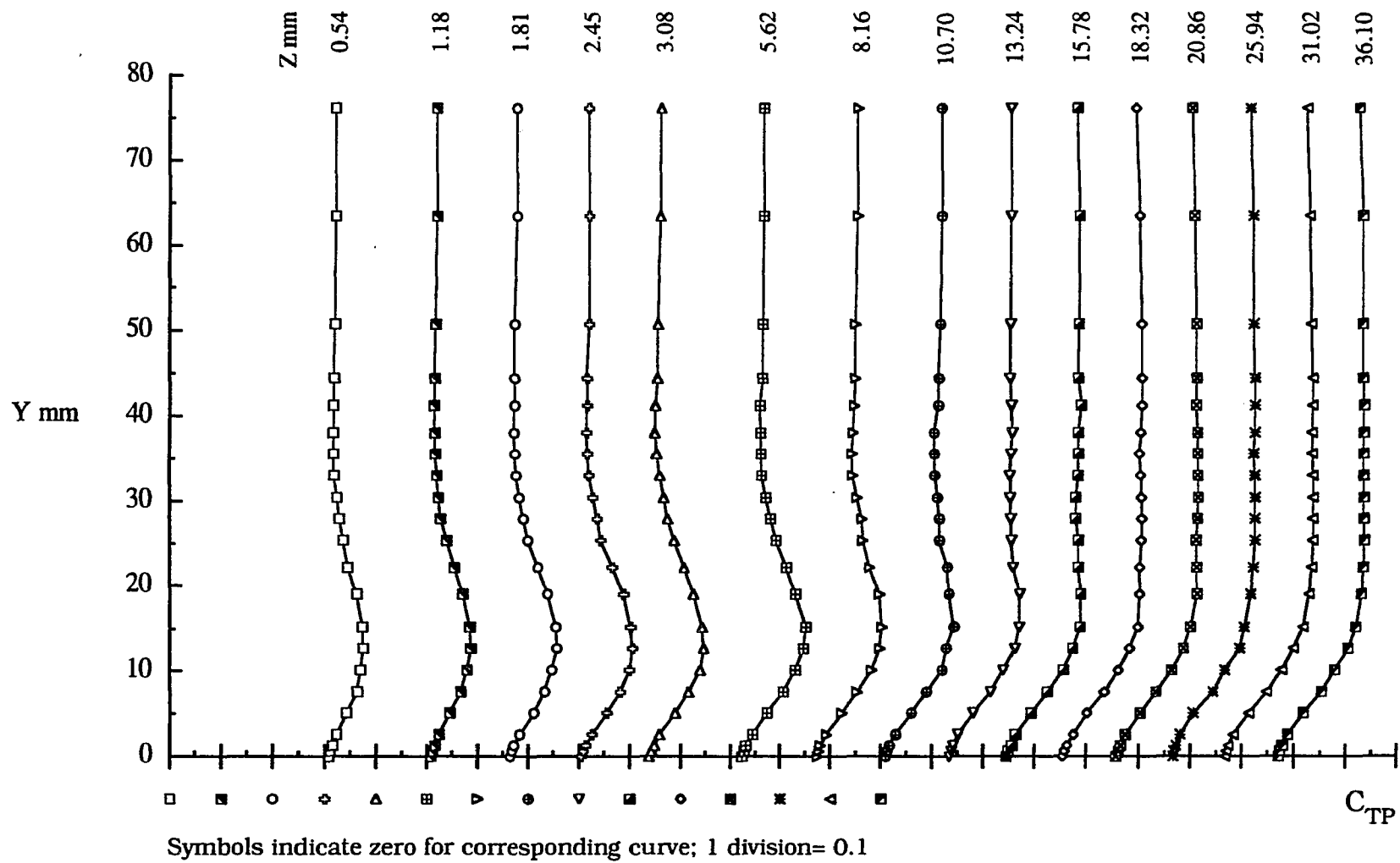


Figure (4.51) Total Pressure Variation Parallel to the Flat Plate at Station #7 (X = 254.00 mm, X/C = 1.00).

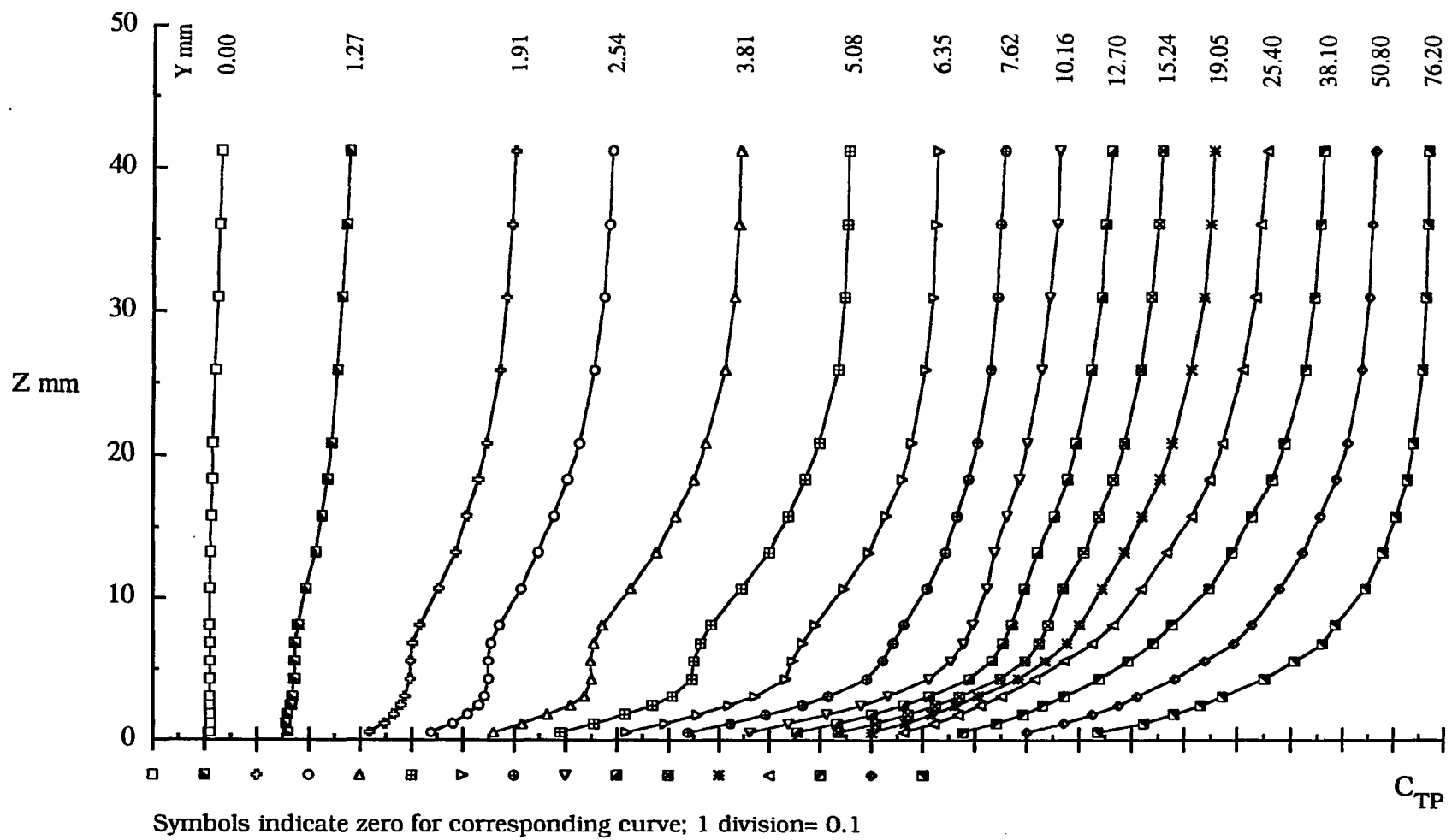


Figure (4.52) Total Pressure Variation Perpendicular to the Flat Plate at Station #0 ($X = 0.00$ mm, $X/C = 0.00$).

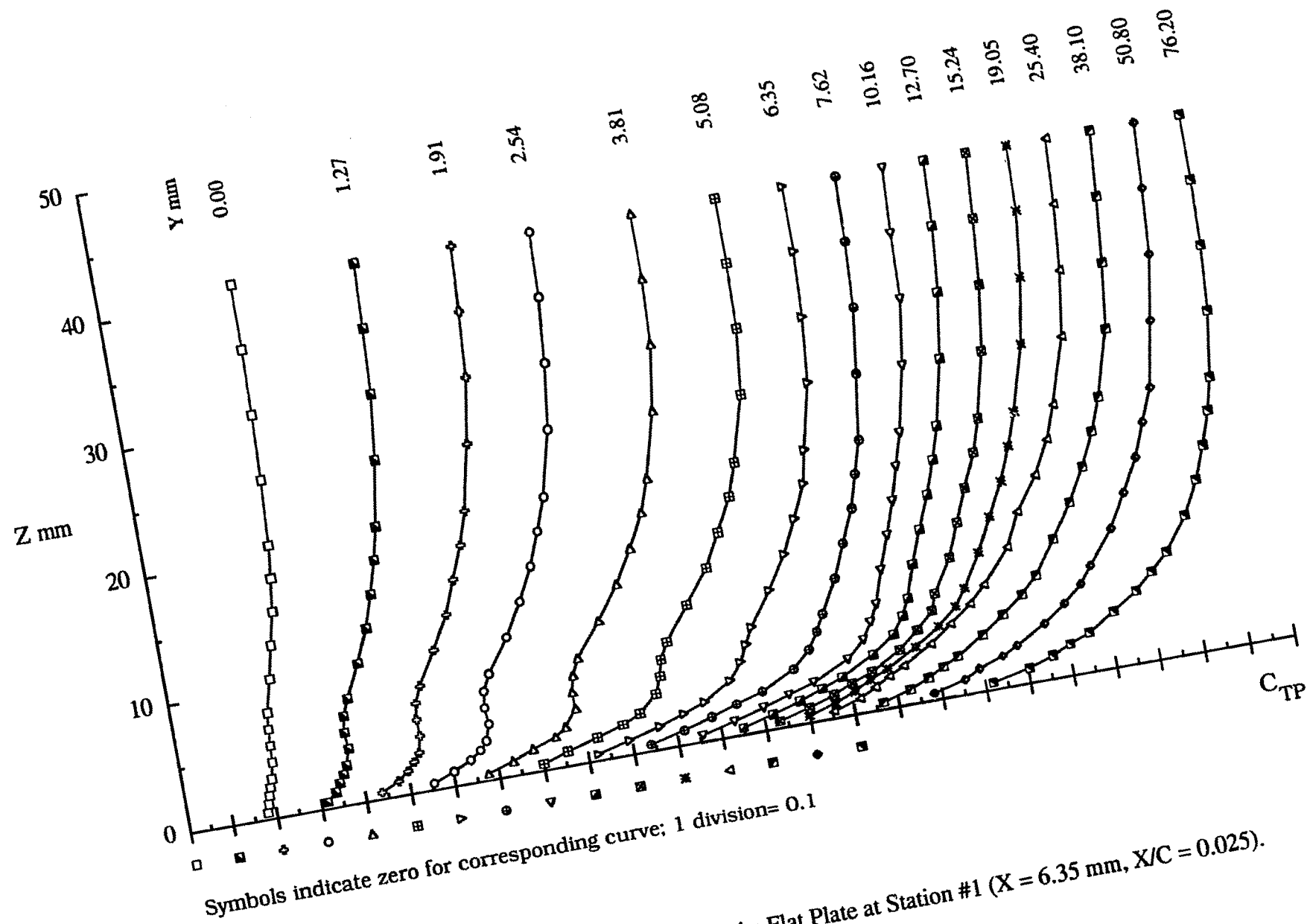
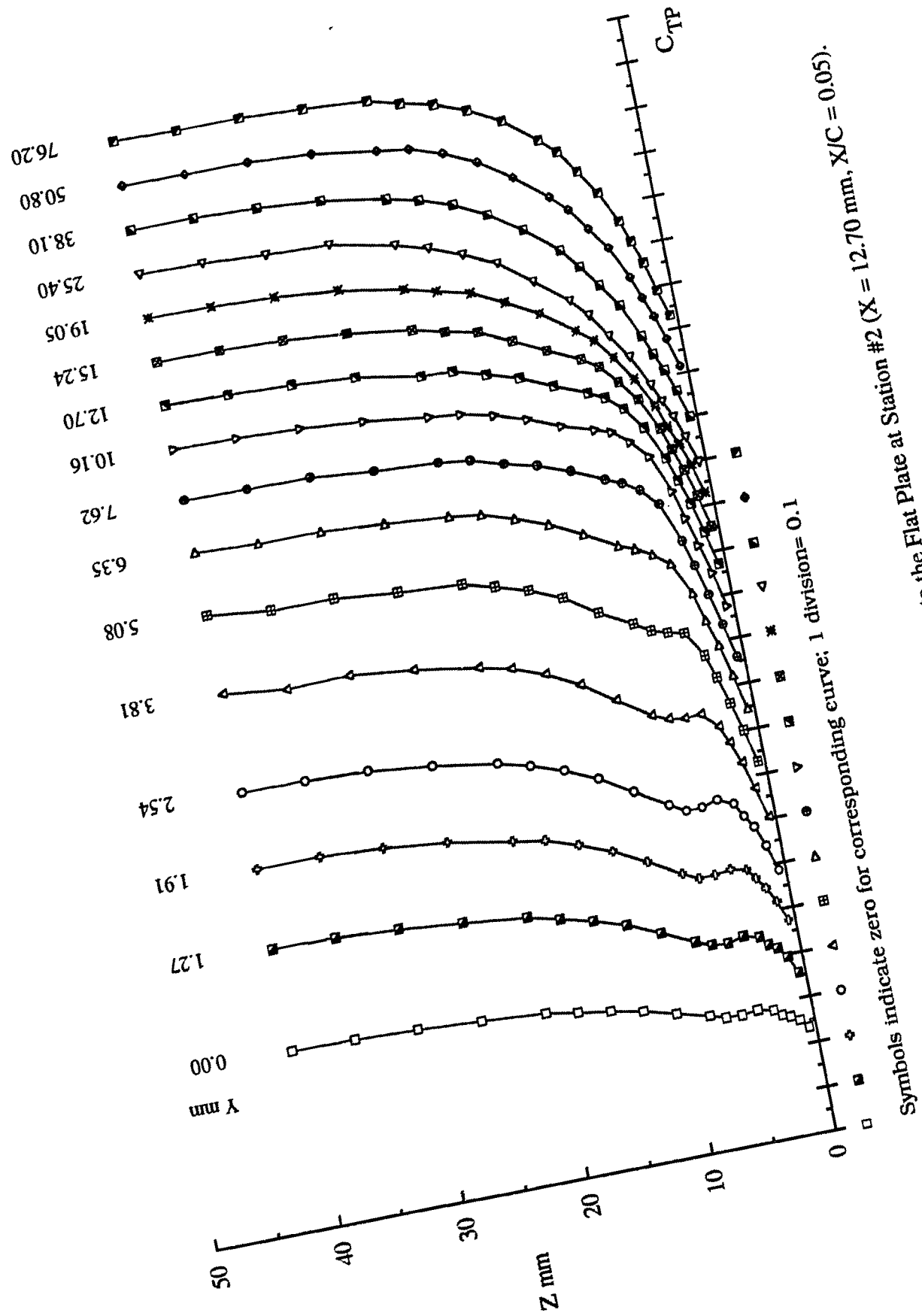


Figure (4.53) Total Pressure Variation Perpendicular to the Flat Plate at Station #1 ($X = 6.35$ mm, $X/C = 0.025$).



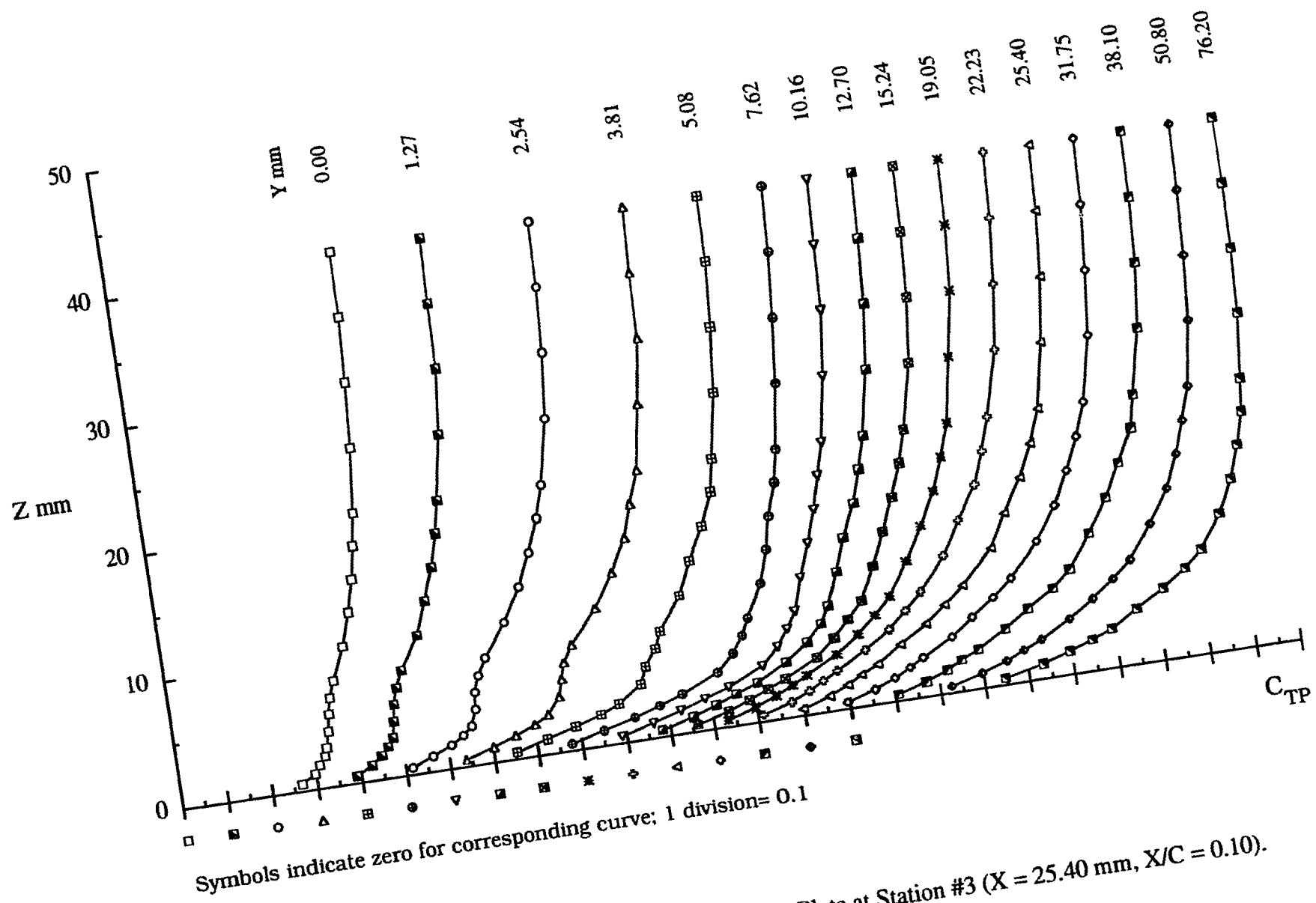


Figure (4.55) Total Pressure Variation Perpendicular to the Flat Plate at Station #3 ($X = 25.40$ mm, $X/C = 0.10$).

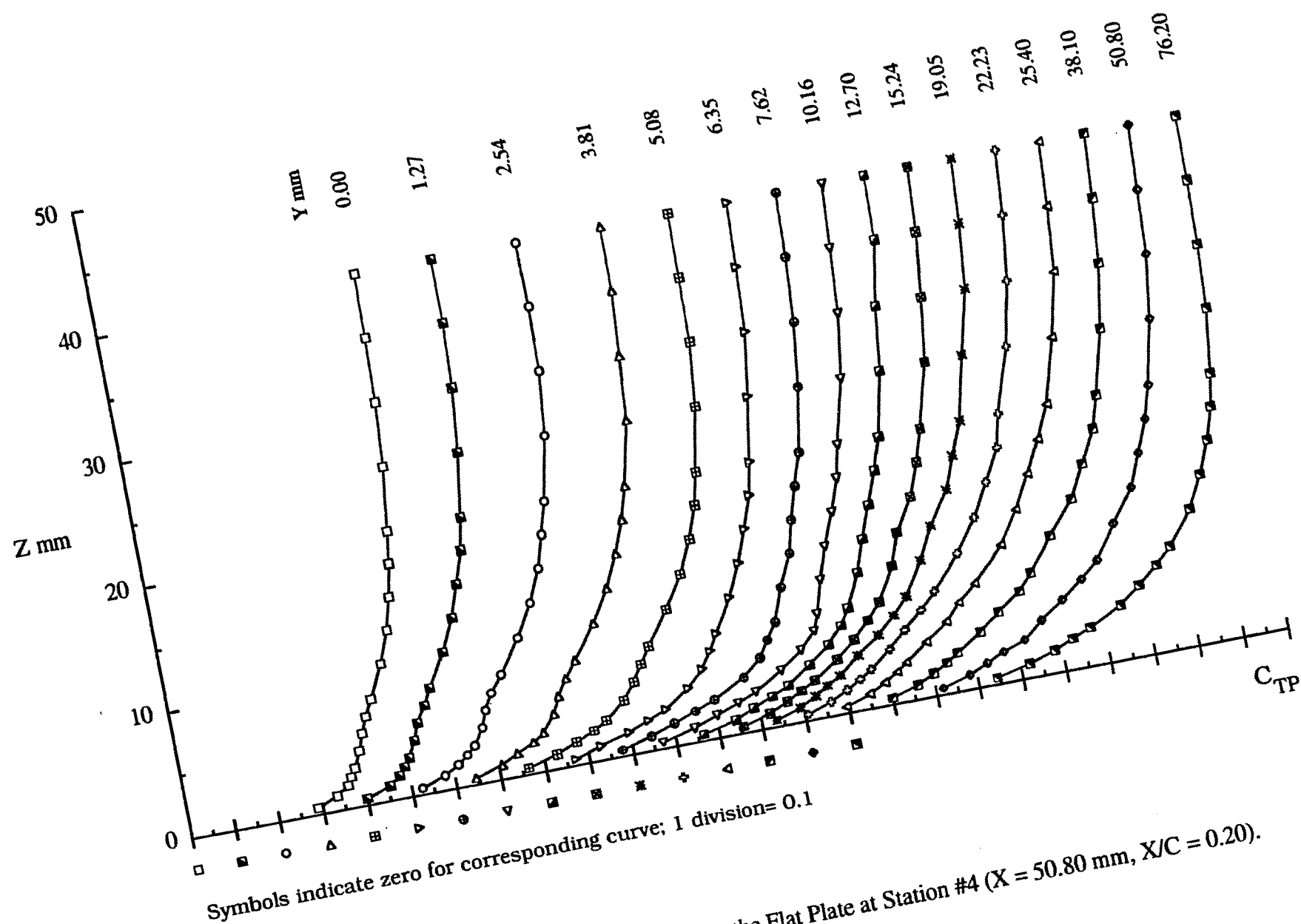


Figure (4.56) Total Pressure Variation Perpendicular to the Flat Plate at Station #4 ($X = 50.80$ mm, $X/C = 0.20$).

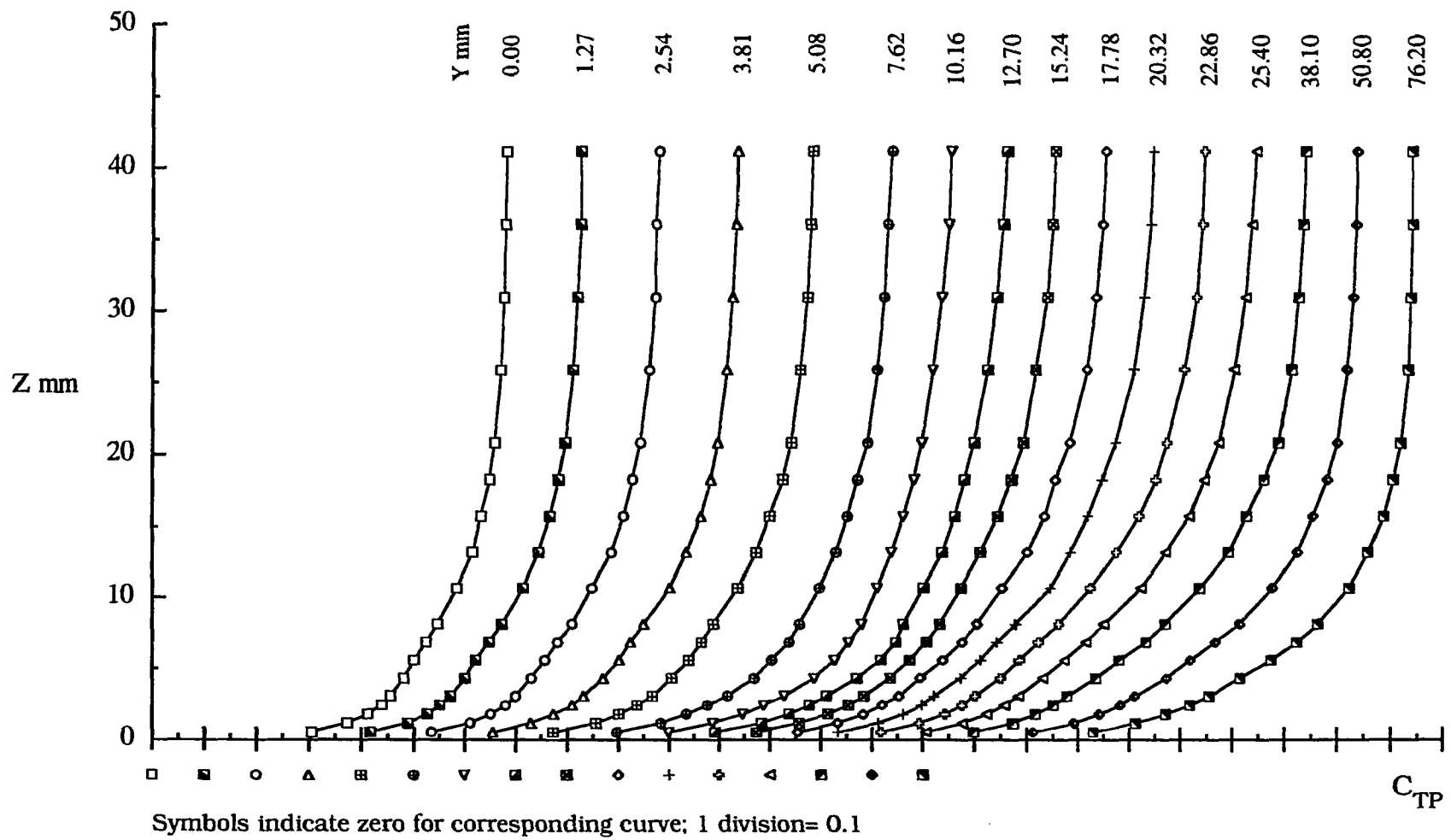


Figure (4.57) Total Pressure Variation Perpendicular to the Flat Plate at Station #5 ($X = 101.60$ mm, $X/C = 0.40$).

