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SEPARATED FLOW AND BOUNDARY LAYER RESEARCH.(U)
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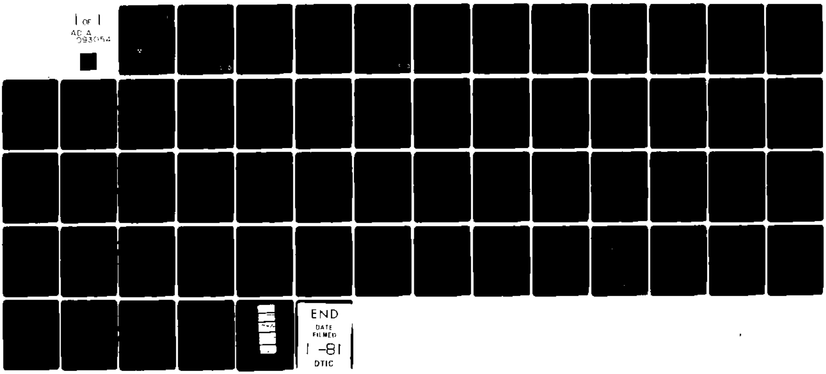

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LEVEL II

SEPARATED FLOW AND BOUNDARY LAYER RESEARCH

S. M. Bogdonoff
G. S. Settles

Final Report For

CONTRACT F44620-75-C-0080
26 APRIL 1975 through 31 JULY 1980

MAE REPORT 1494

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REPORT DOCUMENTATION PAGE

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1. REPORT NUMBER: AFOSR-TR- 80 - 1252	2. GOVT ACCESSION NO.: AD-A093054
	3. RECIPIENT'S CATALOG NUMBER
4. TITLE (and Subtitle): SEPARATED FLOW AND BOUNDARY LAYER RESEARCH	5. TYPE OF REPORT & PERIOD COVERED: Final rept. 26 Apr 75 - 31 Jul 80
	6. PERFORMING ORG. REPORT NUMBER
7. AUTHOR(s): Seymour M. Bogdonoff, Gary S. Settles	8. CONTRACT OR GRANT NUMBER(s): F44620-75-C-0080
9. PERFORMING ORGANIZATION NAME AND ADDRESS: Gas Dynamics Laboratory, Forrestal Campus, Princeton University, Princeton, NJ 08544	10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS: 2307/A1 61102F
11. CONTROLLING OFFICE NAME AND ADDRESS: Air Force Office of Scientific Research, Building 410, Bolling AFB, Washington, D. C. 20332	12. REPORT DATE: October 1980
	13. NUMBER OF PAGES: 55
14. MONITORING AGENCY NAME & ADDRESS (if different from Controlling Office)	15. SECURITY CLASS. (of this report): Unclassified
	15a. DECLASSIFICATION DOWNGRADING SCHEDULE

16. DISTRIBUTION STATEMENT (of this Report)

Approved for public release; distribution unlimited.

17. DISTRIBUTION STATEMENT (of the abstract entered in Block 20, if different from Report)

18. SUPPLEMENTARY NOTES

19. KEY WORDS (Continue on reverse side if necessary and identify by block number)

supersonic, turbulent, boundary layer, shock wave, compression corner, separation, flowfield, reattachment, hot-wire anemometry, hypersonic

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The work carried out under the subject contract has generated many new views of high speed fluid mechanics. The two-dimensional corner study provides a unique framework bridging attached to separated flows. It only needs the measurements of local heat transfer and flowfield fluctuating measurements to be a complete description of an important shock wave boundary layer configuration.

The study of a two-dimensional reattaching free shear layer has provided a →

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unique series of measurements, including the first turbulence data, at one flow condition and geometry. Reynolds number and geometry effects have yet to be studied, but the physics of reattachment have been explored in a way not heretofore examined.

The investigation of three-dimensional shock wave turbulent boundary layer interactions, specifically of the swept wedge, provides new insights into these complex flows with strong lateral gradients. The studies have revealed many new elements, but "separation" and "reattachment" take on new meanings which have yet to be clarified.

Hypersonic turbulent boundary layers have been found to have very complex structures with very wide density variations and orders of magnitude unit Reynolds number changes across the layers. Highly viscous, laminar sublayers, are bounded by turbulent layers in which fluctuations of 50% are experienced.

Present facilities, instrumentation, and data handling techniques (developed during this contract) have been adequate to explore the selected regimes and geometries.

Many of the results of the present studies were used to test, validate, and guide major efforts in computational fluid dynamics. To date, the computations adequately describe attached flows, but do not capture the details of separated flows, although general characteristics of the flow field are predicted. The results of computations, sensitivity analysis, and testing of various models (turbulence) were important guides for the experimental studies.

SEPARATED FLOW AND BOUNDARY LAYER RESEARCH

S. M. BOGDONOFF
G. S. SETTLES

FINAL REPORT FOR

CONTRACT F44620-75-C-0080
26 APRIL 1975 THROUGH 31 JULY 1980

MAE REPORT 1494

APPROVED FOR PUBLIC RELEASE; DISTRIBUTION UNLIMITED

PREPARED FOR

AIR FORCE OFFICE OF SCIENTIFIC RESEARCH
BOLLING AIR FORCE BASE,
WASHINGTON, D.C.

PREPARED BY

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DEPARTMENT OF MECHANICAL AND AEROSPACE ENGINEERING
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OCTOBER 1930

PREFACE

This final report is a summary of the experimental research program in supersonic and hypersonic flows carried out by the staff of the Gas Dynamics Laboratory, Department of Mechanical and Aerospace Engineering, Princeton University. The investigations were sponsored by the Air Force Office of Scientific Research under Contract Number F44620-75-C-0080. The program was monitored by Dr. James D. Wilson, Program Manager, Directorate of Aerospace Sciences.

This report covers the work performed during the period 26 April 1975 through 31 July 1980.

CONTENTS

LIST OF ILLUSTRATIONS

NOMENCLATURE

C	corner location
C_f	skin friction coefficient based on freestream conditions
L_M	swept compression corner upstream influence length
M	Mach number
Nu	Nusselt Number
P	Pressure
PØ	wind tunnel stagnation pressure
R	reattachment location
Re	Reynolds number
S	separation location
$S\rho u$	hot-wire mass flow sensitivity coefficient
S_{TT}	hot-wire total temperature sensitivity coefficient
u	mean velocity component parallel to model surface
x	distance along model surface measured from corner, taken as positive in the downstream direction
y	distance normal to model surface
α	corner angle, deg
β_k	equilibrium pressure gradient parameter (see Ref. 13)
δ	boundary-layer thickness
z	spanwise coordinate
η	yaw angle
λ	compression corner sweepback angle; also incoming shear layer width (see Ref. 13)
Π	wake component strength parameter (also PI)
σ	free shear layer spreading rate parameter (see Ref. 13)
τ_{wr}	$(T_w - T_r)/Tr$, hot-wire overheat ratio

Subscripts

AVG	average
e	boundary layer edge condition
i	incipient separation condition
K_1	critical point downstream of reattachment (see Ref. 12)
L, ℓ, LOCAL	local condition at any point in the flowfield
r	recovery
w	wall
0	initial incoming boundary-layer condition
∞	initial incoming freestream condition (also INF)
δ	based on δ

I. INTRODUCTION

High speed turbulent boundary layers and their characteristics under a wide range of imposed conditions, particularly adverse pressure gradients, are of key importance in a wide range of fluid mechanical problems ranging from high speed maneuvering aircraft to compressors and turbines and inlets. The interactions of boundary layers with shock waves poses a major problem in our predictions of many complex high speed flows. Although a considerable amount is known about the general characteristics of equilibrium high speed turbulent boundary layers, many elements of interaction phenomena, turbulent boundary layers in strong gradients and with separation, are not well understood and are the subject of intensive theoretical computation and experimental study. In the supersonic regime, two-dimensional flows have been studied extensively and much of the physical phenomenon is understood, although computations still cannot predict flows with separation. For three-dimensional flows, very little fundamental information is available and, it is our belief, much of the "understanding" is open to question. At hypersonic speeds, although many practical problems have been solved, the fundamental understanding of the unique characteristics of hypersonic turbulent boundary layer is still open to considerable discussion.

The studies undertaken under the present contract have concentrated on two main thrusts; one, to try to understand the physics involved in the flow interactions and thus to understand the key parameters which govern the flows and, two, (more recently) to provide classical "experiments" of critical flows in sufficient detail and redundancy to provide a guide for and evaluation of numerical computations. To carry out this program, the Laboratory developed facilities, instruments, and techniques to provide the results that were required. For the sort of major experimental program described herein, these developments are essential to generating the advanced experimental results needed.

The initial phase of this contract was primarily concerned with supersonic turbulent boundary layers. Shortly after its initiation, it was merged with a major program (which had been underway for some time, also under the sponsorship of the Air Force Office of Scientific Research) on the study of hypersonic boundary layers. For the first three years, therefore, the present contract covered both supersonic and hypersonic studies. The initial work was concerned with mean flow characteristics of several configurations. As time went on, the supersonic work concentrated primarily on mean two-dimensional flows, while the hypersonic work progressed to the point where details of the hypersonic boundary layers required the development of new tools to measure fluctuating quantities. The program has changed substantially over the period of this contract. After three years of work, the hypersonic flow studies were phased out in the light of major efforts to increase progress in key supersonic areas. The phasing out of the hypersonic work was accentuated by the departure of one of the key researchers, who became Technical Director of the Office of Naval Research. He had been carrying a major responsibility in the use of new techniques for hypersonic turbulent boundary layer investigations. The hypersonic work on turbulent boundary layers, using hot wires and electron beams, required the development of new data acquisition and processing techniques. This development, including a new high-speed mini-computer, became the basis for the continuing work at supersonic speeds and the ability to make fluctuating measruements under high Reynolds number supersonic conditions. The two-dimensional supersonic work expanded into the three-dimensional area, and a new series of studies was dictated by the major progress in computational fluid dynamics. Part of the original effort on computations was carried out in the Gas Dynamics Laboratories under other contracts, but the major thrust in this area has come from NASA and industry. The present program has been developed in close contact with these efforts. New experiments were developed to provide a base for the validation of these new computations which specifically lacked a method of handling the turbulence "closure" problem in the full solution of the Navier Stokes equations.

Significant progress has been made during the period of this contract. A brief review of the goals and results which were obtained are given in the following sections.

II. RESEARCH GOALS

Although the general goals of the present studies were primarily associated with enhancing the understanding of high speed turbulent boundary layers and separation, more specific goals were placed on various elements of the program. This was dictated, in large part, by the "maturity" of our understanding of the phenomena and the status of the research at a particular time.

A. Two-Dimensional Shock Wave Boundary Layer Interactions

This phenomenon has been most intensively studied, not only by the Gas Dynamics Laboratory, but by a wide range of other investigators in other laboratories. The present program has aimed at a specific set of details which were designed to clarify key elements of the interaction.

1. The detailed response of the turbulent boundary layer during and downstream of an interaction.

2. An examination of the response of the outer potential flow and the generation of shock structures which are of primary importance in the generation of the detailed flow field.

3. The development and scaling of flow separation, the most critical phenomenon in the understanding of turbulent boundary layer interactions.

4. The examination of the effect of Reynolds number on the interactions which were studied.

5. The examination of the physics of the shear layer reattachment which is the key element in the determination of the scaling and downstream effects of separated flows.

B. Three-Dimensional Shock Wave Boundary Layer Interactions

In contrast to two-dimensional interactions, three-dimensional flows have received much less attention. The understanding of such flows is in a rudimentary state and, as a result, the overall goals of these studies differ considerably from that of Section A above.

1. The study of systematic variations from the supposedly well known two-dimensional "base" flow.

2. An examination of the overall behavior of three-dimensional interactions in terms of basic interaction parameters, for example, scaling.

3. A detailed examination of the physics of three-dimensional separation and reattachment, which are quite different from the well understood two-dimensional cases.

C. Hypersonic Turbulent Boundary Layer Studies

Although the differences between incompressible and compressible turbulent boundary layers are reasonably well understood, the extension to hypersonic speeds makes very significant changes. The details of hypersonic turbulent boundary layers have not been examined in great detail. Since there is no sharp demarcation between supersonics and hypersonics, variations in boundary layer characteristics occur gradually from incompressible to hypersonic flows. An examination of details under different conditions is required to see when approximations or assumptions, valid in one region, must be modified to obtain adequate results in others. The detailed studies of the structure of hypersonic turbulent boundary layers were undertaken to examine these variations.

D. Comparison with Theory and Computation

Although the primary work under the subject contract was experimental, the program was significantly influenced by attempts to work with the current state-of-the-art in theory and, more recently, in computational fluid dynamics. Both theory and computation have been used as guides in the development of critical experiments. In recent years, the experiments have been designed as critical tests, particularly for complex computations, where the adequacy of current turbulence models and the lack of ability to check numerical techniques requires such validation experiments. Both the theory and computation and comparison with experiment provide key information on future developments in all areas.

E. Facility and Instrumentation Development

The present program has used state-of-the-art instrumentation, data acquisition, and reduction techniques in carrying out its major experimental programs. However, in many elements of the programs, extensions, expansions, and new developments were required to make the data obtained more pertinent to the subject under investigation or to provide the capability required for specific tests. An important goal was to have the capacity to make the critical measurements under conditions which were most appropriate. As an example, the study of the hypersonic turbulent boundary layer required the development of a test channel in which such a boundary layer could be generated with a minimum normal pressure gradient. The hypersonic hot-wire and electron beam techniques had to be developed to make measurements in heretofore unexplored regions. The computer development was required to make data acquisition and processing possible under conditions which were far from the usual "mean" studies of the past. Many of these techniques, developed for hypersonic flows, were adapted to much of the current work in supersonic flows.

