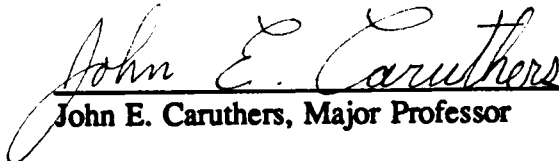
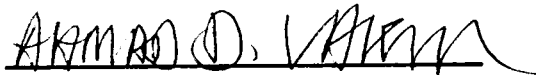


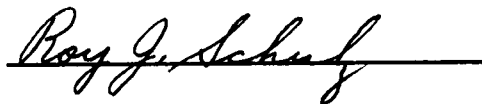
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**An Experimental Study of Jet/Diffuser
Aeroacoustics**

**A Thesis
Presented for the
Master of Science
Degree
The University of Tennessee, Knoxville**

**Paul Steven Cooke
August 1992**

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ABSTRACT

Jet engines are tested in facilities (engine test cells) that simulate high altitude conditions. If the natural resonant modes of the facility geometry couple with the jet's resonant modes, there is the potential for extremely high amplitude sound levels. This investigation examines the sensitivity of the coupling mechanism to alterations in the jet's mean flow field and shear layer. Subscale models of a jet engine test cell and three supersonic nozzles (two converging/diverging rectangular nozzles and one convergent conical nozzle) were tested. The model designs were based on a full-scale facility that had experienced very high amplitude sound levels. Tests were conducted for many different operating nozzle pressure ratios and area ratios (for the two rectangular nozzles). The acoustic response to mean flow field alterations was examined in two ways. First, a sharper angled convergent section in one of the rectangular nozzles introduced more non-uniformities and, hence, instabilities into the jet's mean flow field. Second, a series of tests were conducted with the honeycombs (which are located immediately upstream of the nozzle and are used to produce a straighter and more uniform flow) removed. The acoustic response to alterations in the jet's shear layer was examined by running many tests with various boundary layer "trips", designed to thicken the shear layer, constructed along the nozzle surfaces.

The rectangular nozzle with the sharper angled convergent section produced higher amplitude sound levels (by 10-15dB) compared with the other rectangular nozzle. However, the amplitudes were still about 20dB lower than those observed at the full-scale

facility. Removing the honeycombs generally resulted in higher amplitudes, particularly at an operating pressure ratio of 2.35, where a relatively high amplitude of 150dB at 1250Hz was recorded. These results indicate that a less uniform and, therefore, less stable mean flow field does produce higher amplitude acoustic tones. However, the amplitudes were still significantly lower than those recorded at the full-scale facility.

The boundary layer manipulations generally resulted in lower acoustic amplitudes. Hence, the higher amplitude tones recorded at the full-scale facility could not be attributed to a thicker shear layer.

The findings above indicate a shortcoming in the subscale facility's ability to replicate the acoustic characteristics of the full-scale facility. Since the subscale facility used a cold air jet while the turbines tested in the full-scale facility produce very hot jets, it is suspected that the jet's temperature (particularly, the temperature gradient across the jet's shear layer) is important to the coupling mechanism which produces the high amplitude tones.

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INTRODUCTION

Jet engines are tested in facilities which typically employ a diffuser to aid in the establishment of altitude pressure. The diffuser consists of a conically shaped inlet region which opens into a cylindrical duct. The diffuser is encompassed by a larger cylindrical duct which constitutes the outer portion of the test facility. For certain facility geometries there is the potential for coupling between the jet and the ordinary acoustic modes of the diffuser [1]. If coupling occurs, the resulting high amplitude sound pressures can undermine the structural integrity of the test facility. A subscale model of a testing facility which has experienced very high amplitude fluctuating pressures has been constructed at the University of Tennessee Space Institute. An experimental investigation, using the model, was initiated to determine the physics important to the generation of such high amplitude sound levels. The subscale testing was conducted at the Space Institute's gas dynamics lab under the direction of Professors John E. Caruthers and Ahmad D. Vakili.

The first phase of the investigation, reported in Caruthers et al. [2], confirmed that the high amplitude tones observed in the full scale facility occurred at frequencies corresponding to the natural resonant modes of the facility geometry. The resonant modes were identified experimentally (with the subscale model) and numerically. However, the amplitudes observed in the subscale model were not nearly as high (by 30dB) as those observed in the full scale facility. A summary of the findings from the first phase of the investigation is presented in section 4.1.

Since the subscale facility had not fully reproduced the acoustic characteristics of the full scale facility, it was suspected that small scale effects or minor differences in the nozzle convergent section and throat might be responsible. Such scaling and nozzle contour influences would be manifested in the jet's mean flow field and boundary layer. The purpose of the investigation reported herein was to determine how alterations in the mean flow field and boundary layer of the subscale jet would affect the jet's acoustic output. Such a determination could reveal flow and/or geometric parameters critical to the coupling mechanism which produced the high amplitude pressure fluctuations observed in the full scale facility. Details of the investigation are reported in chapters 4, 5 and 6. Chapters 1, 2 and 3 provide the background information necessary to understanding jet acoustics and aspects of the acoustic interaction between a jet and a diffuser.

CHAPTER 1

BASIC FLUID MECHANICS OF JETS

1.1 Overview

Since jet noise is the product of fluid dynamics, an understanding of those dynamics is essential to a study of jet noise. The purpose of this chapter is to give a physical description of the basic flow features of a jet, particularly those features pertinent to jet noise. Since both the flow structure and the sound field are fundamentally different for subsonic jets and supersonic jets, the two cases are considered separately.

1.2 Fluid Mechanics of Subsonic Jets

A schematic of the flow structure for a subsonic jet is shown in figure 1-1¹. The proportions are based on experimental data taken from a high-Reynolds-number air jet issuing from a convergent nozzle, as presented by Goldstein [3].

There is a region of laminar flow called the potential core which makes up a conically shaped region with its base at the nozzle exit. As the jet exits the nozzle, a mixing layer develops between the moving fluid and its surroundings. Generally, the flow in the mixing layer becomes turbulent no farther than one-half of a jet diameter downstream of the nozzle exit [3,4]. This turbulent mixing layer spreads linearly outward

¹ All figures are located in Appendix A.

and inward, gradually encroaching on the inner laminar flow, until it fills the entire jet at about four to five diameters downstream. This region is called the mixing region.

After the mixing layer has filled the jet, it continues to grow linearly outward. However, the outward expansion occurs at a greater rate than it did in the mixing region. At about eight diameters downstream the turbulent flow is fully developed, i.e., there are no remaining effects on the flow from the potential core. Between the mixing region and the fully developed region lies a transition region.

The turbulent eddies in the mixing layer tend to be elongated in the flow direction; The dimension of an eddy in the axial direction is about twice its dimension in the radial direction [3]. Furthermore, the eddies grow in size as they progress downstream.

Figure 1-2, taken from Goldstein [3], shows the mean jet velocity and the mean square turbulence velocity as functions of radial position. The peak turbulence intensity occurs along the centerline of the mixing layer and remains fairly constant the full length of the mixing region and part of the transition region. Further downstream in the transition region the turbulence peak moves to the center of the jet and decays with axial distance [4]. Thus, most of the turbulent energy is confined to a narrow region at the center of the mixing layer surrounding the potential core [3].

1.3 Fluid Mechanics of Supersonic Jets

The most dramatic difference between a subsonic jet and a supersonic jet is the presence of shock waves in the latter. These shock waves, found in imperfectly expanded supersonic jets, profoundly affect the sound field produced by a jet. There is, however,

another difference that, while not as visible as shock waves, can have just as much effect on the sound field. This difference is not manifest until the plume Mach number is high enough to produce a supersonic mixing layer. As the turbulent structures in the mixing layer move downstream at supersonic speeds, they emit sound in the form of "ballistic" shock waves similar to those produced by a missile traveling at supersonic speed. This phenomena is termed eddy Mach wave radiation. In this section the flow structure of the supersonic jet is discussed, leaving the associated acoustics to a fuller development in chapter three.

1.3.1 Perfectly Expanded Supersonic Jets

Figure 1-3, [3], shows the basic characteristics of a perfectly expanded supersonic jet. For a jet to be perfectly expanded, assuming the nozzle is contoured properly, the nozzle exit pressure must be equal to the ambient pressure of the medium into which the jet is exhausting. Under these conditions, there are no shock waves or expansion waves generated at the jet boundary to relieve pressure differences. As shown in the figure, the flow structure is similar to that of a subsonic jet except that the potential core and the mixing layer immediately surrounding the core are supersonic. Also, unlike subsonic jets, experimental evidence [3] has shown that the length of the potential core increases with increasing Mach number for supersonic jets.

The supersonic and subsonic mixing regions are composed of turbulent eddies which emanate from the nozzle lip and propagate downstream. An eddy may start out as a vortex ring at the nozzle exit, becoming weaker and less cohesive as it proceeds

downstream. The eddies are part of larger scale turbulent flow structures originating at the nozzle lip and extending downstream through the jet mixing layer [5,6]. These large scale flow structures are coherent and spacially quasi-periodic throughout the length of the annular mixing layer [7]. For low and moderate Reynolds number supersonic jets, the large scale disturbances are in the form of instability waves. Initially, close to the nozzle exit, these instability waves grow in amplitude as a result of the steep velocity gradient in the shear layer. As the waves propagate downstream their growth rate is reduced corresponding to a reduction in the velocity gradient associated with the widening mixing layer. Eventually, their growth rate becomes zero, and far downstream, where the mixing layer has filled the entire jet, the waves die out. Liu [8] has explored how such instability waves could exist in turbulent flow and the mechanism for their origin.

The intensity of a turbulent eddy, determined by its root mean square velocity divided by its mean velocity, falls off relative to its convection speed with increasing plume Mach number. At very high Mach numbers, the eddy appears virtually frozen (i.e., like a static blob of fluid) relative to the mean flow. When a turbulent eddy is convected downstream at supersonic speed, it produces a Mach wave in the same way that a solid object, such as a wedge or cone, produces a shock wave when moving at supersonic speed. Eddy Mach wave radiation, discussed more completely in chapter 3, is a significant source of noise in high speed jets.

1.3.2 Imperfectly Expanded Supersonic Jets

When a supersonic jet is expanded such that the nozzle exit pressure does not match the ambient pressure of the environment into which the jet is exhausting, expansion and compression waves will form near the jet boundary downstream of the nozzle exit, and propagate throughout the jet until dissipated by viscous effects. The jet is said to be imperfectly expanded. When the nozzle exit pressure is lower than the ambient pressure, the jet is called overexpanded. Oblique shock waves, emanating from the nozzle exit, are necessary to raise the jet static pressure to the ambient pressure. When the nozzle exit pressure is higher than the ambient pressure, the jet is called underexpanded. Hence, expansion waves, emanating from the nozzle exit, are required to lower the jet static pressure to the ambient pressure.

Figure 1-4, taken from Liepmann and Roshko [9], shows the pressure distributions and the associated wave configurations for several exit pressure ratios. The local pressure ratio, equal to the local static pressure divided by the stagnation pressure, is plotted as a function of nozzle axial position.