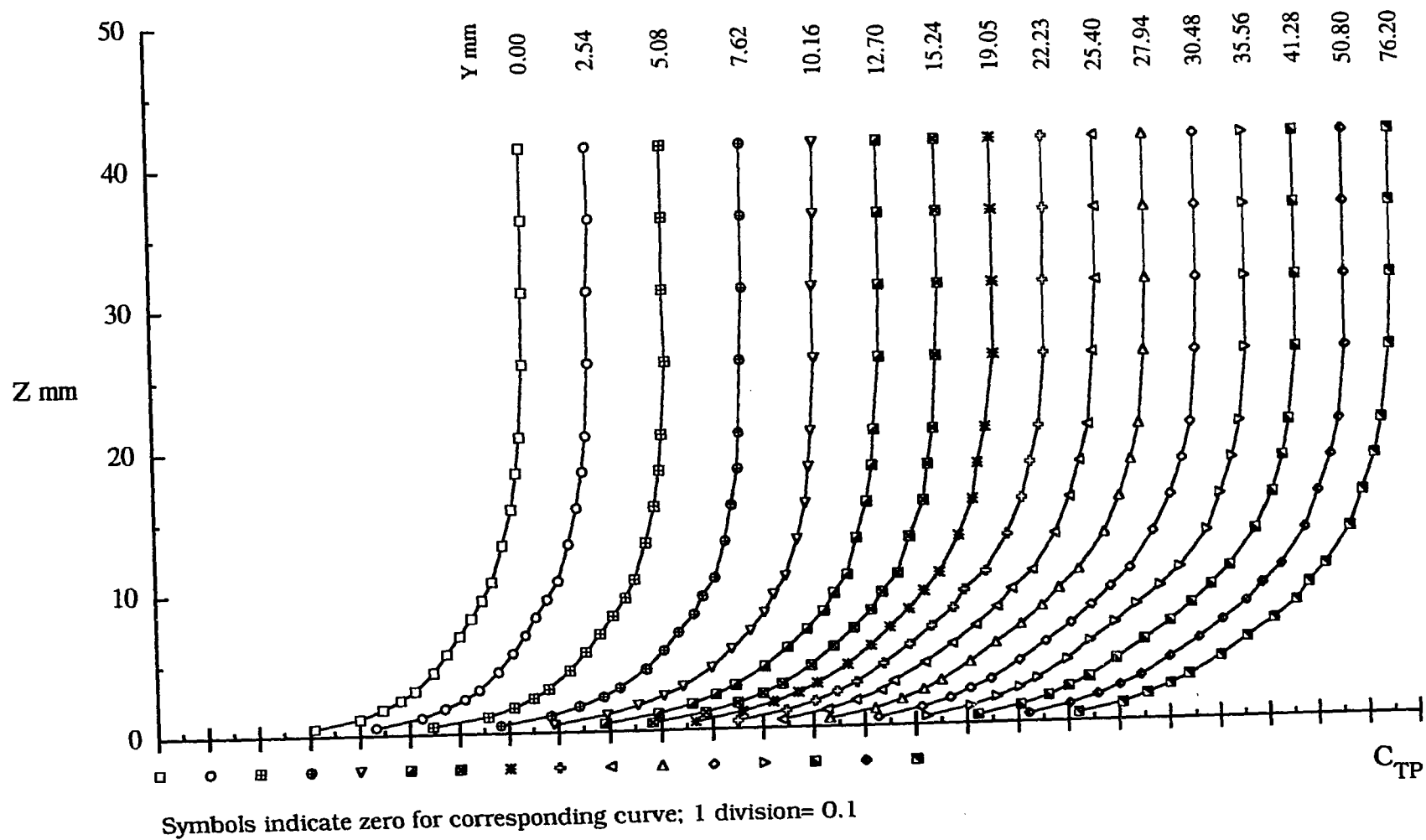


Figure (4.58) Total Pressure Variation Perpendicular to the Flat Plate at Station #6 ($X = 177.80$ mm, $X/C = 0.70$).

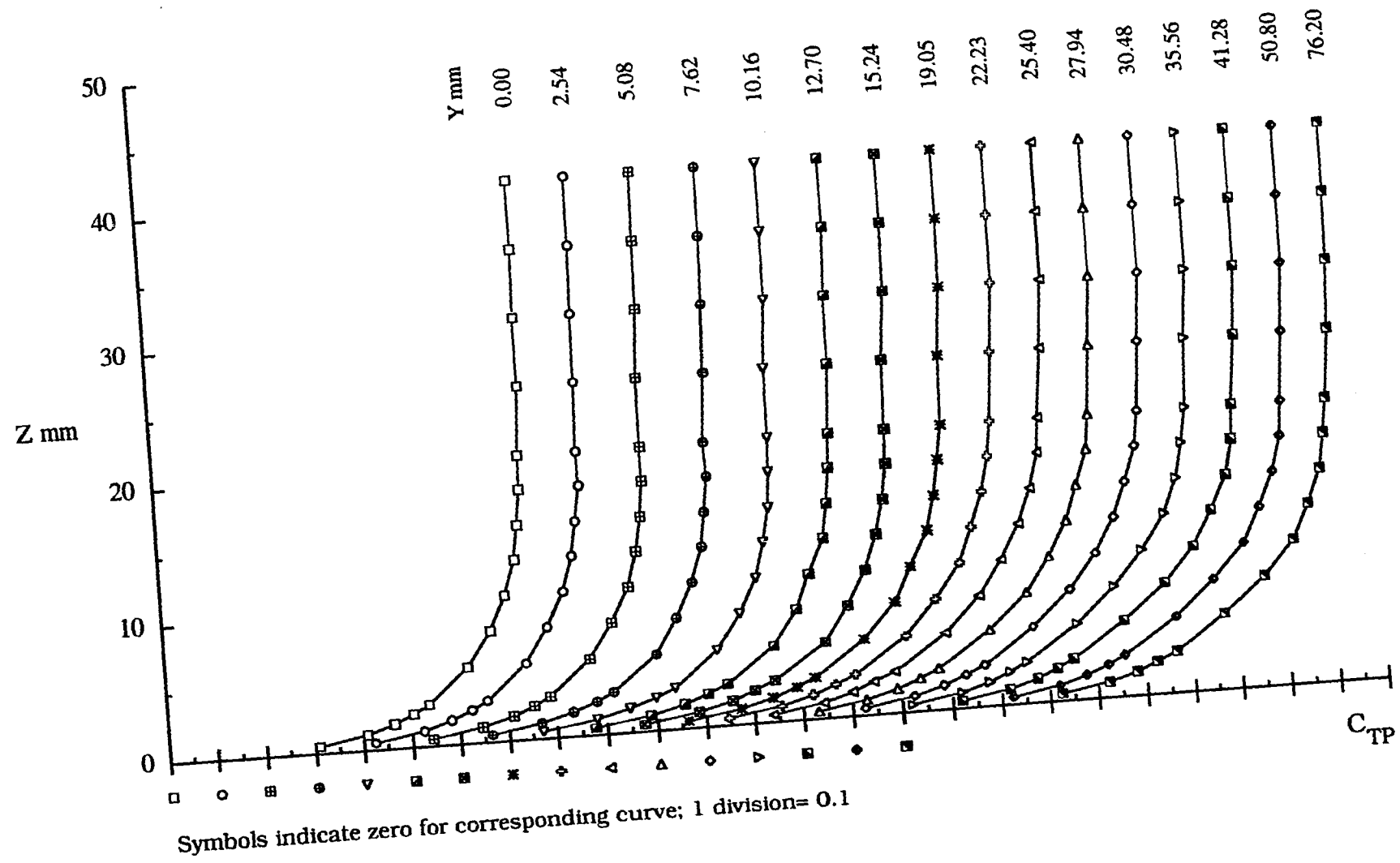


Figure (4.59) Total Pressure Variation Perpendicular to the Flat Plate at Station #7 ($X = 254.00$ mm, $X/C = 1.00$).

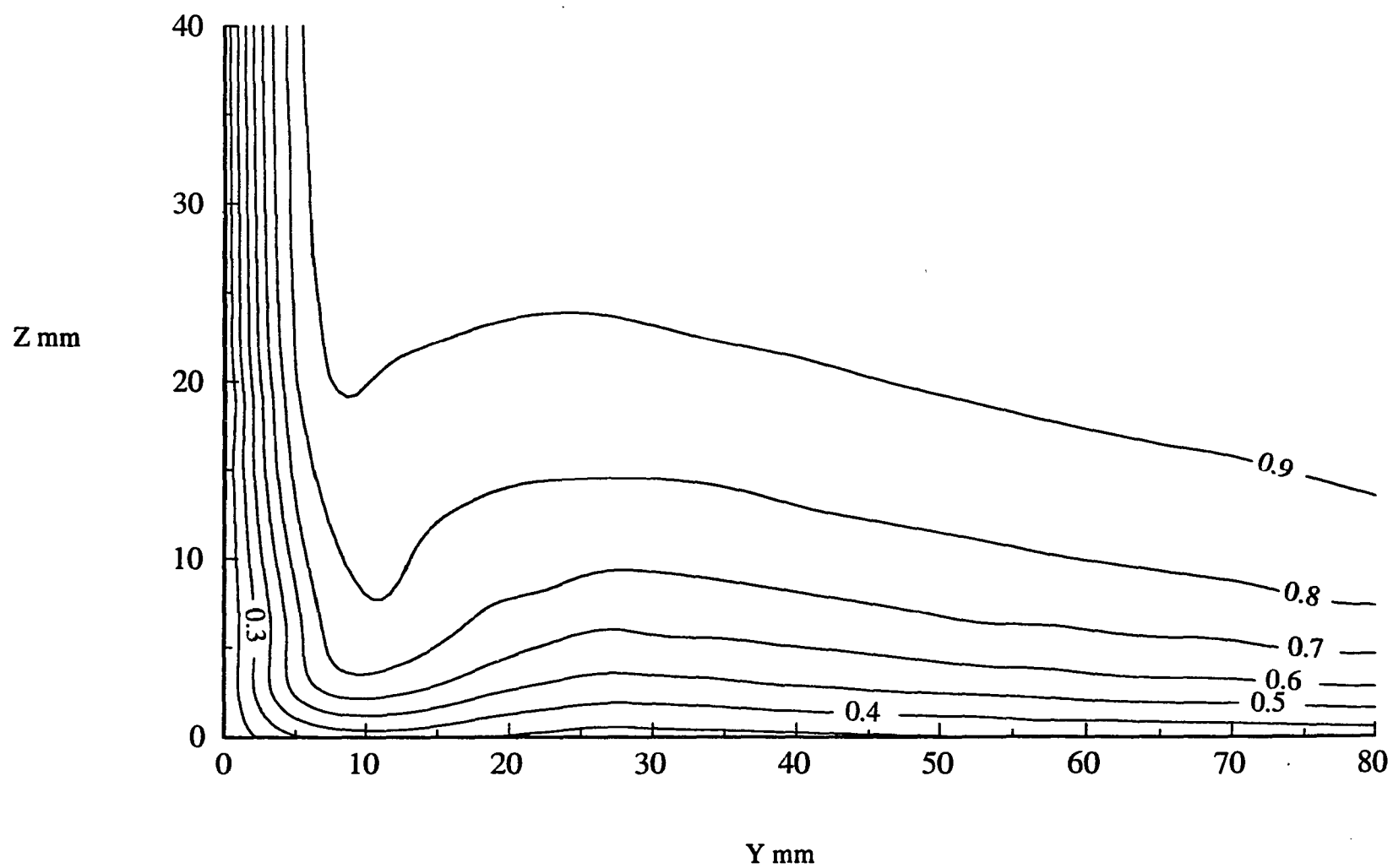


Figure (4.60) Total Pressure Contours at Station #0 ($X = 0.00$ mm, $X/C = 0.00$).

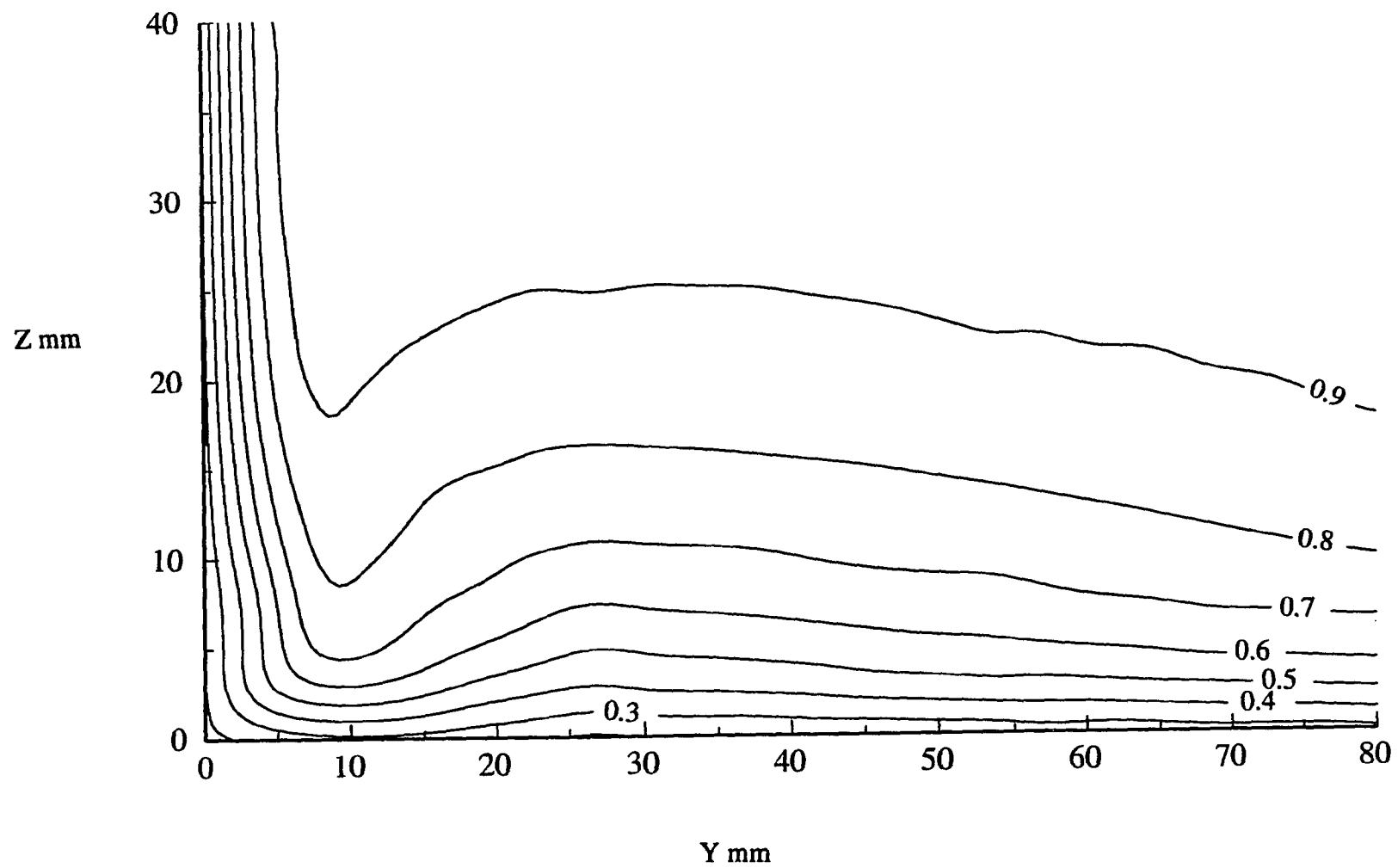


Figure (4.61) Total Pressure Contours at Station #1 ($X = 6.35$ mm, $X/C = 0.025$).

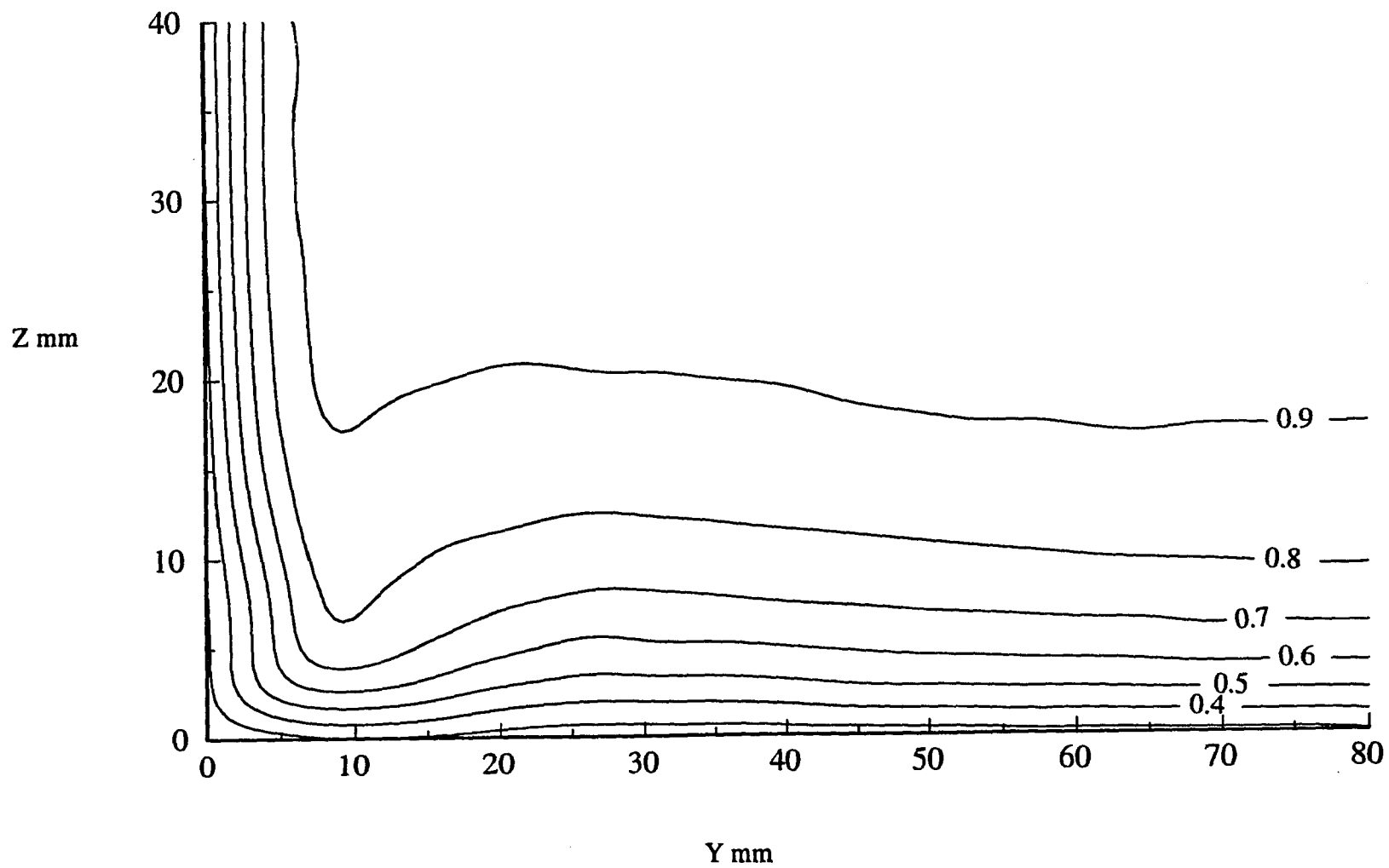


Figure (4.62) Total Pressure Contours at Station #2 ($X = 12.70$ mm, $X/C = 0.05$).

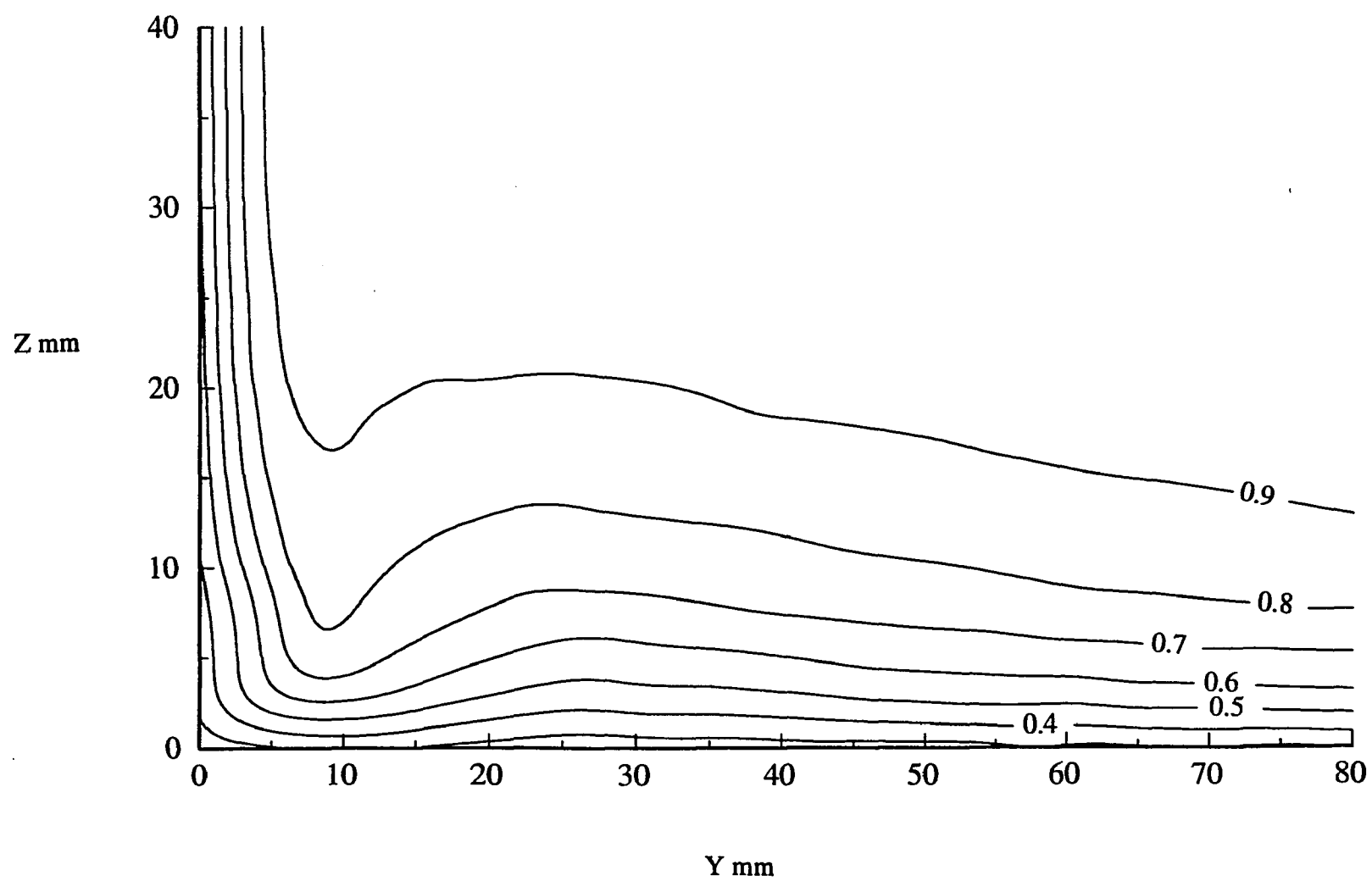


Figure (4.63) Total Pressure Contours at Station #3 ($X = 25.40$ mm, $X/C = 0.10$).

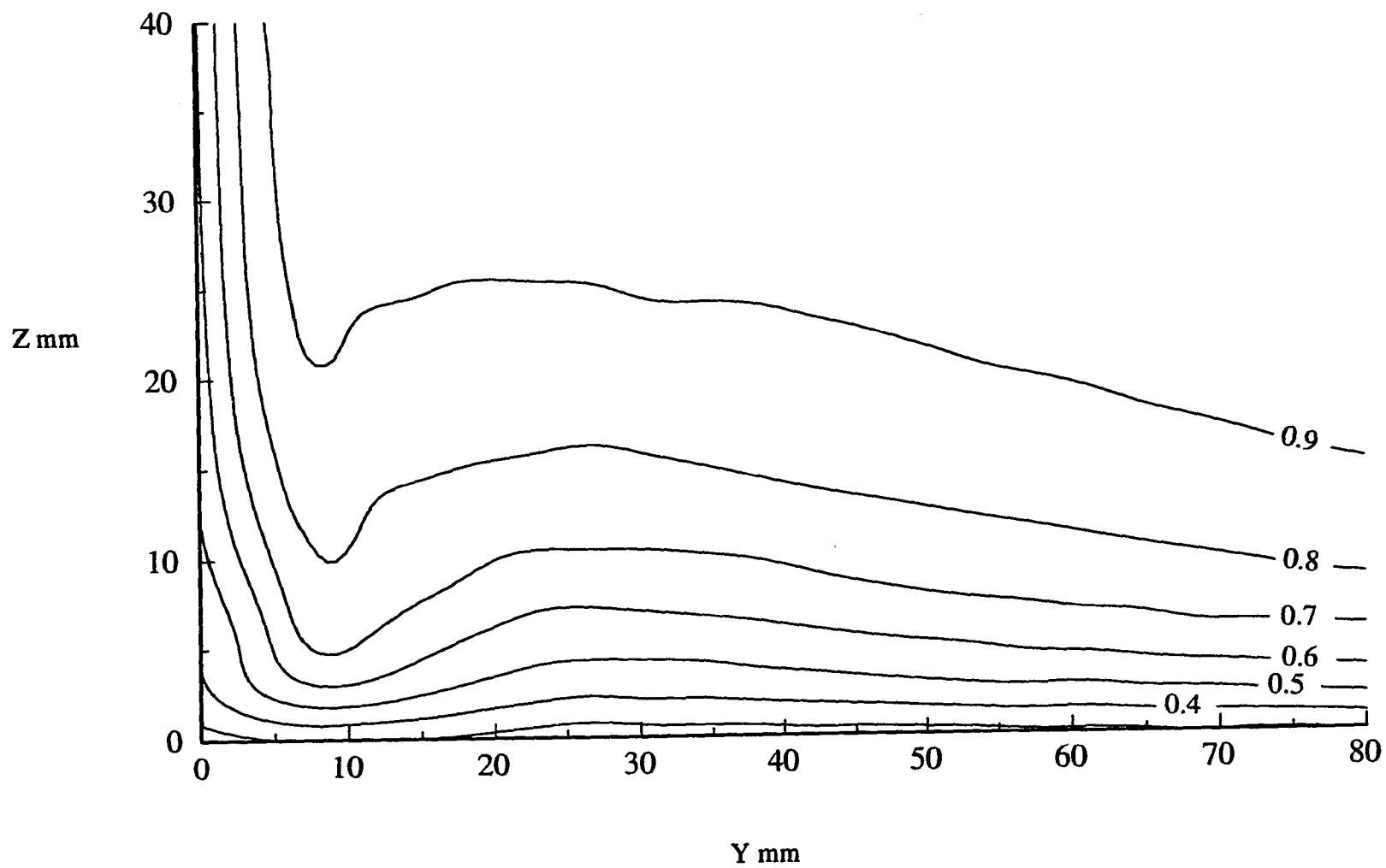


Figure (4.64) Total Pressure Contours at Station #4 ($X = 50.80$ mm, $X/C = 0.20$).

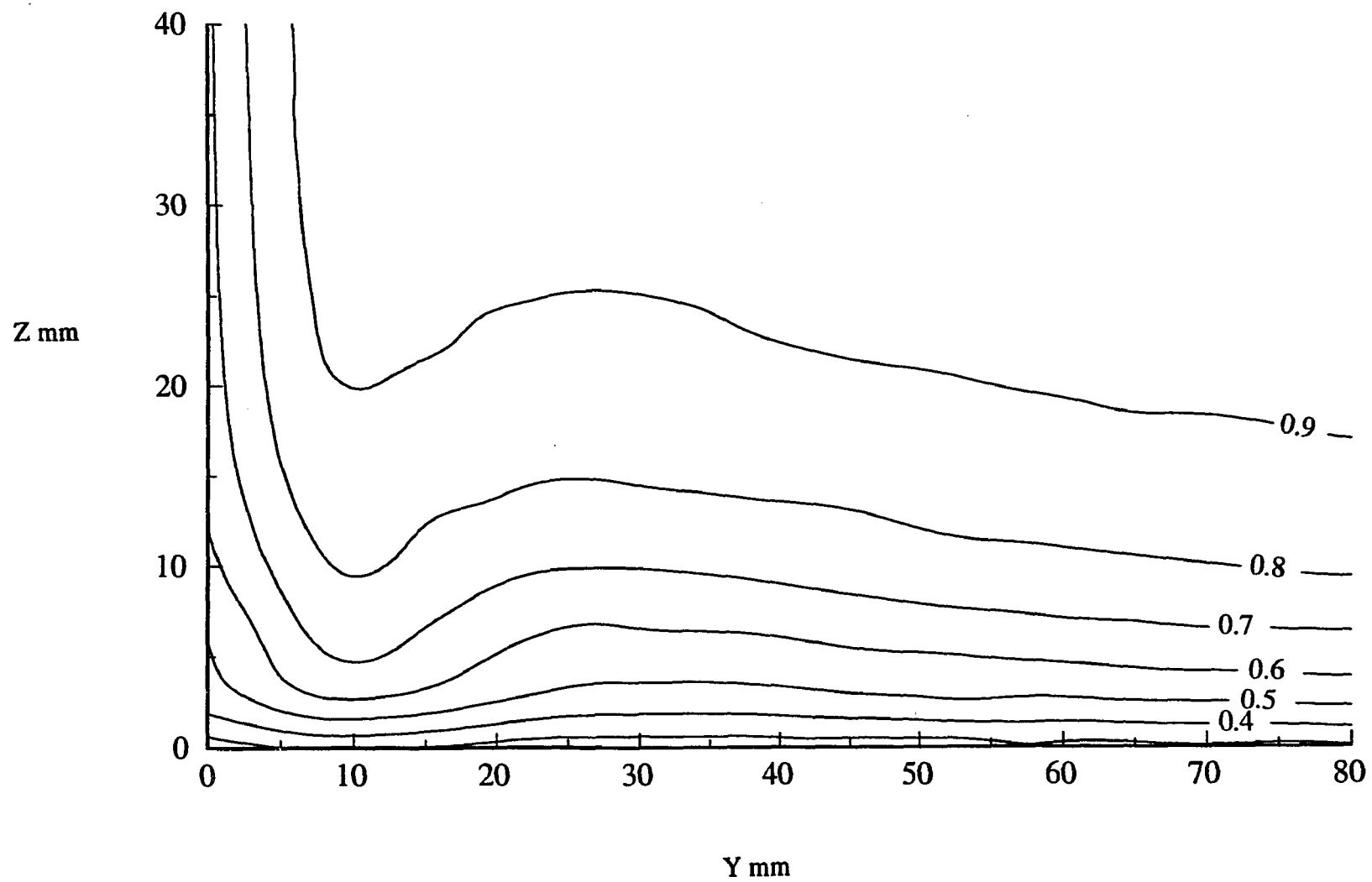


Figure (4.65) Total Pressure Contours at Station #5 ($X = 101.60$ mm, $X/C = 0.40$).