III. SUMMARY OF RESEARCH RESULTS

During the contract period, a number of technical papers and presentations, reports, and student theses were written concerning the research and the results obtained. A complete listing of these publications is given in the Bibliography included in this Final Report. The following "summary of results" briefly reviews the detailed discussions which can be found in the complete publications, and notes some exploratory results which have not reached the stage of publication.

A. Two-Dimensional Compression Corner Experiments

A significant effort during this contract was devoted to the study of shock wave/boundary layer interactions at two-dimensional compression corners. These compression corners produce adverse pressure gradients which must be negotiated by an incoming, equilibrium flat-plate boundary layer. The response of the boundary layer to such disturbances was extensively examined.

The experimental program was carried out in the Princeton 20 x 20 cm (8 x 8 inch) high Reynolds number blowdown wind tunnel, which is sketched in Fig. 1. The compression corner models were mounted on the floor of wind tunnel section number two, where an equilibrium turbulent boundary layer had developed with an edge Mach number of 2.85. The freestream unit Reynolds number was 6.3×10^{7}/meters for all tests, and the incoming boundary layer thickness, δ_o , was 2.3 cm.

Four compression corner models with angles of 8, 16, 20 and 24 degrees were tested. Based on previous experience, these corner angles were chosen to produce fully attached flow (8^o), incipient separation (16^o), and separated flows (20^o and 24^o). Thus, the progressive strengthening of the shock/boundary layer interaction and the development of flow separation could be observed with fixed and known incoming conditions.

Each of these four flowfields was thoroughly surveyed using mean pitot pressure, static pressure, and total temperature probes. The analysis of these results allow one to describe the development of mean flow velocities through an interaction or, at a given position, observe the flow changes with changing interaction strength.

The development of flow separation at the compression corner with increasing interaction strength is shown by surface streak patterns in Figs. 2a-d. The 8° corner produces no separation, but the 16° corner shows evidence of the beginning of a separation bubble just upstream of the corner line. A 20° corner angle generates a noticeable separation bubble and, at 24°, this reverse-flow region has grown to about two initial boundary layer thicknesses in streamwise extent.

In general, these surface streak traces demonstrate that two-dimensionality was preserved throughout the range of test conditions. There are weak three-dimensional perturbations superimposed upon the otherwise straight separation and reattachment lines, which are most noticeable in Fig. 2d. These perturbations have a cellular character which is related to the vortical instability of the turbulent boundary layer when subject to streamline curvature (see, for example, Ref. 1).

An important goal in these experiments was to determine the response of the compressible turbulent boundary layer to the imposed adverse pressure gradients. The measured response is shown graphically in Fig. 3, where the wake-strength parameter of the Coles (Refs. 2, 3) wall-wake velocity profile is plotted versus normalized distance downstream of the compression corner. The experimentally determined wake-strength parameter decreases rapidly with downstream distance for the three largest compression corner angles (incipient and separated flows). In physical terms, the turbulent boundary layer is retarded by the adverse pressure gradient but recovers its "fullness" quickly thereafter. The boundary layer has not regained a flat-plate equilibrium condition after traveling a short distance downstream, but the rate of recovery is quite rapid.

The fact that these downstream boundary layer profiles have not attained equilibrium is illustrated in Fig. 4, which shows the measured static pressure across the boundary layer for all four compression corner angles. There is a lingering gradient of static pressure across each layer, even though the measurements were made downstream of the point where the streamwise pressure gradient had disappeared. This normal pressure gradient is most noticeable in the case of the 24° corner, where it amounts to approximately a 10% drop in static pressure from the wall to the boundary layer edge. The solid lines in Fig. 4 are the results of numerical computations (discussed in the next section) which do not predict the measured pressure gradients.

The experimental compression corner flowfields described have been simulated by numerical computations carried out at the NASA Ames Research Center (Refs. 4 and 5). These solutions were obtained on a high-speed computer using several different assumed turbulence models for closure of the time-averaged compressible Navier-Stokes equations. Specifically, these turbulence models were an algebraic eddy viscosity function, a model using an additional partial differential equation for the kinetic energy of turbulence, and a model using still another partial differential equation for the turbulence length scale. Referred to as the zero-, one-, and two-equation models, all three amount to successively more complex representations of the eddy viscosity concept. The details of these models are available in several publications (e.g., Ref. 5) and are not recounted here.

Comparisons of the computed results with the present experimental data showed that the three turbulence models all lead to similar results. Certain features of the experiment were predicted, and others were not. Overall, the one-equation model seems to have done slightly better than the others and, to avoid confusion in the plotted comparisons, solutions using this model are the only ones illustrated here.

The computed and experimental surface static pressure distributions on the compression corner models are shown in Figs. 5a and 5b. The agreement of the one-equation model results with the 8° and 16° corner data, in Fig. 5a, is excellent. However, Fig. 5b shows that the computational predictions of the 20° and 24° surface pressure data are not good. This contrast is believed to reflect an inadequacy of the turbulence modeling schemes where significant flow separation is involved (as it is in the 20° and 24° data, but not in the 8° and 16° data). Comparisons with computations with zero- and two-equation turbulence models yielded the same overall conclusion.

A further comparison of measured and computed results is shown in Fig. 6, which contains the boundary layer velocity profiles at selected stations along the 24^{o} compression corner interaction. While the incoming boundary layer profile is matched exactly by the computed solutions, the agreement between the two becomes progressively worse with increasing distance downstream. The computed solutions indicate a substantial reverse-flow region at the experimentally-observed reattachment point. Further, the rapid recovery of the downstream turbulent boundary layer is not simulated by solutions with any of the several turbulence models which were tried.

These results call attention to a critical inadequacy in the current state-of-the-art methods for Navier-Stokes modeling of compressible turbulent flow interactions. The computed solutions are only accurate for attached flows, where most of the flowfield is essentially inviscid. They fail to capture the details of flows which contain significant regions of viscosity-dominated flow separation, although the general characteristics are simulated.

The two-dimensional compression corner experiments were extended to examine the Reynolds number effect on incipient turbulent boundary layer separation. Incipient separation has been previously studied at the Gas Dynamics Laboratory over a range of moderate to high Reynolds number at Mach 3 (Ref. 6). During the present phase of the program, this study was extended downward in Reynolds number toward the regime of boundary layer transition in an effort to shed light on some long-standing discrepancies there. The techniques used to determine incipient separation at low turbulent Reynolds numbers were the same as those discussed in detail in Ref. 6.

The results of this study (Ref. 7) are summarized in Fig. 7, which is the standard plot of incipient separation compression corner angle, α_i, versus Reynolds number. The results of a number of investigators (using a variety of incipient separation criteria) are shown. All results were obtained at or about Mach 3.

The most direct conclusion of the present work is that, across three decades of Reynolds number variation, the incipient separation conditions do not vary greatly. The early study by Kuehn (Ref. 8) and that of Kessler

(Ref. 9) are suspect because of boundary layer tripping. A dip in the incipient separation level, followed by a subsequent rise, is noticed in the range of Reynolds numbers between 10^4 and 10^5 (based on δ_o). This phenomenon is thought to be due to the nonequilibrium development of the turbulent boundary layer downstream of transition, which is known to take place in this Reynolds number range.

B. Two-Dimensional Free Shear Layer Reattachment Experiments

Although there have been many studies of the separation of a turbulent boundary layer, the reattachment of a separated flow has usually been examined as part of an overall interaction. Seldom has the experiment been designed to control and identify the characteristics of the shear layer which are the initial conditions of the reattachment. As a result, reattachment studies have not been able to characterize or describe the details of reattachment with the same clarity as the separation process. The research program on shear layer reattachment carried out under this contract was designed to avoid these difficulties.

This work began with an axisymmetric double-cone test model in the Princeton 20 x 20 cm tunnel. The turbulent boundary layer separated at the base of the first cone, bridged a cavity, and reattached upon the second cone of larger total angle. By optimizing the cone angles and separation distance, a constant-pressure free shear layer was generated with no wave disturbance from the separation point. This preliminary study (Ref. 10) demonstrated the feasibility of the concept and identified its advantages and disadvantages. To overcome the most serious disadvantage, inadequate shear layer dimensions for detailed measurements, the double-cone test model was replaced by the two-dimensional geometry illustrated in Fig. 8. A turbulent boundary layer develops on a flat plate, separates over a cavity, and reattaches on an inclined planar ramp. The initial conditions for this flow were $M = 2.92$, $Re/m = 6.7 \times 10^7$, and $\delta_o = 0.3$ cm. The position of the ramp was adjusted so that there was no disturbance at the separation corner. Complete mean flowfield surveys were made. Fluctuation measurements were also made using hot-wire anemometry.

Figure 8 illustrates that the transition from a turbulent boundary layer to a free shear layer occurred without flow turning. The shear layer developed at constant pressure above a large, low-speed, recirculation region. The reattachment of the shear layer upon the ramp was accompanied by a distributed compression in which both the wall pressure and the skin friction rose smoothly, then leveled off in the downstream region of boundary layer re-development.

The free shear layer reached an asymptotic growth rate prior to its reattachment. As shown in Fig. 9, this growth rate agreed with that found in previous experiments of other investigators (Ref. 11). This asymptotic growth rate is the main condition required to establish mean equilibrium conditions of the shear layer, thus providing a well-defined incoming boundary condition for the reattachment process.

It was first suggested by Sirieix, et.al. (Ref. 12) that reattachment follows a similarity scaling similar to that of flow separation in a "free interaction." The results of the experiments carried out under the current contract confirmed that hypothesis. Figure 10 illustrates the wall pressure distributions of the Sirieix experiments and the present experiment in similarity coordinates. The correlation of different experiments with different Mach numbers, geometries, and downstream conditions is clearly shown.

Downstream of reattachment, a new turbulent boundary layer develops on the ramp. The state of this relaxing layer can be examined by correlating the shape of its velocity profile with the local strength of the adverse pressure gradient. This is done in Fig. 11, where the velocity profile wake-strength parameter, Π, is plotted versus the square root of the compressible equilibrium pressure gradient parameter, β_k. This comparison showed that the boundary layer relaxes along the same trend established by previous experiments, both compressible and incompressible. This implies a specific relationship between Π and β_k, wherein the boundary layer is in a local equilibrium condition, unaffected by its upstream history.

Since the adverse pressure gradient dissipates soon after reattachment, it followed that the local-equilibrium boundary layer must also recover rapidly from the affects of the pressure gradient. This rapid recovery is illustrated in Fig. 12, where several boundary layer velocity profiles are shown for stations between reattachment and the downstream end of the test model. The early profiles following reattachment bear the inflected wake character of the free shear layer, but the "filling out" of the profiles with downstream distance is quite rapid.

In addition to the mean profile measurements, a series of turbulent fluctuation surveys was carried out using hot-wire anemometry techniques to be described later in this Final Report. An overall result of these measurements is shown in Fig. 13. The local maximum of normalized mass flow fluctuations is shown as a function of distance along the flowfield in the streamwise direction. The maximum fluctuation level is observed to rise gradually along the shear layer, then more sharply in the compression region. A peak fluctuation level of about 45% is reached just after reattachment, followed by a rapid decrease which, however, remains much higher than that in the original boundary layer.

The complete details of this experimental study of shear layer reattachment may be found in Refs. 13 and 14.

C. Three-Dimensional Experiments with Swept Compression Corners

Under other Air Force sponsorship, the study of three-dimensional shock wave/boundary layer interactions was started before the initiation of the present contract. This effort, supplemented by other government support, revealed that the three-dimensional interaction appears to have many new and different phenomena than experienced in two-dimensional interactions. In an attempt to link the extensive two-dimensional work under the present contract with these three-dimensional flows, a new series of studies was initiated with the basic two-dimensional corner configuration.

Swept compression corners were examined in order to study the transition from two-dimensional to three-dimensional shock/boundary layer interactions in a systematic way. As shown in Fig. 14, these swept corner models were mounted on a planar test surface upon which an equilibrium turbulent boundary layer had developed. A series of compression corner models was built, in which the sweepback angle, λ, was varied incrementally from 0° to 60°, while the streamwise corner angle, α, was held constant. Three values of α, 10°, 16°, and 24°, were tested at an incoming Mach number of 2.95 and at several different Reynolds numbers (Ref. 15). The measurements included surface pressures and streaklines, and a few exploratory flowfield surveys.

A general sketch of a surface flow pattern is shown in Fig. 15, along with the definitions of some parameters derived from such patterns. This sketch illustrates a flow near the corner which develops through an "inception region" to a state of cylindrical symmetry with increasing spanwise distance. Such cylindrical symmetry was observed only for low to moderate sweepback angles, depending on the streamwise corner angle, α . At sweepback angles approaching 60°, a conical symmetry of the surface streaklines appeared to develop for all the cases tested.

An important question addressed by these experiments is the manner in which the streamwise extent of a three-dimensional shock/boundary layer interaction scales with changes in initial flow parameters. In particular, it is known that the incoming boundary layer thickness, δ , is a primary scaling parameter for two-dimensional interactions, but whether this is also true for three-dimensional interactions has been open to question.

The present results have revealed that scaling on boundary layer thickness does apply for swept compression corner flows. Figure 16 is a plot of the corner upstream influence distance, L_M , versus the spanwise coordinate, z , both normalized by the average incoming boundary layer thickness ahead of the corner line, δ_{AVG} . Data are shown for several different incoming boundary layer thicknesses, all of which collapse on a single curve in these normalized coordinates. The particular case illustrated in Fig. 16 ($\alpha = 24^{\circ}$, $\lambda = 40^{\circ}$) is representative of the boundary layer thickness scaling for the other corner angle combinations as well.

Another question arises concerning the effect of a change in the sweepback angle, λ , with other parameters held constant. The result for $\alpha = 16^{\circ}$ is shown in Fig. 17, where the normalized upstream influence length, L_M/δ_{AVG} , is plotted versus λ for two values of Reynolds number. The upstream influence length grows as λ is increased, though this growth rate is small near the zero-sweep limit of two-dimensional flow.