The waves, originating at the nozzle exit, become a self-preserving system of expansion and compression waves extending throughout the supersonic length of the jet. The system is self-preserving because there are two boundary conditions that cannot be satisfied simultaneously. One boundary condition involves the pressure equilibrium requirement already discussed: Any part of the jet which is in free contact with the ambient environment around the jet must have a static pressure equal to the ambient pressure. This is the cause of the initial set of waves emanating from the nozzle exit.

The other boundary condition involves flow direction. Jet symmetry requires a central streamline to exist. Any fluid in free contact with the central streamline must have a velocity tangent to that streamline. The initial set of waves, in addition to satisfying the pressure equilibrium requirement, have the side effect of altering the flow direction. Hence, the flow is no longer parallel to the central streamline. The only way to satisfy the flow direction requirement along the central streamline is to reflect the initial set of waves off the central streamline so that the flow is turned back in the direction parallel to the jet axis. Waves incident upon the central streamline reflect in kind (i.e., compression waves reflect as compression waves and expansion waves reflect as expansion waves). Waves striking the free jet boundary are also reflected, though these are reflected in opposite kind (i.e., compression waves reflect as expansion waves and expansion waves reflect as compression waves). Clearly, the cycle must be repeated, since the reflected waves alter the local pressure in addition to correcting the flow direction.

Figure 1-5, taken from Zucker [10], shows an overexpanded jet. For simplicity only the wave system for the inviscid structure of the jet is shown, neglecting boundary layer and mixing effects. An underexpanded jet, figure 1-6, has a wave pattern identical to the overexpanded jet downstream of the initial set of oblique shock waves in the overexpanded jet. In both cases the two boundary conditions mentioned above are satisfied by a system of expansion and compression waves.

The simplified depiction of imperfectly expanded jets, given above, is insufficient for an understanding of the noise produced by these jets. Boundary layer mixing effects

are an important part of the noise production process. The general structure of the imperfectly expanded jet is similar to that of the perfectly expanded jet with the expansion/compression wave system superimposed on the supersonic portion of the jet.

Seiner's review article [11] provides an illustration of the complete shock containing supersonic jet plume, including mixing layer effects. The illustration, reproduced in figure 1-7, is the result of a numerical model developed by Dash and Wolf [12] that explicitly accounts for turbulent mixing influences. For clarity the radial axis is multiplied by a factor of 2.5. Figure 1-7, representing the underexpanded flow from a Mach two nozzle, shows a decrease in wave spacing with increasing distance from the nozzle exit. Though not very clear in this figure, all waves in the supersonic mixing layer are curved since they propagate through regions of nonuniform flow. This contributes to the observable decrease in wave spacing, as well as the degradation of wave strength, with increasing distance downstream of the nozzle exit. The decrease of wave strength in the axial direction is confirmed by experimental evidence which shows the periodic variation in static pressure, associated with the compression/expansion cycle of the underexpanded jet, to decay with distance downstream [11].

The decay process of the expansion/compression wave system is governed by the large scale coherent flow structures in the mixing layer. These structures control the rate of plume spreading and provide a mechanism for the conversion of jet kinetic energy to radiated acoustic energy. M.S. Howe and Ffowcs Williams [13] have advanced a theoretical model describing this conversion of jet kinetic energy to acoustic energy for imperfectly expanded jets. Their idealized model consists of a continuum of

axisymmetric vortex rings located within a thin mixing layer surrounding the potential core. The vortex rings, mimicking the large dominant eddies which are convected downstream, act as a catalyst enabling the energy of the expansion/compression wave system to radiate out as sound. Essentially, the energy of the imperfectly expanded supersonic jet is radiated out by an interaction between the mixing layer and the expansion/compression wave system. While this idealized model of axisymmetric vortex rings provides a useful insight into the energy conversion process, it does not possess the quasi-periodicity and wave-like characteristics of the large scale flow structures and, therefore, does not completely describe the true mechanism for energy conversion. The shock noise mechanism is examined more thoroughly in chapter 3.

CHAPTER 2

BASIC PRINCIPLES OF AERODYNAMIC NOISE GENERATION

2.1 Overview

Jet noise, as a branch of theoretical acoustics, emerged in the early 1950's when Lighthill [14,15] introduced his acoustic analogy. Lighthill's theory, developed to deal with the problem of jet noise, stimulated study in many other aspects of aerodynamic noise. Before considering jet noise specifically, it is helpful to review the fundamental concepts of flow noise.

Any form of noise is generated by forcing mechanisms which introduce very small amplitude disturbances, i.e., sound waves, into the pressure field of the surrounding medium. The study of aerodynamic noise, aeroacoustics, concerns the sound generated by aerodynamic forces or motions originating in a flow rather than by vibrating solid surfaces that generate externally applied forces or motions, such as those that characterize classical acoustics. Hence, the noise generated by a vibrating string, such as a guitar string, or a loud speaker, falls into the category of classical acoustics [3]. Contrastingly, noise generated by turbulent flows or by unsteady aerodynamic forces on solid objects falls under the domain of aeroacoustics. Lighthill's acoustic analogy, arising from the need to understand the noise emitted by the turbulent flow of subsonic jets, has been extended to include the effects of flow interaction with solid boundaries.

Some examples of noise production through flow interaction with solid boundaries include the noise associated with turbine blades, airplane propellers, and helicopter rotors. Nature provides many examples of flow noise. Wind blowing around tree limbs produces a howling noise. A tornado is a huge noise producing vortex. Although this paper is focused on the noise produced by air flow, water flow provides some clear examples of flow generated noise. Ocean waves crashing on a beach and a river cascading over rocks are examples of noise produced by flow interactions with solid boundaries.

The purpose of this chapter is to review the mathematical basis for some fundamental mechanisms of aerodynamic sound production. Lighthill's acoustic analogy, the foundation of aerodynamic noise theory, is presented within the context of classical acoustics. A couple of restrictions on Lighthill's theory should be noted at the outset. First, the theory is simply an explanation of how noise is generated by turbulence. Hence, such an explanation precludes the shock noise phenomena of imperfectly expanded supersonic jets to be discussed in chapter 3. Second, a premise of the theory is that the sound radiated from the flow does not react back on the flow. In fact resonant noise phenomena do occur, such as jet screech (discussed in chapter 3), which are based on just such a back reaction. Nevertheless, Lighthill's theory provides a useful insight into fundamental mechanisms of aerodynamic sound generation.

2.2 The Homogeneous Wave Equation

Before considering aerodynamic noise, particularly Lighthill's acoustic analogy, a few ideas from classical acoustics need to be stated. Many of the ideas and concepts

which have been developed in the study of aerodynamic noise have their roots in classical acoustic theory. With this as a premise, the homogeneous wave equation, fundamental to classical acoustics, is outlined below.

The homogeneous wave equation is derived from a linearization of the equations of mass and momentum conservation. Since the study of acoustics is almost exclusively concerned with sound propagation through fluids such as air and water where the effects of viscosity and thermal conductivity are very small [3], the terms associated with these effects can be neglected from the conservation equations. Then, neglecting external forces, the exact equations of mass and momentum conservation, respectively, can be written in tensor form as

$$\frac{\partial \rho}{\partial t} + \frac{\partial}{\partial x_i}(\rho V_i) = 0 \quad (1)$$

$$\frac{\partial}{\partial t}(\rho V_i) + \frac{\partial}{\partial x_j}(\rho V_i V_j + P_{ij}) = 0 . \quad (2)$$

P_{ij} is the compressive stress tensor, representing the force per unit surface area in the x_i direction with inward normal in the x_j direction [14]. The other element of momentum transport, $\rho V_i V_j$, represents direct convection of the momentum ρV_i by the velocity component V_j . The continuity and momentum equations, respectively, relate the rate of change of mass or momentum in a small volume to the total mass transport or momentum transport out of the volume.

Assuming the fluid undergoes no entropy gradient, the pressure is a function of density alone. It follows that $dp=c^2 d\rho$, where c is the local speed of sound. Then, upon

linearizing equations (1) and (2) about a state of rest in which the fluid has uniform density ρ_0 and sound speed c_0 , the acoustic approximations are [16]

$$\frac{\partial \rho}{\partial t} + \rho_0 \frac{\partial v_i}{\partial x_i} = 0 \quad (3)$$

$$\rho_0 \frac{\partial v_i}{\partial t} + c_0^2 \frac{\partial \rho}{\partial x_i} = 0 . \quad (4)$$

Equations (3) and (4) are combined to form the homogeneous wave equation

$$\frac{\partial^2 \rho}{\partial t^2} - c_0^2 \nabla^2 \rho = 0 , \quad (5)$$

where

$$\nabla^2 = \frac{\partial^2}{\partial x_i \partial x_i} = \frac{\partial^2}{\partial x_i^2} . \quad (6)$$

Equation (5), fundamental to classical noise theory, governs the propagation of sound in a uniform medium without sources of mass or external forces.

2.3 The Inhomogeneous Wave Equation

Now, with a view towards Lighthill's acoustic analogy, attention is given to the case where the right-hand side of equation (5) is non-zero. Such is the case when there is a source distribution within the acoustic medium. Suppose, for instance, that sources of mass are distributed throughout the medium which inject mass at a rate $Q(\bar{x}, t)$ per unit volume. This would add an additional term, $Q(\bar{x}, t)$, to the right-hand side of the

continuity equation, resulting in the inhomogeneous wave equation [16]

$$\frac{\partial^2 \rho}{\partial t^2} - c_0^2 \nabla^2 \rho = \frac{\partial Q}{\partial t} . \quad (7)$$

The density field of equation (7) is the well known monopole field of classical acoustics. Therefore, the introduction of a source of mass, Q , is like adding a monopole source which generates sound waves that propagate throughout the acoustic medium. Combustion noise is an example of a monopole generated sound field.

Next, consider the case where there is a distribution of external forces acting on the fluid with a force per unit volume equal to $\bar{F}$. Such a force distribution would add an additional term, F_i , to the right-hand side of the momentum equation resulting in the inhomogeneous wave equation [16]

$$\frac{\partial^2 \rho}{\partial t^2} - c_0^2 \nabla^2 \rho = -\frac{\partial F_i}{\partial x_i} . \quad (8)$$

The density field of equation (8) is that produced by a dipole source distribution. A model used commonly for a dipole source of sound is a rigid sphere oscillating back and forth along a line. Hence, instead of a mass influx, there is a force necessary to move the sphere back and forth. The strength of the dipole is equal to this force and, like the force, has direction as well as magnitude. A dipole can be viewed as two simple sources a small distance apart, each of equal strength but opposite sign [17]. Then the direction of the dipole strength points from the negative source to the positive source, and the magnitude of the dipole strength is equal to the simple source strength times the distance separating the sources [18].

Dipole source distributions can result from solid objects in an air flow which impose force distributions on the air equal and opposite to the aerodynamic force on the object. For example, a wire in an air flow creates a dipole distribution resulting in the famous aeolian tones. Boundary layer noise is another example of a dipole generated sound field.