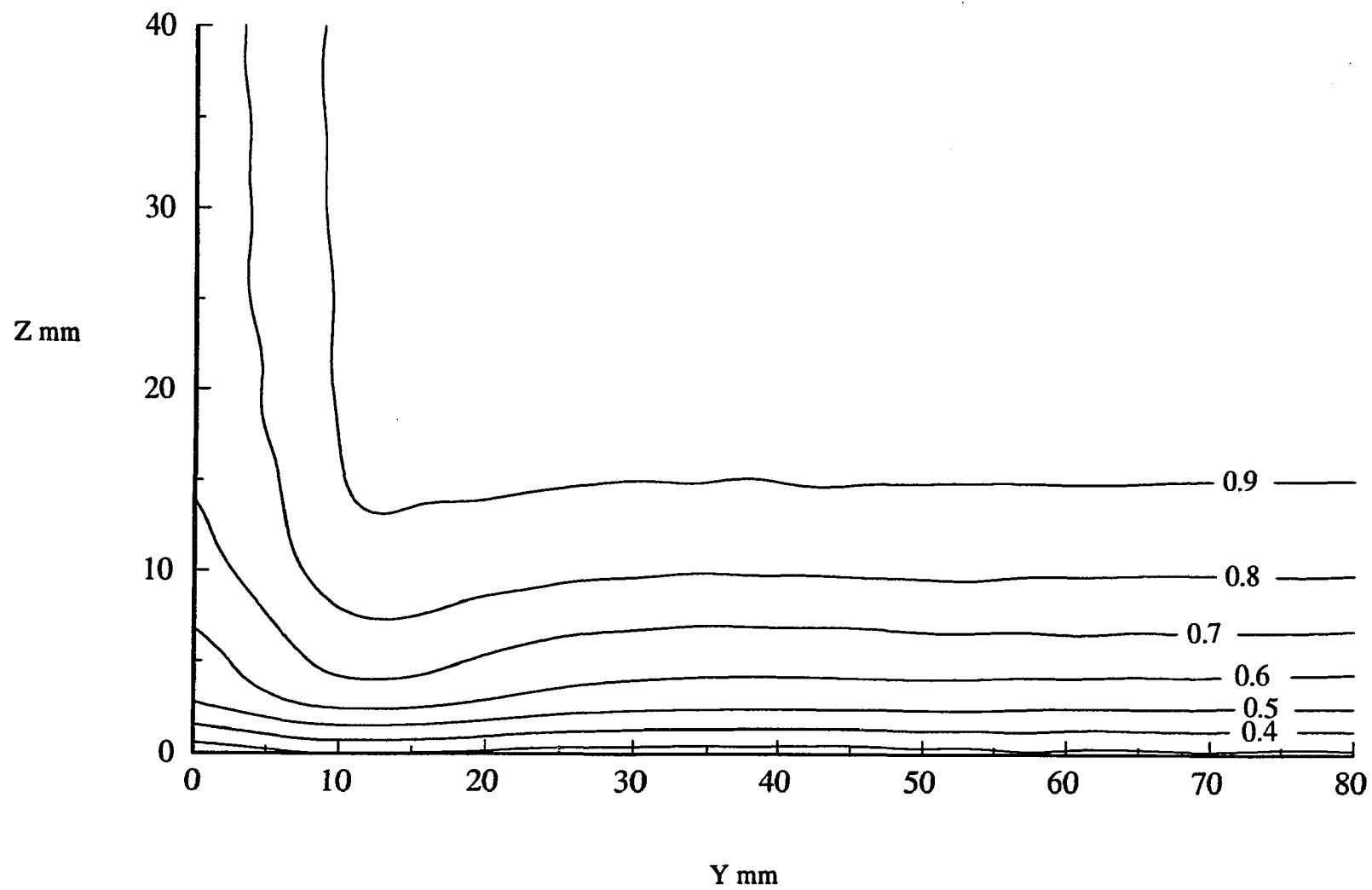


Figure (4.66) Total Pressure Contours at Station #6 ($X = 177.80$ mm, $X/C = 0.70$).

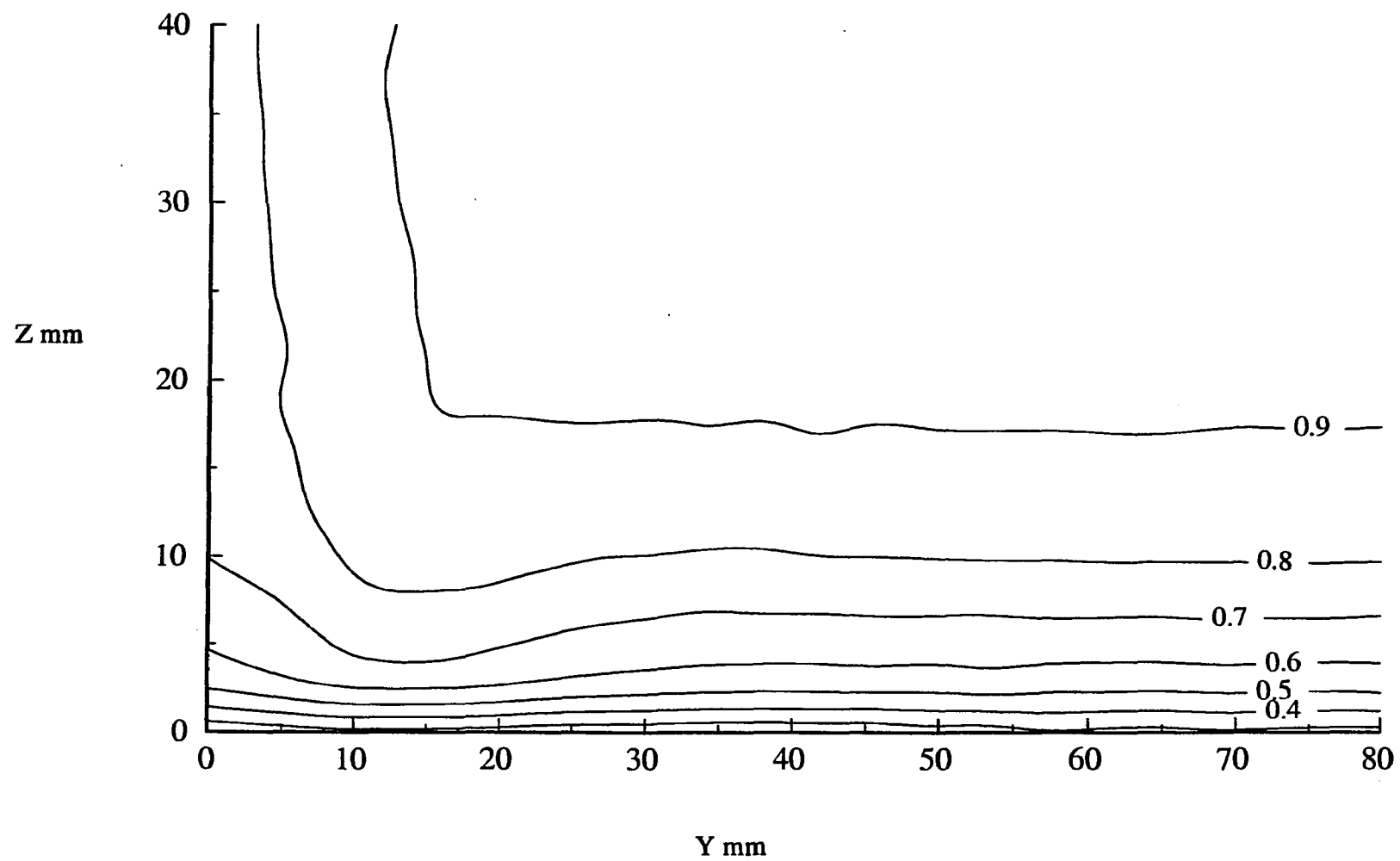


Figure (4.67) Total Pressure Contours at Station #7 ($X = 254.00$ mm, $X/C = 1.00$).

of the boundary layer thickness at that axial station.

Departing from the flat plate, the gradient of rise from C_{TPmin1} to C_{TPmax} increases in its steepness. This increase diminishes as the flow proceeds downstream and it ceases at axial location of approximately $X/C = 0.4$. Furthermore, the rate of fall from C_{TPmax} to C_{TPmin2} first increases in its steepness to a certain height above the flat plate, then it decreases at heights farther from the flat plate. The two values (i.e., C_{TPmax} and C_{TPmin2}) alleviate and become equal to that of the free stream. At traverses exterior to the flat plate boundary layer thickness, pressure profiles become similar to that of a two dimensional wake profile.

Total pressure variations perpendicular to the flat plate (i.e., variable Z) at different distances from the centerplane (i.e., constant Y) are shown in Figure (4.52) to Figure (4.59). It is evident that these profiles are distorted in comparison to the undisturbed two dimensional boundary layer profiles on the flat plate.

At axial stations closer to the trailing edge (Station #0, Station #1, Station #2, and Station #3; $X/C = 0$, $X/C = 0.025$, $X/C = 0.05$, and $X/C = 0.1$, respectively), there are three distinct total pressure profiles. The first profile, in regions far away from the centerplane, resembles that of the undisturbed two-dimensional boundary layer over a flat plate. With decreasing distances in Y , i.e., moving towards the centerplane, a second profile appears and total pressure profiles become "less full" as compared to the first profile. A third profile appears in the vicinity of the centerplane ($Y = 0$ mm to $Y \cong 7$ mm) with an inflection point. This "inflectional profile" is pronounced at the first four axial stations namely, Station #0, Station #1, Station #2, and Station #3 and it diminishes at Station #4. In other words, the "inflectional profiles" exist in the regions bounded by $X/C = 0$ to $X/C \cong 0.1$, $Y/C = 0$ to $Y/c \cong 0.03$ and $Z/C = 0$ to $Z/C \cong 0.04$ (i.e., between the centerplane, $X \cong 25$ mm, $Y \cong 7$ mm and $Z \cong 11$ mm).

The distortion of total pressure profiles are caused by the presence of the secondary flows. Secondary flows redistribute momentum content in regions under their domain. In other words, it transports fluid with a definite momentum content to regions with different momentum content. This redistribution of momentum causes the appearance of regions with total pressure values higher than that of, otherwise undisturbed, boundary layer at that level. It also causes the appearance of inflectional profiles (lower momentum than, otherwise undisturbed, boundary layer) and the appearance of the "less full" profiles.

The horseshoe vortex, which is the dominant transporter of momentum, on one side it transports low momentum fluid from regions near the flat plate away from it. This causes the appearance of the "less full" total pressure profiles. On the other side, it has a dual effect; the first it transports the high, near free stream, momentum fluid towards the plate and the second, it downwashes the low momentum blade wake towards the wall.

The corner vortex, bounded by the centerplane (region of low momentum wake) and the flat plate, has much smaller domain. It transports the low momentum wake fluid away from the centerplane. This causes the inflectional profile in the total pressure (see Figure (4.52) to Figure (4.55)). It also transports fluid from regions above the plate with relatively high momentum, towards the flat plate. The steepness of inflection (severity of distortion) reduces as the flow proceeds downstream and it diminishes at axial location of $X/C \cong 0.2$. This might suggest that the corner vortex gets weaker and dissipates as the flow proceeds downstream and ceases to exist at axial location of $X/C \cong 0.4$. It is also interesting to note that the range, in which the "less full" profiles of total pressure exist, grows wider as the flow proceeds downstream. This suggest that the horseshoe vortex gets wider, and hence its domain, as it proceeds downstream.

Total pressure contours (also called Bernoulli's surfaces or contours) grant a better understanding of secondary flow effects. These are contours on which the total pressure is constant. The distortions of these contours gives a tangible interpretation of the direction, strength and size of secondary flow currents. These contours were extracted from total pressure values presented before in Figure (4.44) to Figure (4.59). The corresponding total pressure contours for the eight axial stations are presented in Figure (4.60) to Figure (4.67).

It is evident from these figures that total pressure contours are distorted in comparison to the flat plate boundary layer and to the blade wake profiles. The largest of the distortions (in size) is a result of the counter-clockwise rotation (looking downstream) of the horseshoe vortex. Parallel to the flat plate, the upward bulges in these contours are a result of the upwash current of the horseshoe vortex. In this region, the horseshoe vortex convects low momentum fluid, from regions near the flat plate, upwards. Hence, it thickens the boundary layer and causes the upward bulges. At larger distances in Y , the flat plate boundary layer recovers its undisturbed two-dimensional thickness, and the contours become parallel to the flat plate. The downward bulges in these contours are a result of the downwash effect of the horseshoe vortex. The horseshoe vortex convects the high momentum fluid, from regions above the flat plate, downward causing a reduction in the boundary layer thickness.

It is interesting to note another bulge in the contours in regions near the centerplane (see Figures (4.60), (4.61) and (4.62) the quadrant bounded by $Y < 10\text{mm}$ and $Z < 10\text{mm}$). This outward bulge in contours is believed to be caused by the action of the corner vortex. The clockwise rotation of the corner vortex on one side, carries the low-momentum wake fluid away from the centerplane, causing otherwise parallel contours to bulge. The other side of the corner vortex has the same sense of direction

as that of the horseshoe vortex and it strengthens its effect in shortening the thickness of the flat plate boundary layer. At heights far above the flat plate ($Z > 20\text{mm}$), these contours become parallel to the centerplane. The bulges caused by the corner vortex are prominent in axial stations near the trailing edge. It can be seen clearly at Station #0, Station #1, Station #2 and Station #3 ($X/C = 0$, $X/C = 0.025$, $X/C = 0.05$ and $X/C = 0.1$, respectively). It diminishes at Station #4 ($X/C = 0.2$), an indication that the corner vortex gets dissipated by this downstream axial location. At station #0 ($X/C = 0$), in regions far from the corner formed by the centerplane and the flat plate, total pressure contours are parallel to the centerplane (blade two-dimensional wake region) and parallel to the flat plate (flat plate two-dimensional boundary layer). It is seen that spacing between blade wake contours are smaller than that of flat plate boundary layer contours. This is a result of a smaller blade wake width as compared to a thicker boundary layer on the flat plate. As the flow proceeds downstream, the spacing between blade wake contours increase at a faster rate than that of the flat plate boundary layer, indicating a faster wake recovery than boundary layer growth. For example, at Station #0, the minimum contour value parallel to the centerplane is 0.2. At Station #4, the minimum value parallel to the centerplane is 0.7 and by axial Station #7 it is 0.8. This is an indication of a wake recovery, i.e., wake deficit is reduced. It is also interesting to note that the distance (in Y), of the maximum value in total pressure contours, from the centerplane increases as the flow proceeds downstream. This is an indication of growth in the blade wake width. Proceeding downstream of the blade trailing edge, total pressure contours become less distorted. This is an implication of flow recovery and a sign that secondary flows become weaker.

In summary, the distortion of total pressure contours, caused by the horseshoe vortex, can still be seen at downstream axial location of one chord length ($X/C = 1$), but the distortion is much weaker at this axial location. As far as the corner vortex, its

effect on total pressure contours diminishes by axial location of $X/C = 0.2$. There is no evidence, as far as the distortion in total pressure contours is concerned, of the stress induced vortex reported by Hazarika and Raj [1]. It is possible that the size and strength of this vortex is small (an order of magnitude less than the horseshoe vortex [1]) and could not be detected by the total pressure probe. It is also possible that this stress induced vortex has dissipated by the time it reached the trailing edge.

4.5 Axial Wake Centerline Static and Total Pressure Variations:

Blade wake centerline static pressure coefficient (C_p) variation, at various spanwise (Z) distances, with the streamwise (X) distances is shown in Figure (4.68). As it was mentioned in Section 4.4 and sub-section 4.4.1, static pressure coefficient (C_p) is at its highest in the vicinity of the blade trailing edge and it reduces in the streamwise (X -) direction. Except for stations #1 and #2, there is no significant static pressure variation in the spanwise (Z -) direction. For Stations #1 and #2, the maximum increase of C_p in the spanwise (Z -) direction is of the order of 7%. One can conclude that no significant static pressure variation, in the spanwise (Z) direction would result when a two-dimensional wake is submerged in a boundary layer.

Blade wake centerline total pressure variation, at various spanwise (Z) distances, with the streamwise (X) distances is shown in Figure (4.69). A general trend is clear for all spanwise (Z) distances above the flat plate. Wake centerline starts with a minimum value of approximately 0.12 at the trailing edge location ($X/C = 0.0$). It increases in the streamwise (X -) direction to a local asymptotic value, for that height, at one chord length ($X/C = 1.0$). It can be noted from this figure that the wake centerline total pressure recovers faster at heights away from the flat plate, as compared to those close to the flat plate. In other words, the wake deficit recovery

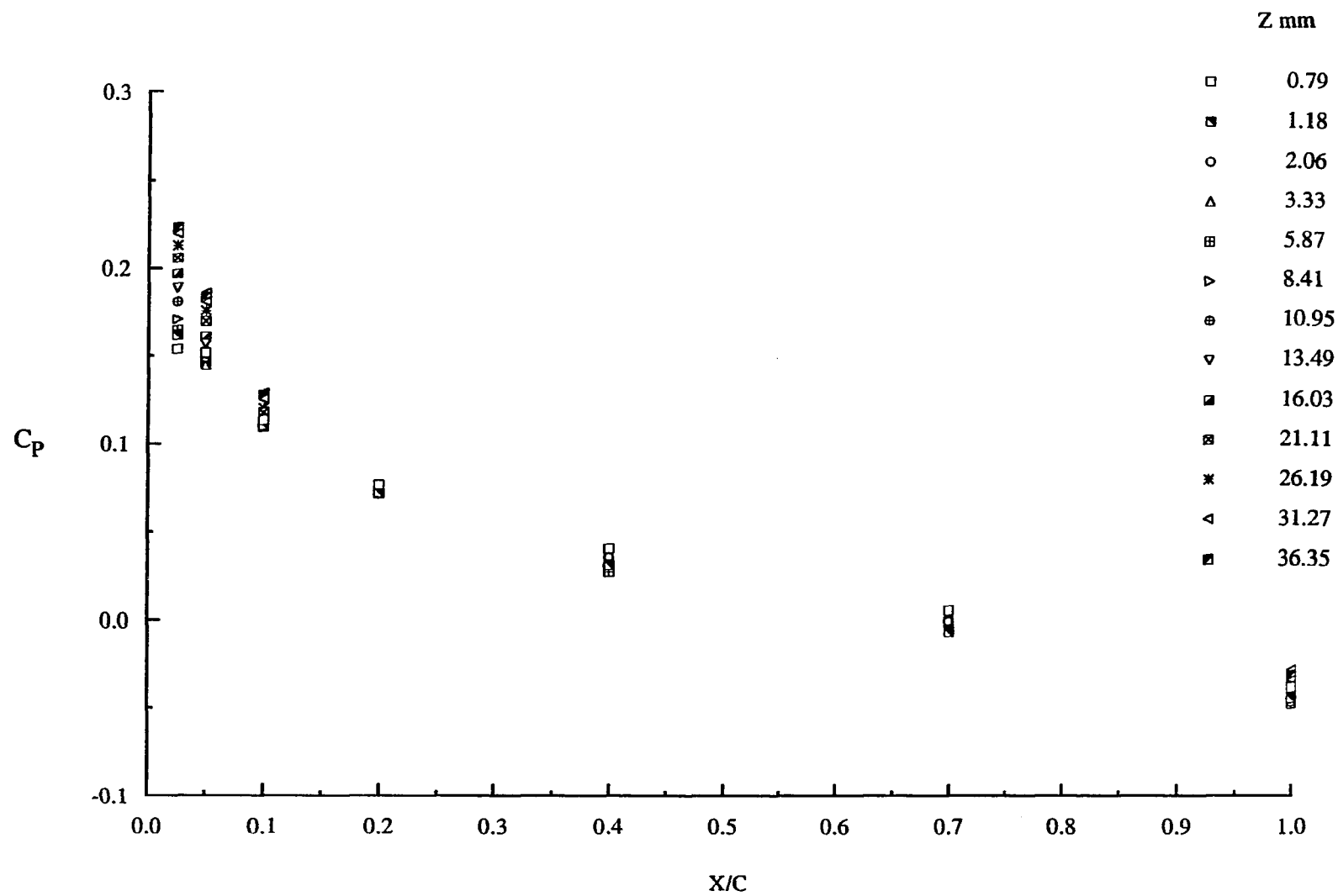


Figure (4.68) Wake Centerline Static Pressure Variation.

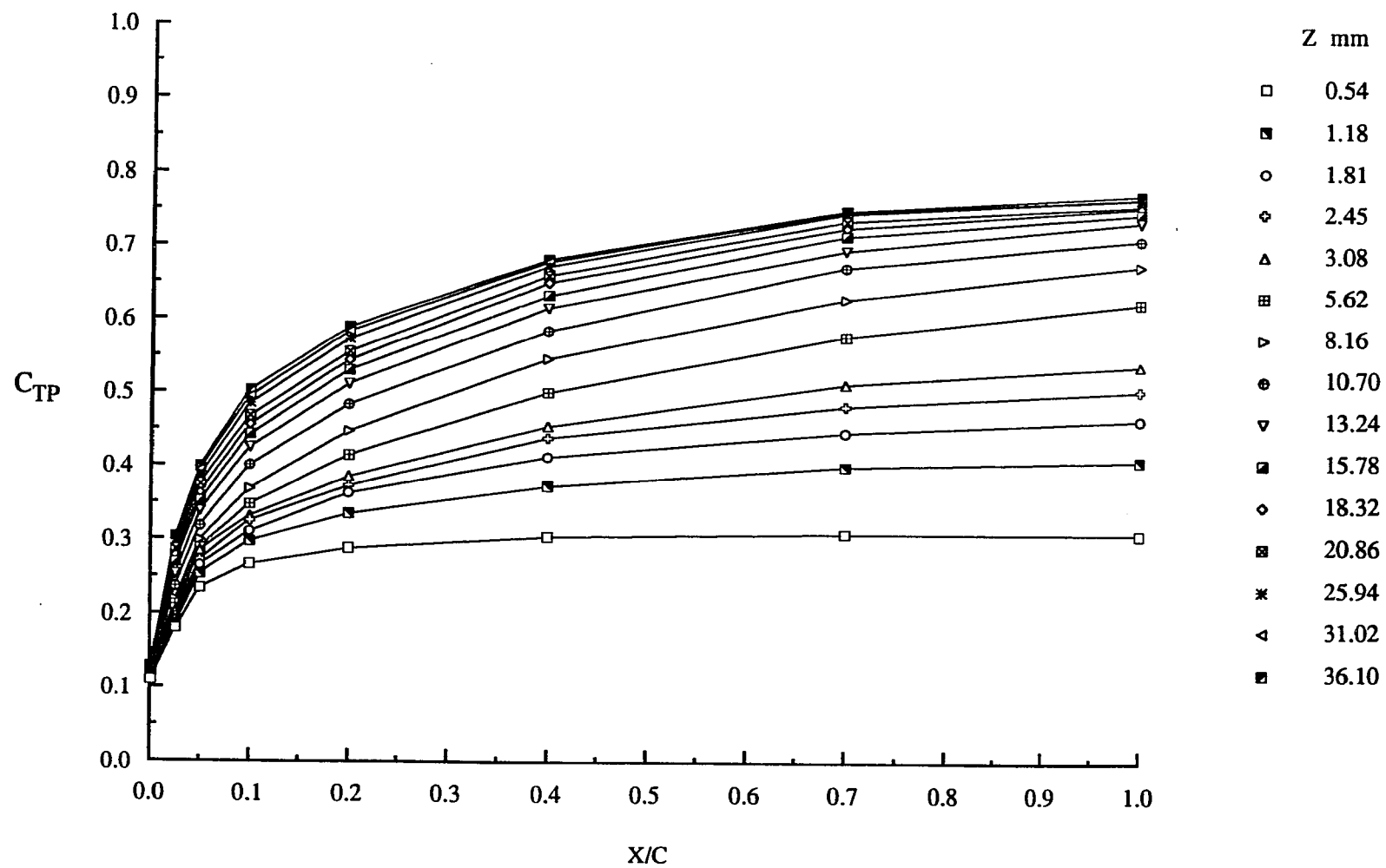


Figure (4.69) Wake Centerline Total Pressure Variation.

rate, in the streamwise (X-) direction, increases as one moves away from the flat plate. This suggests that a blade wake submerged in a boundary layer recovers at a slower rate than the undisturbed two-dimensional wake.

4.6 Mean Velocity Measurements:

The three components of mean velocity, measured by the five hole probe, downstream of the blade trailing edge, are presented in this section. Near the blade trailing edge, there was flow unsteadiness. This was probably caused by the sudden removal of the no-slip condition at the blade finite-thickness trailing edge and the consequent interaction of the two legs of the horseshoe vortex. As a result, instruments could not be stabilized for steady state data collection. However, this phenomena persisted up to Station #1 ($X/C = 0.025$). Therefore, data of reasonable accuracy could only be obtained beyond Station #1. The five hole probe calibration procedure, apparatus and discussion of the results are provided in Appendix (A). Calibration characteristics varied with Reynolds number and were different from the calibration data supplied by the manufacturer. Reynolds number effect was pronounced at low Reynolds numbers (Reynolds number less than 2500, based on the probe sensing head diameter) and at high pitch angles (greater than +20 degrees and less than -20 degrees). Calibration data obtained from two calibration sets were averaged. The averaged values were fitted with a fifth order spline curves and used in a FORTRAN program written to process the measured data. The FORTRAN program is presented in Appendix (B).

The variations of the streamwise component (V_x) of the mean velocity, parallel to the flat plate (variable Y) and at different heights above the flat plate (constant Z), for six downstream axial stations are shown in Figure (4.70) to Figure (4.75). It is

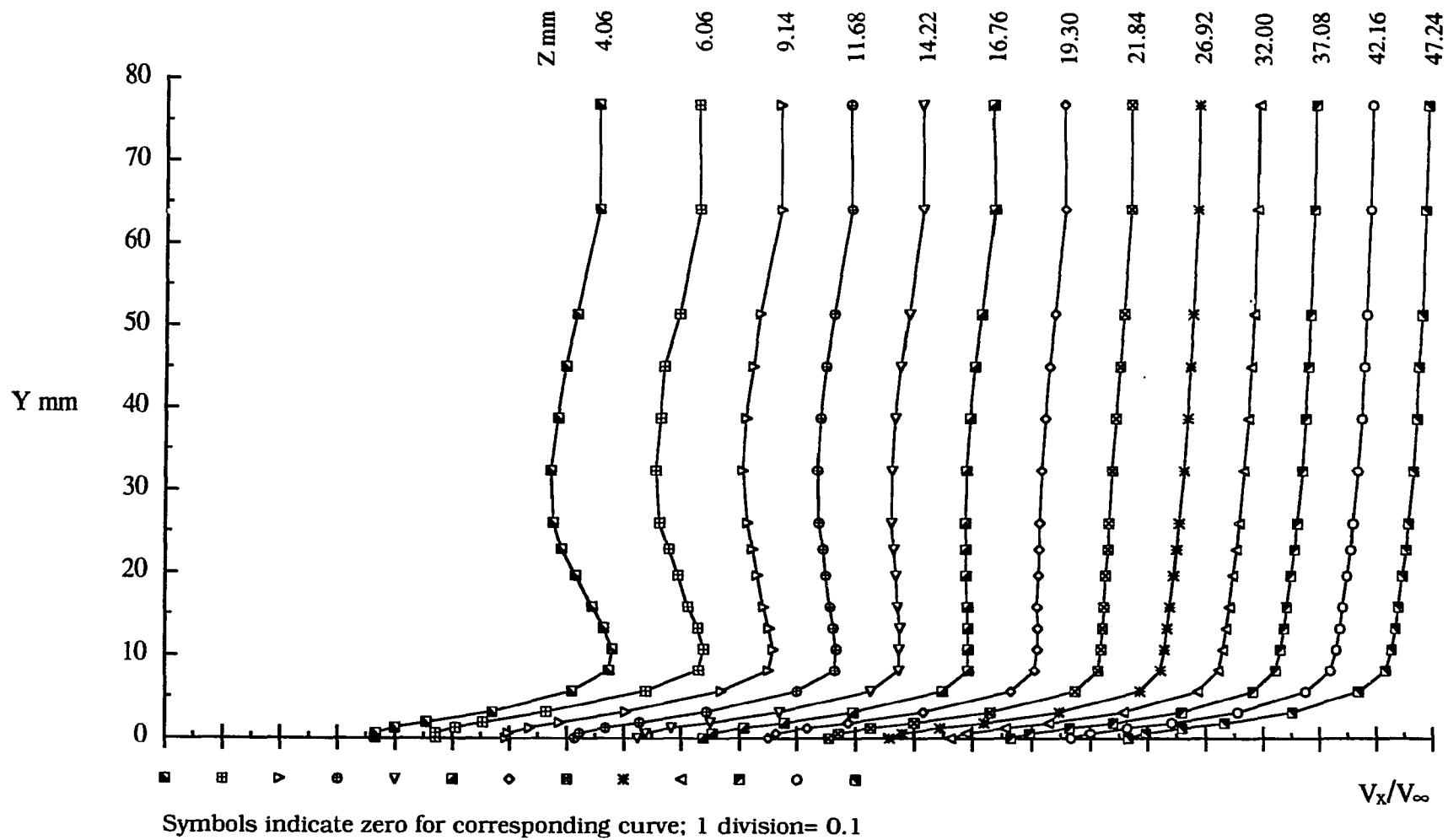


Figure (4.70) Streamwise Velocity Component Variation Parallel to the Flat Plate at Station #2 ($X= 12.70$ mm, $X/C = 0.05$).