For comparison, a two-dimensional empirical formula for upstream influence, due to Roshko and Thomke (Ref. 16), is also shown in Fig. 17. This formula was evaluated using the normal components of freestream Mach number and the compression corner angle at each value of λ . This comparison shows that the observed trend of upstream influence with λ is in qualitative agreement with the assumption of two-dimensional flow normal to the corner but it appears to break down at higher sweep angles.

Finally, some exploratory flowfield surveys of the $\alpha = 24^{o}$, $\lambda = 40^{o}$ interaction using a three-hole yaw probe revealed the flowfield which is shown in Fig. 18. The most significant result of these measurements is the discovery that the yaw angle, η , of the flow away from the streamwise direction is small everywhere except in the immediate vicinity of the model surface. This same result was also noted in measurements of a different three-dimensional interaction by Oskam (Ref. 17 and 18) and by Kussoy and Horstman (Ref. 19).

The most recent experiments in this test series were conducted near the end of the contract period, and are still being analyzed. A technical paper on these results is planned for the near future (Ref. 20).

D. <u>Hot-Wire Anemometry in High Reynolds Number Supersonic Flow</u>

The development and use of hot-wire anemometry techniques in high Reynolds number supersonic flows has progressed for several years in the Gas Dynamics Laboratory under the current contract support. The hot-wire is potentially a powerful tool in the measurement of such flows, since it has a high frequency response and good spatial resolution. The problems encountered in this application of hot-wire anemometry were mainly concerned with wire survival, signal interpretation, and data acquisition and analysis.

The combination of very high Reynolds number and high turbulence intensities in supersonic shock/boundary layer interactions proved to be a harsh environment for a hot-wire probe. Many wires were lost before a sufficiently rugged construction technique was evolved. The technique which worked best involved welding the 5 micron tungsten wires to the stainless steel support prongs using a tungsten electrode and a carefully-controlled capacitor discharge. Some slack was given to the wire to avoid strain-gaging, and rubber cement "shock absorbers" were applied to the welds. Wires installed in this manner generally survived a number of surveys in an interacting flowfield.

The technique used for calibrating the wires is sketched in Fig. 19. The hot-wire probe and a parallel pitot pressure probe were mounted in the test section of a variable-density supersonic calibration wind tunnel. The pitot pressure, along with the tunnel stagnation pressure and temperature, were converted to analog signals in appropriate transducers, and then digitized and stored in the memory of a Hewlett-Packard 1000 minicomputer. The output of the DISA 55M10 Constant Temperature Hot-Wire Anemometer was digitized directly at a 500 kHz sampling rate, and also stored. A data analysis program then analyzes these results to yield tabulated and plotted values of the Nusselt number-Reynolds number relationship and the mass flow and total temperature sensitivity coefficients of the hot-wire.

The hot-wire signal interpretation is somewhat simplified by maintaining the wire overheat ratio, τ_{wr} , at relatively high values, between 1.0 and 1.3, during an experimental test. As shown in Fig. 20, this insured that the ratio of total temperature to mass flow sensitivity coefficients, $S_{TT}/S_{\rho u}$, is less than one, or, in other words, that the hot-wire signal is primarily a function of mass flow rate.

While work continues on the problems of hot-wire anemometry in high Reynolds number supersonic flow, the technique is now at a level where useful measurements can be made in shock/boundary layer interactions. Such measurements have been made in the shear layer reattachment study described earlier in this Final Report (see also Ref. 13).

E. Studies of Hypersonic Turbulent Boundary Layers

Extensive hypersonic boundary layer studies were carried out under previous sponsorship of the Air Force. These studies covered many configurations and test conditions in helium and nitrogen wind tunnels at Mach numbers from 11 to 25. The hypersonic work included in the present contract was primarily focussed on the specific problem of the structure of the turbulent hypersonic boundary layer. This study was carried out in the Hypersonic Helium Wind Tunnels because of their high Reynolds number capacity.

The pacing item in the measurements of fluctuating quantities in hypersonic turbulent boundary layer was the development of instrumentation to make the critical measurements and the development of a data acquisition system which would handle the type of data obtained. The classical hot-wire techniques of low speed flows were brought to bear on the special problems of hypersonic flow with its wide variation of density and speed. The fine wire total temperature probe, which had also been used at quite different conditions, was also applied. Finally, and probably most important, the electron beam, developed for measurements in highly rarefied flows, was extended to use in high density flows (quite different than the original conception of the electron beam development). The development of these instruments, and their application to hypersonic turbulent boundary layers, were carried out in several of the Gas Dynamics Laboratory's High Reynolds Number Helium Hypersonic Tunnels at a nominal Mach number of 16. The initial studies were carried out in the boundary layer on the wall of an available test section. The boundary layer was about 1.0 inch thick (2.5 cm) with a free stream Reynolds number of 174,000/inch (68,500 cm). Once the instrumentation and techniques were developed, a new test channel was built to generate a turbulent boundary layer with as small a normal pressure gradient as was possible to achieve. This development, which took several years, managed to generate a hypersonic turbulent boundary layer which was close to "relaxed-equilibrium" so that the results obtained can be directly correlated with supersonic turbulent boundary layer equilibrium results.

<u>Hot-wire in hypersonic helium</u>

Studies with the hot-wire resulted in the development of a hypersonic hot-wire and a demonstrated ability to use this hot-wire in a helium turbulent boundary layer. The details in Reference 21 show that, because of the very large density variation involved, special importance had to be attached to the recovery ratio of the wire, and to the conductive heat losses out of the ends of the wire. It was found that the recovery ratio of the wire in helium flow differed from that in air, and the experimental calibrations of the density sensitivity agreed with the predictions based on calculation of the effects of the end losses.

Electron beam fluorescent studies at hypersonic conditions

The electron beam fluorescent technique was developed to the point where it could be realistically used as a means of measuring density, temperature, and other thermodynamic state variable fluctuations in a compressible turbulent flow. The technique was primarily limited by electron beam operation at high number densities, and by the sensitivity of the fluorescent intensity to the flow quantitites of interest. This study added a valuable new tool, of non-intrusive character, which, combined with the hot-wire, gave the capability of making redundant measurements under the conditions of high Reynolds number hypersonic turbulent flows in helium.

Mini-computer development

As an essential element in the use of hot-wires and electron beams noted above, the computer system used for data acquisition and processing for mean flow measurements had to be replaced. The new one, with high frequency characteristics appropriate for the acquisition and processing of turbulence data was installed in 1977. The new system consisted of a Hewlett-Packard 1000 and a Preston Scientific A-D converter. There was some considerable difficulty in getting the original installation to meet the specifications but, by November 1977, the new fast data acquisition system was utilized in the time resolved turbulent measurement and has provided the basis for the Laboratory's continuing work in high frequency measurements.

Hypersonic turbulent boundary layer measurements

Both the hot-wire and the electron beam were used extensively in measuring the characteristics through a Mach 16 hypersonic turbulent boundary layer. The results presented in Refs. 21 and 22 show that the hypersonic turbulent boundary layer has characteristics quite different than a supersonic turbulent boundary layer. There is a large viscous sublayer, which is laminar. There are very strong fluctuations in the outer part of the flow. From the hot-wire results mean properties of the layer were derived showing the density varies by a factor of over 40 across the layer, Fig. 21. Most of the variation takes place in the outer half of the boundary layer with a very low density region close to the wall. The turbulence measurements showed that the RMS mass flux fluctuations were as large as 50% of the local mean values, Fig. 22. This level of fluctuation is much larger than those found at lower Mach numbers. From the electron beam, the fluctuations in density, temperature, and pressure were all found to be very large. They are so large and, in many cases, asymmetrically distributed about the mean, that it appears that a linearized treatment of them is impossible. In direct contrast to observations at low supersonic Mach number boundary layers, the pressure fluctuations observed in the present study are too large to be neglected in any theoretical treatment. The "intermittancy" of the flow at the outer edge of the boundary layer (fraction of time the flow is laminar) which has been seen in lower speed boundary layers, is also observed deep within the hypersonic layer at the edge of the laminar sublayer. This is clearly seen in Fig. 23 where the output from the electron beam is recorded for several stations across the boundary layer.

The work on hypersonic turbulent boundary layers was phased out in 1978 because of the requirement of increased emphasis on supersonic flows, and the departure of Professor Smith, who was primarily responsible for the development of the electron beam technique. The hot-wire facilities and experience, and the major development of the computer for high frequency measurements, has formed the base for the continuing studies of supersonic turbulent boundary layers. The hypersonic facilities and instrumentation were disassembled, but the system could be reactivated rather quickly.

IV. CONCLUDING REMARKS

The work carried out under the subject contract has generated many new views of high speed fluid mechanics. The two-dimensional corner study provides a unique framework bridging attached to separated flows and only needs the measurements of local heat transfer and flowfield fluctuating measurements to be a complete description of an important shock wave boundary layer configuration.

The two-dimensional reattaching free shear layer has provided a unique series of measurements, including the first of the fluctuating quantities, at one flow condition and geometry. Reynolds number and geometry effects have yet to be studied, but the physics of reattachment have been explored in a way not heretofore examined.

The three-dimensional shock wave turbulent boundary layer interactions continue to provide the major area for future work. Details of these three-dimensional flows have revealed many new elements, but "separation" and "reattachment" take on new meanings which have yet to be clarified.

Hypersonic turbulent boundary layers are very complex structures with very wide density variations and orders of magnitude unit Reynolds number changes across the layers. Highly viscous, laminar sublayers, are bounded by turbulent layers in which fluctuations of 50% are experienced. There is much to be done in studies to better understand these layers - and their response to gradients - but present priorities seem to relegate this work to the future.

Present facilities, instrumentation, and data handling techniques (developed during this contract) have been adequate to explore the selected regimes and geometries. Facility geometries and instrumentation will be limiting in the future, and continued advancements in these areas are required for future studies.

Many of the results of the present studies are and will be used to test, validate, and guide the major efforts in computational fluid dynamics.

At the same time, the results of computations, sensitivity analysis, and testing of various models (turbulence) will be important guides for future experimental studies.

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M.S.E. Thesis in Progress
R. Benvenuti
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F I G U R E S

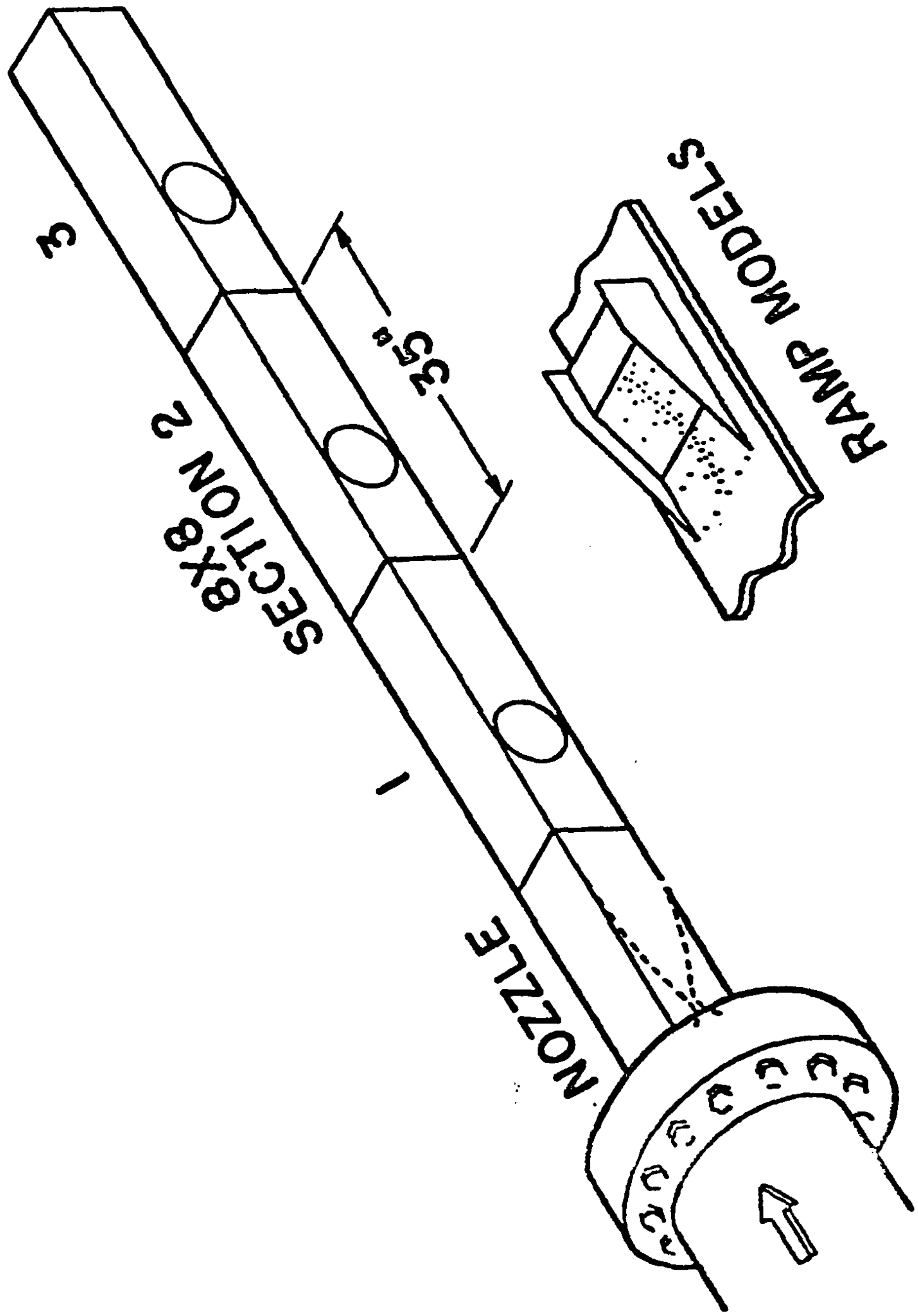


Figure 1 - Sketch of the Princeton 20x20cm (8x8 inch) Supersonic Wind Tunnel and Ramp Model Installation.