2.4 Lighthill's Acoustic Analogy

After becoming accustomed to viewing the inhomogeneous term in the wave equation as an acoustic source term, it is easier to understand Lighthill's [14,15] acoustic analogy. The monopole and dipole sources from classical acoustics have been used to describe the noise generation due to an injection of mass into the flow or due to an applied external force on the flow, as would be the case with a solid body impinging on the flow field. While these noise generating mechanisms fall into the category of sound generated aerodynamically, there is another source of flow noise which occurs independent of mass injection or solid boundary impingement. Lighthill's theory describes the source distribution within a fluctuating flow surrounded by a large volume of fluid at rest. Such a source distribution would, for example, account for the noise generated by a turbulent jet exhausting into the atmosphere. In fact, the jet noise problem was the primary motivation for Lighthill's work.

Lighthill's theory of aerodynamic noise is based on an analogy drawn between the full non-linear flow problem and the linear theory of classical acoustics. The full non-linear momentum and continuity equations govern the density field in the real fluid,

including both the fluctuating flow region and the surrounding stationary acoustic medium. However, the acoustic approximations of the momentum and continuity equations are also applicable to the region consisting only of the stationary acoustic medium. The difference between these two sets of equations, the full non-linear set and the acoustic approximation set, is considered as if it were the effect of a fluctuating stress field external to the stationary acoustic medium which radiates sound according to the laws of classical acoustics [14].

The full non-linear continuity and momentum equations, respectively, are given by equations (1) and (2). Equations (3) and (4) are the corresponding acoustic approximations. Now, the full momentum equation is essentially rewritten in a form like that of the acoustic approximation so the difference between the two equations can be explicitly identified. To this end the full momentum transport ($\rho V_i V_j + P_{ij}$) is written as the sum of the acoustic approximation, $c_0^2 \rho \delta_{ij}$, of the momentum transport (where δ_{ij} is the Kronecker delta) plus an additional term, T_{ij} , which takes the form [14]

$$T_{ij} = \rho V_i V_j + P_{ij} - c_0^2 \rho \delta_{ij} . \quad (9)$$

Then the full non-linear momentum equation can be written as [14]

$$\frac{\partial(\rho V_j)}{\partial t} + c_0^2 \frac{\partial \rho}{\partial x_j} = -\frac{\partial T_{ij}}{\partial x_j} . \quad (10)$$

Combining equation (10) with the continuity equation leads to an inhomogeneous wave equation [14]

$$\frac{\partial^2 \rho}{\partial t^2} - c_0^2 \frac{\partial^2 \rho}{\partial x_i^2} = \frac{\partial^2 T_{ij}}{\partial x_i \partial x_j} . \quad (11)$$

The term on the right-hand side of equation (11) contains two space derivatives, and therefore, represents neither a monopole nor a dipole source distribution. Rather, it represents a quadrupole distribution derived from dipoles in the same way dipoles are derived from monopoles. Since a system of externally applied forces is represented by a dipole distribution, it follows that a system of externally applied stresses is represented by a quadrupole distribution. As shown in figure 2-1, a fluid element under stress experiences equal and opposite forces on two sides. The forces are equivalent to dipoles, and therefore the combination of equal and opposite forces is equivalent to a quadrupole. An element under compressive stress, figure 2-1a, is represented by a longitudinal quadrupole, while an element under shear stress, figure 2-1b, is represented by a lateral quadrupole. Figure 2-2 shows the sound field produced by a lateral quadrupole. Each of the nine elements in the stress tensor T_{ij} is a quadrupole with axes in the x_i and x_j directions. Three of the nine are longitudinal quadrupoles (where $i=j$), and the remaining six are lateral quadrupoles.

Since Lighthill's inhomogeneous wave equation is based on the full continuity and momentum equations, all real flow effects must be taken into account by the quadrupole source distribution T_{ij} . Outside the flow region, where the classical laws of acoustics hold, these sources vanish, and equation (11) reduces to the homogeneous wave equation.

This implies that the fluctuating flow, confined to a limited part of an otherwise stationary acoustic medium, generates the same fluctuations of density as that produced by a system of externally applied stresses, T_{ij} , acting on the stationary acoustic medium. T_{ij} , representing the difference between the real momentum transport and its acoustic approximation, is dominated by the convective momentum transport term $\rho V_i V_j$ for subsonic cold air jets [18].

The solution to equation (11), obtained using the Green's function, is given by the volume integral [16]

$$\rho(\bar{x}, t) - \rho_0 = \frac{1}{4\pi c_0^2} \frac{\partial^2}{\partial x_i \partial x_j} \int T_{ij}(\bar{y}, t - \frac{|\bar{x} - \bar{y}|}{c_0}) \frac{d^3 \bar{y}}{|\bar{x} - \bar{y}|}, \quad (12)$$

where the solution represents the density field at a distance $|\bar{x} - \bar{y}|$ from the source at $\bar{y}$. The integration is carried out with respect to the source volume element $d^3 \bar{y}$. In the far field, where the distance from the flow region is large compared to the dimensions of the flow region, $|\bar{x} - \bar{y}|$ may be approximated by x , provided the origin is taken in the flow field. Hence, the far field approximation of equation (12) may be written as

$$\rho(\bar{x}, t) - \rho_0 = \frac{1}{4\pi c_0^2} \frac{\partial^2}{\partial x_i \partial x_j} \int T_{ij}(\bar{y}, t - \frac{x}{c_0}) \frac{d^3 \bar{y}}{x}. \quad (13)$$

Also, since [16]

$$\frac{\partial}{\partial x_i} = -\frac{x_i}{x} \frac{1}{c_0} \frac{\partial}{\partial t}, \quad (14)$$

the form appropriate for dimensional analysis is

$$\rho(\bar{x},t) - \rho_0 = \frac{1}{4\pi c_0^4} \frac{x_i x_j}{x^3} \frac{\partial^2}{\partial t^2} \int T_{ij}(\bar{y}, t - \frac{x}{c_0}) d^3 \bar{y} . \quad (15)$$

Notice that the density field fluctuates in time in the same way as T_{ij} , except for the time delay x/c_0 . Thus, the sound field has the same frequency as the source, but a phase difference.

If this result is applied to a region of simple turbulence with zero mean flow, the velocity fluctuations can be characterized by the root-mean-square (r.m.s.) velocity u . The volume scale of the source region may be approximated by the dimensions of the turbulent mixing region. The turbulent region is composed of eddies having a random motion uncorrelated to that of other eddies, although the motion within an individual eddy is well correlated. Hence, each eddy may be thought of as an independent quadrupole source distribution [18]. The total acoustic power emitted by the turbulent region is the sum of that radiated from each quadrupole distribution.

In order to avoid the tedious statistical treatment necessary for the random source distributions associated with the complete turbulent region, attention is narrowed to the sound field radiated by a single eddy. Consider a typical eddy with a size on the order of L moving around at an r.m.s. velocity of order u . Then the frequency of the quadrupole distribution associated with this eddy is of order u/L . Assuming $T_{ij} \sim \rho_0 u_i u_j \sim \rho_0 u^2$, $\partial^2 T_{ij} / \partial t^2$ is proportional to $(u/L)^2 \rho_0 u^2$. Applying these approximations to

equation (15), the density field may be estimated as

$$\rho(\bar{x}, t) - \rho_0 \sim \frac{1}{c_0^4} \frac{1}{x} \left(\frac{u}{L}\right)^2 \rho_0 u^2 L^3 - \rho_0 \left(\frac{L}{x}\right) \left(\frac{u}{c_0}\right)^4. \quad (16)$$

The sound intensity is c_0^3/ρ_0 times the mean square fluctuation of the density field [14].

Consequently, the intensity may be estimated by

$$I \sim \frac{c_0^3}{\rho_0} \rho_0^2 \left(\frac{L}{x}\right)^2 \left(\frac{u}{c_0}\right)^8 \sim \rho_0 \left(\frac{L}{x}\right)^2 \frac{u^8}{c_0^5}, \quad (17)$$

and the power radiated by the eddy is

$$P \sim \rho_0 L^2 \frac{u^8}{c_0^5}. \quad (18)$$

So, the power radiated by the quadrupole distribution associated with a single eddy is proportional to the eighth power of the eddy's r.m.s velocity. Referred to as Lighthill's eighth power law, this result is applied in chapter 3 to the quadrupole distributions of a jet mixing layer, where the source velocities are on the order of the mean jet velocity.

Finally, it should be noted that the actual sound field produced aerodynamically may be due to a combination of monopole, dipole, and quadrupole source distributions. Such a combination of source distributions will occur if there is a solid boundary interacting with the flow. Monopoles, distributed along the boundary, have a strength proportional to the variation of local mass outflow from the boundary. Dipoles, also distributed along the boundary, have a strength proportional to the local rate of

momentum output from the boundary. Lastly, quadrupoles, distributed throughout the flow, have a strength proportional to the local stress tensor.

The primary aim of chapter two has been to describe various aerodynamic noise mechanisms from a very fundamental starting point. While such an approach has many limitations, as stated at the outset, it still provides a foundation necessary to understanding jet noise. Chapter 3 focuses on the specific problem of jet noise and, particularly, supersonic jet noise.

CHAPTER 3

JET ACOUSTICS AND THE ACOUSTIC INTERACTION BETWEEN A JET AND A CYLINDRICAL DUCT ENCOMPASSING THE JET

3.1 Overview

After describing the fluid dynamics important to jet noise in chapter 1 and the basic principles of flow noise in chapter 2, specific features of jet noise are presented below, followed by a discussion of the acoustic interaction between a jet and a cylindrical solid wall encompassing the jet. The encompassing cylindrical duct is large enough relative to the jet so that the jet flow does not impinge on the duct surface (hence, such acoustic interaction is not related to dipole generated noise).

This chapter begins with an explanation of so called jet mixing noise which can be explained by Lighthill's acoustic analogy. Supersonic turbulent convection and the resulting eddy Mach wave radiation are discussed in section 3.3. Shock noise and jet screech are addressed in sections 3.4 and 3.5 respectively. Section 3.6 describes the acoustic interaction between a jet and a cylindrical duct encompassing the jet. Finally, chapter 3 is concluded with a discussion of temperature influences on jet acoustics.

3.2 Jet Mixing Noise

All the noise phenomena discussed in this chapter originate in the turbulent mixing layer of the jet. Jet mixing noise refers to the noise due exclusively to turbulent mixing

effects. Hence, any noise associated with shock waves is precluded from a discussion of jet mixing noise. For subsonic and low supersonic jets, where the turbulent eddies in the mixing region are convected at subsonic speeds, Lighthill's acoustic analogy has been applied with great success. When the plume Mach number is high enough that the turbulent eddies are convected at supersonic speed, further explanation of the noise mechanism is needed. Section 3.3 explores the noise of supersonically convected eddies, termed eddy Mach wave radiation. Therefore, the jet mixing noise discussed herein refers to the noise of subsonic and low supersonic jets for which Lighthill's acoustic analogy is applicable.