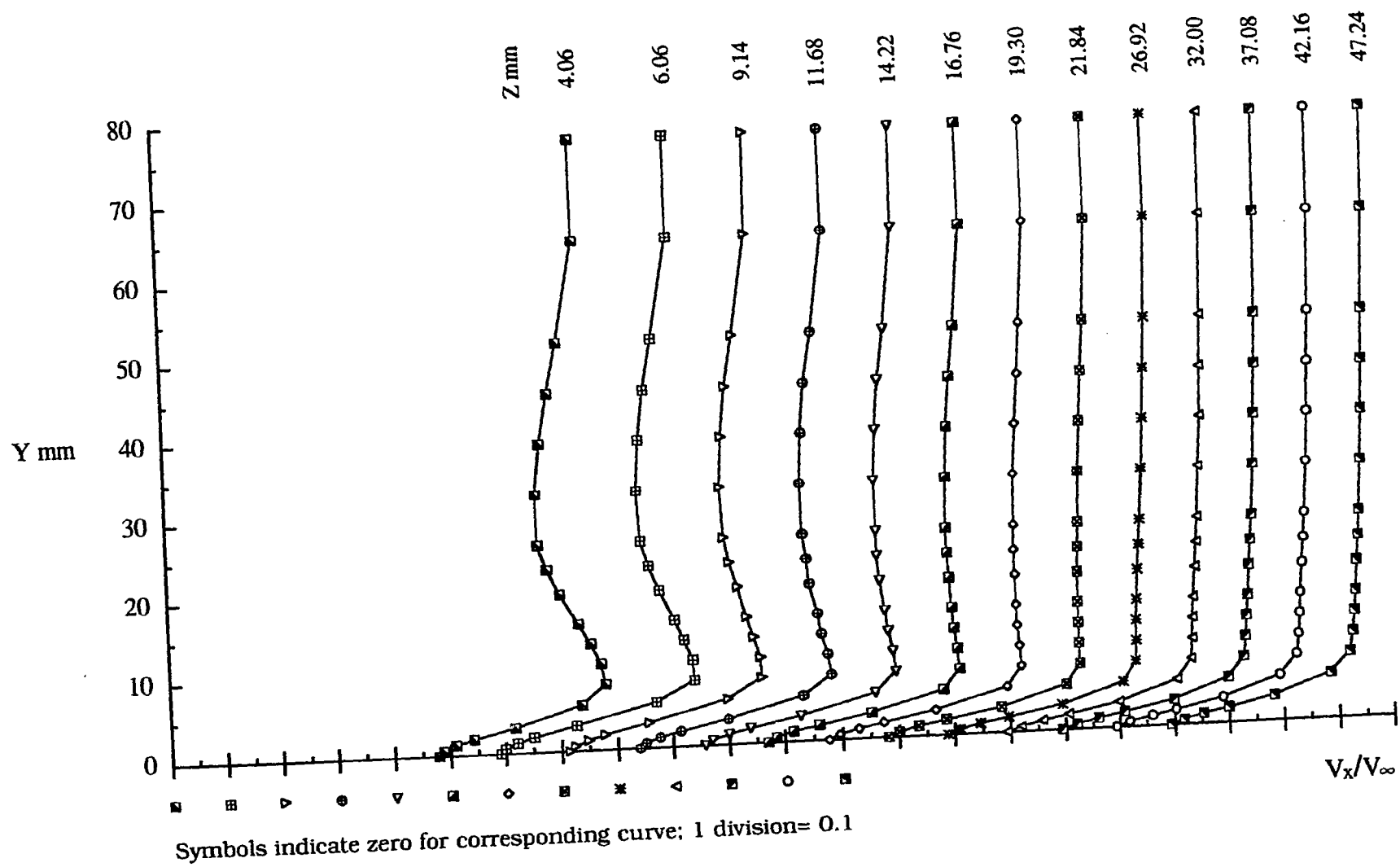


Figure (4.71) Streamwise Velocity Component Variation Parallel to the Flat Plate at Station #3 ($X = 25.40$ mm, $X/C = 0.10$).

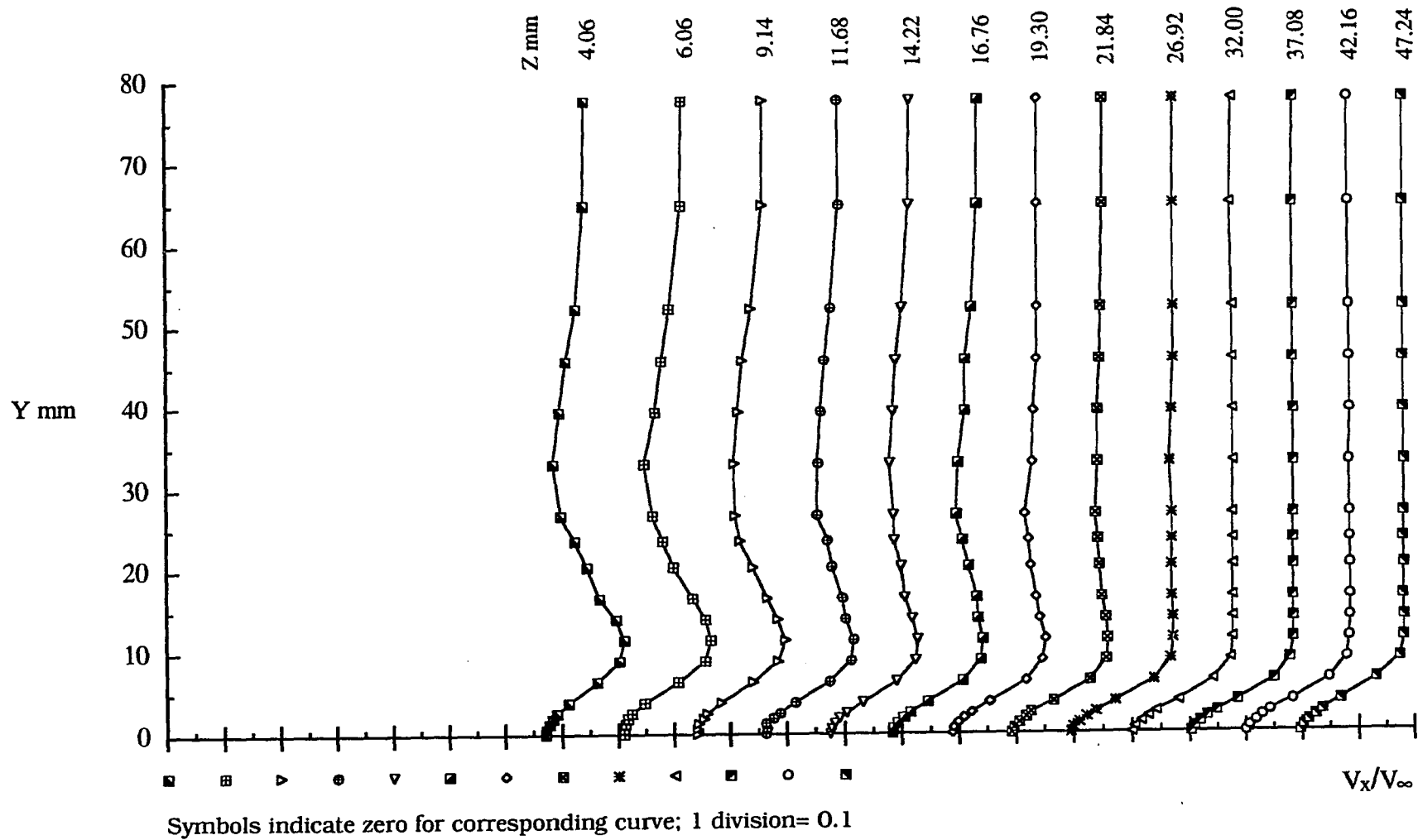


Figure (4.73) Streamwise Velocity Component Variation Parallel to the Flat Plate at Station #5 ($X = 101.60$ mm, $X/C = 0.40$).

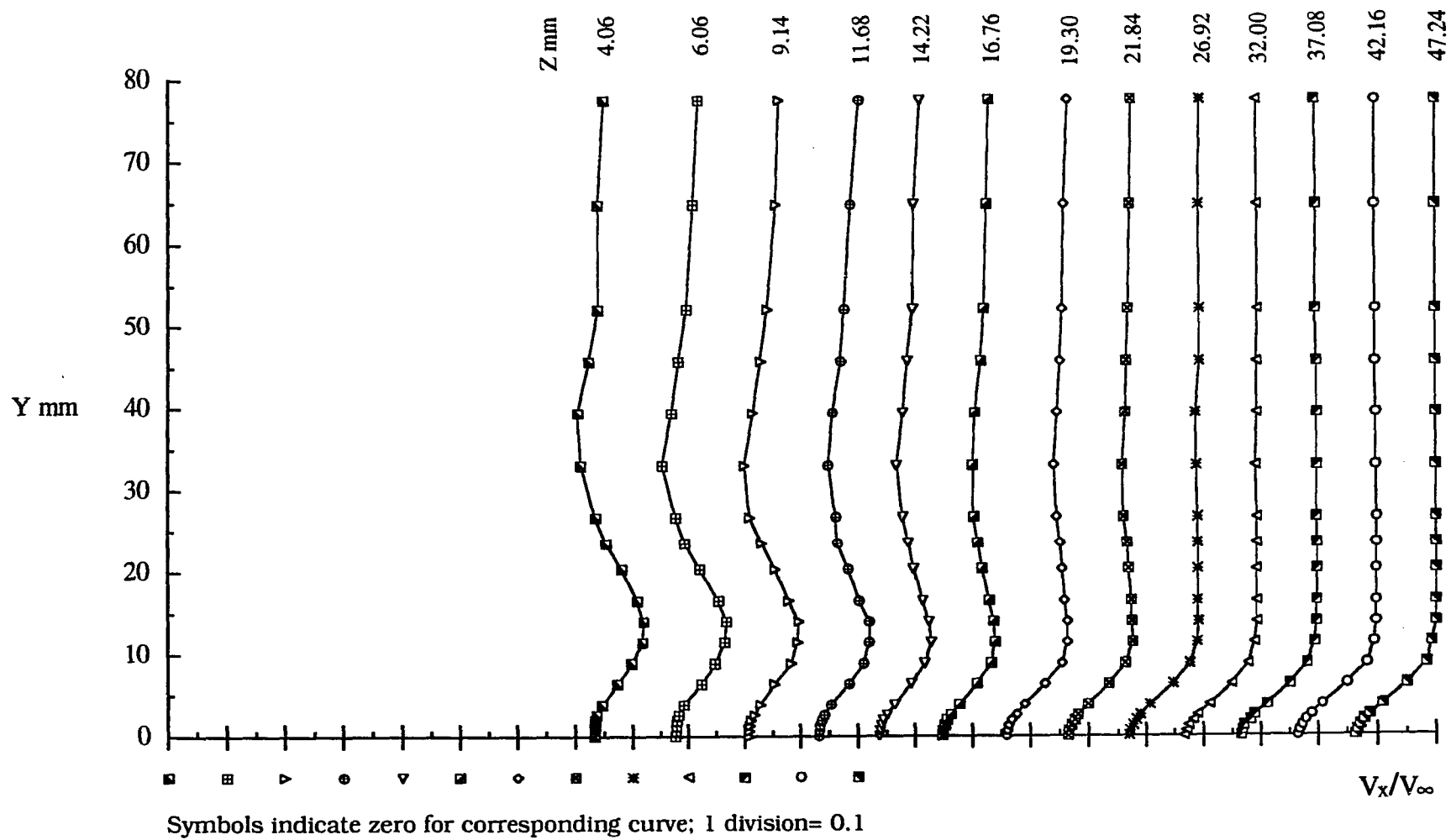


Figure (4.74) Streamwise Velocity Component Variation Parallel to the Flat Plate at Station #6 ($X = 177.80$ mm, $X/C = 0.70$).

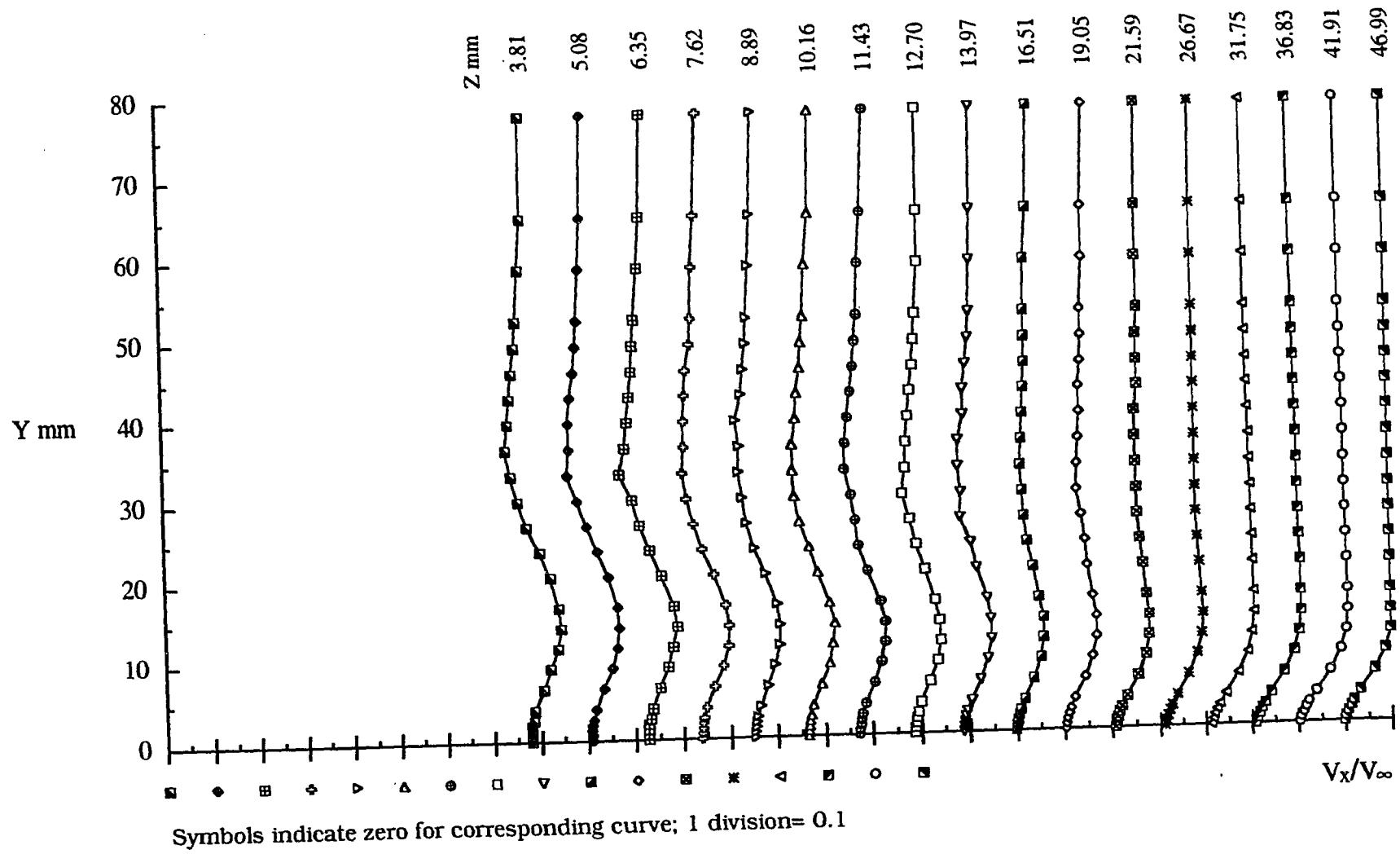


Figure (4.75) Streamwise Velocity Component Variation Parallel to the Flat Plate at Station #7 ($X = 254.00$ mm, $X/C = 1.00$).

observed that the shape of streamwise velocity component profiles closely resemble the corresponding shape of total pressure profiles. Therefore, the discussion of the total pressure profiles and contours, presented in sub-section 4.4.2 should apply equally well to the streamwise velocity results.

Secondary flow velocity vectors at six axial stations, namely; Station #2, Station #3, Station #4, Station #5, Station #6, and Station #7, are shown in Figure (4.76) to Figure (4.81). The length of each vector in these figures was calculated from: $V_s = (V_y + V_z)^{0.5}$. Where V_y is the yaw velocity and V_z is the pitch velocity. The direction of these vectors is equal to $\tan^{-1} (V_y/V_z)$ with respect to the Y-direction. Velocity vectors near the flat plate ($Z/C < 0.005$ i.e., when the probe was resting on the flat plate) has been removed from these figures. Due to the presence of the probe near the plate, the flow accelerates (wall proximity effects), resulting in an increased and unrealistic value in the pitch velocity V_z .

These figures clearly show, as inferred from the total pressure profiles and contours, the presence of the horseshoe vortex with an anti-clockwise rotation (looking downstream). The horseshoe vortex has an elliptical shape and its size scales that of the boundary layer thickness of the flat plate. As it was postulated before, from the distortions in total pressure profiles and contours, the horseshoe vortex has the dominant effect. On one side, it transports the low momentum fluid from near the flat plate surface and away from it. On the other side, it has a dual contribution. Firstly, it transports the high momentum, near free stream, fluid downward towards the flat plate surface. Secondly, it downwashes the low momentum fluid, in the blade wake, to the flat plate surface.

At Station #2 ($X/C = 0.05$) and Station #3 ($X/C = 0$), there are traces of the corner vortex (see Figure (4.76) and Figure (4.77) at $Y/C \cong 0.03$, i.e., $Y = 7$ mm).

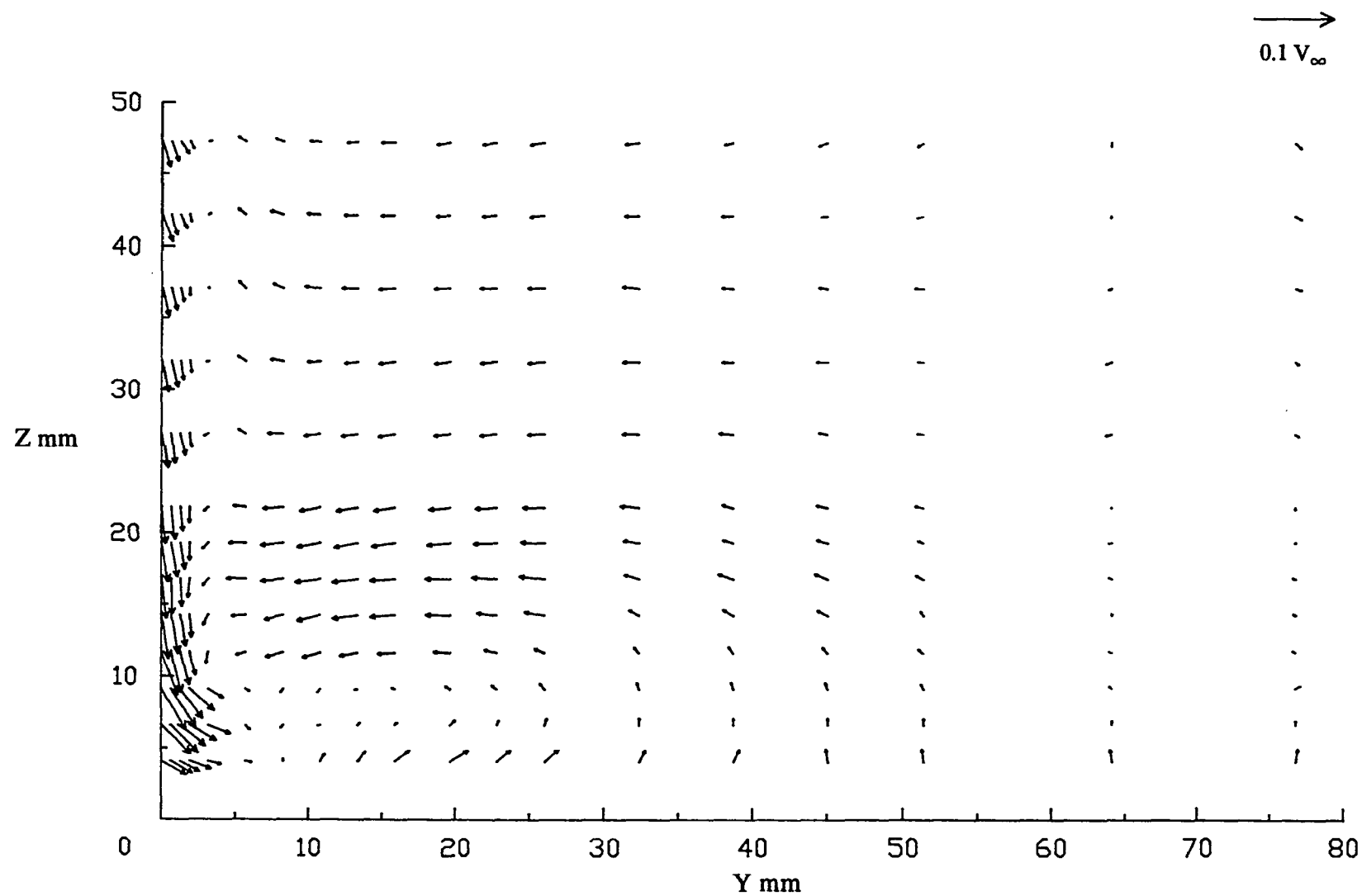


Figure (4.76) Secondary Flow Velocity Vectors at Station #2 ($X = 12.70$ mm, $X/C = 0.05$).

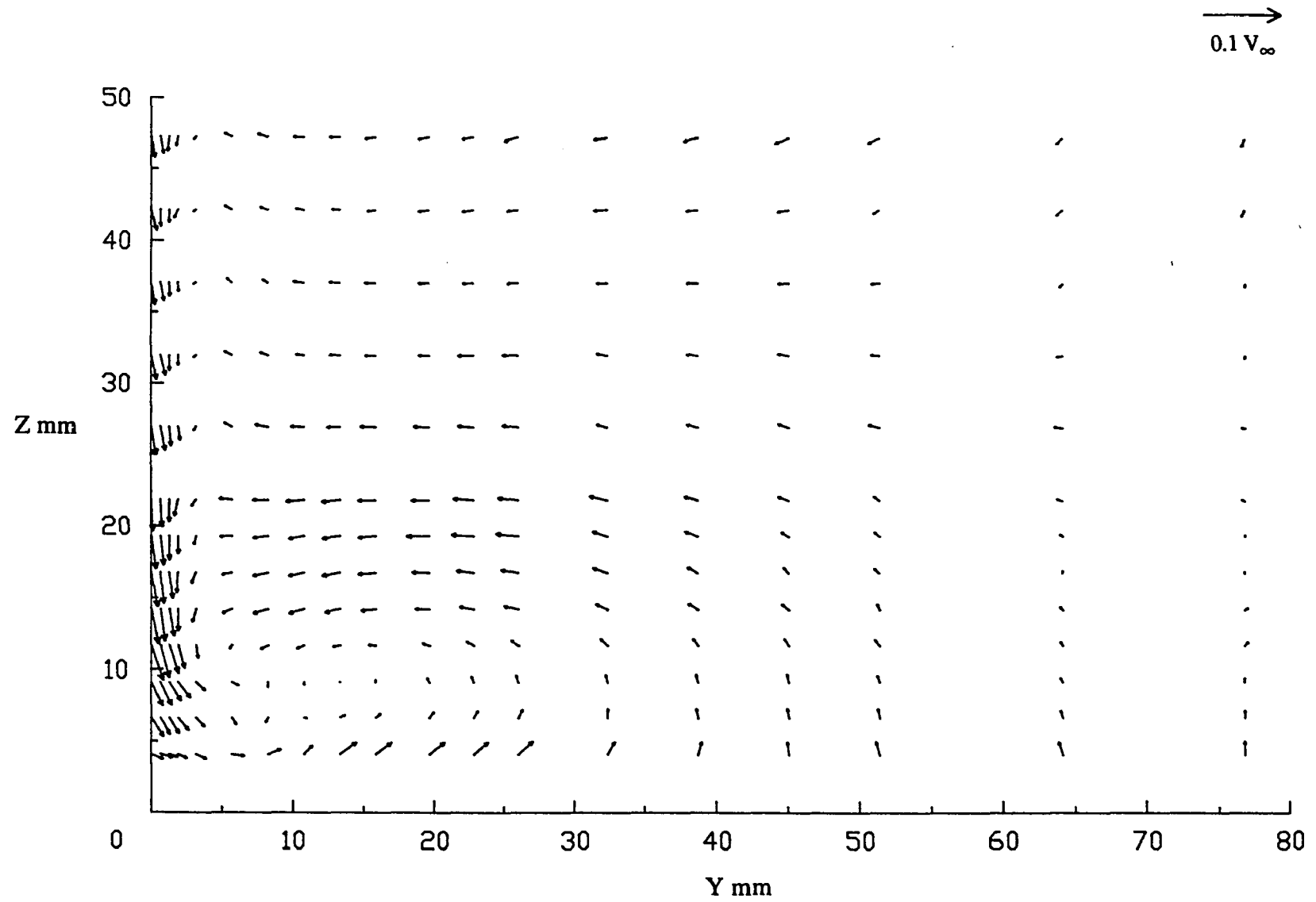


Figure (4.77) Secondary Flow Velocity Vectors at Station #3 ($X = 25.40$ mm, $X/C = 0.10$).