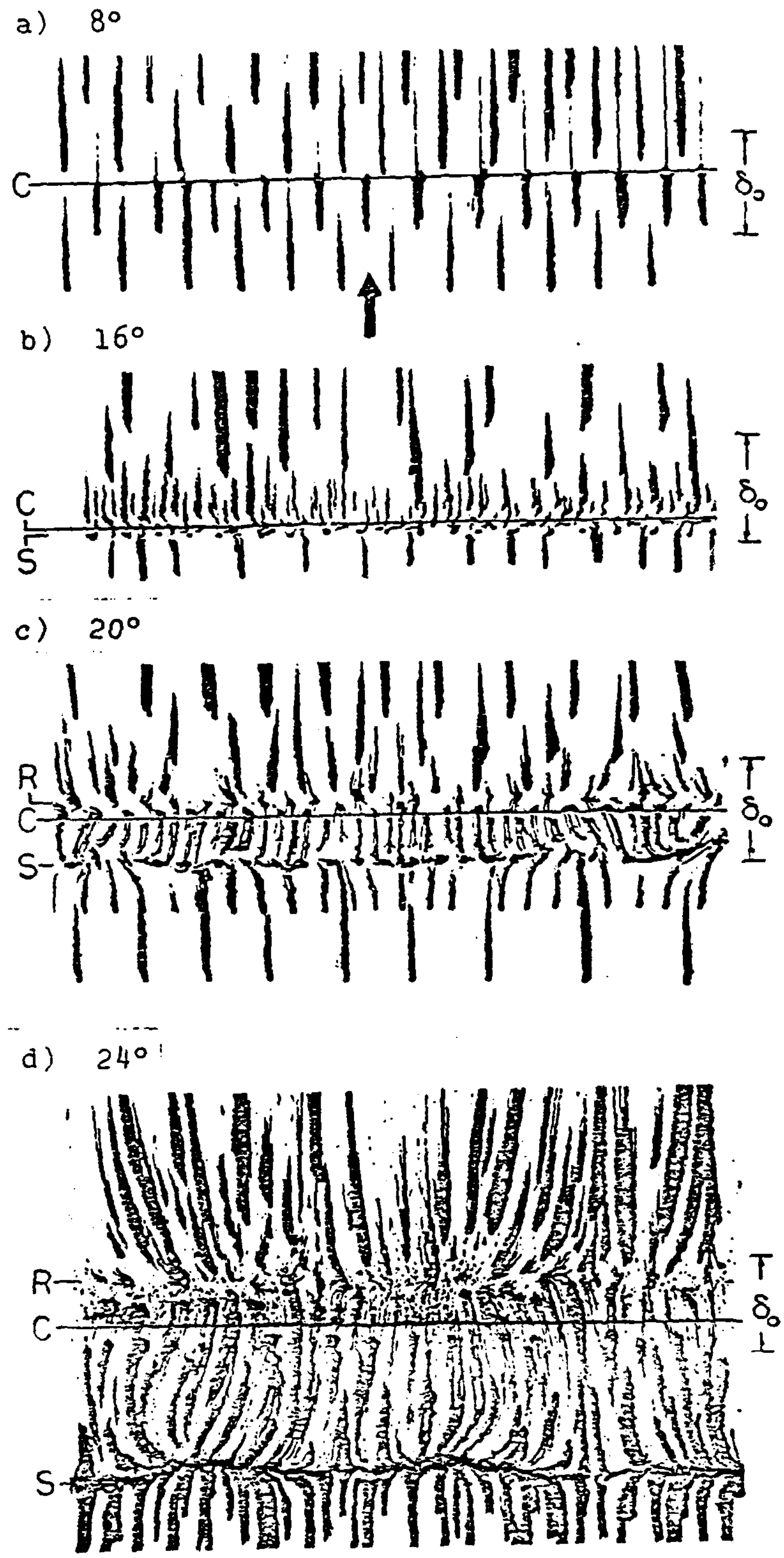


Figure 2a-d - Surface Streak Patterns From the 8°, 16°, 20°, and 24° Compression Corner Flowfields. (S,C, and R Denote Separation, Corner, and Reattachment Locations, Respectively. The Streamwise Direction is From Bottom to Top in Each Case. The Incoming Boundary Layer Thickness, δ_o, is noted for a Scaling Reference.

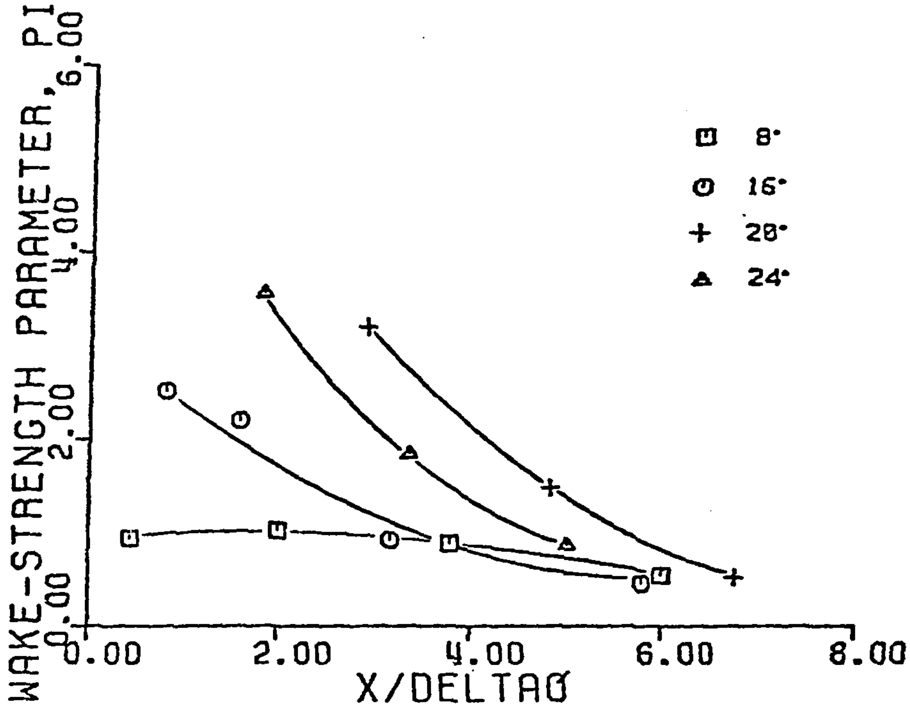


Figure 3 - Decay of the Velocity Profile Wake-Strength Parameter with Distance Downstream of Each Compression Corner.

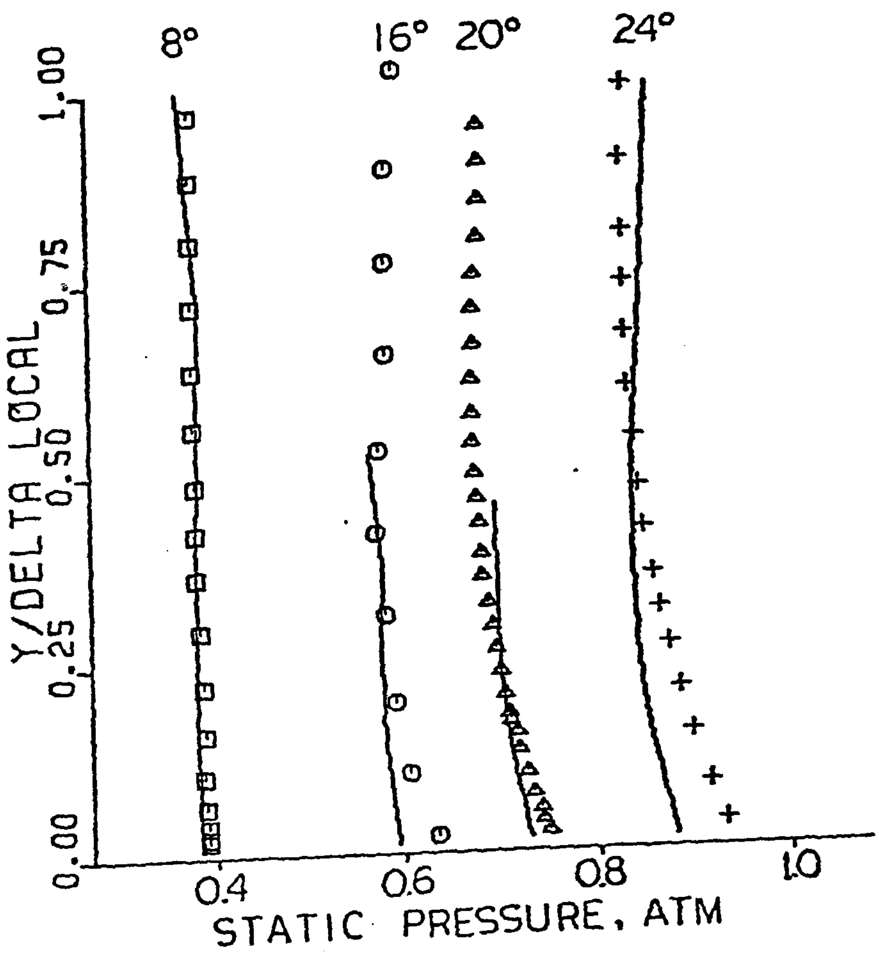


Figure 4 - Comparison of Experimental and Computed Downstream Static Pressure Profiles for the Four Compression Corner Flowfields. (Profile Locations are at $x/\delta_0 \approx 5$ for $\alpha = 8$, 16, and 24 deg and $x/\delta_0 = 2.75$ for $\alpha = 20$ deg.)

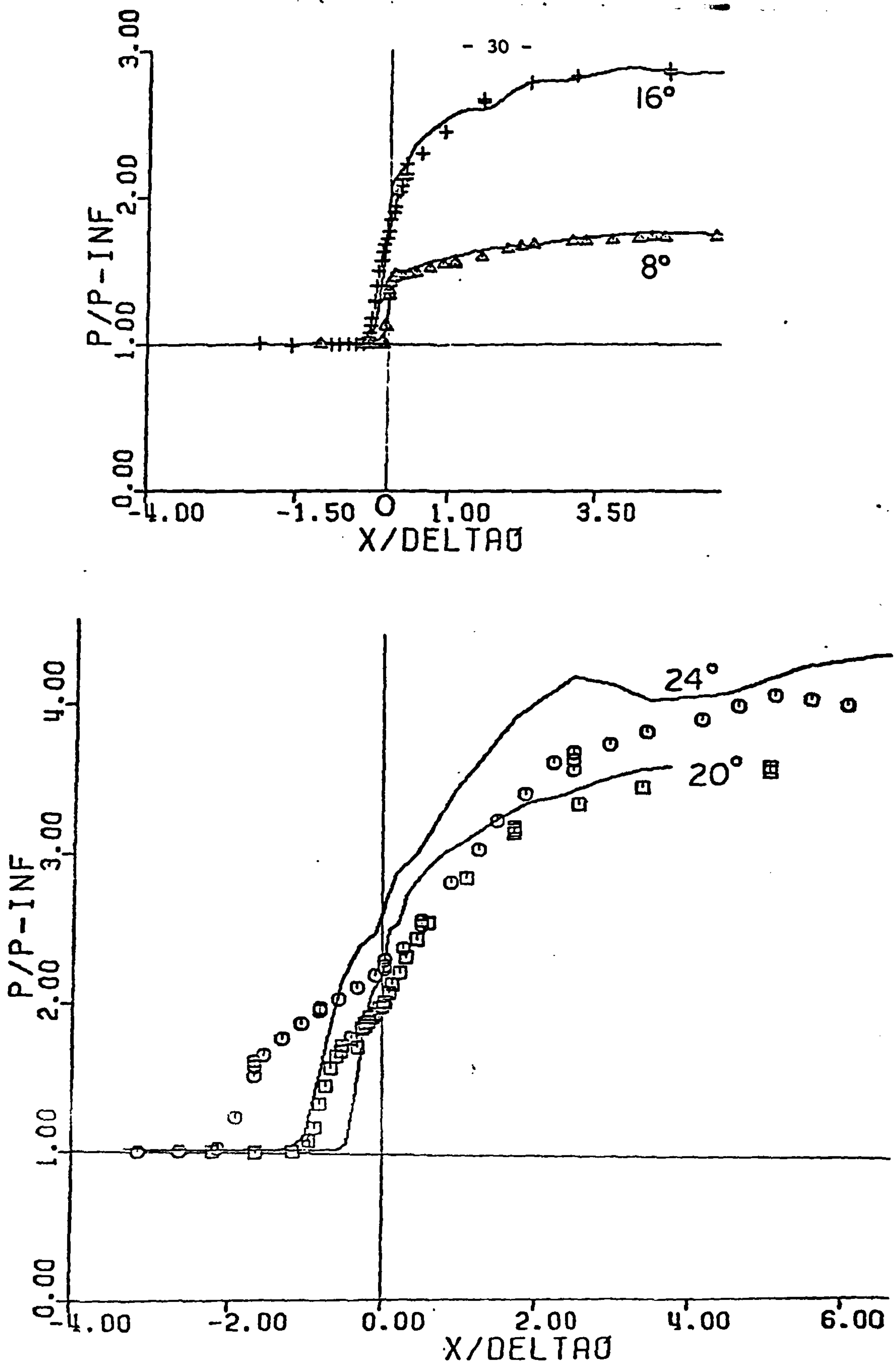


Figure 5a-b - Comparison of Experimental Surface Static Pressures on the Four Compression Corner Models with Computations Using a One-equation Turbulence Model.