In section 2.4 Lighthill's theory was applied to a single quadrupole distribution associated with a turbulent eddy. The concept of viewing a turbulent eddy as a quadrupole distribution has strong support [16,19]. On the other hand it is far less certain, from a fundamental point of view, that Lighthill's theory can be applied directly to an extensive region of turbulence. Crighton [16] argues that the effect of eddies scattering the sound waves generated by other eddies and the effect of mean flow gradients refracting the sound waves should be accounted for explicitly, rather than assuming the stress tensor T_{ij} includes all such effects. Nevertheless, Lighthill's theory has proved very accurate in predicting the noise emitted by the turbulent mixing layer of shock free jets up to Mach numbers approaching two [16].

Applying Lighthill's theory directly to jet noise, the source velocity is on the order of the mean flow velocity V . The source volume, generally determined by the external

mechanism which generates the turbulence, is on the scale of the nozzle diameter D . Thus, equation (18) becomes

$$P \sim \rho_0 D^2 \frac{V^8}{c_0^5} \quad (19)$$

for the acoustic power output of a jet. The fact that equation (19) should follow experimental observations so well [3,15,16], even up to high subsonic and low supersonic speeds, actually surprised researchers who thought the effect of source convection would increase the acoustic power output. In fact, although source convection does increase the power output, its effect is offset by other factors.

Source convection results in a Doppler shift in the wavelength. Ahead of the source the wavelength is contracted by the factor $(1-M_c \cos \theta)$, where M_c is the convection Mach number and θ is the observation angle with respect to the jet axis. Clearly, if the wavelength is contracted by the factor $(1-M_c \cos \theta)$, the frequency must be increased by the factor $(1-M_c \cos \theta)^{-1}$. This leads to the factor $(1-M_c \cos \theta)^{-2}$ for the source term $\partial^2 T_{ij} / \partial t^2$ in equation (15). Also, the effective volume of convected sources is increased by the Doppler factor $(1-M_c \cos \theta)^{-1}$ [17]. Together, the increase in effective source volume and the wavelength contraction lead to the modification of equation (16) for convected eddies

$$\rho(\bar{x}, t) = \rho_0 = \rho_0 \frac{L}{x} \frac{u^4}{c_0^4} (1-M_c \cos \theta)^{-3} . \quad (20)$$

The acoustic power output of a convected eddy is then

$$P_e \sim \rho_0 L^2 \frac{u^8}{c_0^5} (1 - M_e \cos \theta)^{-6} . \quad (21)$$

However, considering the power output of a fixed volume of convected turbulence, like that of a jet, the number of eddy sources must decrease as their effective size is increased. Hence, Lighthill [19] shows that the convection effect should be modified by the factor $(1 - M_e \cos \theta)$, and so the acoustic power output of a jet is given by

$$P \sim \rho_0 D^2 \frac{V^8}{c_0^5} (1 - M_e \cos \theta)^{-5} . \quad (22)$$

While the effect of convection on the total acoustic power output is offset by other factors, as will be shown, the convection factor does explain the directionality of jet noise. At lower angles the convection effect is more pronounced which explains the downstream bias of jet mixing noise. Note that the convection factor increases the sound emitted in the downstream direction more than it decreases the sound emitted in the upstream direction. Consequently, source convection increases overall sound power output.

As already stated, the eighth power law, as given by equation (19), compares favorably with observation even at high subsonic and low supersonic jet velocities where a pronounced convection effect is expected. Hence, there must be an offsetting factor. Powell [17] and Lighthill [19] attribute the offsetting factor to reduced turbulent intensities at higher Mach numbers. As pointed out in section 1.3.1, the r.m.s. velocity of an eddy is reduced relative to its convection speed with increasing Mach number. In

fact at supersonic Mach numbers the eddy is virtually frozen relative to the mean flow (i.e., the eddy looks like a blob of fluid moving in the jet's shear layer), while at Mach numbers between 0.5 and 1.0 Lighthill [15,18] suggests an r.m.s. velocity proportional to the $3/4$ power of jet velocity. This leads to a sound power output proportional to V^6 instead of V^8 for jet Mach numbers between 0.5 and 1.0. The V^6 result compares well with measurements at $\theta=90^\circ$ from the jet axis, where the convection factor has no effect. The convection factor restores the velocity dependence from V^6 at 90° to about V^9 at 20° , so that the total power output is still closely proportional to V^8 . The balance between increasing convection speed and decreasing turbulent intensity relative to the jet velocity maintains the eighth power up to velocities nearly twice the speed of sound for shock free jets [16].

As indicated by figure 1-2, most of the turbulent energy is confined to a narrow region near the center of the mixing layer. It follows that the dominant noise sources are located in the center of the mixing layer, and since the turbulent intensity remains fairly constant over the first four diameters (for a subsonic jet), the sound power output is expected to be fairly constant over this length. As the jet velocity falls off in the fully developed region, the sound output also decreases. Therefore, most of the sound output comes from the first four diameters downstream of the nozzle exit.

The frequency spectrum of jet mixing noise is very broad with a flat ill-defined peak [15]. The high frequency sound emanates from the heavily sheared mixing layer near the nozzle exit, where strong lateral quadrupole distributions are expected, based on Lighthill's theory, due to the dominant transverse momentum transport. Low frequency

sound originates predominantly in the turbulence further downstream, where the r.m.s. velocities and the associated frequencies of the turbulent eddies have decreased.

3.3 Eddy Mach Wave Radiation

The mixing layer of a supersonic jet has convection speeds typically ranging from 20% to 90% of the mean jet velocity, depending on the radial distance from the potential core. As stated in section 1.3, eddies convected in this mixing layer at supersonic speed are nearly frozen relative to the mean flow and can emit sound in the form of ballistic Mach waves. The eddy may be viewed as a quadrupole source distribution whose emitted sound waves coalesce into a Mach wave, as shown in figure 3-1. The Mach angle (the angle the Mach wave makes with respect to the axis of source convection) is given by the inverse sine of the reciprocal of the convection Mach number. Figure 3-2 is a schematic of the Mach wave field of a supersonic jet. The propagation direction of the Mach waves is given by the complement to the Mach angle.

The origin of a Mach wave inside the mixing layer of a jet is more complicated than figure 3-1, since the eddy is surrounded by a convection velocity gradient instead of a stationary environment. Hence, the eddy is never supersonic relative to its immediate surroundings, and the emitted sound waves do not initially coalesce into a Mach wave. A Mach wave emerges only when the sound waves propagate into a region relative to which the source velocity is supersonic.

Therefore, a very small eddy, though convected at supersonic speeds, may not produce a Mach wave, since its disturbance may die out before reaching a location in the

mixing layer where they will form a Mach wave. The quadrupole source distribution, characteristic to an eddy, must be of sufficient strength to produce sound waves that penetrate through the mixing layer in order for it to produce Mach waves. A very thin supersonic mixing layer cannot support the large dominant eddies required for eddy Mach wave radiation.

Based on measurements of the Mach angle taken from Schlieren photographs, the convection velocity of the Mach wave bearing eddies has been determined to be about 75% of mean jet velocity [8,11]. This supports the notion that a very thin supersonic shear layer, which has convection speeds no lower than 80% of the mean jet velocity, cannot support the large eddies necessary to produce Mach waves.

The strength of eddy produced Mach waves is proportional to the convection velocity of the eddy. This is because higher speed jets have a steeper velocity gradient in the mixing layer. Hence, there is less of a spatial expanse for the sound waves to decay before coalescing into a Mach wave. Eddies convected at speeds only slightly above the sonic speed are unlikely to produce Mach waves, since their sound waves, decaying exponentially in the mixing layer, have to propagate all the way to the exterior of the jet before forming a Mach wave. The waves are so weak by the time they emerge from the jet that any Mach wave formed is almost non-effectual. When the characteristic Mach wave bearing eddy, traveling at 75% of the mean jet velocity, is convected at higher supersonic speeds, a steeper velocity gradient results in less signal decay before coalescence into a Mach wave. In fact, for higher source convection speeds, the Mach wave originates deeper in the mixing layer and then curves upstream through the

remaining velocity gradient before becoming straight in the stationary environment around the jet.

With the statements above in mind, it is not surprising that eddy Mach wave radiation does not become a significant source of jet noise until the mean jet Mach number approaches two [8,11]. Jet mixing noise, discussed in section 3.2, dominates at Mach numbers lower than two for shock-free jets. Eddy Mach wave radiation and jet mixing noise are comparable at Mach numbers close to two. At higher Mach numbers near 2.5 the eddy Mach wave mechanism is the dominant source of noise for shock-free jets [11,16]. These guidelines are for unheated jets. Eddy Mach wave radiation becomes important at lower Mach numbers for heated jets, where there is a temperature gradient as well as a velocity gradient in the mixing layer.

Ffowcs Williams and other researchers studying jet noise have extended Lighthill's acoustic analogy to explain eddy Mach wave radiation [1]. The acoustic analogy, as presented in section 3.2, would predict an infinite sound intensity in the direction of eddy Mach wave radiation. This can be seen from equation (22), where the angle θ is the complement to the Mach angle. Although the intensity is very high in this direction, the infinite prediction results from an assumption, incorporated in the derivation of the convection factor, that retarded time variations within an eddy can be neglected [19]. By removing this assumption the acoustic analogy approach has been used to predict the noise of eddy Mach wave bearing jets, and the results, according to Lighthill [19] and Crighton [16], show good agreement with observation. More recently, however, Seiner [11] has pointed out several flaws in this extension of the acoustic analogy approach and,

in fact, has concluded that all current extensions of the acoustic analogy have not evolved to the point where they can predict the dominant noise radiated by the eddy Mach wave mechanism. This conclusion is a consequence of the fact that these analogy models predict the dominant noise sources to be located near the nozzle lip, while, based on observations, the dominant noise sources are expected to be far downstream for eddy Mach wave radiation.

Liu [8] has developed an instability wave model to explain the mechanism for eddy Mach wave radiation. As pointed out in section 1.3, the turbulent eddies in the mixing layer are part of larger scale coherent fluid dynamic structures [7]. These coherent structures are quasi-periodic and extend the length of the jet mixing layer, before dying out in the transition region. The quasi-periodic structures can be viewed as traveling instability waves. The instability waves, initiated at the nozzle exit, rapidly grow as they are convected downstream. Liu's model, which assumes that the strength of the radiated Mach waves is linked to the local instability wave strength, allows for the peaking of noise intensity downstream from the nozzle lip. Tam and Burton [5] have also developed an instability wave model to predict the noise emitted via Mach waves and report good agreement with observation. The Mach wave radiation due to an instability wave traveling at supersonic speed is analogous to the wavy wall radiated field of compressible flow.

Seiner [11] contends that the instability wave model, which recognizes the role of large scale coherent structure, offers the most promise for a unified theory to physically interpret and predict the generation of jet noise. The large scale coherent structures

appear to be the common denominator for all the mechanisms of jet noise, including the mechanism for shock noise.