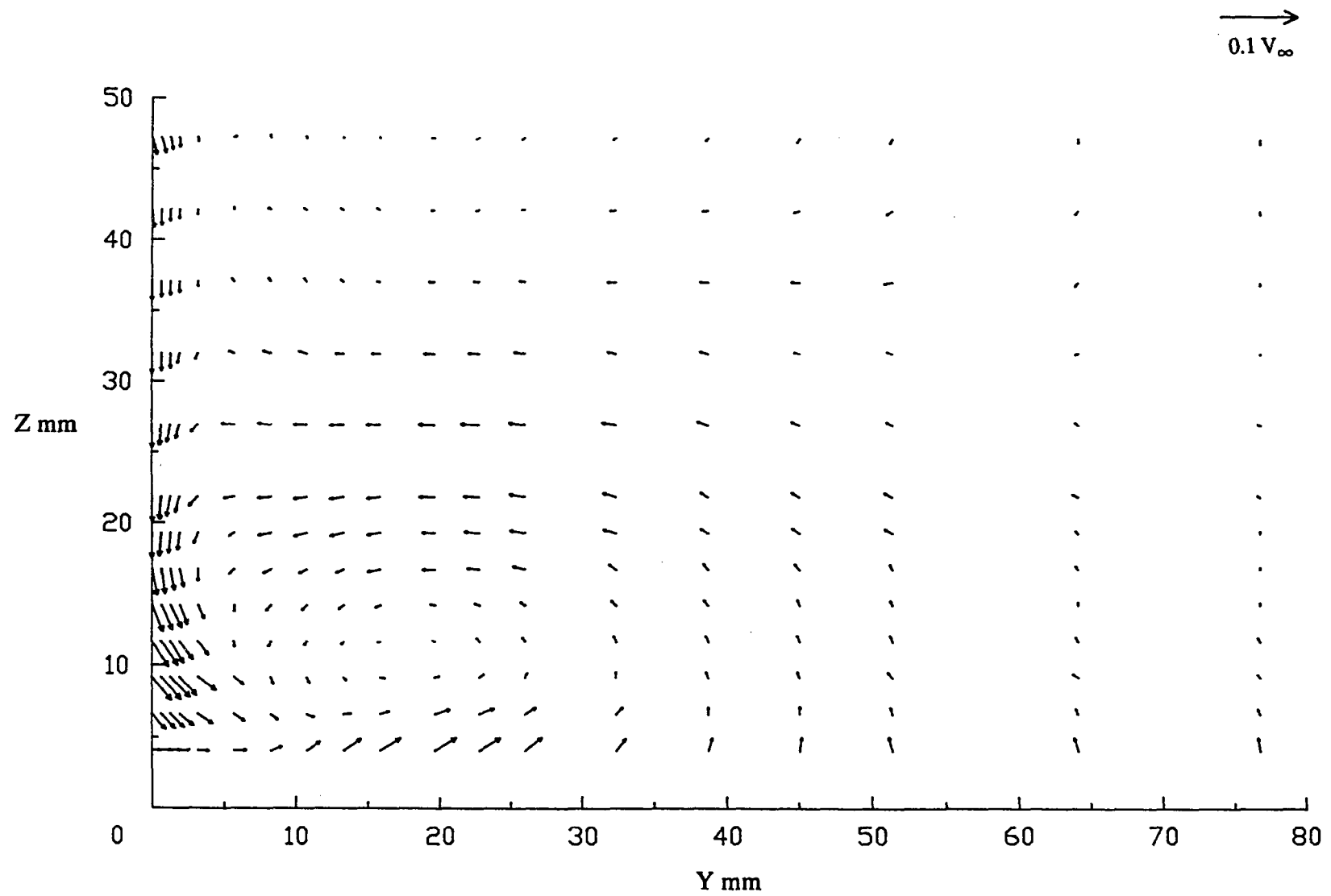


Figure (4.78) Secondary Flow Velocity Vectors at Station #4 ($X = 50.80$ mm, $X/C = 0.20$).

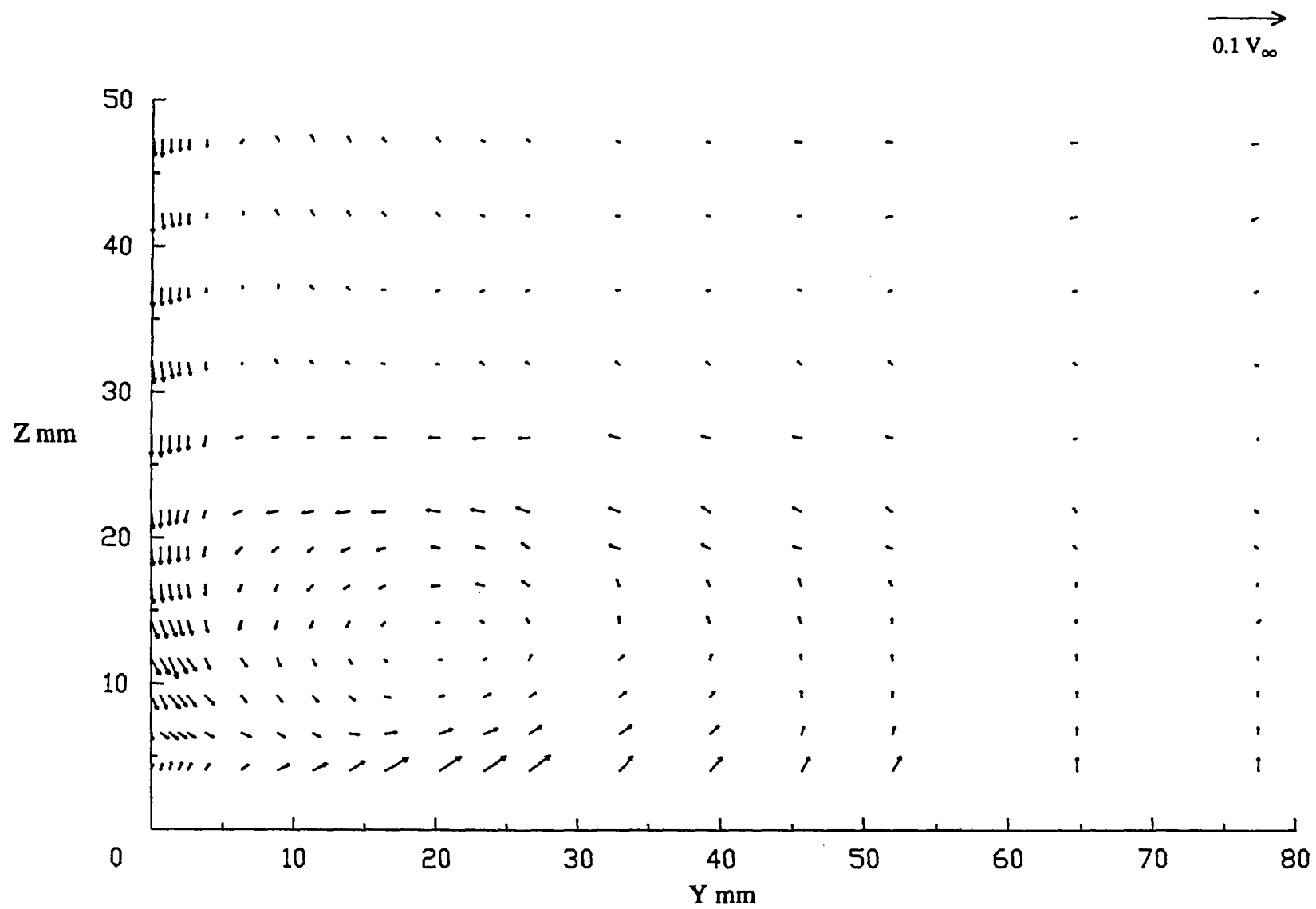


Figure (4.79) Secondary Flow Velocity Vectors at Station #5 ($X = 101.60$ mm, $X/C = 0.40$).

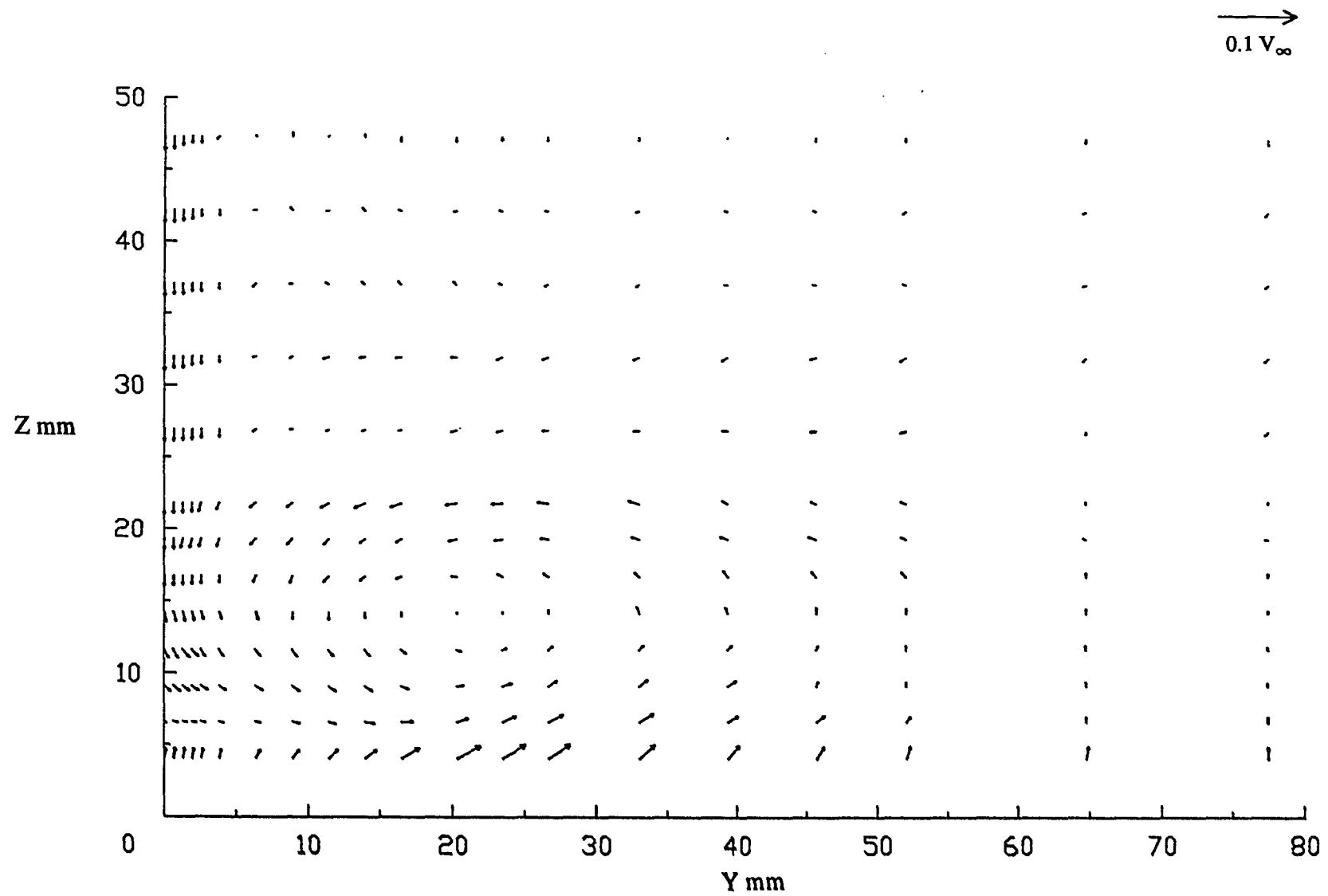


Figure (4.80) Secondary Flow Velocity Vectors at Station #6 ($X = 177.80$ mm, $X/C = 0.70$).

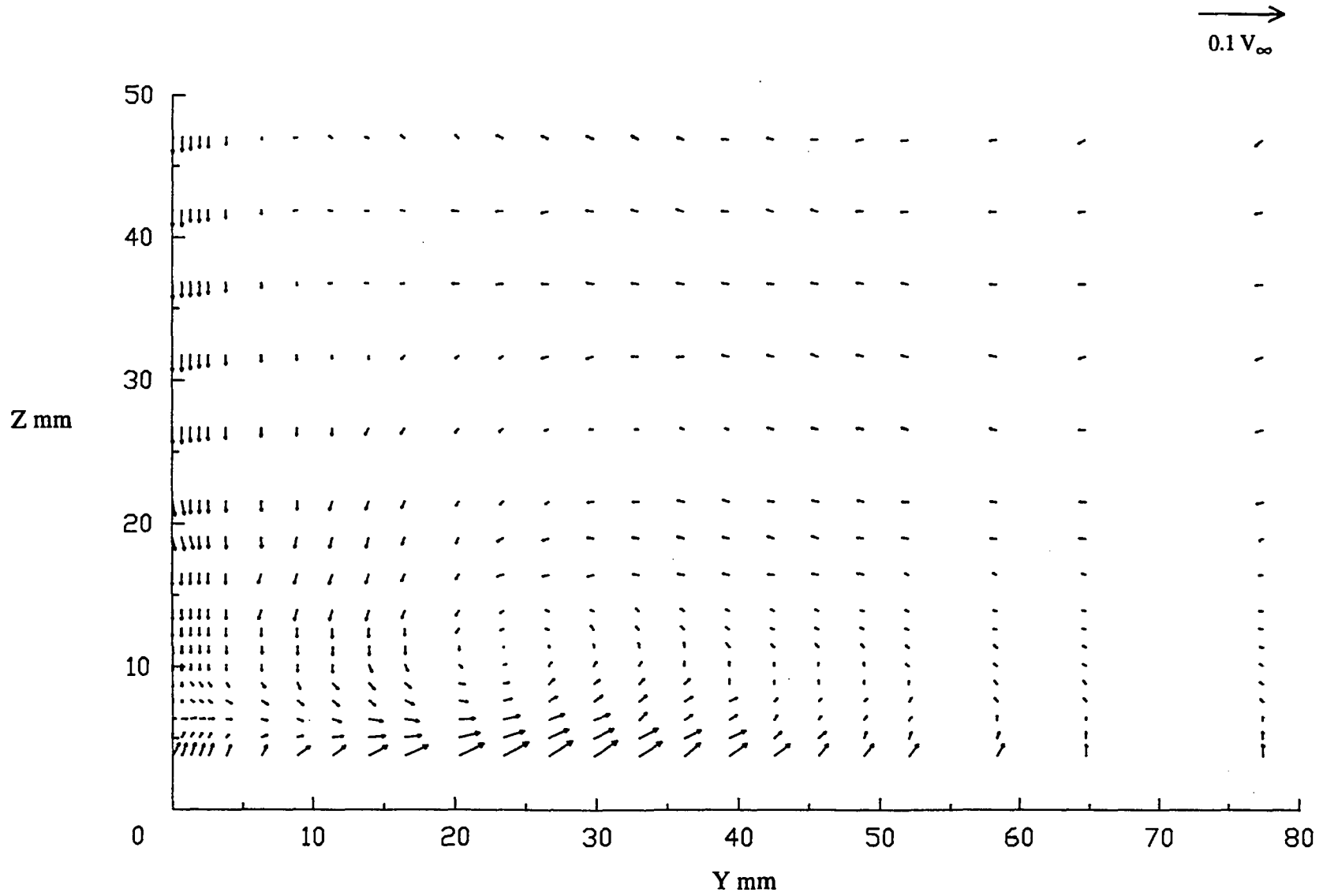


Figure (4.81) Secondary Flow Velocity Vectors at Station #7 ($X = 254.00$ mm, $X/C = 1.00$).

The presence of the corner vortex is partially masked by the strongly downwashed wake. The weak clockwise vectors of the corner vortex is seen at these two stations only, which supports earlier postulation that the corner vortex gets dissipated by axial location of $X/C \cong 0.2$ (i.e., Station #4). This vortex system is depicted schematically in Figure (4.82)

As the flow proceeds downstream, the length of the secondary flow velocity vectors gets shorter. This is an indication that the horseshoe vortex gets attenuated and diffused with increasing axial distances from the blade trailing edge. Also, the horseshoe vortex moves away from the centerplane and away from the flat plate surface as it proceeds downstream. The apparent location of its core moves away from the centerplane at a faster rate than its movement away from the flat plate surface. For example at Station #2 ($X/C = 0.05$) the approximate location of the core is at $Y \cong 13$ mm and $Z/C \cong 8$ mm. At Station #7 ($X/C = 1$) it is at $Y \cong 25$ mm and $Z \cong 11$ mm.

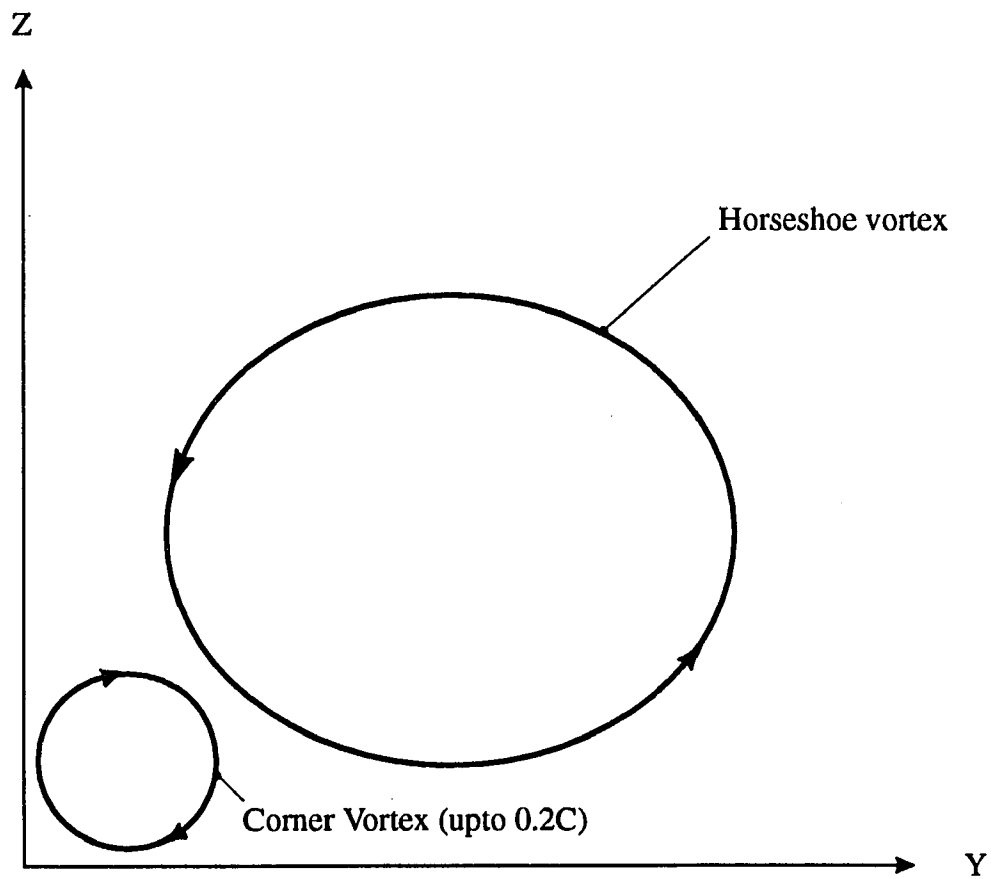


Figure (4.82) Secondary Flow System Downstream of the Blade Endwall Corner.

CHAPTER V

OBSERVATIONS AND CONCLUSIONS

The three dimensional flow characteristics downstream of a corner formed by a NACA 65-015 base profile airfoil and a flat plate have been studied experimentally. The study included surface flow visualization, surface static pressure local wall shear stress, static and total pressure survey and the three components of mean velocity. The characteristics of the Preston tube were also investigated. Based on the analysis of the results of the above mentioned measurements, the following observations and conclusions are made:

1. Total pressure, flow visualization and secondary flow velocity vectors confirmed the presence of a single dominant horseshoe vortex. The formation of the horseshoe vortex is a result of the skewing and separation of the flat plate boundary layer ahead of the blade leading edge. The horseshoe vortex redistributes the mean flow momentum content and distorts total pressure profiles and contours. In planes parallel to the flat plate, total pressure values were found to be higher than the undisturbed flat plate boundary layer at that height. Their presence is confined to the flat plate boundary layer thickness.
2. Downstream of the corner, there was evidence of the presence of a corner vortex. The corner vortex is smaller and weaker than the horseshoe vortex. It dissipates and ceases to exist by downstream axial location of $X/C \cong 0.2$. There was no evidence that suggests the existence of a stress induced vortices.

3. As the flow proceeds downstream of the corner, the effective core of the horseshoe vortex moves away from the flat plate and away from the centerplane. Its rise from the flat plate is much shorter than the distance it moved away from the centerplane.
4. The three dimensionality in the flow exists far downstream the trailing edge of the blade. It extends more than one chord length in the stream wise direction. In the transverse direction, secondary flow effects dominated up to $0.3C$ away from the centerplane.
5. Surface static pressure varies significantly on the flat plate. It was found to be maximum at the blade trailing location and it decreases in both the downstream and transverse directions. There was no significant static pressure variation in the spanwise (Z-) direction.
6. Downstream the blade trailing edge, under the domain of the horseshoe vortex, local wall shear stress increased to values exceeding the values found in the undisturbed boundary layer at that axial location.
7. The presence of high magnitude of wall shear stress on the flat plate between the blade leading edge and the point of maximum blade thickness is confirmed, a by-product of flow visualization.
8. It was difficult to set up this experiment with absolute flow symmetry on the two sides of the centerplane. Although the blade was set at zero geometrical angle of incidence, manufacturing inaccuracies make absolute symmetry a difficult task to achieve.

9. Values of the local wall shear stress, measured by the three sizes of Preston tubes, compared well and were within the experimental error (< 5 percent). This indicated that Patel's calibration equations adequately accounted for the tube size.
10. Preston tubes were found to be insensitive to flow angle of ± 6 degrees. Therefore, the direction of the local wall shear stress vector should not be determined using this probe in situations where the flow turning is of this order.
11. Five hole probe calibration characteristics varied with Reynolds number and were found to be different from the calibration data supplied by the manufacturer.

APPENDIX A

CALIBRATION OF THE FIVE HOLE PROBE

A-1 Discussion

It would be convenient if the calibration equations of the five hole probe could be established analytically. Potential flow solution for spherical sensing head probes developed by Wright (107) is valid for inviscid flow of restricted angularity. Analytical solution of any type for other probe geometries (e.g., angle tube and prismatic) is difficult. These probe geometries characterized by sudden changes in contour are subject to flow separation and viscous effect which make it hard to model. In addition to manufacturing inaccuracies, operating range and accuracy requirements encountered in test conditions makes the calibration a must.

The five hole probe can be calibrated and employed in three different modes; namely, nulling mode, semi-nulling mode and non-nulling mode. If the probe is mechanically aligned with the flow direction (by equalizing pressures across the symmetrically arranged pressure holes) the angles of yaw and pitch can be read on protractors and the reading of the center hole will represent the stagnation pressure at that point. This is the nulling mode technique which was been previously used by Winternitz (108).

This mechanical probe alignment is rarely used nowadays, as it is time consuming and requires a large carriage arrangement. The semi-nulling mode,

which is a more convenient method, applies mechanical alignment of the probe in one plane in conjunction with calibration results of the plane perpendicular to it. Usually, the probe is aligned by rotation with the velocity vector in the yaw plane (by equalizing pressures P_2 & P_3 , (Figure (3.12))), then the yaw angle could be read on protractor. This requires the determination of the relationships between the pressure read by the other three sensing holes (namely P_1 , P_4 and P_5), the true local total, static pressures and the pitch angle. These relationships are obtained by calibrating the five hole probe in a known field.

Recently, the non-nulling technique (or fixed position) has been used extensively (e.g., Wright (107), Treaster and Yocum (109), Matsunaga et al. (110) and Sitaram et al. (111)), when space limitations or other considerations (e.g., probe rotating with the rotor of a compressor) make the nulling or semi-nulling modes impractical. This requires the determinations of measured pressures at the five holes and the true local total and static pressures. These desired relationships are usually expressed as dimensionless pressure coefficients, which are functions of the flow angularity. Since, when in use, the flow angles are unknown, it also requires the determination of relationships between the five measured probe pressures and flow direction on both Pitch and Yaw planes. The application of the calibration curves obtained by this technique requires interpolation procedure and is described by Treaster et al. (109).

A-2 The Five Hole Probe Calibration Equipment

The five hole probe was calibrated in the wind tunnel test section which has the dimensions of 46 cm x 46 cm x 149.9 cm with the model and the grids removed. On one side of the test section, the last 55 cm was removed and replaced

with the calibration stand (described in Section (A.2.1)) and an opening of 371 cm x 5 cm was cut on the lower side of the test section to allow ± 35 degree rotation of the five hole probe in the pitch plane as well as an outlet for the pressure hoses connected to the probe.

The calibration equipment consists of:

1. Calibration stand, capable of rotating the five hole probe about its measuring point, i.e., the geometrical center of the sensing head surface.
2. Calibration apparatus, which consisted of tubing connections, pressure transducers, demodulators and voltmeters.
3. The five hole probe.
4. Pitot static probe.

A.2.1. Calibration Stand

In order to be able to rotate the five hole probe in the pitch plane around its tip, an attachment was designed to be used in the wind tunnel test section. The stand, used to position the five hole probe during calibration, consisted of a wooden base (which was placed on the side of the wind tunnel test section), a two-piece swing arm and a probe support holder as shown in Figure (A-1). The base was made from plywood with dimensions of 55 cm x 49.5 cm and had a 1 cm wide semi-circular arc groove allowing 180 degree of rotation with a 5 degree graduation. A zero pitch angle was marked on the base (middle of the arc) which corresponds to a position at which the probe stem was at a vertical position (i.e., at a right angle with the test section axis). The swing arm consists of two pieces, one

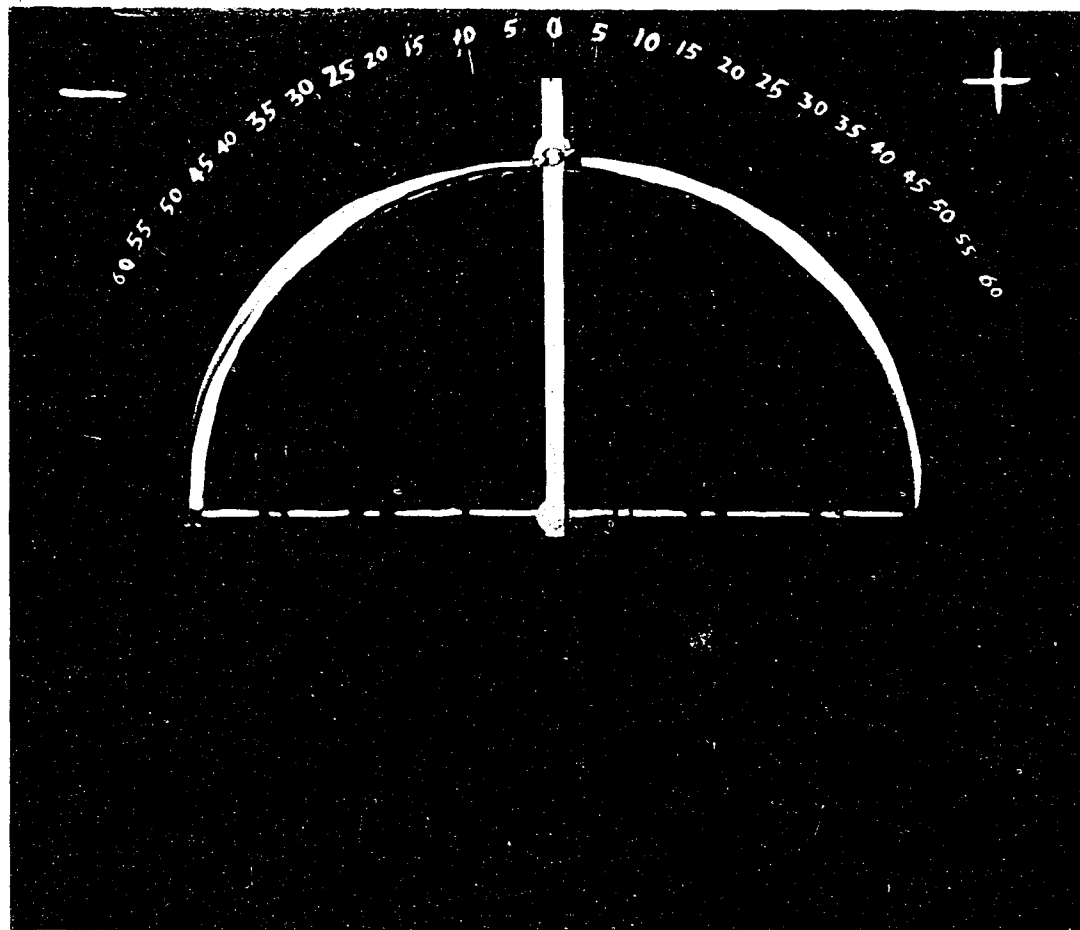


Figure (A.1) Photograph of the Five Hole Probe Calibration Stand.

on each side of the base, both were made from 0.95 cm cross section and 19.4 cm long brass bar. The two pieces were connected together at two points. The first, acted as a pivot and was located at the center of the base (center of the test section). The second point was through the 1 cm groove via a screw with a wing nut. This two point connection allowed the rotation of the two pieces together along the groove with the pivot as the center of the semicircular arc (when the wing nut was loose) and could be tightly fixed at any position of pitch angle (when tightened). The inner piece was connected to the probe holder, the outer piece was used as an indicator for the position of the inner piece (as they are fixed together). The outer piece has the pivot on one end and the other end was shaped like a knife which contains a V-notch in the middle for exact location of the angle of rotation.

The probe holder, made from 0.95 cm x 0.95 cm cross section and 15.2 cm long brass bar, was screwed at one end to the inner piece of the swing arm at a distance of 8.6 cm from the pivot axis, parallel to the swing arm. At the other end of the probe holder is 3 cm long and 0.3 cm wide cut which ends with a 0.63 cm through hole (the size of the five hole probe stem diameter). The hole was made in the plane containing the swing arm and perpendicular to it. In the plane perpendicular to the cut plane another hole was drilled (1 cm from the end) through which a screw-nut was placed (to control the width of the cut). This assembly of the probe holder facilitated a snug placement of the five hole probe. At this position the probe stem axis (which is parallel to the base) is in the vertical plane of the test section (i.e., the pitch plane). The axis swing arm pivot and probe holder were in the horizontal plane of the test section (i.e., the yaw plane).