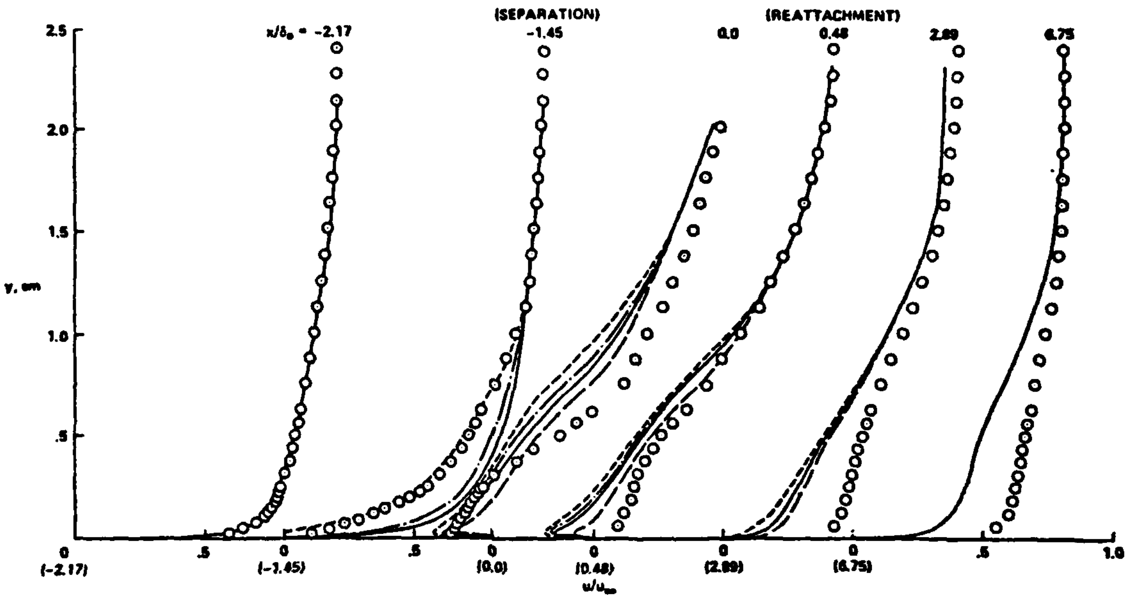


Figure 6 - Comparison of Computations and Velocity Profile Measurements at Selected x-stations; $\alpha = 24^{o}$, $Re_{\delta_o} = 1.33 \times 10^6$.

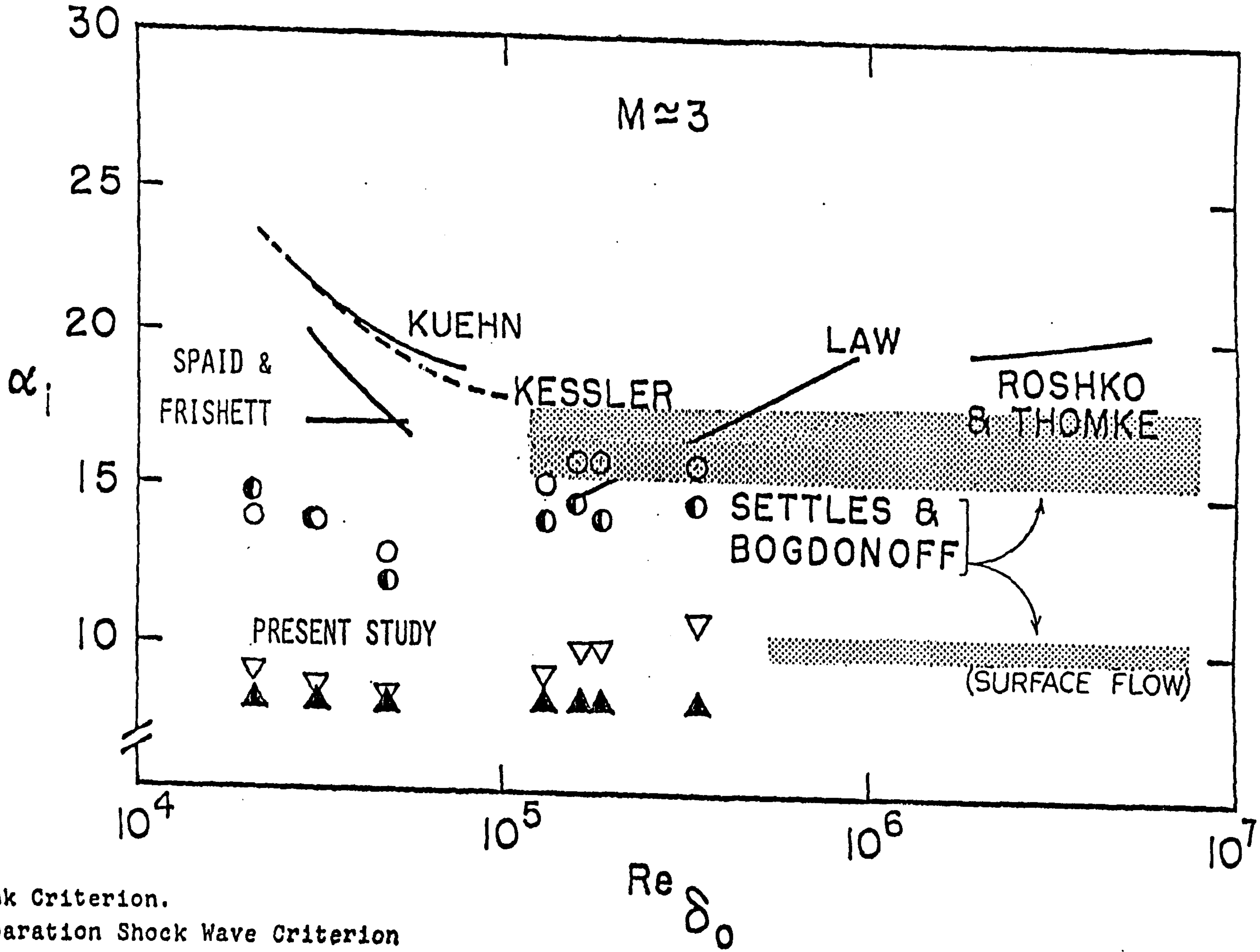
o EXPERIMENT

—— BASELINE MODEL

—— RELAXATION MODEL

— — PRESSURE GRADIENT MODEL

— . — RELAXATION PLUS PRESSURE GRADIENT MODEL



○ Kink Criterion.
◐ Separation Shock Wave Criterion
▲ Surface Flow Patterns, $\Delta Sep/\delta_0 = 0$
▽ Surface Flow Patterns, $\Delta Sep/\delta_0 \approx 0.1$

Figure 7 - Conventional Plot of Incipient Separation Corner Angle vs. Reynolds Number at $M \approx 3$ Showing Present Results and Those of Other Investigators.

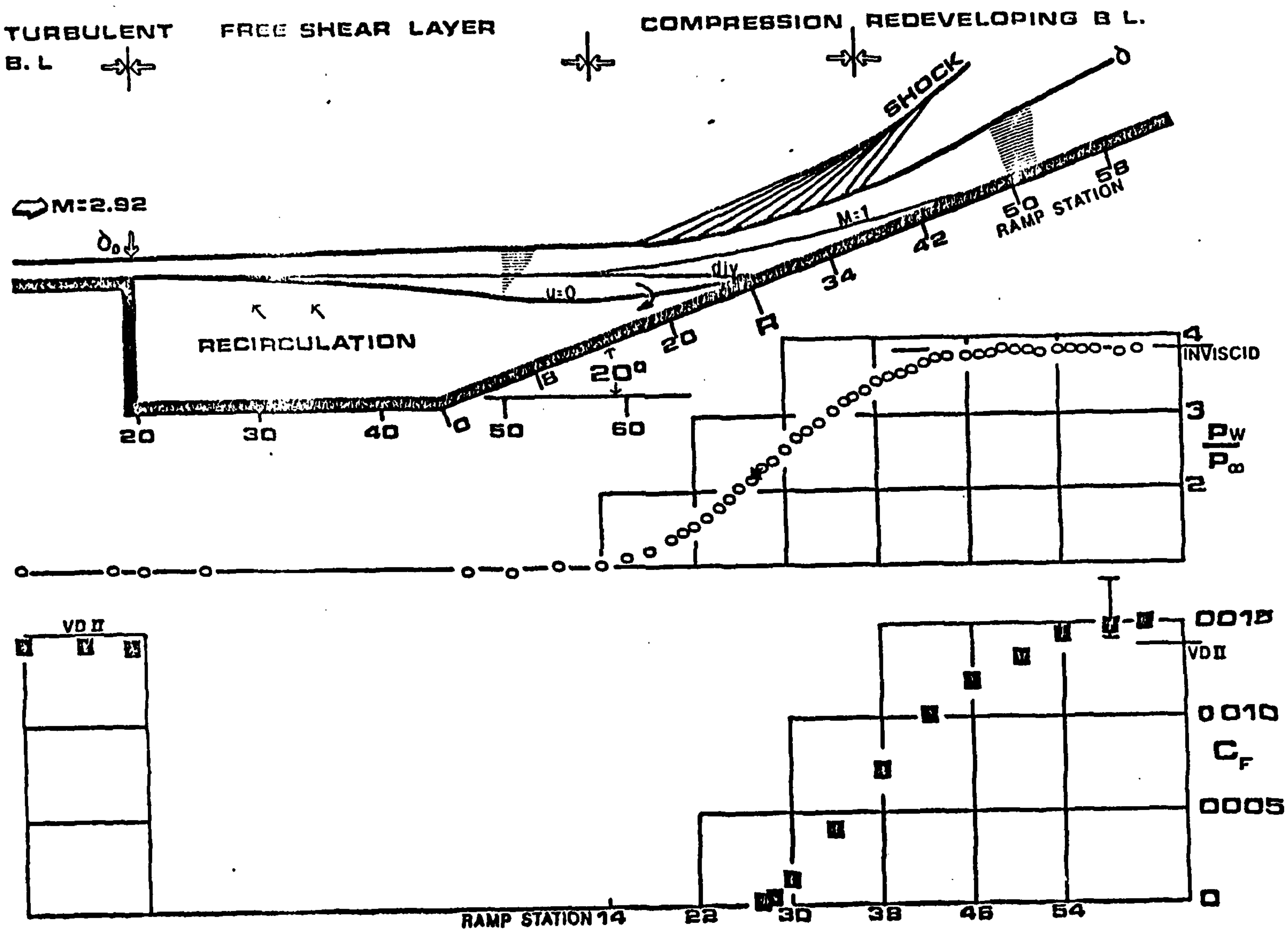


Figure 8 - Experimental Model of the Reattachment Interaction With Wall Pressure and Skin Friction Distributions.

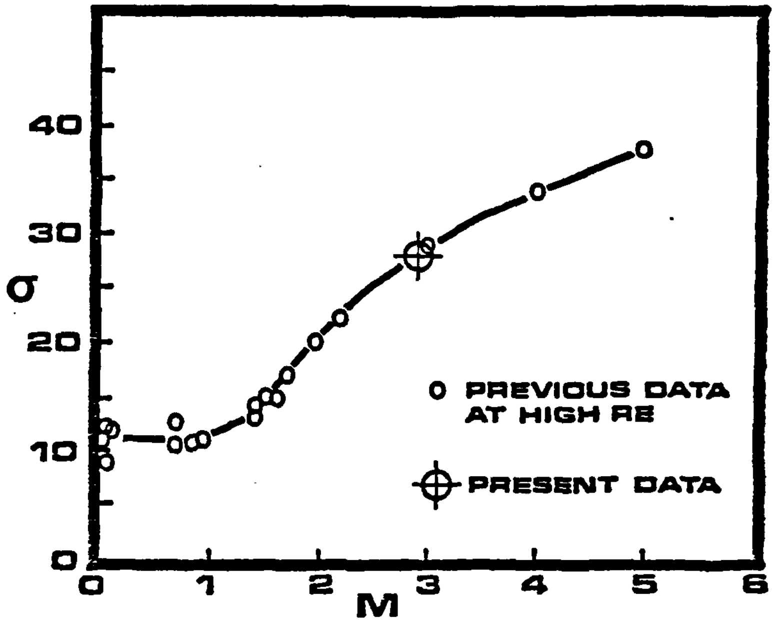


Figure 9 - Variation of Shear Layer Spreading Parameter, Sigma, vs. Mach Number.