3.4 Shock Noise

As demonstrated in section 1.3.2, imperfectly expanded supersonic jets contain an expansion/compression wave system extending almost the complete supersonic length of the jet. The diamond shaped regions outlined by the waves are called shock cells, and the noise produced by the interaction of the shock cells with convected turbulence in the supersonic mixing layer is called shock noise. Shock noise dominates the sound radiated from an imperfectly expanded jet at all angles not associated with eddy Mach wave radiation and, in fact, dominates the entire sound field at lower supersonic Mach numbers for which eddy Mach wave radiation is not present.

Numerical predictions and experimental evidence [11] indicate that the intensity of the turbulence in the mixing layer is not reduced by interactions with the expansion/compression wave system. The intensity level of turbulence rises above that for the perfectly expanded condition at the end of a compression system but drops back to the level of the perfectly expanded condition at the end of an expansion system. Based on this turbulence data, the fluid dynamic structures which govern eddy Mach wave radiation and jet mixing noise are expected to be unaffected by the presence of shocks in the jet. Experimental observation has confirmed that jet mixing noise and eddy Mach wave radiation are not reduced by the presence of oblique shocks in the jet. However,

if the jet is operated at extreme off design conditions such that normal shocks are present in the jet plume, these forms of jet noise are reduced [11].

Since the intensity level of the turbulence in the mixing layer is not degraded by the shock/turbulence interactions of an imperfectly expanded jet, the energy radiated as sound via these interactions must come from the shock waves. The loss of energy from the periodic wave system is reflected in an incremental weakening of wave strength until, eventually, the wave system dies out far downstream of the nozzle exit. Experimental observations indicate that the periodic wave system dies out prior to the termination of the supersonic region [13]. These observations fit Seiner's numerically generated model [11], shown in figure 1-7, and are further proof that the periodic wave system supplies the energy which is radiated as sound. Hence, the sound power radiated is proportional to the power contained in the periodic wave system.

Several models have been developed to explain the mechanism for shock noise generation. Harper-Bourne and Fisher's [20] model, as described by Seiner [11], consists of an array of monopole sources located at the ends of the periodic shock cells in the mixing layer. These sources are assumed to be well correlated such that the noise radiated is subject to constructive interference. Their model, based on Powell's earlier model for jet screech, predicts peaks in the shock noise spectrum at frequencies given by

$$f_p = \frac{nV_e}{L(1-M_e \cos \theta)}, \quad n=1,2,3,\dots, \quad (23)$$

where θ is the observation angle with respect to the flow direction, V_e is the eddy convection velocity, L is the shock cell spacing, and $M_e = V/c_s$ with c_s equal to the

ambient speed of sound. Based on Harper-Bourne and Fisher's experiments, the eddy convection velocity is estimated to be about 70% of the mean jet velocity. The shock cell spacing, L , is taken as an average of the shock cell spacing in the region where the dominant shock noise originates. Experimental observations show that the most intense shock noise comes from the region near the end of the annular mixing layer. The average shock cell spacing can either be obtained numerically or empirically [11]. Harper-Bourne and Fisher's experiments on choked jets confirm that the dominant noise is associated with downstream shocks and show that the intensity of the shock noise, which is proportional to the shock wave strength, can be represented by

$$I_{(\text{shock noise})} \propto (M_j^2 - 1)^2 . \quad (24)$$

This formula is only applicable to choked jets that do not have the strong imbedded normal shocks present in the exhaust of jets operating at extreme off design conditions.

The theoretical model of Howe and Williams [13], consisting of a continuum of axisymmetric vortex rings which interact with the shocks to produce sound, compares favorably with Harper-Bourne and Fisher's model for shock noise generation. Both models predict a sound power output that is proportional to shock wave strength. Hence, the sound power output largely depends on the pressure ratio at which the jet is operated, essentially independent of the details of the shock/turbulence interaction mechanism.

There is evidence that shock noise, like eddy Mach wave radiation, is somehow linked to large scale coherent structures in the mixing layer. Harper-Bourne and Fisher's model envisions an array of monopole sources whose emitted noise is well correlated. They do not explain the phase correlation of the emitted sound, except to say that it is

necessary for a strong sound field via constructive interference. This phase correlation indicates that large scale coherent structures in the mixing layer may play an important part in the shock noise mechanism. Observations [11] show the large scale coherent structures to be phase locked to the periodic shock waves. A further evidence that a relationship exists between the quasi-periodic coherent structures and the shock noise mechanism is their strength correspondence. As stated in section 1.3, the quasi-periodic coherent structures, which can be viewed as large scale instability waves, reach their maximum strength at the end of the mixing layer surrounding the potential core. Observations show that the principle region of shock noise emission occurs near the end of the annular mixing layer. A possible explanation for this correspondence is that the larger and stronger coherent structures found in this region allow more of the periodic shock system's energy to be tapped and converted to sound.

Tam and Tanna [21] have developed a shock noise theory which includes the role of large scale coherent structures. They propose that broadband shock associated noise is generated by a weak interaction between downstream propagating large turbulence structures, or instability waves, and the shock cell structure of the jet. Their theory, even though derived in a way fundamentally different from that of Harper-Bourne and Fisher, nevertheless, recovers the same sound intensity and peak frequency predictions for the sonic nozzle.

The Tam and Tanna frequency prediction is given by

$$f_p = \frac{V_c k_n}{2\pi(1-M_c \cos\theta)}, \quad n=1,2,3,\dots, \quad (25)$$

where V_c is the phase velocity of the instability wave of frequency f_n and k_n is the wave number of the n^{th} Fourier component of the shock cell structure. Experimental evidence and numerical models [21] indicate that the phase velocity of instability waves is about 70% of the jet velocity. The phase velocity and the wave number are evaluated at the position in the jet where the amplitude of the instability wave is a maximum. The wave number is obtained using analytical model solutions to represent the shock cell structure and the large turbulence structures. According to these solutions, the amplitudes of the higher order Fourier components of the shock cell structure decrease as n increases. Since the shock noise amplitude is proportional to the amplitude of the Fourier component, the amplitudes of the higher order peaks are relatively small compared to that of the fundamental. This is consistent with experimental observations which show the higher harmonics to be mostly weak and not easily observable. Also, the Tam and Tanna sound intensity prediction, given by

$$I_{(\text{shock noise})} \propto (M_j^2 - M_d^2)^2, \quad (26)$$

is more general than that of Harper-Bourne and Fisher, having been derived for convergent-divergent nozzles. Harper-Bourne and Fisher's prediction for sonic nozzles, equation (24), is recovered as a special case for which the design Mach number, M_d , is equal to one.

The intensity proportionality predicted by Tam and Tanna is a reflection of the idea, stated early in this section, that the level of shock noise radiated from a jet is dependent on the strength of the shock system and, to a large extent, independent of other factors. Therefore, shock noise intensity is a function of the degree to which a jet is operated overexpanded or underexpanded. An exception to this rule occurs when a jet is operated at such extreme off-design conditions that normal shocks, called Mach discs, are imbedded in the jet. When this happens, equation (26) is no longer valid [11].

Equation (26) implies that the sound output is not dependent on temperature; The temperature independence is consistent with experimental observations [11,21] and stands in contrast to the eddy Mach wave phenomena which is sensitive to temperature gradients in the mixing layer.

3.5 Jet Screech

At discrete frequencies the shock noise, discussed above, can be accompanied by much higher amplitude tones called screech tones. Alan Powell [22] first described screech tones in the early 1950's. He attributed the high amplitude tones to an aeroacoustic feedback loop, half the loop consisting of propagating flow disturbances and the other half consisting of propagating acoustic disturbances. The acoustic disturbances are generated by shock/turbulence interactions which, at discrete frequencies, produce sound waves of sufficient strength to propagate all the way upstream to the nozzle exit. Upon reaching the nozzle exit, these sound waves destabilize the jet shear layer enough to produce an embryonic disturbance which has the same frequency as the instigating

sound waves. The embryonic disturbance grows larger as it is convected downstream, eventually obtaining enough strength so that its interactions with shock waves produce sound waves of sufficient amplitude to propagate upstream to the nozzle exit. Two characteristics distinguish this disturbance from other disturbances convected downstream in the mixing layer: First, the disturbance due to feedback is of greater magnitude than other disturbances, having been initiated further upstream at the nozzle exit. Second, the disturbance due to feedback possesses the discrete frequency associated with maximum upstream sound propagation. As a consequence of the first characteristic, a screeching jet has a higher decay and spread rate than a non-screeching jet and produces a higher amplitude sound peak. As a consequence of the second characteristic, the disturbance, when interacting with shock waves, produces sound waves with a preference for propagating back to the nozzle exit thus maintaining the feedback loop.

Therefore, the inner part of the feedback loop consists of flow disturbances convected downstream in the jet shear layer, while the outer loop consists of sound waves propagating from the region of shock turbulence interactions upstream to the nozzle exit. The feedback loop is only maintained at certain discrete frequencies associated with the most intense sound emission in the upstream direction.

Powell's model for the screech mechanism is much the same as Harper-Bourne and Fisher's model for shock noise. Hence, Powell's model, consisting of a stationary array of monopole sources located at the termination of the shock waves, predicts spectral peaks at frequencies given by equation (23) with the angle θ equal to 180° . The angle is a reflection of the upstream propagating waves associated with jet screech.

More recently, Tam, Seiner, and Yu [23] have shown that the screech phenomena, like shock noise, is related to the large scale instability waves in the jet mixing layer. An instability wave, having the same frequency as the screech tones, makes up the inner part of the feedback loop. This instability wave, excited at the nozzle exit by intense upstream propagating sound waves, grows in amplitude as it is convected downstream. Upon reaching a peak amplitude near the end of the mixing layer surrounding the potential core, the instability wave interacts with the shock cell structure generating intense sound waves. These sound waves propagate upstream outside the jet completing the outer part of the feedback loop. The sound waves of sufficient strength to reach the nozzle exit are confined to a narrow frequency band associated with the most unstable mode of the jet shear layer. That frequency band is centered around the fundamental frequency of equation (25) with θ equal to 180° . Hence, the narrow frequency band associated with jet screech is centered around a frequency given by

$$f_s = \frac{V_c k_1}{2\pi(1+M_j)} , \quad (27)$$

where k_1 is the fundamental wave number of the shock cell system. Tam [24], using a vortex sheet to model the jet mixing layer, has derived an analytical prediction for the fundamental wave number.

Equation (27) implies that the screech frequency is a function of the jet Mach number. Experimental observations indicate that heating a jet increases its screech frequency for a given Mach number. However, the effect of heating is apparently accounted for in the models developed by Powell and Tam [11]. Screech tone amplitude

is hard to predict since it partially depends on boundary conditions at the nozzle exit. However, a general guideline for screech amplitude can be obtained from previous observations. Jet screech, like shock noise, increases with increasing shock strength up to the point where normal shocks appear in the flow. Hence, the amplitude of jet screech depends, in part, on the degree to which a jet is operated at off-design conditions.