The five hole probe was positioned on the probe holder with its tip distance from the holder such that the pivot axis will pass through the midpoint of the probe tip (which is exactly at the center of the test section). The movement of the swing

arm makes the probe rotate around its tip in the plane of the stem (i.e., in the pitch plane, Figure (A.2)). This was not very crucial as a uniform flow field was available in the test section.

A.2.2 Calibration Apparatus

A typical installation with pressure transducers for calibrating the five hole probe in the semi-nulling mode is shown schematically in Figure (A.3). The dividing choke provides a high resistance bleed across the differential ($P_2 - P_3$), tapped at the center so that the center tap indicates $(P_2 + P_3)/2$. This average pressure changes only slowly with yaw angle, so (P_2) and (P_3) do not have to be exactly equal when the pressures are observed. Therefore, the probe can be effectively balanced more quickly, speeding up readings considerably.

Six Validyne (DP15) differential pressure transducers with CD15 Carrier demodulators were used with this tubing connections. The output of each transducer was displaced on DISA 55D31 Digital Integrating Voltmeter. Tygon Tubings (rubber hoses) were used in the hook-up, illustrated in Figure (A.3), with oversized T-joints and end connections which make these joints tight enough to prevent any air leaks. After completing the hook-up, leaks were checked by pressuring the tubes and sealing off the ends except one end which was connected to a Miriam Micromanometer. The micromanometer pressure readings should keep steady for about ten minutes, a slow drop is evidence of a leak in the system. The leak can be found quite easily with rubber hoses by working back from either end and sealing off successive portions of the tube. Where this is impossible, it can be found by applying soap solution to all joints. Care must be taken not to get any

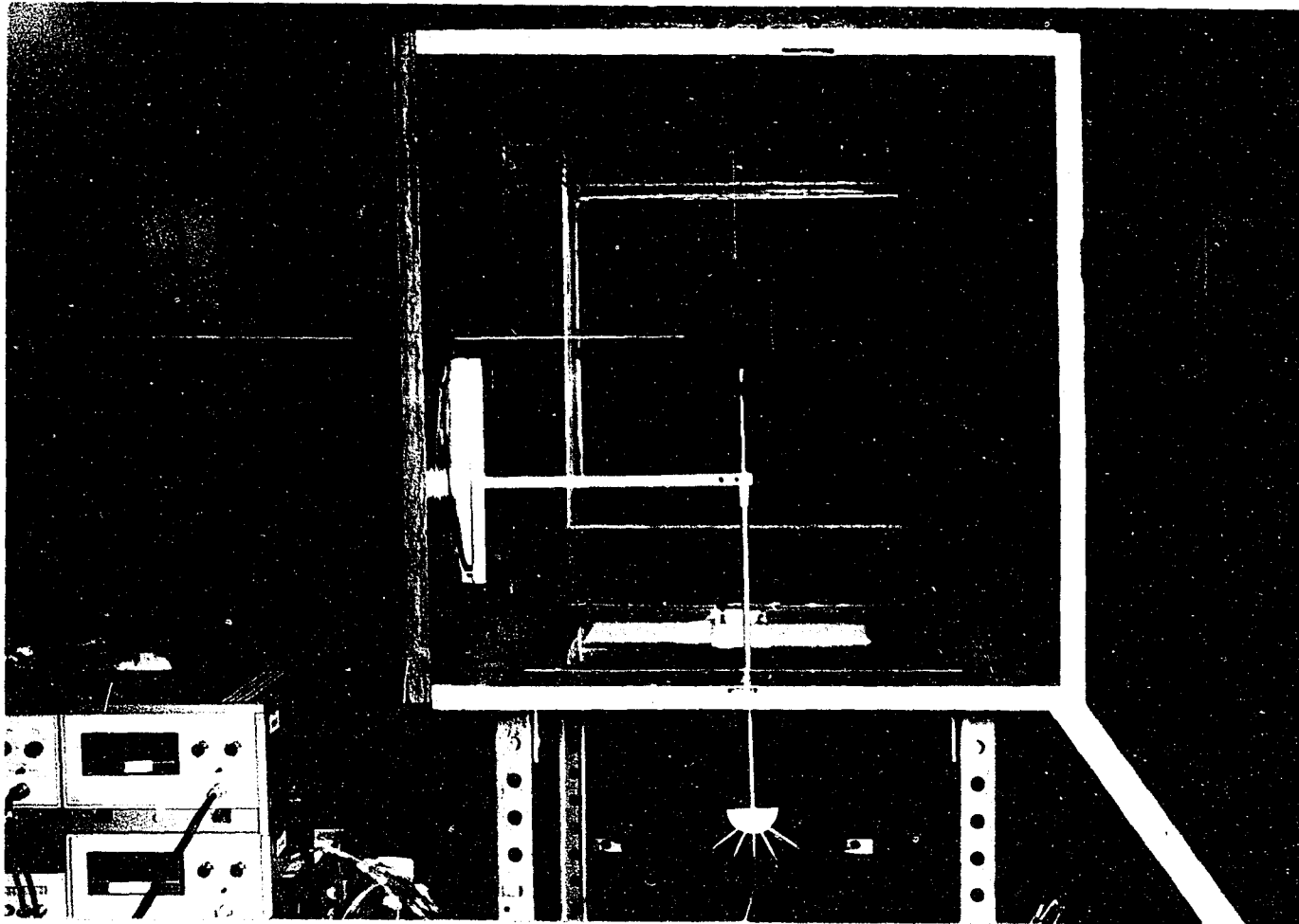


Figure (A.2) Photograph of the Five Hole Probe Calibration Set-up.

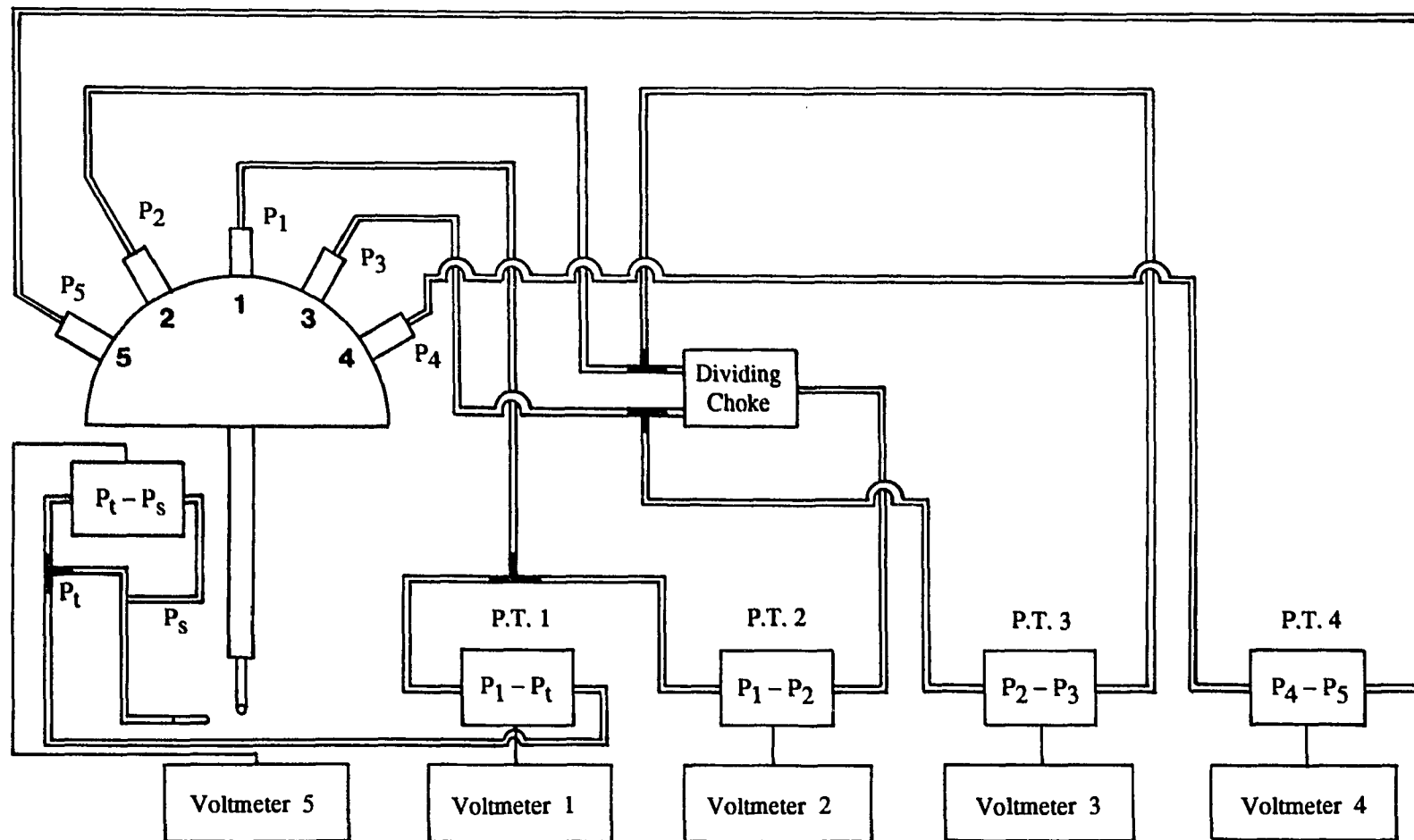


Figure (A.3) Typical Installation of the Pressure Transducers for Calibrating the Five Hole Probe.

liquid drop inside the pressure lines. This procedure was also applied for checking leaks at the joints across the differential pressure transducers.

A.2.3 The Five Hole Probe:

As the name indicates, the five hole probe is characterized by five pressure sensing holes laying in two perpendicular planes with the lines of intersection of the two planes passing through the center hole. The schematic representation of the five hole probe is shown in Figure (3.7) and the details of the sensing head are shown in Figure (3.12). The probe used in this investigation was manufactured by United Sensor Corporation. It was DC-095-250-13-125 type, the smallest available size supplied commercially.

This miniature probe sensing head was chosen not only to minimize the distortion of the flow field due to the probe existence, but to reduce the velocity gradient effects and to measure as nearly as possible at a point.

The pitot-static probe, described in Section (3.1.5) was used to measure the difference between total and static pressures ($P_t - P_s$) used in the calibration calculations. Its sensing head was placed 3.81 cm x 3.81 cm above the sensing head of the five hole probe by inserting it through a cut made in the stand device (described in Appendix A.2.1)

A-3 Calibration Procedure

The five hole probe employed in this study was used in the semi-nulling mode which requires the determination of relationships between the measured pressures at three of the five holes and the actual local total and static pressures. These

relationships are usually expressed in non-dimensionalized pressure coefficients which in turn are functions of flow angularity. Since, when in use, the flow angle (in this case the pitch angle, as the yaw angle is determined by equating two pressures namely, (P_2) and (P_3) , and read on the second rotating device) is unknown, relationships between the three measured probe pressure and the flow direction (pitch angle) are also required.

These relationships are

1. Pitch angle pressure coefficient $[C_{PP} = (P_4 - P_5)/(P_1 - P_2)]$ versus Pitch angle.
2. Velocity pressure coefficient $[C_{PV} = (P_t - P_s)/P_1 - P_2)]$ versus Pitch angle.
3. Total pressure coefficient $[C_{PT} = P_1 - P_t)/(P_t - P_s)]$ versus Pitch angle.

where (P_1) is the pressure recorded by pressure hole located centrally, (P_4) and (P_5) are pressures recorded by pressure holes located above and below the center hole, respectively. The numbering of the holes in the probe sensing head is shown in Figure (3.12). It can be noticed that the pressure differential in the yaw plane $(P_2 - P_3)$ is not included in these relationships as this reading was kept zero (i.e., $P_2 - P_3 = 0$), which corresponds to zero yaw angle at all calibration points. It can be observed, from looking at these relationships, that five pressure differentials should be obtained;

1. $(P_t - P_s)$
2. $(P_1 - P_t)$
3. $(P_1 - P_2)$
4. $(P_4 - P_5)$
5. $(P_2 - P_3 = 0)$

The calibration was carried out at air speed range of 38.3 fps to 103.4 fps which corresponds to Reynolds number range of 1722.4 to 4767.6 (based on probe sensing head diameter). The probe was calibrated twice to verify the repeatability of the results. The results were averaged to reduce the scattering in calibration data.

Sign convention for the yaw and pitch angles is shown in Figure (3.12). The pitch angle (θ) was measured in a plane containing the tunnel test section axis and probe stem. The Yaw angle (ϕ), measured in plane normal to the probe stem, was kept zero at all points at which the probe was calibrated.

The probe was positioned in the calibration stand described in Section (3.1.6.1) with a distance such that the tip of the sensing head (where hole No.1 is located) was in the center of the wind tunnel test section, for the various pitch angles. The stand device permitted a rotation of -35° to $+35^\circ$ in the pitch plane with 5 degrees graduation (i.e., a set of 15 readings for each run of Reynolds number). Pressure differential readings ($P_t - P_s$), ($P_1 - P_2$), ($P_4 - P_5$), ($P_2 - P_3 = 0$) and ($P_1 - P_t$) were read (after the system reached equilibrium) from the digital voltmeters at the various pitch angles. Then each of the above pressure differentials was multiplied by its differential pressure transducer coefficient. Thereafter, the three pressure coefficients (C_{pp} , C_{pv} and C_{pt}) were calculated. Table (A.1) shows a sample of these calculations at air speed of 103 fps (which corresponds to Reynolds number of 4767.6, based on the probe sensing head diameter). The results of table (A-1) were plotted in three graphs, which are;

1. C_{pp} versus (θ); Figure (A.4.1)
2. C_{pv} versus (θ); Figure (A.4.2)
3. C_{pt} versus (θ); Figure (A.4.3)

	Pitch Angle	Pt-Ps	P1-P2	P4-P5	P1-Pt
1	0	0.962	1.110	-0.052	0.007
2	-5	0.960	1.105	0.098	0.004
3	-10	0.961	1.095	0.229	-0.005
4	-15	0.960	1.085	0.331	-0.041
5	-20	0.957	1.038	0.369	-0.109
6	-25	0.954	1.014	0.565	-0.209
7	-30	0.950	0.980	0.761	-0.330
8	-35	0.950	0.929	0.962	-0.490
9	0	0.959	1.103	-0.038	0.006
10	+5	0.959	1.111	-0.200	0.003
11	+10	0.960	1.116	-0.348	-0.023
12	+15	0.960	1.108	-0.492	-0.074
13	+20	0.960	1.096	-0.621	-0.144
14	+25	0.962	1.070	-0.766	-0.257
15	+30	0.963	1.042	-0.954	-0.405
16	+35	0.963	0.999	-1.130	-0.565

Table (A.1.1) Data Output of the Calibration.

	Ct5*(Pt-Ps)	Ct2*(P1-P2)	Ct4*(P4-P5)	Ct1*(P1-Pt)	$\frac{Ct1*(Pt-Ps)}{Ct2*(P1-P2)}$	$\frac{Ct3*(P4-P5)}{Ct2*(P1-P2)}$	$\frac{Ct4*(P1-Pt)}{Ct5*(Pt-Ps)}$
1	0.399	0.453	-0.021	0.003	0.881	-0.046	0.007
2	0.398	0.451	0.040	0.002	0.882	0.089	0.004
3	0.398	0.447	0.094	-0.002	0.890	0.210	-0.005
4	0.398	0.442	0.135	-0.017	0.900	0.305	-0.043
5	0.396	0.423	0.151	-0.045	0.936	0.357	-0.114
6	0.395	0.414	0.231	-0.086	0.954	0.558	-0.218
7	0.394	0.400	0.311	-0.136	0.985	0.777	-0.345
8	0.394	0.379	0.394	-0.203	1.040	1.040	-0.515
9	0.397	0.450	-0.016	0.002	0.882	-0.036	0.006
10	0.397	0.453	-0.082	0.001	0.876	-0.181	0.003
11	0.398	0.455	-0.142	-0.010	0.875	-0.312	-0.024
12	0.398	0.452	-0.201	-0.031	0.881	-0.445	-0.078
13	0.398	0.447	-0.254	-0.060	0.890	-0.568	-0.151
14	0.399	0.436	-0.313	-0.106	0.915	-0.718	-0.266
15	0.399	0.425	-0.390	-0.167	0.939	-0.918	-0.419
16	0.399	0.407	-0.462	-0.234	0.980	-1.135	-0.586

Table (A.1.2) Data Processing for Generating the Calibration Curves.

Table (A.1) Sample of Data Output and Processing for Generating the Calibration Curves of the Five Hole Probe.

Reynolds number = 4767.6

$$y = -0.0424 - 0.0249x - 1.549e-4x^2 + 3.157e-6x^3 + 1.264e-7x^4 + 0.000e+0x^5 \quad R = 1.00$$

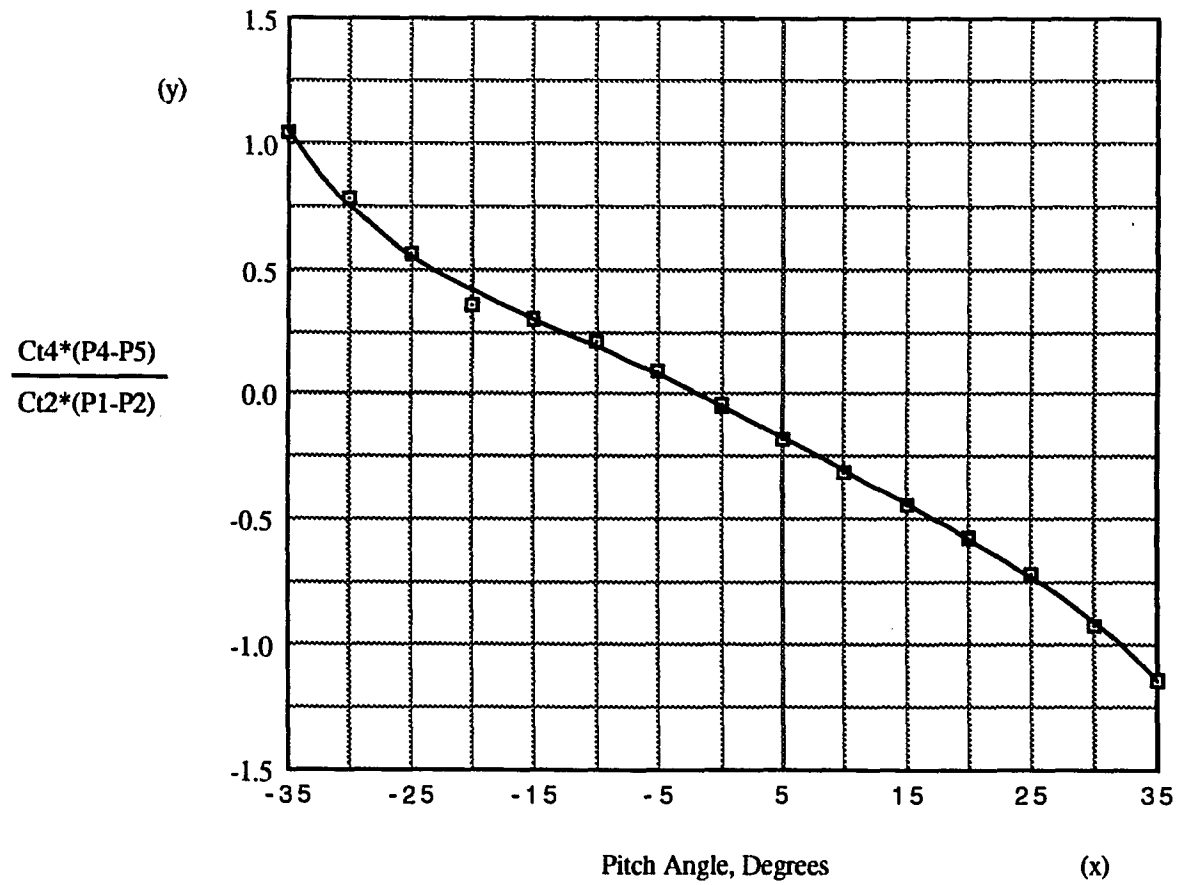


Figure (A.4.1) Pitch Angle Pressure Coefficient versus Pitch Angle.

Reynolds number = 4767.6

$$y = 0.878 - 8.463e-4x + 6.546e-5x^2 + 3.163e-8x^3 + 3.411e-8x^4 + 0.000e+0x^5 \quad R = 1.00$$

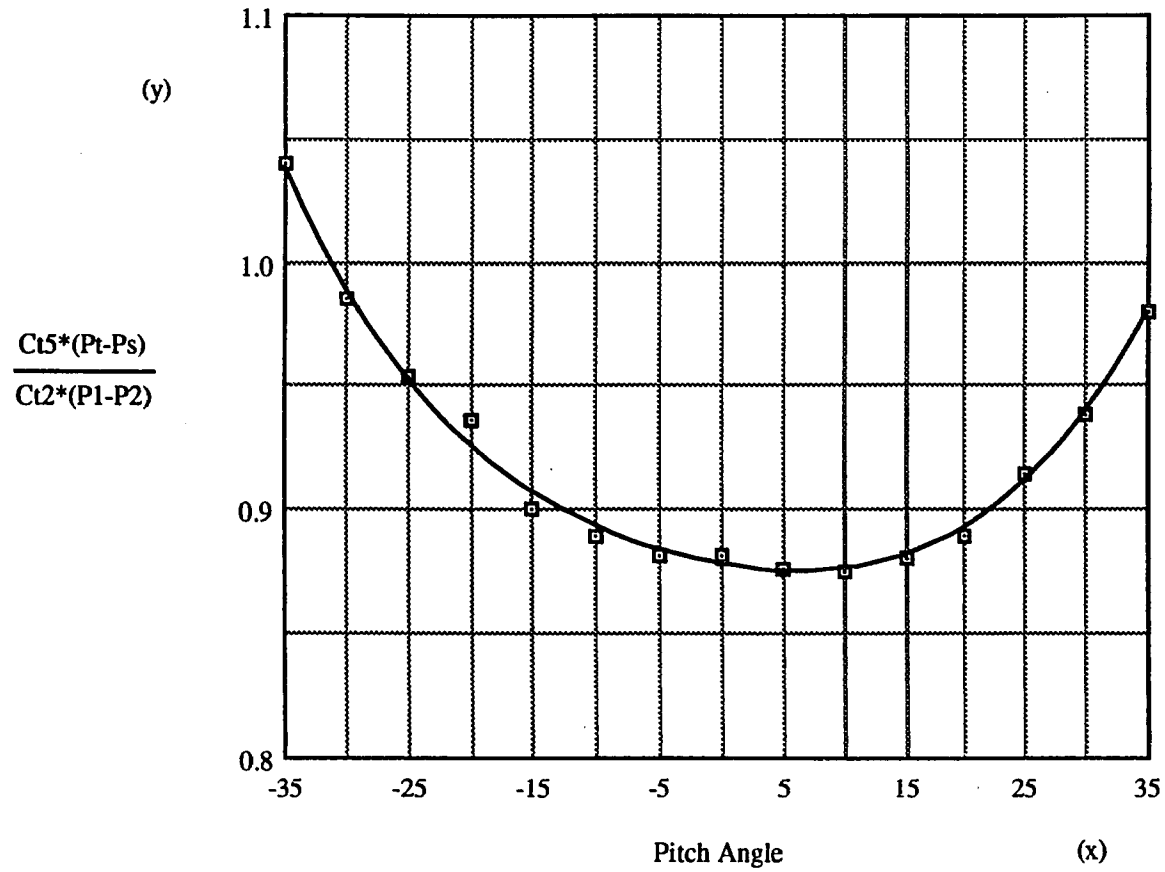


Figure (A.4.2) Velocity Pressure Coefficient versus Pitch Angle.

Reynolds number = 4767.6

$$y = 0.0146 - 7.513e-4x - 3.406e-4x^2 - 8.750e-7x^3 - 1.016e-7x^4 + 0.000e+0x^5 \quad R = 1.00$$

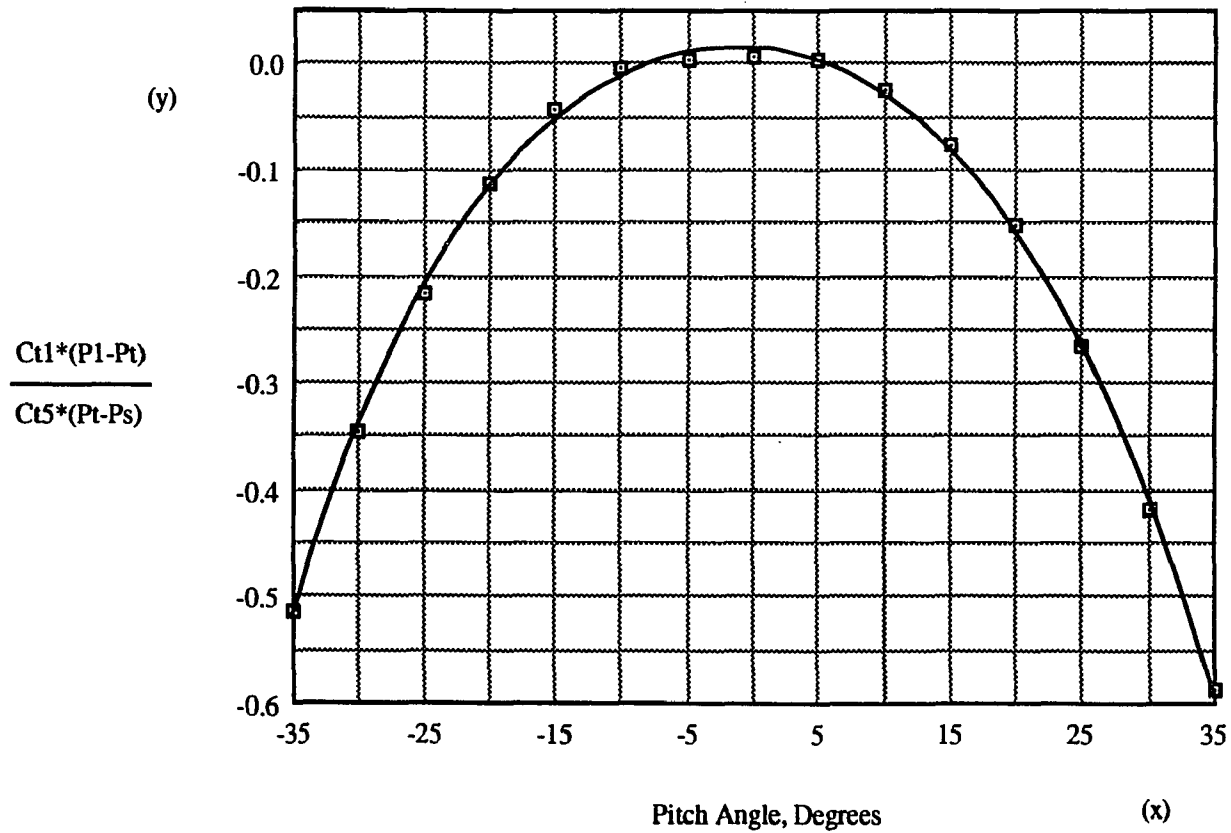


Figure (A.4.3) Total Pressure Coefficient versus Pitch Angle.

A fifth order polynomial curve fitting was used for the above three calibration graphs. The equation for each of these curves was utilized in the interpolation procedure used in performing the calculation of the three components of mean velocity. A FORTRAN program is presented in Appendix (B).

A. 4. Results of the Calibration

A complete set of calibration data for the five hole probe at air speed of 103 fps is presented in Figures (A.4.1, A.4.2 and A.4.3), where the graphical data have been represented by fifth order spline curves passing the individual data points. While calibrating the probe, it was observed that the calibration characteristics vary with Reynolds number. Figure (A.5.1) shows the effect of Reynolds number on pitch angle pressure coefficient (C_{pp}) for the various pitch angles. The variation in C_{pp} is pronounced up to Reynolds number of 2500 and for pitch angles greater than +20 degrees and lower than -20 degrees, then it levels off around the average values. This could be attributed to the position of pressure sensing holes with respect to the flow direction. At a large pitch angle, one of the two holes in the pitch plane (say P_4) is approximately aligned with the flow, senses pressure near the free-stream total pressure, whereas, the other hole (P_5) senses a pressure much less than the free stream static pressure. This is due to the acceleration of the flow around the probe sensing head and also possibly due to flow separation at low Reynolds number with lower momentum flow.