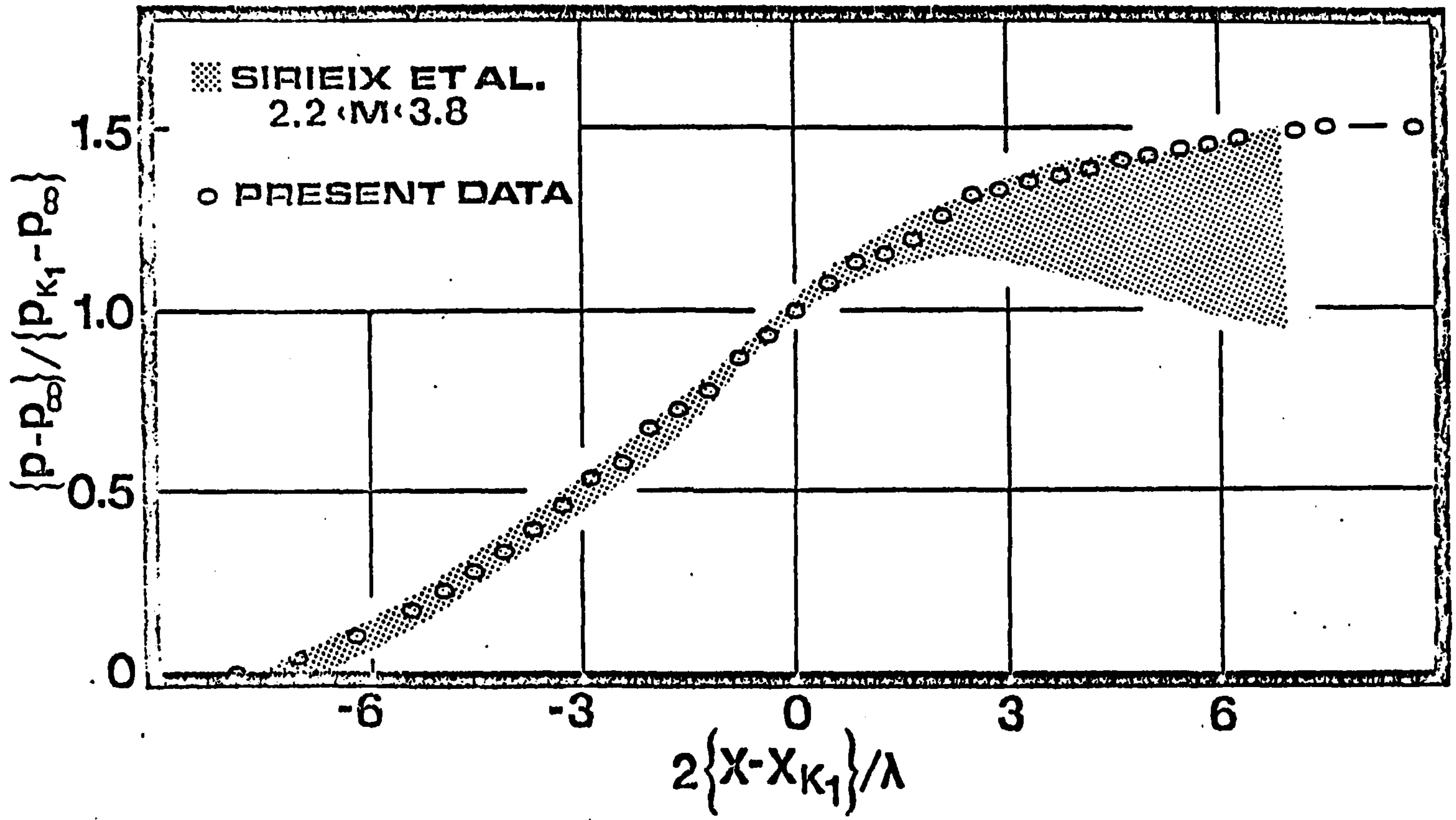


Figure 10 - "Free Interaction" Correlation of Reattachment Wall Pressure Distributions for Various Mach Numbers and Geometries, Including Present Data.

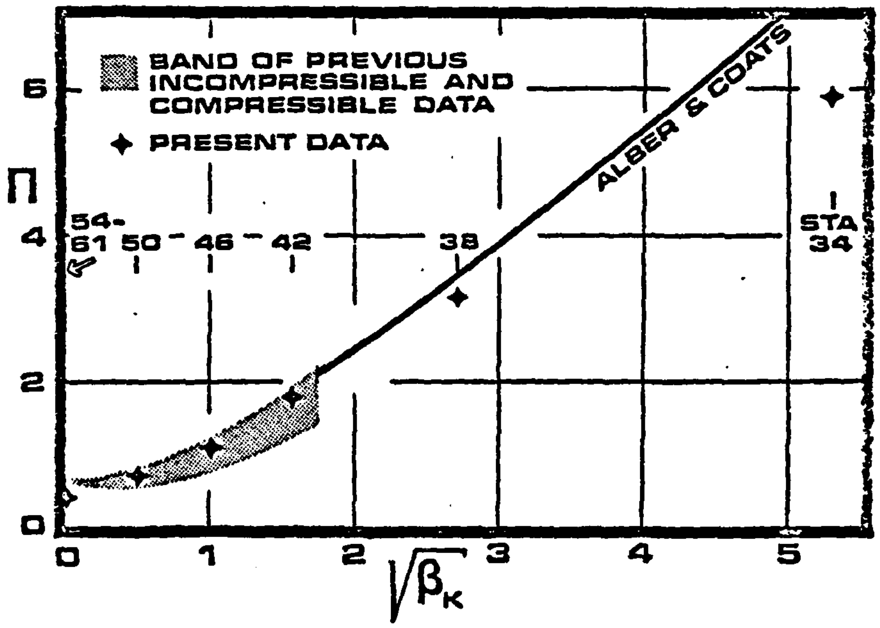


Figure 11 - Wake-Strength Parameter vs. Equilibrium Pressure Gradient Parameter in Redeveloping Boundary Layer, Compared with Previous Data.

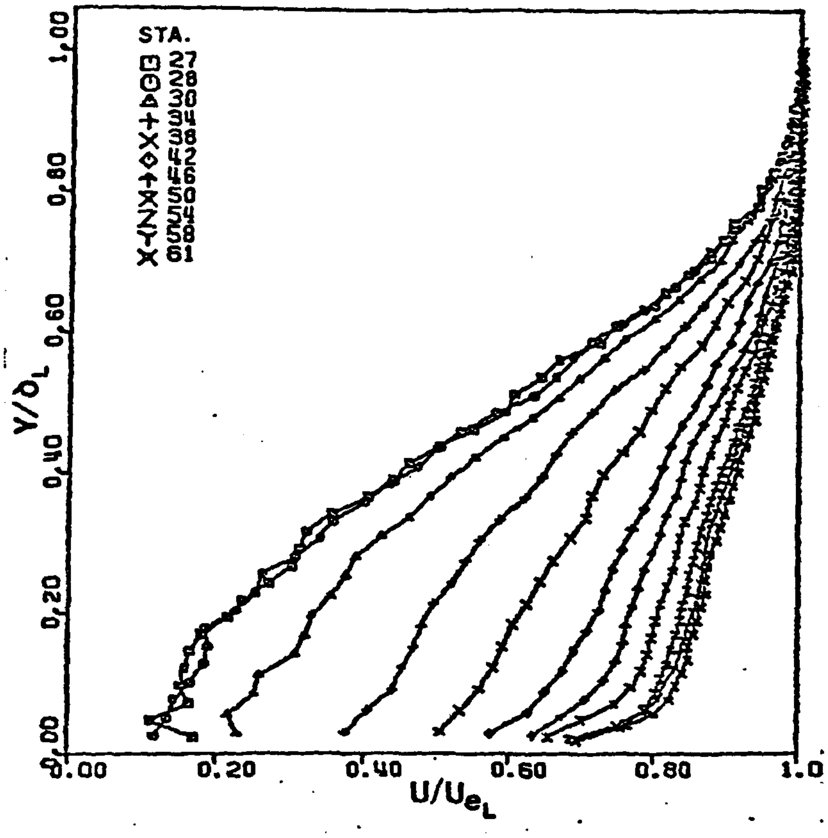


Figure 12 - The Redevelopment of Normalized Mean Velocity Profiles in the Boundary Layer Downstream of Reattachment.

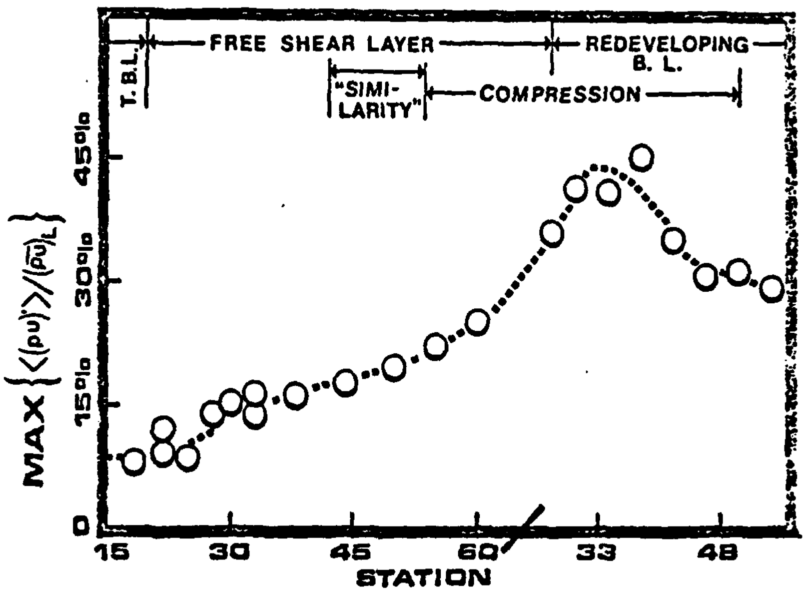


Figure 13 - Plot of Local Maximum RMS Mass Flow Fluctuations vs. Distance Along the Entire Length of the Interaction.

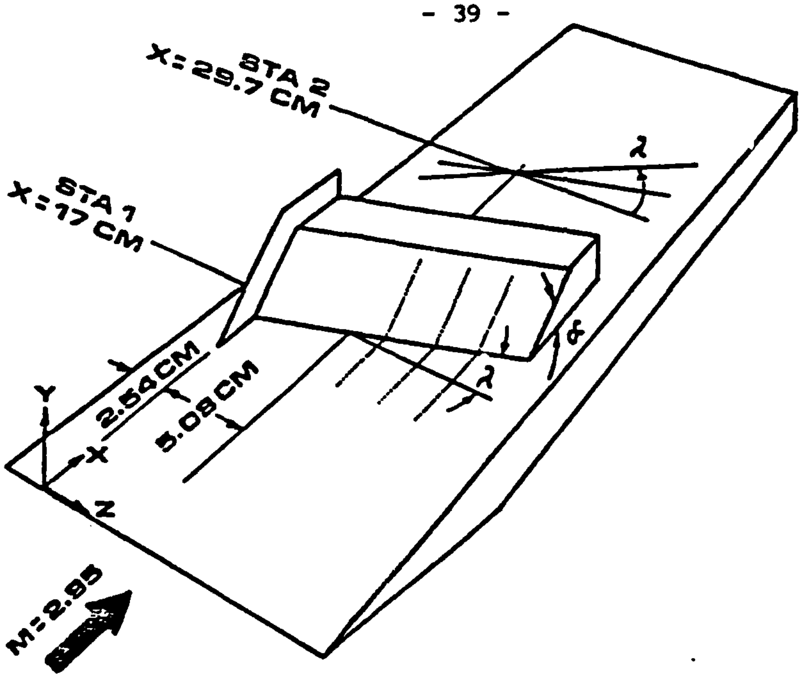


Figure 14 - Sketch of Experimental Configuration

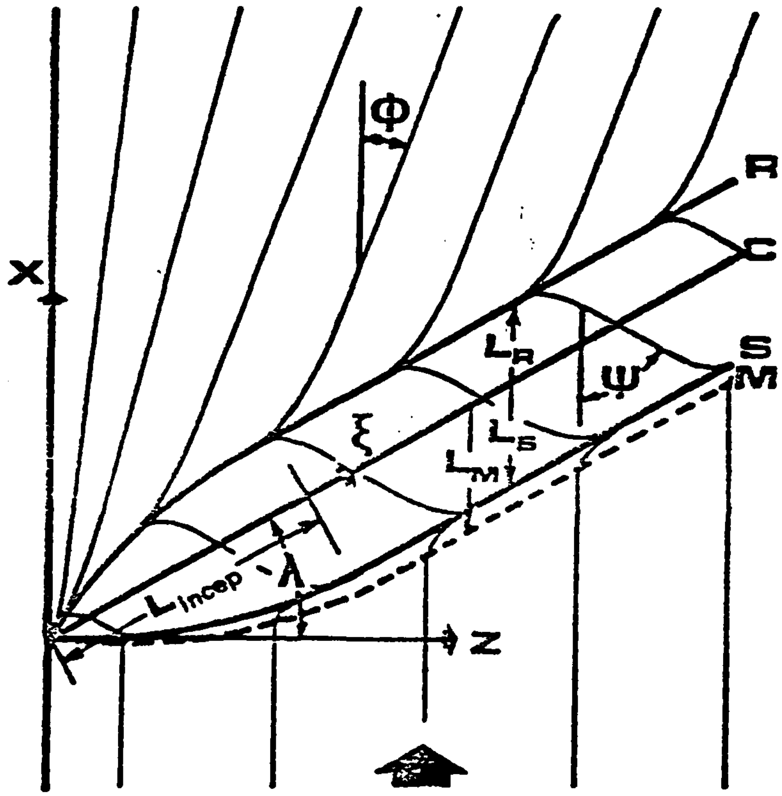


Figure 15 - General Sketch of Surface Flow Pattern on Swept Compression Corner with Definition of Parameters.