3.6 Jet-Duct Acoustic Interaction

From the above discussion of jet screech it is clear that jets are sensitive to the acoustic field with which they interact. The feedback acoustic signal associated with jet screech destabilizes the jet shear layer and, as Seiner [11] has observed, alters the spacial structure of the jet's large scale flow disturbances. Another way in which a jet can be subjected to acoustic radiation is through a reflection of the jet's acoustic field back on the jet. Feedback loops can develop, utilizing the reflecting surface, which amplify preferred resonant modes of the jet/reflector geometry. This is the phenomena at work when a jet is encompassed by a cylindrical hard wall.

Abdel-Fattah and Favaloro's [25] experimental work, involving spectral analysis and schlieren photography, shows that discrete tones of acoustic radiation are emitted by supersonic jets confined within cylindrical ducts. However, unlike the discrete tone fundamental frequency associated with jet screech, the fundamental frequency of the duct enclosed jet exhibits discontinuities in its variation with the jet Mach number. More precisely, the discrete tone fundamental frequency of the confined jet displays a staging behavior in its variation with the jet Mach number such that the fundamental frequencies

are clustered around the resonant modes of the duct. Hence, for a range of Mach numbers the discrete tone fundamental frequency shows little variation, having been determined by a particular resonant mode of the duct. Another range of Mach numbers excites a different resonant mode of the duct so that, overall, there is a step variation in tonal frequency with Mach number. Abdel-Fattah and Favaloro show that two discrete tone frequencies, together with their harmonics, appear at Mach numbers close to the points of discontinuity.

The spectral analysis of Abdel-Fattah and Favaloro on ducts of various size reveals that the discrete tone frequency, associated with a particular resonant mode of the duct, varies in inverse proportion to the duct diameter. This is a consequence of the fact that the Helmholtz number associated with a particular resonant mode, defined as the tonal frequency times the duct diameter divided by the local speed of sound ($H=fD/c$), is the same for ducts of different diameter. Therefore, even if ducts are scaled such that they have the same ratio of duct to nozzle cross sectional area, larger diameter ducts will produce discrete tones at lower frequencies. However, the Helmholtz numbers associated with the discrete tones would be the same for any duct diameter. The dependence of discrete tone frequency on duct diameter implies that it is the duct's radial modes which are excited to the point of resonance.

Spectral data also show that the jet Mach number at which a certain discontinuity occurs (ie, the pressure ratio corresponding to a change in the preferred resonant mode) varies linearly with the duct to nozzle area ratio. Hence, increasing the duct diameter for a given nozzle shifts the excited resonant modes with their characteristic tonal frequencies

to higher Mach numbers. Increasing the duct length for a given duct to nozzle area ratio is shown to increase the distortion and dissipation of the basic shock structure of the jet and to shift the discontinuities to lower Mach numbers.

The schlieren photographs of Abdel-Fattah and Favaloro reveal weak shock waves propagating in the upstream direction. It is postulated that these are due a coalescence of sound waves reflected off the duct wall. These reflected waves interact obliquely with the jet, distorting the cellular shock structure and affecting the overall jet behavior before eventually reaching the nozzle exit. Hence, there is a feedback loop at work similar to Powell's proposed screech mechanism with the duct geometry determining the preferred resonant modes.

The acoustic interaction between a jet and a cylindrical duct encompassing the jet, discussed above, is similar to the jet-diffuser resonant phenomena recently investigated at the University of Tennessee Space Institute. The diffuser geometry is similar to a cylindrical duct except for a conical portion at the nozzle inlet. The diffuser is encompassed by a larger cylindrical duct. A schematic of the diffuser/duct geometry is given in chapter 4. The jet-diffuser resonant phenomena is the subject of the experimental investigation reported in chapters 4, 5 and 6.

3.7 Temperature Influences on Jet Acoustics

A temperature gradient in the jet's shear layer could have a significant impact on the jet's acoustic output. A temperature gradient would produce a corresponding gradient in the speed of sound which is proportional to temperature. Hence, as velocity

decreases across a hot jet's shear layer, there is a simultaneous decrease in sound speed. Consequently, the local Mach number, which is the ratio of velocity and sound speed, decreases less rapidly across a hot jet's shear layer than it does across that of a cold jet.

Since the local Mach number at a given convection velocity in the shear layer is higher for a hot jet, eddy Mach wave radiation, produced by turbulent structures convected at supersonic speeds in the jet's shear layer, can occur at lower operating pressure ratios with a hot jet. Similarly, for a given operating pressure ratio, the eddy Mach waves produced by a hot jet are expected to be stronger than those produced by a cold jet.

As reported in section 3.5, experimental observations have shown that heating a supersonic jet increases its screech frequency for a given operating pressure ratio. This is consistent with the theory discussed in sections 2.4 and 3.2 which defines frequency, f , as a function of source velocity, V_s , which in turn is a function of local sound speed, c , and, hence, a function of temperature, T . These relationships can be described in terms of proportionalities as

$$f \propto V_s \propto c \propto \sqrt{T} . \quad (28)$$

Also, the tonal frequencies associated with duct resonant modes are dependent on temperature. This follows from the discussion in section 3.6, where the Helmholtz number, H , essentially a reduced frequency, is shown to be constant for a particular duct resonant mode. Hence, $H=fD/c$ which is equal to a constant for all duct diameters, D ,

implies that the modal frequencies are proportional to the sound speed and, thus, are proportional to the square root of the duct temperature.

From the discussion above it is clear that a correspondence between the discrete tone frequencies associated with jet noise and the natural modes of a particular duct geometry depends on the temperature difference between the jet and the surrounding duct environment. Such a dependence on the temperature difference can be expressed in terms of proportionalities as

$$\frac{f_j}{f_s} \propto \frac{c_j}{c_s} \propto \sqrt{\frac{T_j}{T_s}}, \quad (29)$$

where f_j refers to the jet's tonal frequencies and f_s refers to the natural resonant frequencies of the duct geometry.

CHAPTER 4

EXPERIMENTAL INVESTIGATION OF THE ACOUSTIC INTERACTION BETWEEN A JET AND A DIFFUSER

4.1 Overview

As stated in the introduction, a subscale model of a jet engine testing facility which has experienced very high amplitude discrete tones was constructed at the University of Tennessee Space Institute's gas dynamics lab. A schematic of the test cell geometry is given in figure 4-1. An investigation was initiated to determine whether the high amplitude tones were associated with the natural resonant modes of the testing facility's geometry. Professors John E. Caruthers and Ahmad D. Vakili directed the investigation. The first phase of the investigation is reported in Caruthers, et. al. [2], and the results are summarized below.

In order to verify the suggested coupling between the jet and the acoustic modes of the test cell geometry it was necessary to identify these acoustic modes. The complex test cell geometry, consisting of a diffuser and an encompassing cylindrical duct, necessitated the use of a numerical technique [26] to determine the acoustic modes. Hard wall boundary conditions were used on solid surfaces and at the inlet of the diffuser. Radiation boundary conditions were applied at the exit of the diffuser and also at the exit of the outer cylindrical duct. The conical portion of the diffuser wall was forced to oscillate at a given frequency, while the pressure amplitude response was recorded. This

was repeated over a range of frequencies for each of four circumferential modes. The resonant modes identified in this fashion showed agreement with experimental observations in the full scale facility, thereby supporting the coupling hypothesis.

As part of the same investigation, Caruthers, et. al. [2] performed experiments on the subscale model of the test cell, shown in figure 4-1. A converging-diverging rectangular nozzle was used to simulate the jet engine. The jet exhausted into a diffuser which is similar to the cylindrical ducts tested by Abdel-Fattah and Favaloro [25] except for a conically shaped region at the diffuser inlet. The diffuser is encompassed by a cylindrical duct. The aim was to reproduce the resonant phenomena observed in the full scale facility and to precisely identify the physics associated with the coupling mechanism. The predicted fundamental acoustic modes of the diffuser/cell geometry were verified using a frequency generator and a speaker placed at many locations in the diffuser. Frequency and phase information for the emitted sound, obtained from dynamic pressure transducers located at positions A, B, C, D, E, F and G in figure 4-1, were used to determine the fundamental acoustic modes of the diffuser/cell geometry. The fundamental modes determined in this way agreed with the computational predictions. It should be noted that the discrete tone frequencies observed for the model were higher than those observed in the full scale facility, which is in agreement with the observations reported in section 3.6. However, the Helmholtz number, essentially a reduced frequency, was found to be the same for the resonant modes observed in subscale model and for those observed in the full scale facility. Having verified the acoustic modes of the scaled model to be the same as those of the full scale facility, tests were conducted with the jet

operating. The fundamental modes of duct resonance, identified when the jet was turned on, were the same as those found in the no flow case and in the full scale facility. However, the discrete tone amplitudes observed in the scaled model, associated with the resonant modes of the test cell geometry, were not nearly as high (by 30db) as those observed in the full scale facility, indicating that the scaled model had not fully reproduced the physics of the full scale facility. In order to resolve this discrepancy a continuation of the experimental work was performed. The remainder of this paper is concerned with experimental work aimed at resolving the discrepancy between the duct resonant mode amplitudes of the subscale facility and those of the full scale facility.

4.2 Objectives

The primary motive for continued subscale testing was to determine the reason(s) why the first phase of testing did not completely reproduce the phenomena observed in the full scale facility. Such a determination could reveal the flow and/or geometric parameters critical to the occurrence of the phenomena.

There were several possible explanations for the amplitude discrepancy. First, the subscale nozzle did not have as sharp a convergent section as that of the full scale nozzle. This difference, whose effect had initially been neglected, may have altered the mean jet flow enough to affect the coupling mechanism. Second, since the Reynolds numbers of the subscale and full-scale jets were not matched, the boundary layer may have been thinner and more stable than that of the full scale nozzle. A more stable boundary layer would weaken the large scale coherent structures which are convected in the jet's shear

layer, allowing less of the periodic shock system's energy to be converted to sound. Finally, since the subscale model had only recently been constructed, there was some uncertainty about the quality of the air supply's velocity profile and about the ability of the facility to reproduce more classic supersonic jet noise phenomena.

Since the full scale jet is exhausted at very high temperatures, a possible temperature influence on sound output was acknowledged but consideration of such an effect was deferred due to the high cost involved.

With the above considerations in mind, the following objectives were identified for further study:

- (1) Verify the scaling principles employed in the design of the subscale facility. Since detailed jet acoustic data is not available for the converging/diverging rectangular nozzle (the subject of our investigation), benchmark experiments would be conducted on a conical nozzle designed under the same scaling principles for which detailed jet acoustic data does exist.
- (2) Verify that the subscale test facility is operating properly (i.e. show that the mean steady flow of the subscale jet is symmetric).
- (3) Investigate the sensitivity of the jet's acoustic output to variations in the throat contour of the rectangular nozzle (ie. investigate the impact of changing to a sharper corner throat section which is more like that of the full scale nozzle).
- (4) Investigate the impact of thickening the boundary layer of the subscale jet by adding upstream trips and other disturbances to the nozzle surfaces.