Figures (A.5.2) and (A.5.3) show the effect of Reynolds number on total pressure coefficient C_{PT} and on velocity pressure coefficient C_{PV} for the various pitch angles. Also, noticed is the higher scattering of data at high pitch angles and lower Reynolds numbers which could be due to the separation that might occur at

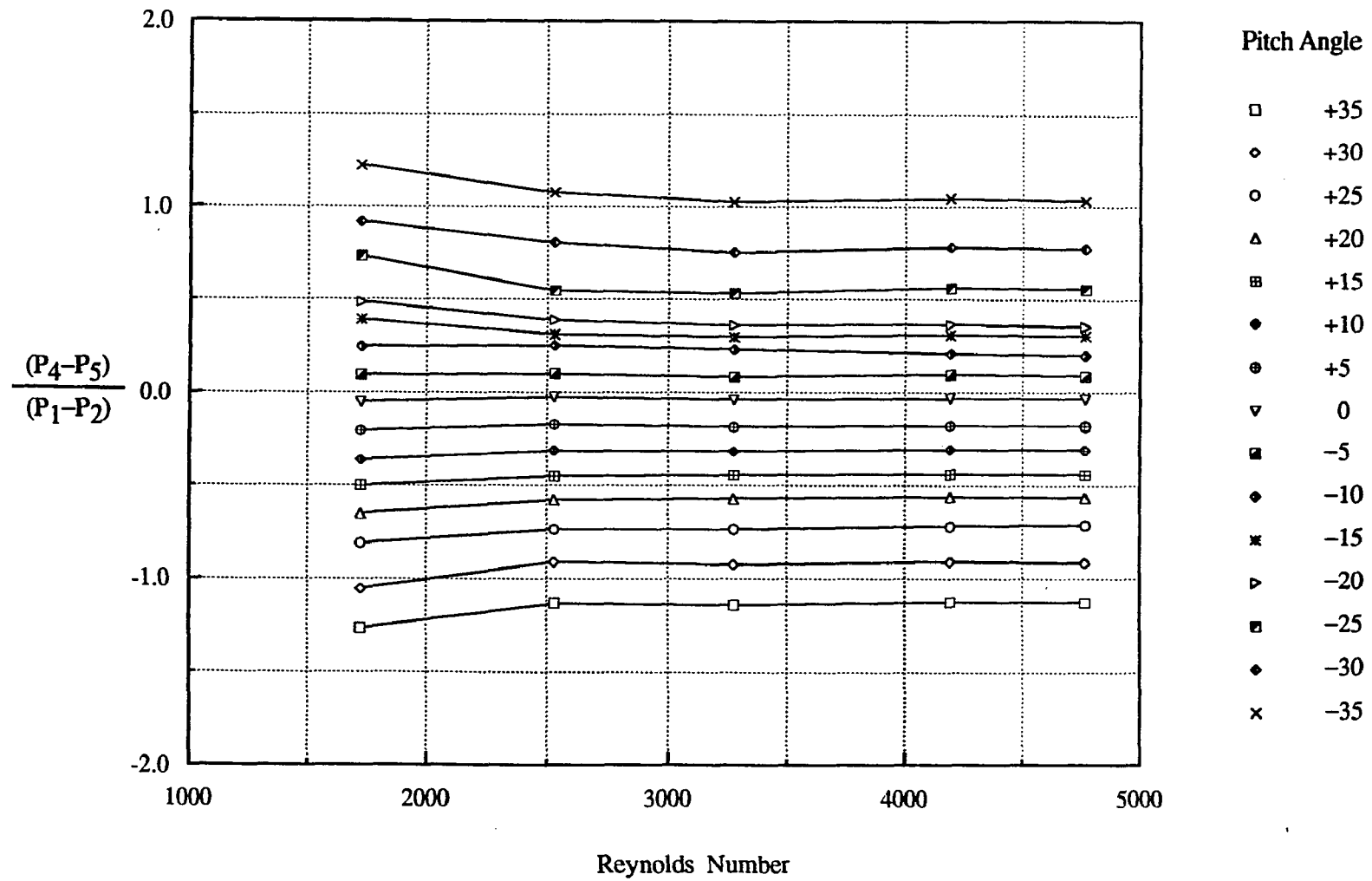


Figure (A.5.1) Reynolds Number Effect on Pitch Angle Pressure Coefficient.

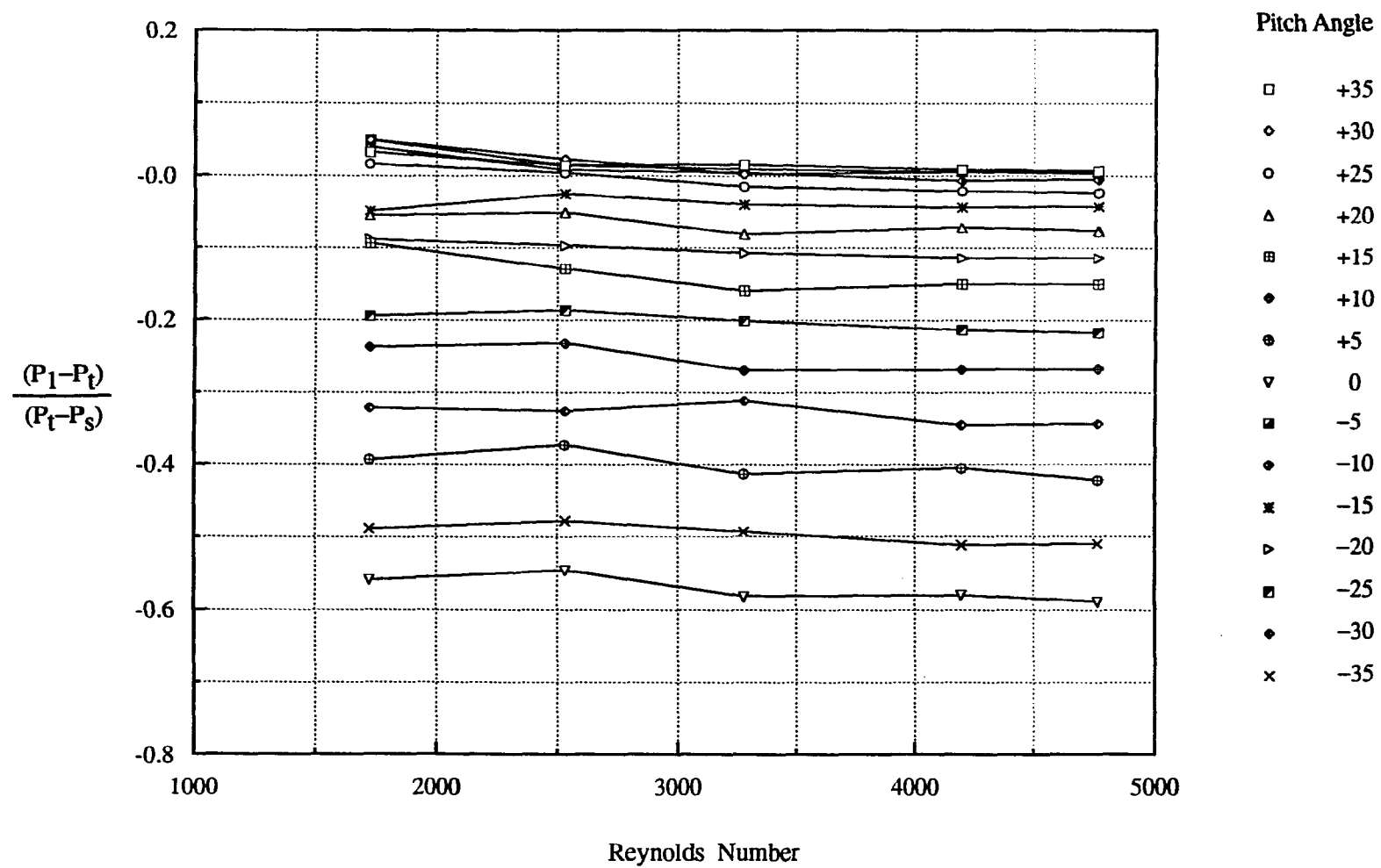


Figure (A.5.2) Reynolds Number Effect on Velocity Pressure Coefficient.

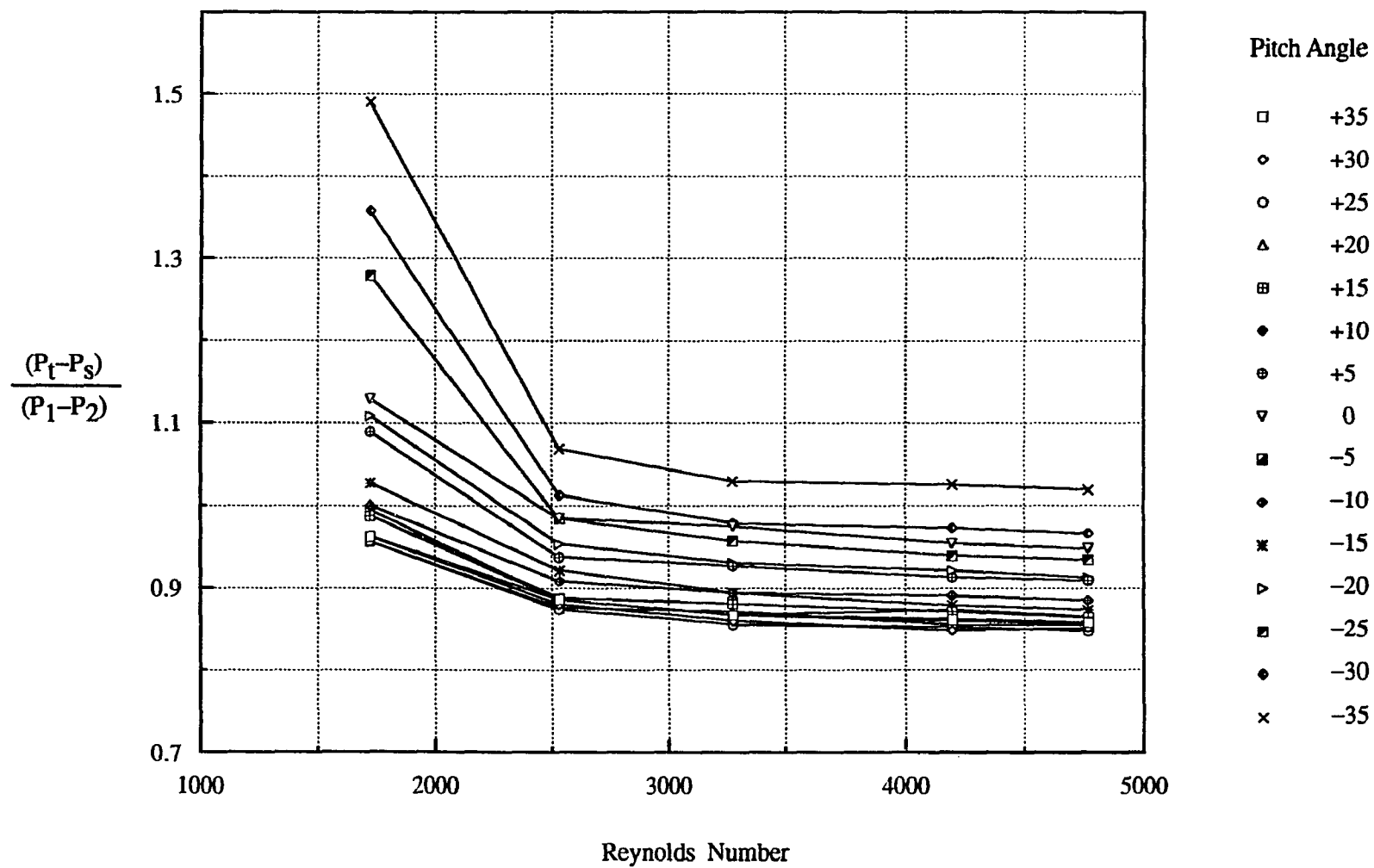


Figure (A.5.3) Reynolds Number Effect on Total Pressure Coefficient.

lower Reynolds numbers. Figure (A.5.3) also shows that the velocity pressure coefficient (C_{PV}) has a different response to a low Reynolds number depending on positive or negative pitch angle. For instance at Reynolds number of 1722; a higher scatter of C_{VP} at $+35^\circ$ of pitch angle compared to relatively lower scatter of -35° angle. This could be attributed to an asymmetric probe manufacturing. In application, a corresponding iterative procedure in terms of the unknown Reynolds number would be required. This limits the practicality of this probe when used in studies having a wide variation of Reynolds number. However, one can calibrate the probe at the expected Reynolds number, or, one can average the data obtained over a small range of Reynolds numbers.

The calibration, data obtained from the two sets of calibration for the Reynolds number range of 1700-4800, were averaged. This is justified by the relatively weak dependence of the calibration coefficients on Reynolds numbers higher than 2,000 (which is the expected range for this investigation). Moreover, pitch angles of the investigated flow are expected to be much less than 35 degrees.

Figures (A.6.1, A.6.2 and A.6.3) show the comparison between the average results obtained from the two calibration carried out at the Turbomachinery Laboratory and the data supplied by United Sensor Corporation (with calibration air speed of 186 fps which corresponds to Reynolds number of 8,725). The prominent difference can be seen (graphically or by comparing the coefficients in the equations of the curve fitting) in the velocity pressure coefficient C_{PV} (Figure (A.6.2)). Also, in Figure (A.6.1) one can see at pitch angle = 0, C_{PP} has different value. This is important in the initial placement of the second rotary device holding the probe (Section 3.1.3.2) with the traverse mechanism (Section 3.1.3.1) as described in Section (3.2.3.)

United Sensor $y = -0.0754 - 0.0225x - 1.793e-4x^2 - 6.676e-6x^3 + 1.159e-7x^4 + 0.000e+0x^5$ $R = 1.00$

Average $y = 0.0186 - 0.0236x - 2.166e-4x^2 + 1.316e-6x^3 + 2.038e-7x^4 + 0.000e+0x^5$ $R = 1.00$

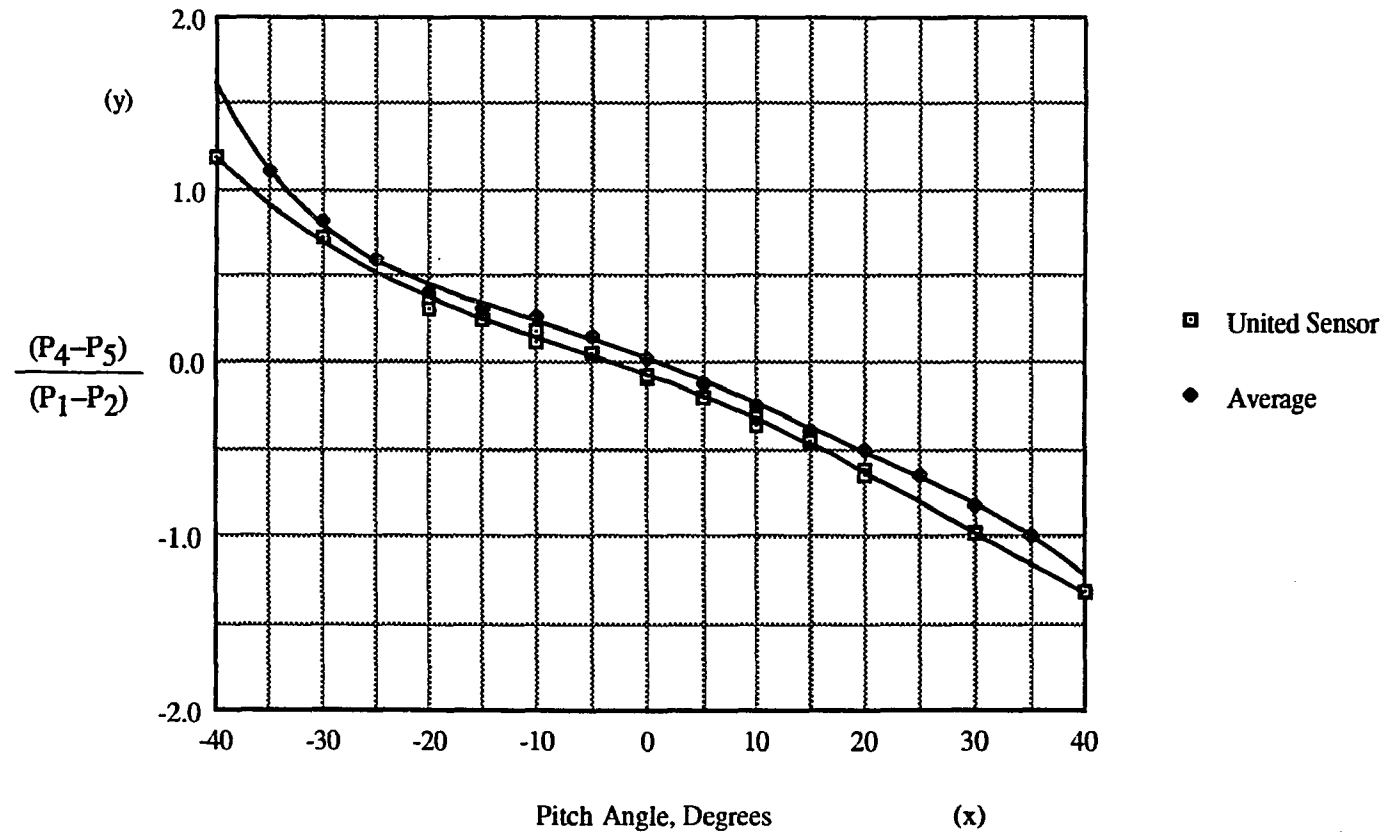


Figure (A.6.1) Comparison Between the Data Supplied by United Sensor and the Averaged Values for the Pitch Angle Pressure Coefficient.

United Sensor $y = 0.7878 + 1.193e-4x + 1.489e-5x^2 - 1.159e-6x^3 + 3.483e-8x^4 + 5.045e-10x^5$ $R = 0.99$

Average $y = 0.8078 - 0.0012x + 6.880e-5x^2 - 1.225e-6x^3 + 2.069e-8x^4 + 0.000e+0x^5$ $R = 1.00$

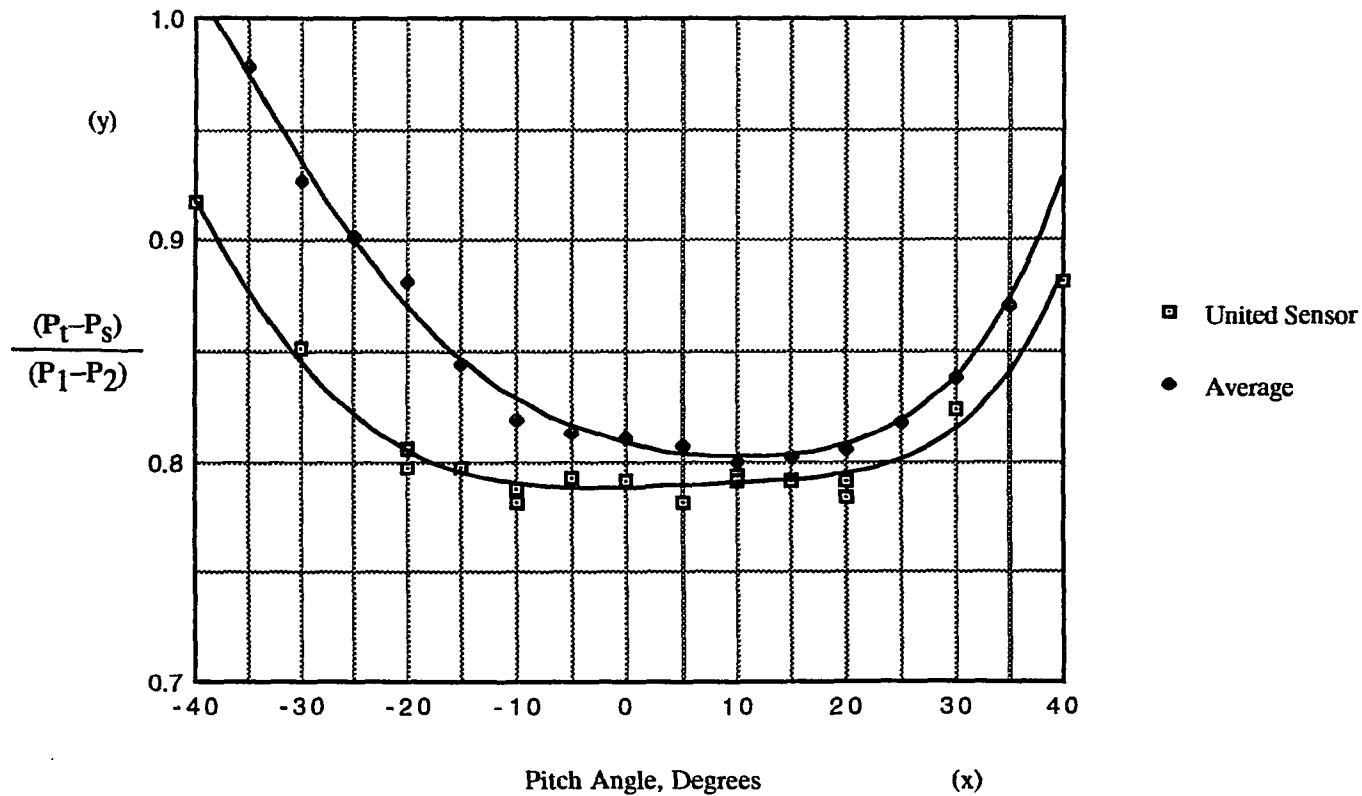


Figure (A.6.2) Comparison Between the Data Supplied by United Sensor and the Averaged Values for the Velocity Pressure Coefficient.

United Sensor $y = 0.0102 - 0.0022x - 4.587e-4x^2 - 1.096e-6x^3 - 9.025e-9x^4 + 0.000e+0x^5$ $R = 1.00$

Average $y = 3.344e-4 + 7.438e-4x - 3.176e-4x^2 - 5.790e-7x^3 - 9.611e-8x^4 + 0.000e+0x^5$ $R = 1.00$

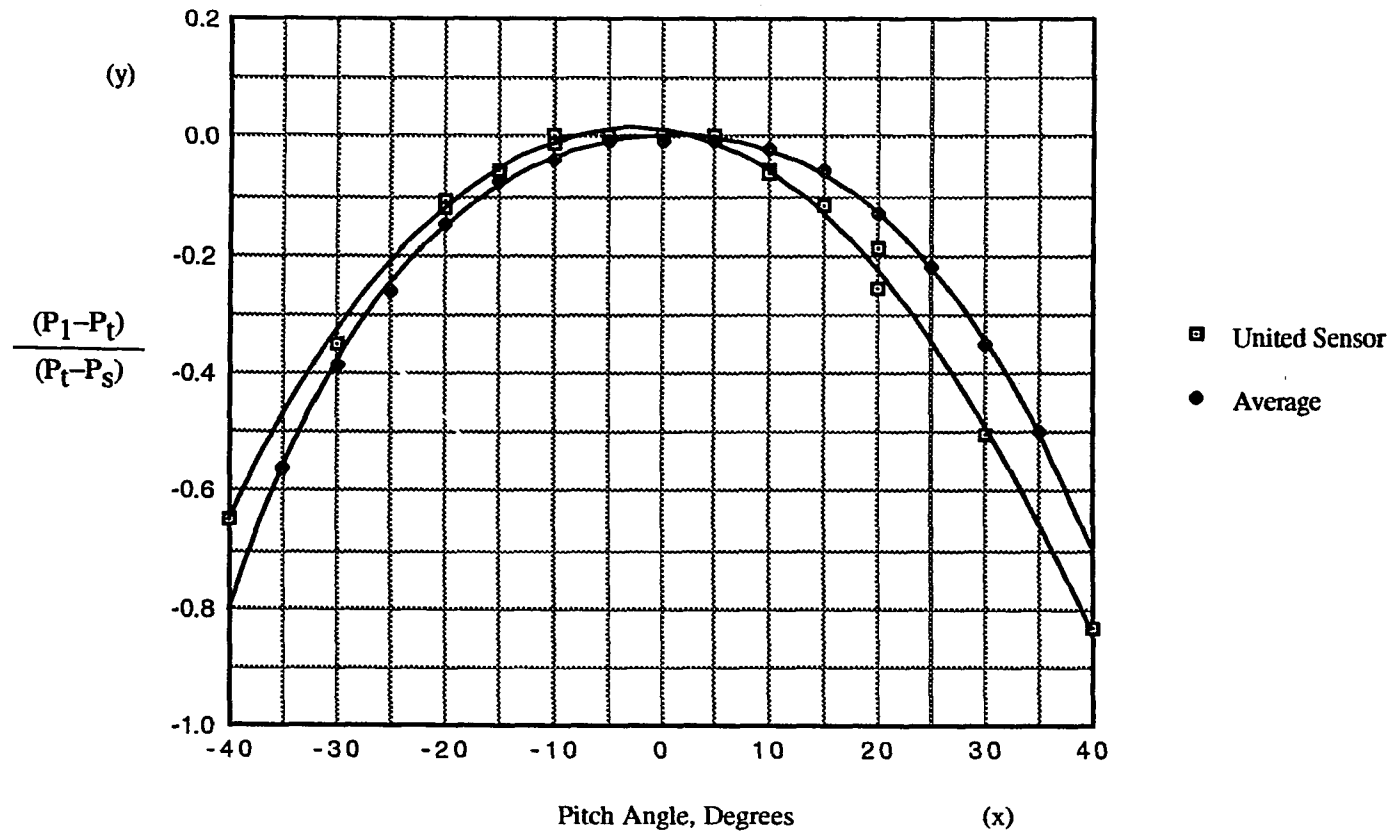


Figure (A.6.3) Comparison Between the Data Supplied by United Sensor and the Averaged Values for the Total Pressure Coefficient.

The averaged values were used in a FORTRAN program written to calculate pitch angle, total and static pressure, total velocity vector and the three components of mean velocity (Appendix B).