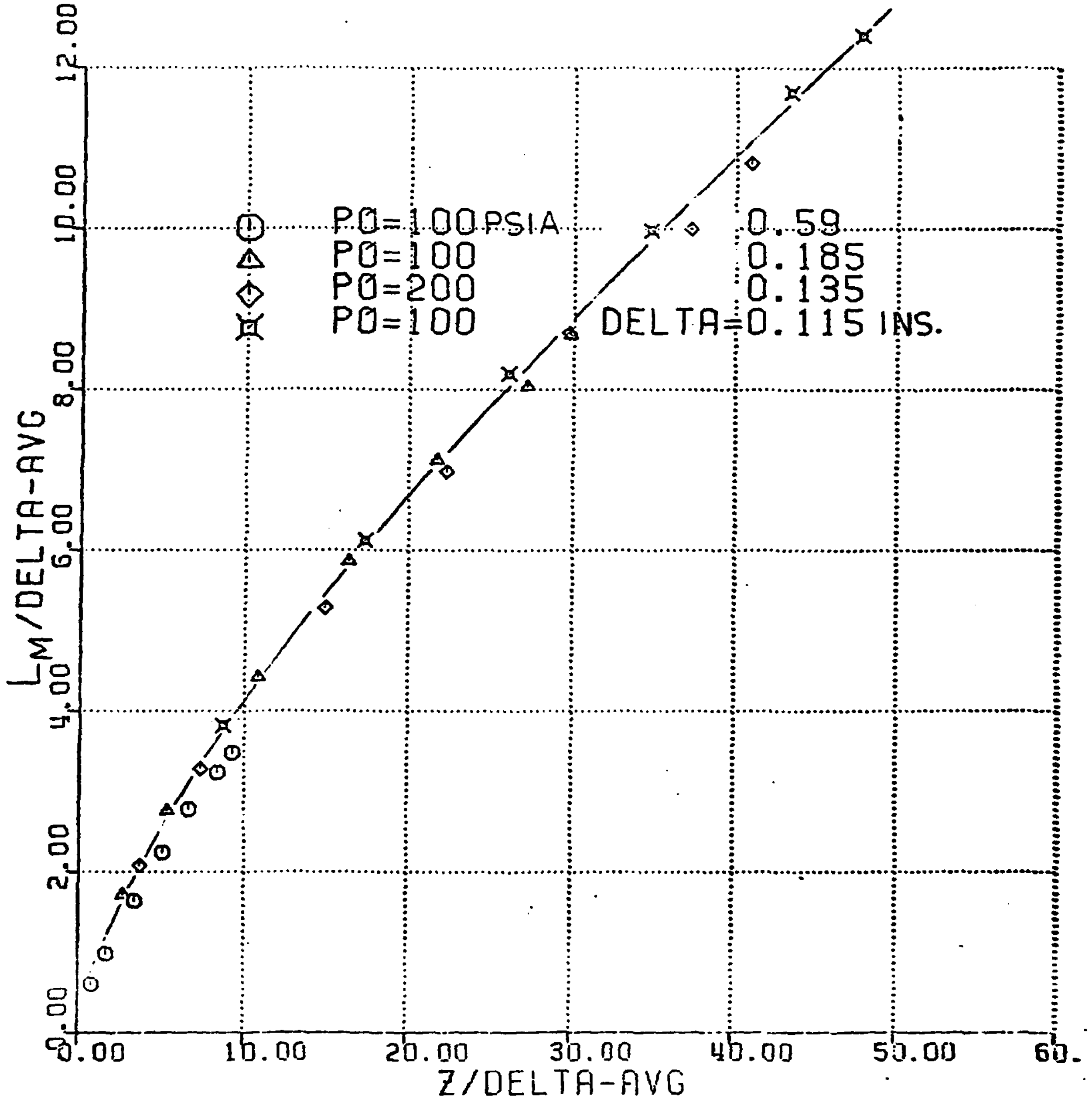


Figure 16 - Plot of Normalized Upstream Influence vs. Normalized Spanwise Dimension, Illustrating Boundary Layer Thickness Scaling.

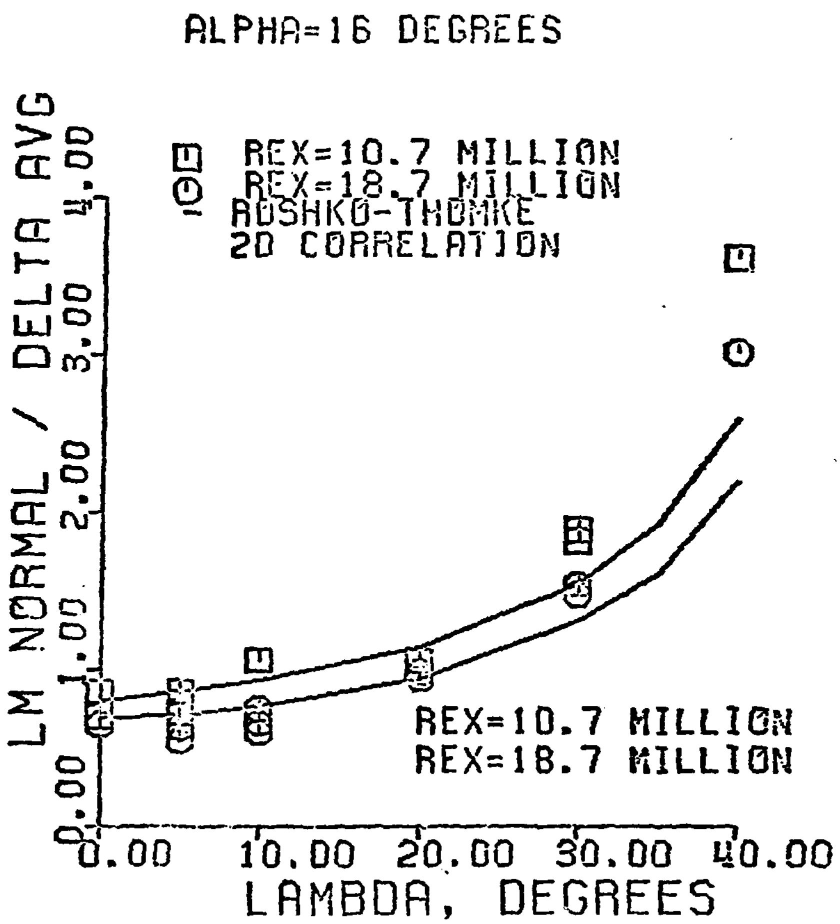


Figure 17 - Comparison of Swept Corner Upstream Influence Parameter and 2D Correlation Resolved Normal to the Corner Line.

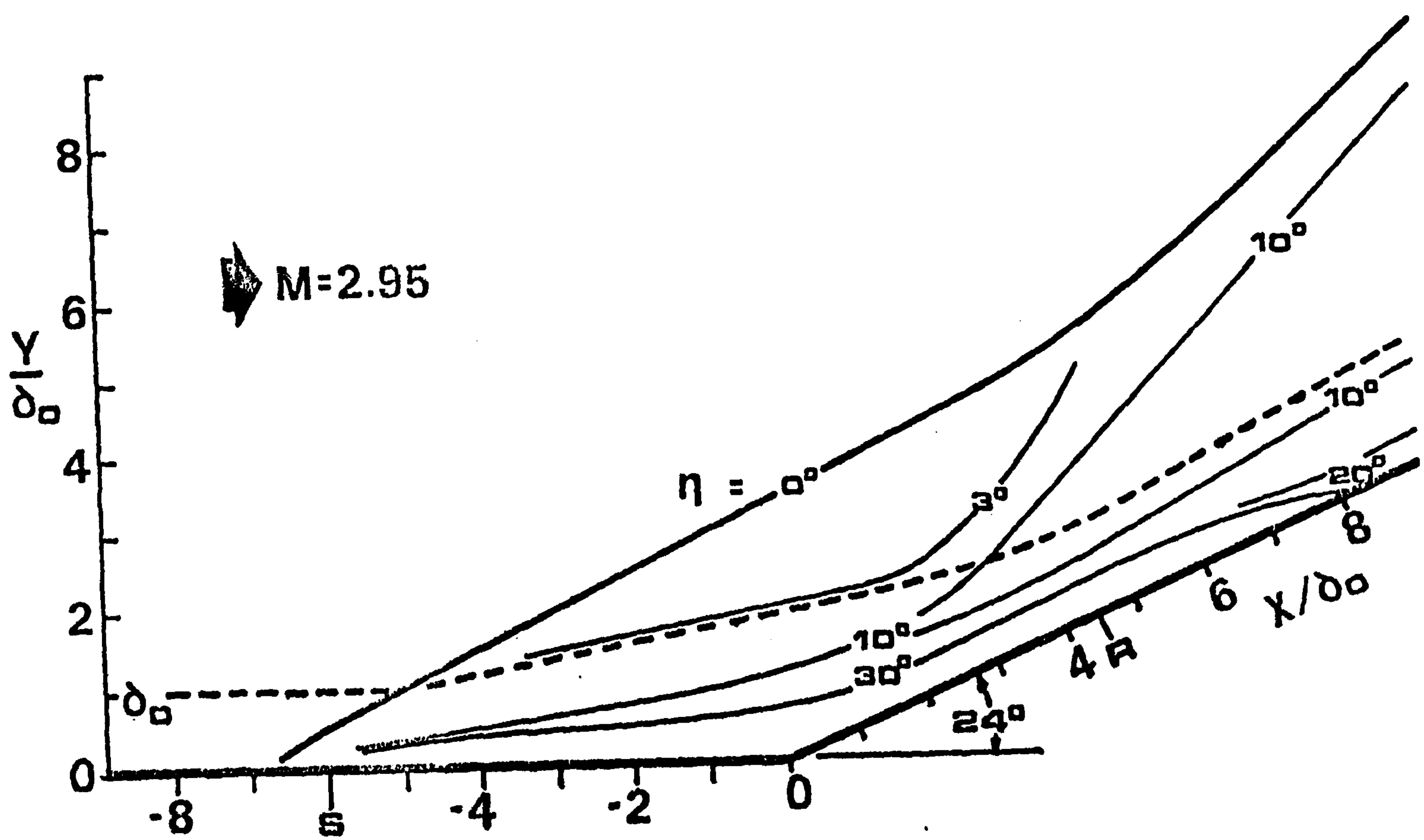


Figure 18 - Yaw Angle Contours of the Flowfield in an xy-plane, $\alpha = 24^{\circ}$, $\lambda = 40^{\circ}$.