4.3 Facility Description

A schematic of the subscale facility is given in figure 4-1. A, B, C, D, E, F and G represent the port locations on the diffuser for dynamic pressure transducers. The

nozzle shown in figure 4-1 is the converging/diverging rectangular nozzle used in the first phase of testing to simulate a jet engine's exhaust. High pressure air, supplied by storage tanks, is throttled through a two inch control valve. Three honeycombs located in the plenum immediately upstream from the nozzle are used to produce a uniform flow. The nozzle and diffuser are encompassed by two large cylindrical ducts bolted together along the circular flange, as shown in figure 4-1. The upstream conically shaped portion of the diffuser is also bolted to the circular flange, while the downstream portion of the diffuser is welded to the flange. Though not shown in the schematic, plexiglass windows are installed in the outer duct and the diffuser to allow for flow visualization.

The rectangular nozzle used in the first phase of subscale testing, henceforth referred to as RN1, was designed for continuous variation of the nozzle exit-to-throat area ratio. RN1 has a throat width of 2.6 times the throat height, $TH1$. The length of the divergent section of RN1 is also $2.6TH1$. The diameter of the diffuser inlet is $7TH1$; the diameter of the downstream portion of the diffuser is $9.1TH1$; the outermost cylinder has a diameter of $22.2TH1$. These dimensions translate into area ratios, relative to the nozzle throat, of 14, 25 and 146, respectively, for the diffuser inlet, the diffuser outlet and the outermost cylinder. The diffuser length is $7.6TH1$; the outermost cylinder extends $110.3TH1$ downstream of the nozzle exit. Twenty-six holes, equally spaced ($0.22TH1$ apart) along the centerline of the nozzle's upper and lower surfaces, served as ports for measuring static pressure. As shown in figure 4-2a, these ports extended from the throat region nearly to the nozzle exit. Tygon tubes connected these ports to a pressure transducer which provided continuous measurements recorded on a personal computer.

The second rectangular nozzle tested, RN2, has a much sharper angle between the convergent and divergent portions of the nozzle. Consequently, RN2's throat height, TH2, is one-fifth less than TH1. RN2's throat width is $3.4TH_2$, while the length of RN2's divergent section is $2.8TH_2$. The area ratios of the diffuser inlet, the diffuser outlet and the outermost cylinder relative to RN2's throat are 19, 32 and 188, respectively. Twenty-two static pressure ports, distributed along the upper and lower surfaces (11 on the upper and 11 on the lower) of the nozzle's divergent section, provided continuous measurements recorded on a personal computer. Figure 4-2b shows the location of these ports for RN2.

Since acoustical data for a convergent conical nozzle was available in the literature, such a nozzle was constructed for testing in the subscale facility. The intent was to verify certain scaling assumptions employed in the facility design by reproducing acoustical data published by other researchers. The conical nozzle had an entrance-to-exit area ratio of 3.5. The exit diameter of the conical nozzle was $1.5TH_1$. This translated into area ratios (relative to the nozzle exit) of 18, 31 and 186, respectively, for the diffuser inlet, the diffuser outlet and the outermost cylinder.

Frequency and phase information from dynamic pressure transducers located in any two of the seven ports (A through G) could be viewed on a Hewlett-Packard 3582A Spectrum Analyzer. When the diffuser/cell was removed, a dynamic pressure transducer was mounted immediately above the nozzle exit. There were two working pressure transducers at the start of the experimental work. However, one was damaged just prior to completing the experimental work and, so, the last few tests were conducted with only

one dynamic pressure transducer. Pressure and temperature transducers installed in the air supply line provided total temperature and total pressure measurements which were recorded continuously during each test via a personal computer. For purposes of Mach number calculations (obtained from the operating pressure ratio) the static pressure was taken to be equal to the ambient pressure.

A pitot probe mounted on a computer controlled traversing system provided a profile of the total pressure distribution at various locations downstream of the nozzle exit. These total pressure profiles gave a direct indication of flow symmetry or asymmetry. The traversing system was adapted so that it could be mounted to a window seal of the large outer duct and traverses could be taken at the nozzle exit from inside the diffuser. Additionally, a mounting system independent of the test cell was constructed so traverses could be made when the diffuser and test cell were removed.

A Schlieren optical system was set up to view the flow structure of the jet exhausting from the nozzle. The flow could be seen through plexiglass windows when the diffuser/cell was installed. The flow structure, as seen via the Schlieren setup, was video recorded for virtually all the tests.

4.4 Experimental Procedure

The experimental procedure follows the objectives outlined in section 4.2. Three nozzles (the rectangular nozzle of the first phase of study, RN1; a second rectangular nozzle with a sharper throat section, RN2; and a conical nozzle) were investigated. Tests were conducted for a wide range of pressure ratios and area ratios (for the two rectangular

nozzles). The conical nozzle and RN2 were tested with the cell/diffuser installed and with it removed. RN1, having been tested with the cell/diffuser installed in the previous study, was tested with it removed.

The first series of tests were conducted using a conical nozzle constructed according to design specifications obtained from Dr. J. M. Seiner of NASA Langley. Published acoustical data for this nozzle, obtained from reference 10, was compared with data collected at the Space Institute's gas dynamics lab in order to verify scaling principles employed in the subscale facility's design. For comparison purposes, tests were conducted with the diffuser removed. Data was collected for pressure ratios ranging from 2.1 to 7.8. The pitot probe was used to characterize the total pressure distribution at distances of $0.14D$, $1.14D$, $2.14D$ and $3.14D$ downstream of the nozzle exit (where D is the nozzle exit diameter). A dynamic pressure transducer, placed 2.5 diameters above the nozzle centerline (at the exit) for some tests and 11.5 diameters above for others, measured the acoustic output of the jet. Flow visualization was possible via a Schlieren optical setup.

The next series of tests involved the two rectangular nozzles, RN1 and RN2. Both were tested without the diffuser. The intent was to verify that the subscale facility was operating properly in that it produced a symmetric flow exiting the nozzle. Tests were conducted for pressure ratios ranging from 2 to 5.1. The area ratio for both nozzles was set at 1.2. The pressure ports described in section 4.3 provided measurements of the static pressure distribution along the nozzles' upper and lower surfaces. These pressure distributions indicated the location of shock waves and revealed the degree of flow

symmetry inside the nozzle. Vertical traverses, taken with the pitot probe, provided profiles of the total pressure distribution downstream of the nozzle exit. These pressure profiles, along with Schlieren flow visualization, indicated the degree of symmetry in the jet. Acoustic measurements were taken with a dynamic pressure transducer located 2.5 diameters above the jet axis at the nozzle exit.

The third series of tests involved RN2 with the diffuser/cell installed. The intent was to measure the acoustic field produced by RN2 inside the diffuser and to compare such data with similar data taken for RN1 in the earlier investigation. Differences in the acoustic fields produced by RN1 and RN2 would indicate that the "sharper corner" throat of RN2 (which is more like the full scale nozzle) adds enough mean flow instability to affect the jet's acoustic output. Two dynamic pressure transducers, placed in various ports (A-G) provided frequency, phase and amplitude information for the sound field inside the diffuser. Tests were conducted at pressure ratios ranging from 2 to 5.2 and at area ratios of 1.2, 1.5 and 2.3. Plexiglass windows installed in the diffuser and outer cylinder allowed for flow visualization via a Schlieren setup. The static pressure distributions along the nozzle's upper and lower surfaces were recorded continuously. The pitot probe traversing mechanism was mounted to one of the window frames for access to the jet inside the diffuser.

Next, the conical nozzle was tested with the diffuser installed. The aim was to examine the impact, if any, that the diffuser would have on the screech tones observed in the previous tests with the conical nozzle. Tests were conducted for pressure ratios

ranging from 2.7 to 4.2. Acoustic and flow data were collected in the same fashion as described above.

Finally, the impact of a thicker boundary layer (between the mean jet and the nozzle surface) was examined. Four techniques were used to alter the boundary layer of RN2. The first effort to thicken the boundary layer was to construct a ridge (with a height of $0.14TH_2$) in the nozzle throat. Acoustic and flow data were recorded during a series of tests with this ridge in place. Secondly, tests were conducted after the ridge had gaps cut into it such that it had a jagged appearance. A third effort at tripping the boundary layer involved constructing triangular blocks ($0.07TH_2$ on all sides) along the divergent section of the nozzle. Finally, after removing the triangular blocks, black sand was glued to the divergent section of the nozzle. The aim of all these efforts was to mimic the boundary layer characteristics of the full scale jet engine, since there was concern that this might be a critical shortcoming in our subscale model. Schematics of the four boundary layer manipulation efforts are shown in figure 4-3. An additional series of tests were conducted with the three honeycombs (located upstream from the nozzle) removed. This was intended to address a concern that the honeycombs made the flow more uniform and, therefore, more stable than the flow produced by the full scale jet engine.

CHAPTER 5

RESULTS AND DISCUSSION

5.1 Overview

Results of the experimental investigation, including acoustic and flow data, are presented below in the form of graphs and Schlieren pictures. Pitot probe traverses, static pressure ports (located on the diverging nozzle's upper and lower surfaces), and Schlieren flow visualization are used to characterize flow conditions such as jet symmetry. Since data was collected for more than 200 separate cases, results have been consolidated for a more concise presentation. Peak sound amplitudes and their associated frequencies have been extracted from the spectral data recorded on a Hewlett-Packard 3582A Spectrum Analyzer and are presented in a variety of plots that allow for comparison of different cases. The frequency bandwidth is 30Hz, and there is a 1% margin of error for the sound amplitude measurements. The error margin for the static and total pressure measurements is also estimated to be 1%.

5.2 Conical Nozzle

The conical nozzle was tested first to confirm scaling principles employed in the subscale facility's design. The first series of tests were conducted without the test cell installed so that results could be compared to those reported in reference 10. Since there was some concern that the jet's acoustic field might interact with the large circular flange

to which the nozzle was mounted, several tests were conducted with a foam insulation cover over the flange.

Acoustic data was collected at pressure ratios ranging from 2.1 to 7.8 (corresponding to jet Mach numbers of 1.1 to 2.0). The jet's flow field was videotaped over this entire range of operating pressure ratios using a Schlieren system. Pitot probe traverses were taken along the vertical plane of the nozzle axis at distances of 0.14D, 1.14D, 2.14D and 3.14D downstream of the nozzle exit (D = nozzle exit diameter).

Figures 5-1 and 5-2 are the Schlieren pictures of the flow for jet Mach numbers of 1.3 and 1.6, respectively. There is a continuous succession of expansion and compression waves visible in each picture, as expected for such underexpanded nozzles. Three shock cells are visible in figure 5-1, while only two are visible over approximately the same distance in figure 5-2. As discussed in sections 3.4 and 3.5, the discrete tone frequencies for shock noise and jet screech are inversely proportional to the shock cell spacing, L. Since the pictures show shock cell spacing to increase with increasing jet Mach number, it can be anticipated that the discrete tone frequencies will decrease with increasing Mach number. This is consistent with the screech tone data reported in reference 10. Perhaps more important, figures 5-1 and 5-2 reveal a high degree of jet symmetry. The total pressure profiles, shown in Appendix B (B-1 through B-4), show the jet to be very symmetric even at more than three diameters downstream of the nozzle exit.