APPENDIX B

A FORTRAN PROGRAM TO PROCESS THD DATA OF THE FIVE HOLE PROBE

```
// JOB
// *MAIN TIME=1,LINES=5
// EXEC WATFIV,REGION=800K
$JOB      XX,TIME=060,PAGES=100
C      A FORTRAN PROGRAM TO CALCULATE THE MAGNITUDE OF TOTAL PRESSURE,
C      STATIC PRESSURE,PITCH ANGLE AS WELL AS THE TOTAL VELOCITY VECTOR,
C      AND THE THREE VELOCITY COMPONENTS OF FLUID FLOW USING A THREE
C      DIMENSIONAL DIRECTIONAL FIVE-HOLE PROBE IN THE SEMI-NULLING
C      MODE OF OPERATION
C      EXPERIMENTAL DATA THAT SHOULD BE ENTERED;
C      1- YAW ANGLE , 2- DIFFERENTIAL PRESSURES (P1-P2),(P1-PA),AND (P4-P5)
C      3- REFERENCE PRESSURE , 4-TRANSDUCERS CALIBRATION CONSTANTS,
C      5- ATMOSPHERIC PRESSURE AND TEMPERATURE, 6- NUMBER OF DATA POINTS.
C      DEFINITIONS OF THE SYMBOLS USED:
C      TH(I)   : PITCH ANGLE (DEGREE)
C      THR(I)  : PITCH ANGLE (RADIAN)
C      PHI(I)  : YAW ANGLE (DEGREE)
C      PHIR(I) : YAW ANGLE (RADIAN)
C      PI      : CONSTANT , (CIRCUMFERENCE/RADIUS)
C      P12(I)  : DIFFERENTIAL PRESSURE (P1-P2) (VOLTS)
C      P1A(I)  : DIFFERENTIAL PRESSURE (P1-PA) (VOLTS)
C      P45(I)  : DIFFERENTIAL PRESSURE (P4-P5) (VOLTS)
C      PREF    : DIFFERENTIAL PRESSURE (PREF-PA) (VOLTS)
C      P1      : PRESSURE MEASURED BY THE CENTRALLY LOCATED PRESSURE HOLE.
C      P2,P3   : PREASURES MEASURED BY THE TWO LATERAL PRESSURE HOLES.
C      P4,P5   : PRESSURES MEASURED BY THE TWO PRESSURE HOLES LOCATED ABOVE
C               AND BELOW THE CENTRE HOLE.
```

```

C      TA      : ATMOSPHERIC TEMPERATURE (F)
C      PA      : ATMOSPHERIC PRESSURE (PSIA)
C      H       : ATMOSPHERIC PRESSURE (INCH OF MERCURY)
C      ROH     : DENSITY OF AIR (KG/M3)
C      C       : TEMPERATURE CORRECTION FACTOR FOR FORTIN BAROMETER
C              WITH BRASS SCALES
C      Z(I)    : DISTANCES IN NORMAL DIRECTION.
C      M       : NUMBER OF MEASURING POINTS.
C      V(I)    : VELOCITY VECTOR AT THE MEASURING LOCATION (M/SEC)
C      VX , VY & VZ: THE THREE COMPONENTS OF VELOCITY IN STREAM-WISE ,LATERAL
C      AND NORMAL DIRECTIONS OF THE FLOW,RESPECTIVELY ( M/SEC).
C      CT12,CT45,CT1A & CTRE: DIFFERENTIAL PRESSURE TRANSDUCERS CALIBRATION
C      CONSTANTS (INCH OF WATER/VOLT), USED IN MEASURING (P1-P2),(P4-P5),
C      (P1-PA) AND (PREF-PA) RESPECTIVELY.
C

```

```

C
C      #####          M A I N   P R O G R A M          #####
C
C

```

```

      IMPLICIT REAL*8(A-H,O-Z)
      DIMENSION P45(30),P12(30),P1A(30),PHI(30),PHIR(30),Z(30)
1     ,CPTH(30),TH(30),THR(30),CPV(30),CPT(30),P1T(30),PTA(30),
2     PPSA(30),PTAR(30),PSAR(30),PTS(30),V(30),VX(30),VY(30),VZ(30)
      READ (5,1) CT45,CT12,CT1A,CTRE,Y
      READ (5,2) H,TA,PRE,M,N
      READ (5,3) (Z(I),I=1,M)
      READ (5,3) (PHI(I),I=1,M)
      READ (5,3) (P45(I),I=1,M)
      READ (5,3) (P12(I),I=1,M)
      READ (5,3) (P1A(I),I=1,M)
1     FORMAT(5F10.4)
2     FORMAT(3F10.4,I3,5X,I2)

```

```

3  FORMAT(7F10.4)
   WRITE(6,4)N
   WRITE(6,5) CT45,CT12,CT1A,CTRE
   WRITE(6,6) H,TA,PRE,M,Y

```

C

```

4  FORMAT('1',////,25X,'##### I N P U T   D A T A #####',///
15X,'  STATION NUMBER = ',I2)
5  FORMAT(5X,'TRANSD. FOR (P4-P5) CALIB. CONST.,(INCH H20/VOLT)=' ,
1F10.5,/,5X,'TRANSD. FOR (P1-P2) CALIB. CONST.,(INCH H20/VOLT)=' ,
2F10.5,/,5X,'TRANSD. FOR (P1-PA) CALIB. CONST.,(INCH H20/VOLT)=' ,
3F10.5,/
4,5X,'TRANSD. FOR (PR-PA) CALIB. CONST.,(INCH H20/VOLT)=' ,F10.5/)
6  FORMAT(/ 5X,'ATMOSPHERIC PRESSURE (INCH HG)=' , F10.5 /, 5X,
1'ATMOSPHERIC TEMP.(F)=' , F10.5 / 5X,'TOTAL REFERENCE PRESSURE
2(VOLTS)=' ,F10.5,/,5X,'NUM OF DATA POINTS=' ,I3,5X,'Y IN=' ,F10.5//)
   WRITE(6,7)
7  FORMAT(5X,'I',9X,'Z (INCH)',10X, 'PHI',10X,'P4-P5',10X,'P1-P2',
110X,'P1-PATM')
   Y=Y*25.4
   DO 10 I=1,M
10  WRITE(6,8) I,Z(I),PHI(I),P45(I),P12(I),P1A(I)
8   FORMAT(4X,I2,2X,5(5X,F10.5))

```

C
C
C
C
C

CALCULATION OF FREE STREAM VELOCITY

```

C=-((9.08*(TA-28.63)*10**(-5))/(1+1.01*(TA-32)*10**(-4)))*H
TA=TA+460
H=H+C
PA=0.49115*H
ROH=43.2432*PA/TA
VREM=22.2998*DSQRT(PRE*CTRE/ROH)

```



```

C      PTS(I)=CPV(I)*P12(I)
      P1T(I)=CPT(I)*PTS(I)
      PTA(I)=P1A(I)-P1T(I)
      PSA(I)=PTA(I)-PTS(I)
      PTAR(I)=PTA(I)/PRE/CTRE
      PSAR(I)=PSA(I)/PRE/CTRE

C
C
      V(I)=22.2998*DSQRT(PTS(I)/ROH)

C
C
      PI=4.0D0*DATAN(1.0D0)
      THR(I)=TH(I)*PI/180.0D0
      PHIR(I)=PHI(I)*PI/180.0D0

C
C
      VX(I)=V(I)*DCOS(PHIR(I))*DCOS(THR(I))/VREM
      VY(I)=V(I)*DCOS(THR(I))*DSIN(PHIR(I))/VREM
      VZ(I)=V(I)*DSIN(THR(I))/VREM

C
      Z(I)=Z(I)*25.4

C
      WRITE(6,13) I,Z(I),PTAR(I),PSAR(I),PHI(I),TH(I),V(I),VX(I), 1VY(I),VZ(I)
13    FORMAT(4X,I2,1X,F10.5,2F15.5,4X,2F8.3,5X,4F10.4)
100  CONTINUE
      STOP
      END
$ENTRY

```

REFERENCES

- 1) Hazarika, B.K., and Raj, R. S., "An Investigation of the Flow Characteristics in the Blade End wall Corner," NASA Contractor Report 4076, 1987.
- 2) Dickinson, S.C., "Flow Visualization and Velocity Measurements in the Separated Region of an Appendage/Plate-Plate Junction," Paper presented at Ninth Biennial Symposium on Turbulence, University of Missouri-Rolla, Oct. 1984.
- 3) Gorski, J.J., Gonindan, T.R. and Lakshminarayana, B., "Computation of Three-Dimensional Turbulent Shear Flows in Corners," AIAA Journal, Vol. 23, No. 5, May 1985, pp.685-692.
- 4) Sepri, P., "An Investigation of the Flow in the Region of the Junction of a Wing and a Flat Surface Normal to the Wing Span," Queen Mary College, Dept. Report, QMC ER-1002, 1973.
- 5) Chu, J.K. and Young, A.D., "A Further Investigation of Viscous Effects in a Wing Plate Junction," Queen Mary College Department Report, QMC ER-1003, 1975.
- 6) Binder, A. and Romey, R., "Secondary Flow Effects and Mixing of the Wake Behind a Turbine Stator," ASME Paper No. 82-GT-46, 1982.
- 7) Love, R.M., "An Investigation of the Effect of the Wall Boundary Layer With the Wake of an Obstacle Protruding From the Wall," U.S. Government Report No. AD613625, 1963.

8) Armili, R.V., "The Interaction of a Turbulent flat Plate Boundary Layer With the Wake of Circular Cylinder Mounted Normal to the Plate," Ph.D. Thesis, Virginia Polytechnic Institute and State University, 1973.

9) Naranjit, S., "An Investigation of the Flow Over a Wing-Body Combination," Ph.D. Thesis, University of London, 1976.

10) Barber, T.J., "An Investigation of Strut-Wall Intersection Losses," Journal of Aircraft, Vol.15, No.10, Oct.1978, pp.676-681.

11) Shabaka, I.M.M.A., "Turbulent Flow in an Idealized Wing-Body Junction", Ph.D. Thesis, University of London, 1979.

12) Bhargava, R., and Raj, R.S, "An Experimental Study of Near Wall Flow Parameters in the Blade End-Wall Corner Region," NASA Contractor Report 4211, 1989.

13) Counihan, J., Hunt, J.C.R. and Jackson, P.S., "Wake Behind Two-Dimensional Surface Obstacles in Turbulent Boundary Layers," Journal of Fluid Mechanics, Vol. No. 3, 1974, pp.529-563.

14) Olivari, D., "Environmental Wind in Built-Up Areas," Von Karman Institute for Fluid Dynamics, Technical Memorandum 26, Feb.1980.

15) Marumo, E., Suzuki, K. and Sato, K., "A Turbulent Boundary Layer Disturbed by a Cylinder," Journal of Fluid Mechanics, Vol. 87, Part 1, 1978, pp.121-141.

16) Turan, A, Ruggieri, N.G. and Smith, G.E., "Three-Dimensional Modeling of Horseshoe Vortex Flows," ASME Paper No.83-GT-191, 1983.

- 17) Rahai, H.R., "Mean Velocity in Idealized Wing Wake-Boundary-Layer Interaction Region," M.S. Thesis, California State University, Long Beach, 1981.
- 18) Nakayama, A. and Rahai, H.R., "Measurements of Turbulent Flow Behind a Flat Plate Mounted Normal to the Wall," AIAA Journal, Vol.22, No.12, November 1984, pp.1817-1819.
- 19) Patel, V.C., "Calibration of the Preston Tube and Limitations on its use in Pressure Gradients," Journal of Fluid Mechanics, Vol 23, part 1, 1965, pp.185-208.
- 20) Gartshore, I.S., "The Intraction Between Turbulent Wakes and Boundary Layer," Canadian Aeronautics and Space Institute Transaction, Vol. 5, No.2, 1972, pp.49-55.
- 21) Tsiolakis, E.P., Krause, E. and Muller, U.R., "Turbulent Boundary Layer-Wake Interaction," Turbulent Shear Flows, Vol. 4, edited by L.J.S. Bradbury, F. Durst, B.E. Launder, F.W. Schmidt, and J.H. Whitelaw, Springer-Veriag, New York, 1983, pp. 204-215.
- 22) Kreplin, H.-P., Nakayama, A. and Liu, B., "Measurement of Wake/Wake and Double-Wake/Boundary-Layer Interactions in Zero Pressure Gradient Flows," Department of Mechanical Engineering Report No. ME-86-1., California State University, Long Beach 1986.
- 23) Nakayama, A. Akdag, V., Liu, B. and Unt, H., "Experimental Study of Mergrer of Coufluent Wake and Boundary Layer in Constant Pressuree Field," Mechanical Engineering Report No. ME-84-6, California State University, Long Beach 1986.
- 24) Kreplin, H.-P., Liu, B. and Nakayama, A., "Merging of Multiple Wakes with a Zero-Pressure Gradient Boundary Layer," proceedings of the Third Asian Congress of Fluid Mechanics, September 1-5, 1986. Tokyo, Japan, pp. 431-434.

25) Pal, S., "Wake Boundary Layer Interaction in Turbomachinery," Doctoral Dissertation, Department of Mechanical Engineering, The City College of the C.C.N.Y., 1981.

26) Ziada, S. and Rockwell, D., "Vortex-Leading-Edge Interaction," *Journal of Fluid Mechanics*, Vol. 118, 1982, pp. 79-107.

27) Kaykayoglu, R. and Rockwell, D., "Vortices Incident upon a Leading Edge: Instantaneous Pressure Fields," *Journal of Fluid Mechanics*, Vol. 156, 1985, pp. 439-461.

28) Dowling, A.P., "The Effect of Large-Eddy Breakup Devices on oncoming Vorticity," *Journal of Fluid Mechanics*, Vol. 160, 1985, pp. 447-463.

29) Sahlin, A., Alfredsson, P.H. and Johansson, A.V., "Direct Drag Measurements for a Flat Plate with Passive Boundary Layer Manipulators," *Journal of Fluid Physics*, Vol. 29, No. 3, March 1986, pp. 696-700.

30) Browne, M.W., "Design Advances Point to Cheap Supersonic Flight," *The New York Times*, *Science Times*, March 3, 1987, Sec. C.

31) Vukoslavcevic, P., Wallace, J.M., and Balint, J.-L., "Viscous Drag Reduction Using Streamwise-aligned Riblets," *AIAA Journal*, Vol. 30, No. 4: Technical Notes, April 1992, pp. 1119-1122.

32) Mojola, O.O. and Young, A.O., "An Experimental Investigation of the Turbulent Boundary Layer along a Streamwise Corner," *AGARD Conference Proceedings*, No. 93, London 1971, pp. 12.1-12.9.

33) Hunt, J.C.R., Abell, C.J., Peterka, J.A. and Woo, H., "Kinematical studies of the flows around free or surface-mounted obstacles; applying topology to flow visualization," *Journal of Fluid Mechanics*, Vol. 86, Part 1, 1978, pp. 179-200.

- 34) Lakshminarayana, B. and Horlock, J.H., "Review: Secondary Flows and Losses in Cascades and Axial Flow Turbomachines," *International Journal of Mechanical Science*, Vol.5, 1963, pp.287-307.
- 35) Dunham, J., "A Review of Cascade Data on Secondary Losses in Turbines," *Journal of Mechanical Engineering Science*, Vol.12, No.1, 1970, pp.48-59.
- 36) Langston, L.S., "Cross Flow in a Turbine Cascade Passage," *Trans.ASME, Journal of Engineering for Power*, Vol.102, No.4, 1980, pp.866-874.
- 37) Sieverding, C.H., "Recent Progress in the Understanding of Basic Aspects of Secondary Flows in Turbine Blade Passages," *ASME Journal of Engineering for Gas Turbines and Power*, Vol. 107, April 1985, pp. 248-257.
- 38) Chung, J.T., Simon, T.W., and Buddhavarapu, J., "Three-Dimensional Flow Near the Blade/Endwall Junction of Gas Turbine: Application of a Boundary Layer Fence," *ASME Paper No. 91-GT-45*, 1991.
- 39) Tang, Y.P., Chen, F., and Chen, M.Z., "Experimental Investigation of Vortex Structure in Corner Region of a Linear Compressor Cascade," *ASME Paper No. 91-GT-158*, 1991.
- 40) Hornung, H.G. and Joubert, P.N., "The Mean Velocity Profile in Three-Dimensional Turbulent Boundary Layers," *Journal of Fluid Mechanics*, Vol.15, Part 3, 1963, pp.368-384.
- 41) Coles, D.E., "The Law of the Wake of the Turbulent Boundary Layer," *Journal of Fluid Mechanics*, Vol.1, 1956, pp.191-226.

- 42) Johnston, J.P., "On the Three-Dimensional Turbulent Boundary Layer Generated by Secondary Flow," ASME Journal of Basic Engineering , Vol.82, March 1960, pp.233-246.
- 43) Johnston, J.P., "The Turbulent Boundary Layer at a Plane of Symmetry," ASME Journal of Basic Engineering, Vol.82, Sept. 1960, pp.622-628.
- 44) Francis, G.P. and Pierce, F.G., "An Experimental Study of Skewed Turbulent Boundary Layers in Low Speed Flows," Journal of Basic Engineering, Sept.1967, pp.597-608.
- 45) Klinksiak, W.F. and Pierce, F.J., "Simultaneous Lateral Skewing in a Three Dimensional Turbulent Layer Flow," ASME Journal of Basic Engineering, Vol.92, 1970, pp.83-92.
- 46) Stanbrook, A., "Experimental Observation of Vortices in Wing-Body Junctions," RAE Aero. Report 2589, 1957. (Also Aeronautical Research Council , Reports and Memoranda, RM 3114, 1959).
- 47) Shwind, R.G., "Three-Dimensional Boundary Layer Near a Strut," MIT GT Report No.67, 1962.
- 48) Oguz, E.A., "An Experimental Investigation of the Turbulent Flow in the Junction of a Flat Plate and Body of Constant Thickness," Ph.D. Thesis, Georgia Institute of Technology, 1981.
- 49) East, L.F. and Hoxey, R.P., "Boundary Layer Effect in an Idealized Wing-Body Junction at Low Speed," Royal Aircraft Establishment Tech. Report TR68161, 1968.
- 50) East, L.F. and Hoxey, R.P., "Low Speed Three-Dimensional Turbulent Boundary Layer Data-Part2, " Royal Aircraft Establishment Tech. Report TR69137, 1969.

- 51) Prahlad, T.S., "Wall Similarity in Three-Dimensional Turbulent Boundary Layers," AIAA Journal, Vol.6, No.9, Sep.1968, pp.1772-1774.
- 52) Hsing, T.-D. and Teng, H.-Y., "Experimental Study of the Behavior of 3D-Turbulent Boundary Layer in a Simplified Wing/Body Junction," AIAA-84-1529.
- 53) Peake, D.J., Rainbird, W.J. and Atraghji, E.G., "Three-Dimensional Flow Separations on Aircraft and Missiles," AIAA Journal, Vol.10, No.5, 1972, pp.567-580.
- 54) Kaufman II, L.G., Korkeg, R.H. and Morton, L.G., "Shock Impingement Caused by Boundary layer Separation Ahead of Blunt Fins," AIAA Journal, Vol.11, No.10, Oct.1973, pp.1363-1364.
- 55) Han, L.S., Ma, C and Rapp, J.R., "The Endwall Influence on Heat Transfer From a Single Cylinder (The Horseshoe Vortex Effect)," GTSJ/ASME, 1983 Tokyo International Gas Turbine Congress, Paper No.83-TOKYO-IGTC-3.
- 56) Blair, M.F., "Heat-Transfer in the Vicinity of a Large-Scale Obstruction in a Turbulent Boundary Layer," Journal of Propulsion and Power, Vol.1, 1985, pp. 158-160.
- 57) Young, A.D., "Some Special Boundary Layer Problems", Z.Flugwis, Weltraumforsch 1, Heft 6, 1977, pp.401-414.
- 58) Raily, J.W. and Sharma, P.B., "Treatment of the Annulus Wall Boundary Layer Using a Secondary Flow Hypothesis", ASME Paper No.76-GT-52.

- 59) Mellor , G.L .and Wood , G.M., "An Axial Compressor End-Wall Boundary Layer Theory," Journal of Basic Engineering , Transaction ASME, Series D, Vol.93, June 1971, pp.300-318.
- 60) Murray, J.C., "Incompressible Flow Past a Wing-Body Combination Using General Curvilinear Coordinates," Aero Quart., August 1979, pp.451-470.
- 61) Shang, J.S., "Numerical Simulation of Wing-Fuselage Aerodynamic Interaction," AIAA-83-0225. AIAA 21st Aerospace Science meeting, Jan. 10-13, 1983, Reno, Nevada.
- 62) Kubendran, L.R., "Study of Turbulent Flow in a Wing-Fuselage Type Juncture," Ph.D. Thesis, Georgia Institute of Technology, 1983.
- 63) Kubendran, L.R., McMahon H.M., and Hubbart, J.E., "Turbulent Flow Around a Wing/Fuselage-Type Juncture," AIAA Journal, Vol.24, No.9, Sep.1986, pp.1447-1452.
- 64) Thomson, W.G., "A Study of Wing-Body Aerodynamic Interference of a Light Twin Aircraft Utilizing a Panel Method Potential Flow Program", Ph.D. Thesis, Wichita State University, 1982.
- 65) Manor, D., "Low Aspect Ratio Wing/Body Vortex Interaction at Large Angles of Pitch and Yaws," Ph.D. Thesis, Wichita State University, 1983.
- 66) Mehta, R.D., "Effect of Wing Nose Shape on the Flow in a Wing/Body Junction," Aeronautical Journal, Dec.1984, pp.456-460.
- 67) Baker, C.J., "The laminar horseshoe vortex," Journal of Fluid Mechanics, Vol.95, Part 2, 1979, pp. 347-367.

68) Baker, C.J., "The turbulent horseshoe vortex," *Journal of Wind Engineering and Industrial Aerodynamics*, Vol. 18, No. 3, 1985, pp. 263-274.

69) Moor, J., and Forlini, T.J., "A Horseshoe Vortex in a Duct," ASME paper No. 84-GT-202.

70) Eckerle, W.A. and Langston, L.S., "Horseshoe Formation Around a Cylinder," ASME paper No. 86-GT-246 Presented at the International Gas Turbine Conference and Exhibit, Dusseldorf, West Germany-June-8-12, 1986.

71) Scheiman, J. and Kubendran, L. R., "Juncture Flow Measurements Using Laser Velocimetry", AIAA-85-1612, AIAA 18th Fluid Dynamics and Plasmadynamics and Laser Conference, Cincinnati, Ohio.

72) Hasan, M.A.Z., Casarella, M.J. and Rood, E.D., "An Experimental Study of the Wall Pressure Field Around a Wing-Body Junction," *Transactions of ASME, Journal of Vibration, Acoustics, Stress, and Reliability in Design*, Vol. 108, 1986, pp. 308-314.

73) Lai, K.Y.M. and Makomaski, A.H., "Three-Dimensional Flow Pattern Upstream of a Surface-Mounted Rectangular Obstruction," *Journal of Fluids Engineering*, Vol.111, December 1989, pp.449-456.

74) Menna, J.D. and Pierce, F.J., "The Mean Flow Structure Around and Within a Turbulent Junction or Horseshoe Vortex-Part I: The Upstream and Surrounding Three-Dimensional Boundary Layer," *Journal of Fluids Engineering*, Vol.110, 1988, pp.406-414.

75) Pierce, F.J. and Harsh, M.D., "The Mean Flow Structure Around and Within a Turbulent Junction or Horseshoe Vortex-Part II: The Separated and Junction Vortex Flow," *Journal of Fluids Engineering*, Vol.110, 1988, pp.415-423.

- 76) Pierce, F.J. and Tree, I.K., "The Mean flow Structure on the Symmetry Plane of a Turbulent Junction Vortex," *Journal of Fluids Engineering*, Vol.112, 1990, pp.16-22.
- 77) Pierce, F.J., "Standard Test Case for a Low-Speed, Turbulent Junction Vortex Flow," *AIAA Journal*, Vol.29, No.1, 1991, pp.14-15.
- 78) Devenport, W.J. and Simpson, R.L., "LDV Measurements in the Flow Past a Wing-Body Junction," 4th International Symposium on Applications of LDA to Fluid Mechanics", Lisbon, Portugal, 1988.
- 79) Falco, R.E., "An Album of Fluid Motion," Assembled by M. Van Dyke, Parabolic press, Stanford, California, 1982, Photograph No.165.
- 80) Devenport, W.J. and Simpson, R.L., "Time-dependent and Time-averaged Turbulence Structure Near the Nose of a Wing-body Junction," *Journal of Fluid Mechanics*, Vol. 210, 1990, pp. 23-55.
- 81) Eckerle, W.A. and Awad, J.K., "Effect of Free Stream Velocity on the Three-Dimensional Separated Flow Region in Front of a Cylinder," *Journal of Fluids Engineering*, Vol.113, 1991, p.37-44.
- 82) Hung, C.-M., Sung, C.-H. and Chen, C.-L., "Computation of Saddle Point of Attachment," *AIAA*, Vol.30, No.6, June 1992.
- 83) Chen, C.-L. and Hung, C.-M., "Numerical Study of Junction Flows," *AIAA Journal*, Vol. 30, No. 7, July 1992, pp.1800-1807.
- 84) Agui, J.H., and Andreopoulos, J., "Experimental Investigation of a Three Dimensional Boundary Layer Flow in the Vicinity of an Upright Wall Mounted Cyclinder," *ASME, Journal of Fluids Engineering* (in press), 1992.

- 85) Hawthorne, W.R., "The Secondary Flow About Struts and Airfoil," *Journal of the Aeronautical Sciences*-September, 1954. pp.588-606.
- 86) Rood, E.P. and Keller, J.E.C., "Evidence of Large Scale Time Dependent Flow in the Wing-Wall Interaction Wake," *Unsteady Turbulent Boundary Layers and Friction*, ASME FED- Vol.12, 1984, pp.39-44.
- 87) Rood , E.P., "The Separate Spatial Extents of the Trailing Horseshoe Vortex Legs from a Wing and Plate Junction," AIAA-84-1526. AIAA 17th Fluid Dynamics, Plasma Dynamics, and Laser Conference, June-25-27,1984 / Snowmass, Colorado .
- 88) Rood, E.P. and Anthony, D.G., "Tail Profile Effects On Unsteady Large Scale Flow Structure in the Wing and Plate Junction," *Forum on Unsteady Flow*, edited by P.H. Rothe, ASME FGD 27, 1985.
- 89) Kubendran, L.R., and W.D. Harvey, "Juncture Flow Control Using Leading-Edge Fillets," AIAA 3rd Applied Aerodynamics Conference, October 14-16, 1985, Colorado Springs, Colorado, AIAA-85-4097.
- 90) Devenport, W.J., Agarwal, N.K., Dewitz, M.B., Simpson, R.L. and Poddar K., "Effects of a Fillet on the Flow Past a Wing-Body Junction," *AIAA Journal*, Vol.28, No.12, 1990, pp.2017-2024.
- 91) McMahon, H.M., Metari, P. and Yoo, K.M., "Mean Velocities and Reynolds Stresses in the Juncture Flow and in the Shear Layer Downstream of an Appendage," GITAER Report, 1987, Georgia Institute of Technology.
- 92) Ozcan, O. and Olcmen, M.S., "Measurements of Turbulent Flow Behind a Wing-Body Junction," *AIAA Journal*, Vol.26, No.4, April 1988, pp. 494-496.

- 93) Chang, P.S. and Gessner, F.B., "Experimental Investigation of Flow About a Strut-Endwall Configuration," AIAA Journal, Vol.29, No.12, December 1991, pp.2105-2114.
- 94) Settles, G.S., "Modern Developments in Flow Visualization," AIAA-84-1599, Paper presented at AIAA 17th Fluid Dynamics, Plasma Dynamics, and Laser Conference, June 25-27, 1984, Snowmass, Colorado.
- 95) Langston, L.S. and Boyle, M.T., "A New Surface-Streamline Flow Visualization Technique," Journal of Fluid Mechanics, Vol.125, 1982, pp.53-57.
- 96) Brown, K.C. and Joubert, P.N., "The Measurement of Skin Friction in Turbulent Boundary Layers with Adverse Pressure Gradients," Journal of Fluid Mechanics, Vol.35, part 4, 1969, pp.737-757.
- 97) Winter, K.G., "An Outline of the Techniques Available for the Measurements of Skin Friction in Turbulent Boundary Layers," Prog. Aero. Sci., Vol.18, 1977, pp.1-57.
- 98) Prahlad, T.S., "Yaw Characteristics of Preston Tubes," AIAA Journal, Technical Notes, Vol.10, No.3, March 1972, pp.357-359.
- 99) Bragg, G.M., "The Turbulent Boundary Layer in a Corner," Journal of Fluid Mechanics, Vol.36, 1969, pp.485-503.
- 100) Preston, J.H., "The Determination of Turbulent Skin Friction by Means of Pitot Tubes," Journal of Royal Aeronautical Society, Vol.58, Feb. 1954, pp.109-121.
- 101) McAllister, J.E., Pierce F.J. and Tennant, M.H., "Direct Force Wall Shear Measurements in Pressure-Driven Three-Dimensional Turbulent Boundary Layers," Journal of Fluids Engineering, Vol.104, June 1982, pp.150-155.

102) Sproston, J.L. and Goksel , O.T., "The Calibration of a Surface Static Tube," Aero. Journal ,Technical Notes, Feb.1972, pp.101-103.

103) Bhargava, R. K., and Raj, R., "Surface Static Tube Accuracy in Three-Dimensional Turbulent Flows", *Proceeding, 31st Heat Transfer and Fluid Mechanics Institute*, June 1989, pp.231-240.

104) Abdulla, A.K., Bhargava, R.K., and Raj, R., "An Experimental Study of Local Wall Shear Stress, Surface Static Pressure and Flow Visualization Upstream, Alongside, and Downstream of a Blade Endwall Corner," ASME Journal of Turbomachinery, Vol.113, October 1991, pp.626-632.

105) Devenport, W.J. and Simpson, R.L., "Some Time-Dependent Features of Turbulent Appendage-Body-Juncture Flows," Proceedings of the 16th Symposium on Naval Hydrodynamics, National Academy Press, Washington, DC, 1986, pp.312-335.

106) Munukutla , S.S., "Turbulent Wake Development Behind Streamlined Bodies," Ph.D. Thesis, University of Iowa, 1981.

107) Wright, M.A., "The Evaluation of a Simplified Form of Presentation for Five-Hole Spherical and Hemispherical Pitometer Calibration Data," Journal of Physics, E: Scientific Instruments, Vol.3, 1970, pp.356-362.

108) Winternitz , F.A.L., "A Comparative Survey of Different Types of Instrument," Aircraft Engineering, Aug. 1956, pp. 273-278.

109) Traester, A.L., and Yocum, A.M., "The Calibration and Application of Five-Hole Probes," Proceedings of 24th International Instrumentation Symposium, Instrument Society of America, 1978, pp.255-266 .

110) Matsunaga, S., Ishibashi, H. and Nishi, M., "Accurate Measurement of Nonsteady Three-Dimensional Incompressible Flow by Means of a Combined Five-Hole Probe," Proceedings ASME Conference on " Nonsteady Fluid Dynamics ," Edited by D.E. Crow and J.A. Miller, 1978, pp.65-72.

111) Sitaram, N. , Lakshminarayana, B. and Ravindranath, A., "Conventional Probes for the Relative Flow Measurement in a Rotor Blade Passage," ASME Conference on Measurement Methods of Rotary Components of Turbomachinery, 1980, pp.59-69.