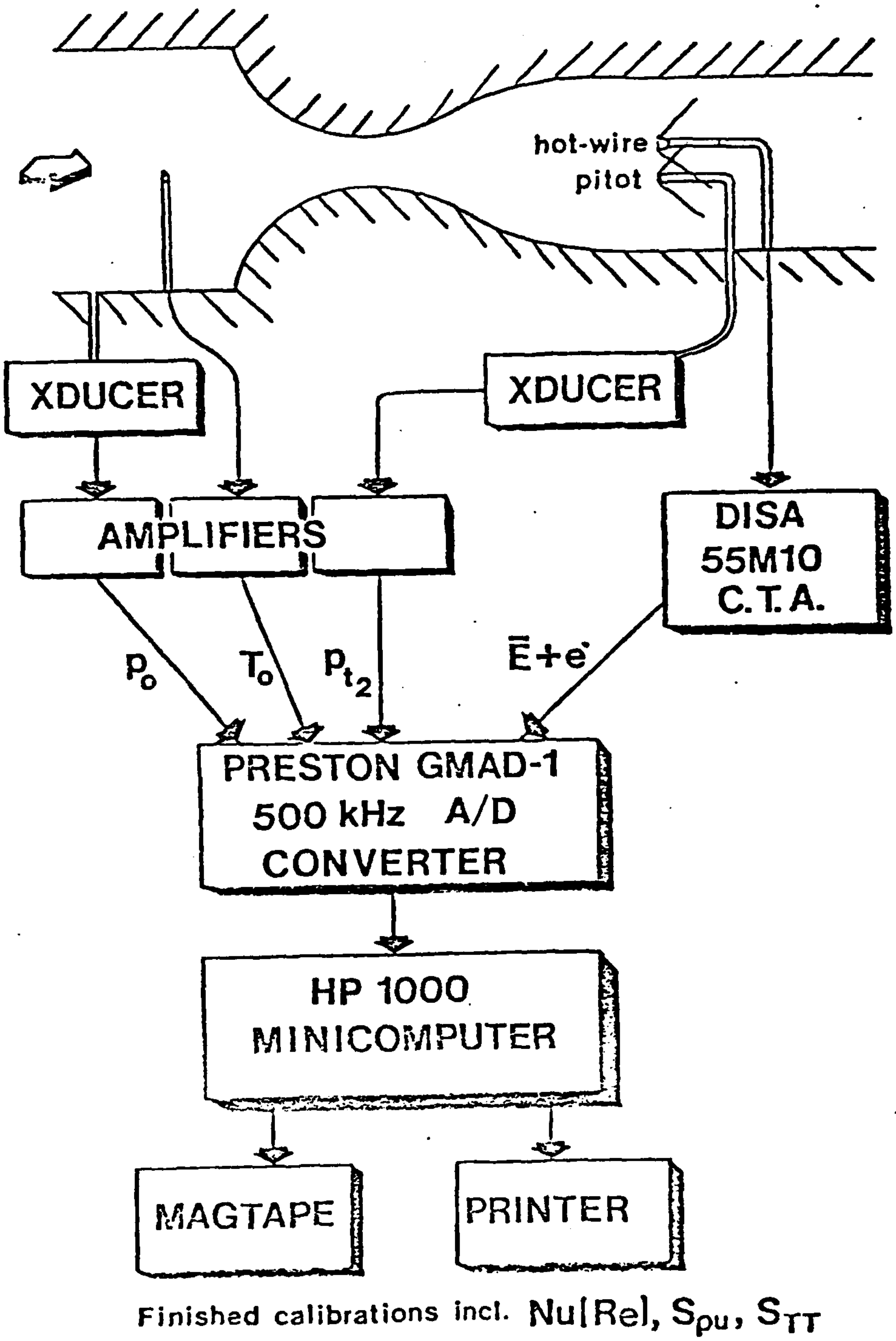


Figure 19 - Block Diagram of Hot-wire Calibration Setup

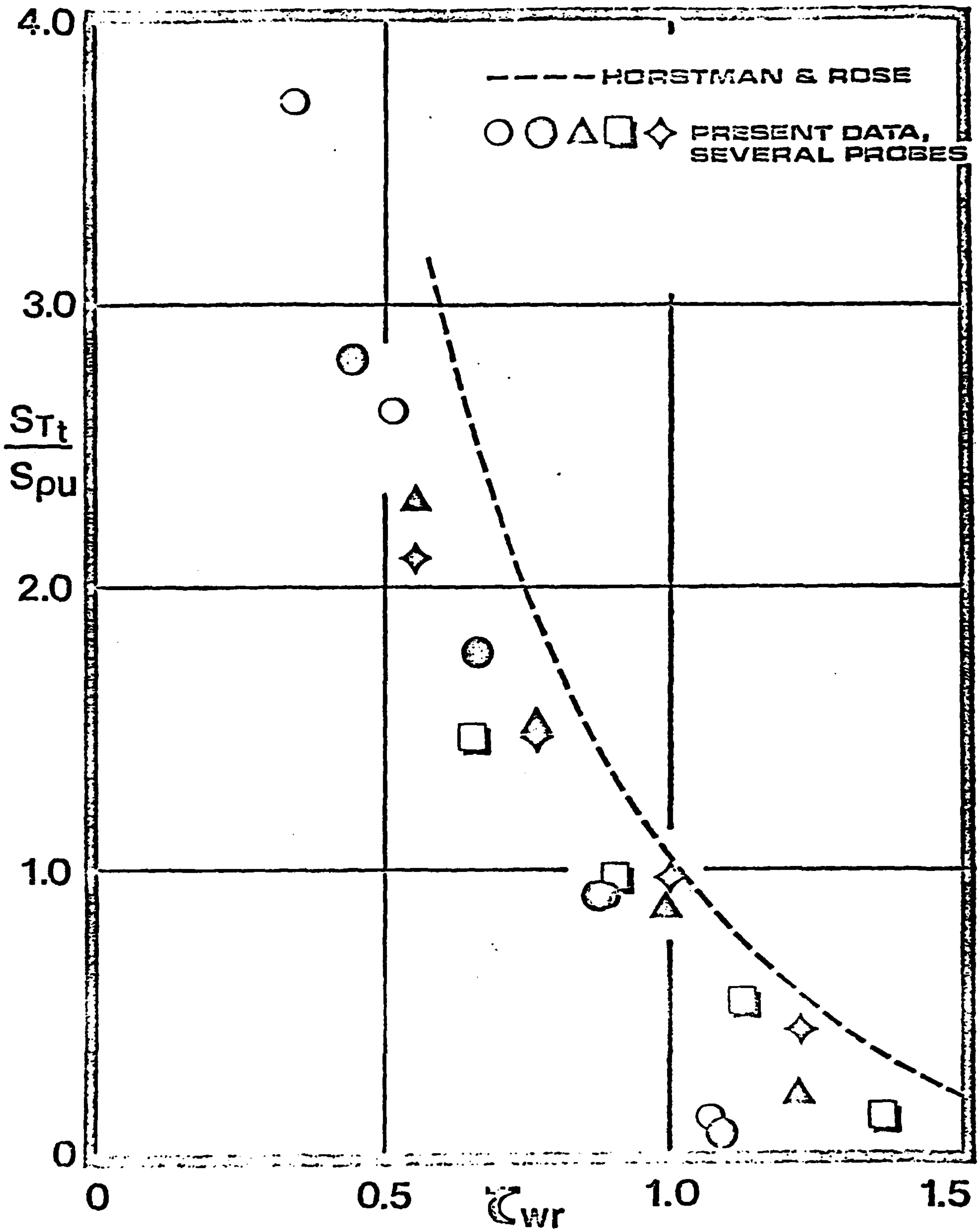


Figure 20 - Plot of Hot-Wire Sensitivity Coefficient Ratio vs. Overheat Ratio.

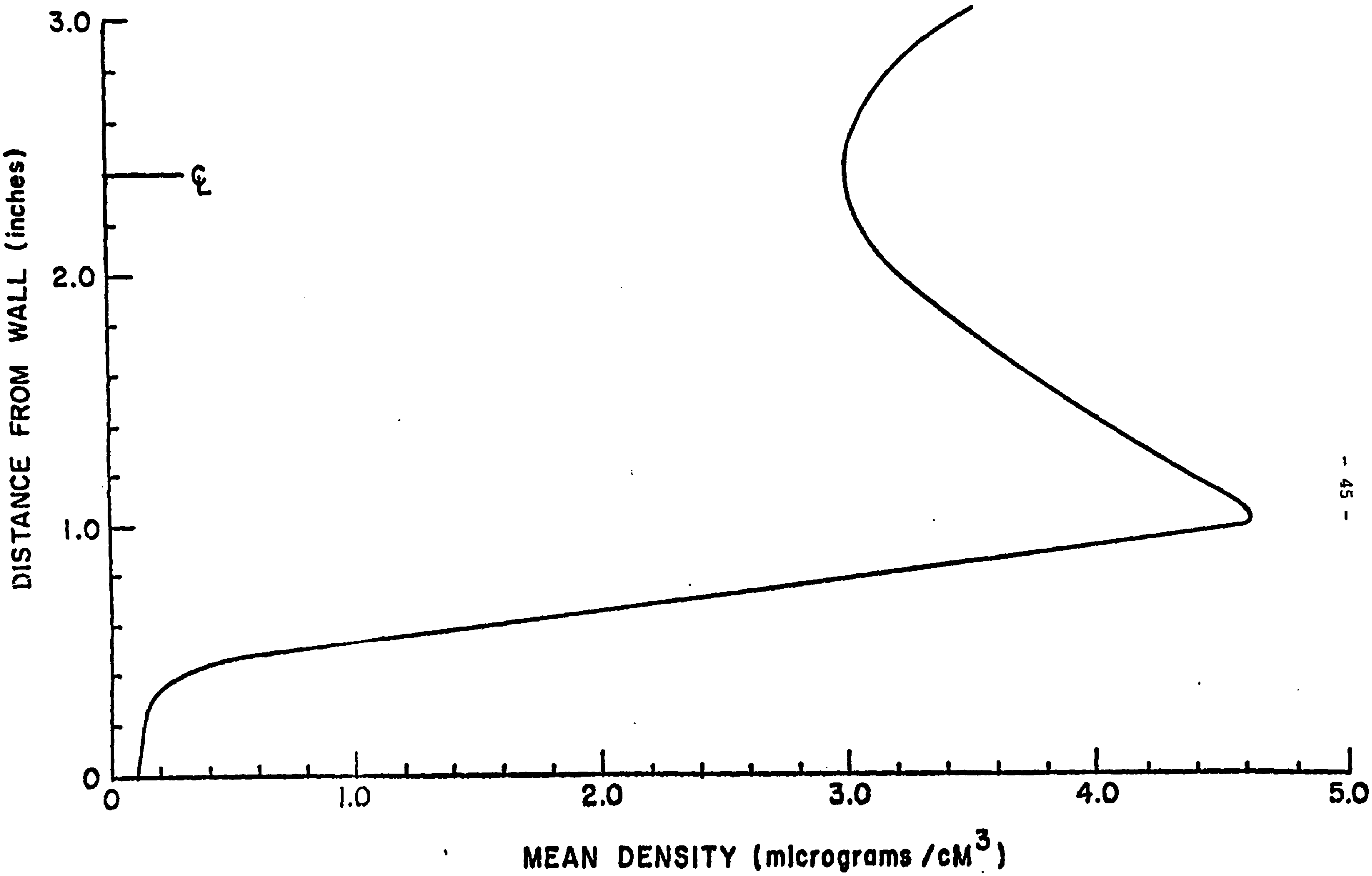


Figure 21 - Mean Density Profile of the Hypersonic Turbulent Boundary Layer.

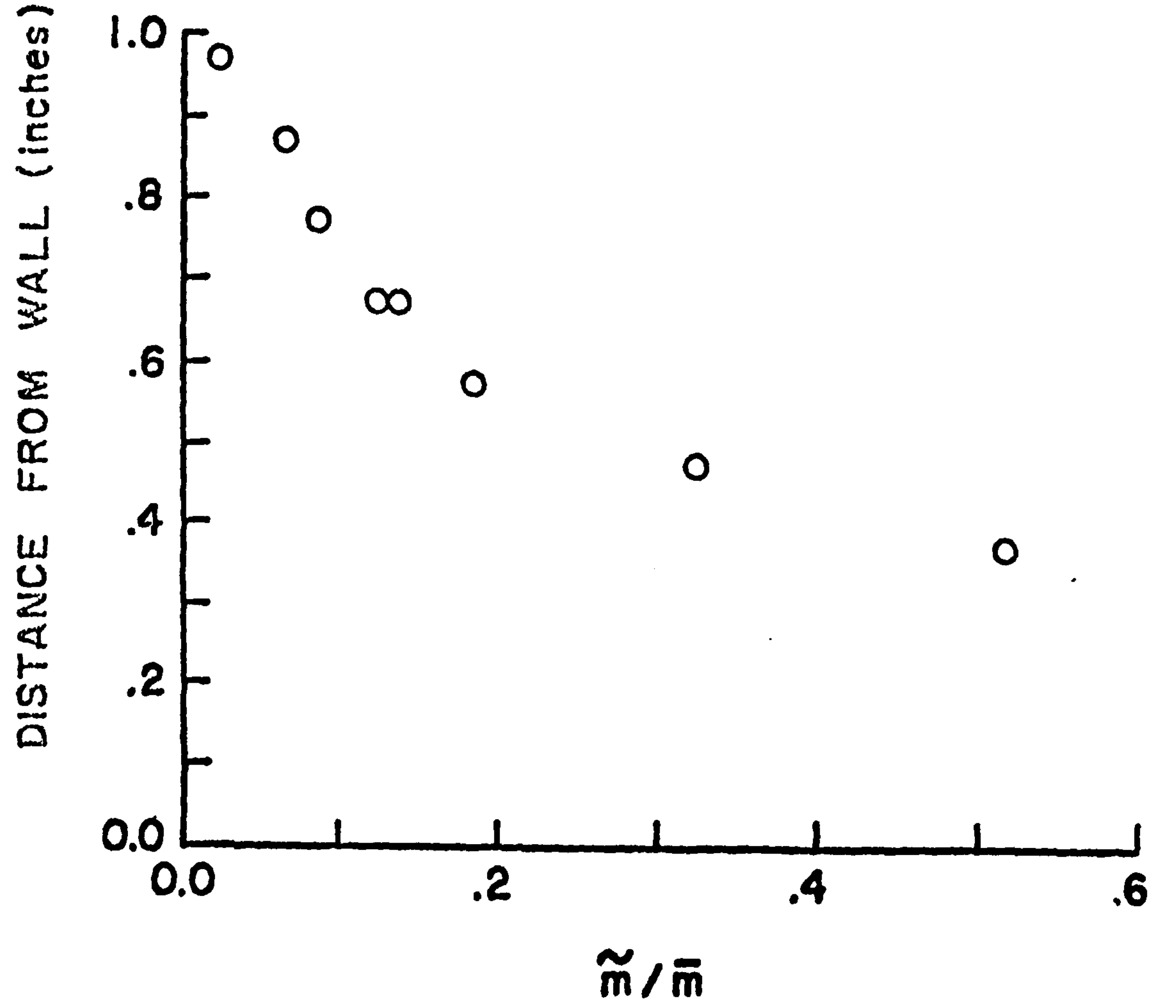


Figure 22 - RMS Mass Flux Readings in the Hypersonic Turbulent Boundary as a Fraction of the Local Mean.

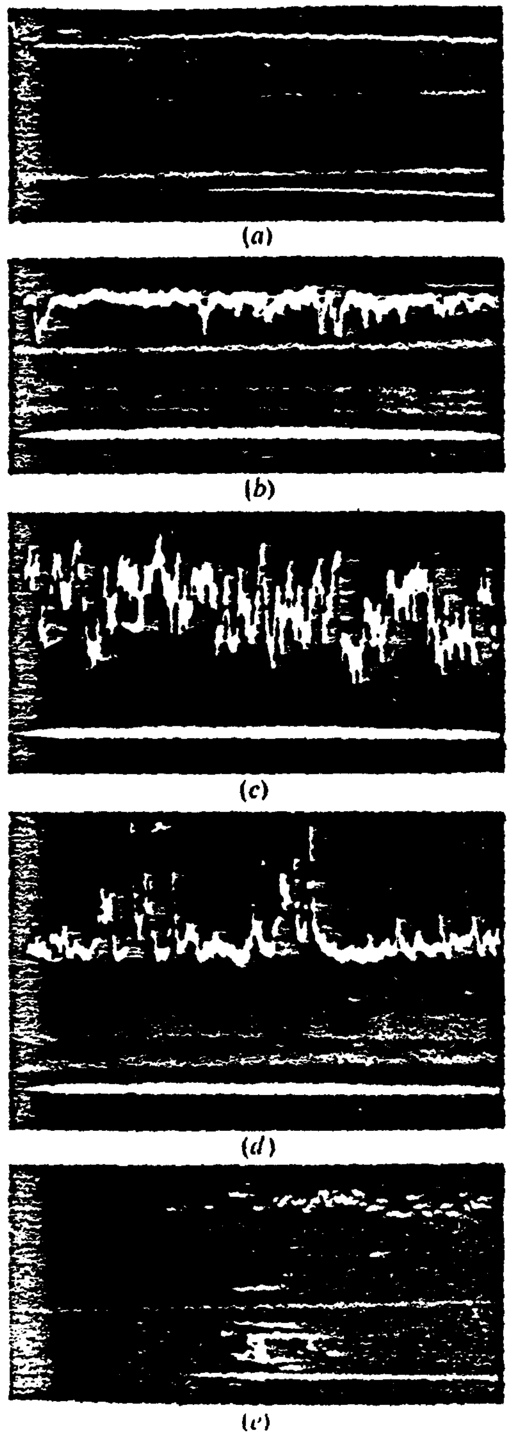


Figure 23 - Representative Intensity Variations with Time for the 5016Å Line at Various Points in the Boundary Layer. 5 ms div^{-1}. (a) Free Stream (y = 3·81 cm); (b) intermittent (2·54); (c) turbulent (1·78); (d) intermittent (0·76); (e) viscous sublayer (0·25).