Figure 5-3 shows the discrete tone frequencies as a function of nozzle pressure ratio for the conical nozzle. As expected, these frequencies decrease with increasing

pressure ratio. A dynamic pressure transducer, located 2.5 diameters above the center of the nozzle exit, provided the acoustic data when the test cell was not installed. The circles, representing data collected when the test cell was installed, only cover the pressure ratio range from 2.8 to 4.2, since this was the range containing the highest amplitude peaks. In fact, there were no distinct peaks at pressure ratios below this range with the test cell installed; above a pressure ratio of 4.2 there was a drop in peak amplitudes. Figure 5-4 shows the amplitude (in pressure dB) of the discrete tones whose frequencies are given by the circles in figure 5-3. The A and C ports on the diffuser, referred to in the figure, are those ports shown in figure 4-1.

With the test cell absent, the most distinct peaks (which were accompanied by an audible screeching sound) occurred at pressure ratios between 3.5 and 5.7. Hence, the data points which lie outside this range in figure 5-3 probably do not represent jet screech. The discrete tone amplitudes for all the data points in figure 5-3 are shown in figure 5-5. The test cell had two effects on the jet noise produced by the conical nozzle: First, the discrete tone amplitudes were about 25dB higher when the test cell was installed. Second, the distinct peaks, characteristic to jet screech, occurred at a lower range of pressure ratios when the test cell was installed (i.e. 2.8-4.2 instead of 3.5-5.7). Again, though data is only shown over the range of pressure ratios between 2.8 and 4.2 (since this range contained the highest amplitude peaks), tests were conducted over a broader range which included the range (3.5-5.7) containing the highest amplitude peaks observed with the diffuser absent. The discrete tones' shift to lower pressure ratios with the test cell installed seems consistent with the experimental work described in section

3.6 (although the two cases are not completely parallel). The experimental work of Abdel-Fattah and Favaloro showed that the duct resonant modes were shifted to lower pressure ratios when the duct diameter was reduced. Hence, it is not surprising that the discrete tones associated with jet screech would occur at lower pressure ratios when the jet is encompassed by a diffuser than would be the case when there is no diffuser (i.e. a diffuser with an infinite diameter). However, the discrete tone frequencies observed when the diffuser was installed do not exhibit the staged variation with pressure ratio which characterizes duct resonant modes. As shown in figure 5-3, the tonal frequencies observed with the diffuser installed exhibit the same continuous variation with pressure ratio as those observed with the diffuser removed. This can be seen clearly in figure 5-6, where only the circles of figure 5-3 are displayed. This continuous variation of tonal frequency with pressure ratio, characteristic to jet screech, indicates that there was no coupling between the jet and the acoustic modes of the diffuser. Hence, the higher amplitudes observed when the test cell was installed are probably the result of the jet noise simply being confined within the diffuser.

Figure 5-7 illustrates a comparison of the data collected at U.T. Space Institute's (UTSI's) subscale facility with that reported by Dr. J. M. Seiner [11]. The regions bounded by the solid lines represent three modes of jet screech which Seiner had observed (Seiner's data represents jet acoustics exclusively, i.e., there was no test cell apparatus). The vertical axis is a non-dimensional frequency parameter (c_0/fD), where c_0 is the speed of sound, f is the discrete tone frequency (cycles/second) and D is the nozzle exit diameter. The discrete tone frequencies are taken from the data presented in figure

5-3 up to a pressure ratio of 5 (corresponding to a Mach number of 1.7). The data collected at UTSI's facility shows reasonable agreement with that reported by Seiner with most points falling on the second or third mode.

All the acoustic data reported above for the conical nozzle without the test cell was taken with a dynamic pressure transducer located 2.5 nozzle diameters above the center of the nozzle exit (position 1). Also, when the test cell was removed, foam insulation covered the flange to which the nozzle was mounted for the data reported above. Figure 5-8 shows data taken with the insulation removed and with the dynamic pressure transducer located 11.5 diameters above the center of the nozzle exit (position 2). No distinct peaks were observed when the foam insulation was covering the flange and the dynamic transducer was located at position 2. Hence, the flange appears to have amplified the sound (at least in the far field), but the absence of the insulation had no clear effect on the tonal frequencies.

5.3 Rectangular Nozzles (RN1 and RN2)

After testing the conical nozzle to confirm the facility's ability to reproduce published acoustical data, tests were conducted on the rectangular nozzle (RN1) which had been the object of the previous phase of experimental work (referred to in section 4.1). These tests were done with the test cell removed, since RN1 had been tested with the diffuser installed in the earlier study. RN1's area ratio was set at 1.2 for all the tests; the operating pressure ratio ranged from 2 to 3.2 (the full scale facility's highest amplitude sound peaks occurred while operating within this range of pressure ratios).

Figure 5-9 shows P/P_t (P is the static pressure measured along the nozzle's upper and lower surfaces; P_t is the total (stagnation) pressure upstream of the nozzle) as a function of x/L (x is the distance from the nozzle throat to the static pressure port; L is the distance from the throat to the nozzle exit). Surface pressure distributions are plotted for operating pressure ratios (total/ambient) of 2.0, 2.4 and 2.8. In all three cases the nozzle is operating in an overexpanded mode, as indicated by the sharp rise in static pressure which signifies the presence of a shock wave. Although there appears to be some asymmetry at a pressure ratio of 2.0, generally the plots reveal a high degree of symmetry between RN1's upper and lower surfaces.

The test cell was absent for the initial series of tests on RN2. Tests were conducted with the exit-to-throat area ratio set at 1.2 and for pressure ratios from 2.0 to 5.1. This was followed by a series of tests on RN2 with the cell/diffuser installed. Tests were conducted at pressure ratios ranging from 2.0 to 5.2 and for area ratios of 1.2, 1.5 and 2.266.

The variation in pressure ratio with distance from the nozzle throat for RN2 is displayed in figure 5-10. The two static pressure ports closest to the nozzle exit were removed to accommodate installation of the test cell. The plots shown in figure 5-10 (for the same pressure ratios as those of figure 5-9) reveal significant asymmetry between RN2's upper and lower surfaces. Comparisons of surface pressure distributions for RN1 and RN2 operating at pressure ratios (total/ambient) of 2.0, 2.15, 2.4 and 2.8 are illustrated in figures 5-11 through 5-14.

Figures 5-15 and 5-16 are the Schlieren pictures of the jet produced by RN1 operating at pressure ratios of 2.15 and 3.18, respectively. The shock waves visible in these pictures reveal a high degree jet symmetry downstream from the nozzle exit. The data collected from pitot probe traverses, shown in Appendix B (B-5 and B-6) provides further evidence of jet symmetry for RN1.

Schlieren pictures for RN2 operating at pressure ratios of 2.0, 2.15, 2.3 and 3.5 are shown in figures 5-17 through 5-20, respectively. All of these pictures were taken with the test cell absent. The asymmetry in RN2's jet is especially noticeable in figure 5-20 (for a pressure ratio of 3.5). Data collected by the pitot probe for operating pressure ratios of 2.0, 2.15, 2.3, 2.8 and 4.75 is shown in Appendix B (B-7 through B-11). The asymmetry in the total pressure distribution along the vertical plain of the jet axis, evident at pressure ratios of 2.15 and 2.3, reflects the asymmetry in shock pattern apparent in the Schlieren pictures. Figures 5-21 and 5-22 are the Schlieren pictures of RN2's jet with the cell/diffuser installed. Although it is difficult to view the jet through the plexiglass window, shock waves are visible through the crevice at the nozzle exit. Oblique shock waves, emanating from the upper and lower nozzle surfaces, are visible converging on a normal shock wave.

Figures 5-23 and 5-24 summarize the acoustic data collected at lower pressure ratios for RN1 and RN2 with the diffuser/cell absent. The plotted points represent a sampling of discrete tone data observed in multiple tests of RN1 and RN2 at each of three pressure ratios (2.15, 2.3, 3.2). For example, the four circles at a pressure ratio of 2.15 represent four peaks observed during two separate tests, while the two triangles also

represent data collected during two tests. Comparing figures 5-5 and 5-24, RN1 and RN2 produced sound levels 8-10dB higher than the conical nozzle (with the diffuser absent) at pressure ratios ranging from about 2 to 3. RN2 produced a relatively high amplitude peak (158dB) at a pressure ratio of 3.2.

Figure 5-25 displays the data for RN1 with the area ratio set at 1.2 and with the diffuser installed (observed by Caruthers et. al. [2] during the earlier investigation). Since the same discrete tone frequencies occurred at all three pressure ratios, it is apparent that the tones are not shock noise or jet screech (each of which exhibits a continuous variation of tonal frequency with pressure ratio). As reported in section 4.1, these frequencies are associated with the resonant modes of the diffuser/cell. However, the amplitudes, ranging from 122 to 140dB, are about 30dB lower than those observed in the full scale facility. Comparing figures 5-23 and 5-24 with figure 5-25 reveals that more low frequency tones were observed when the diffuser was installed. Figure 5-26 shows similar data observed when the nozzle exit-to-throat area ratio set at 1.5. The tonal frequencies are the same as those shown in figure 5-25. However, the amplitudes are slightly higher for an area ratio of 1.5.

Spectral data for RN2 operating at an area ratio of 1.2 (with the diffuser installed) is illustrated in figure 5-27. Comparing figures 5-25 and 5-27, both RN1 and RN2 produced spectral peaks at about 900 and 1400Hz. However, the amplitudes observed for RN2 were higher than those observed for RN1. Also, the spectral peaks at 1120 and 1280Hz (observed at pressure ratios of 2.15 and 2.4) are exclusive to RN2. This may be the result of differences in the two nozzles or simply because data was not available for

RN1 at these pressure ratios. The first cause seems more likely, since the spectral peaks observed for RN1 at a pressure ratio of 1.5 occur at about the same frequencies as those observed at pressure ratios of 2.7 and 4.0.

Figures 5-28 through 5-30 summarize the effect of the diffuser/cell on RN2 at pressure ratios of 2.15, 2.3 and 5.1, respectively. The diffuser/cell appears to have created more tonal frequencies. Also, at frequencies where there is data available for both cases (with and without the diffuser) the tones observed while the diffuser/cell was installed have about the same amplitudes compared to those observed with the diffuser absent. This unexpected result contrasts with the data recorded for the conical nozzle, illustrated in figure 5-5, where the diffuser had the effect of increasing amplitudes by 25dB.

As stated above, RN1's tonal frequencies remain constant over a range of pressure ratios which indicates that these tones are associated with the normal acoustic modes of the diffuser/cell. The discrete tone frequencies observed while RN2 was installed are similar to those observed for RN1, except at pressure ratios of 2.15 and 2.4. However, as figure 5-31 shows, the amplitudes are higher over a similar range of pressure ratios when RN2 is installed.

The implications of the preceding discussion are illustrated clearly in figure 5-32. While RN2 produced higher amplitudes, the discrete tone frequencies were very close to those observed with RN1 installed. A noteworthy exception, again, is the 1120 and 1280Hz tones observed for RN2 at pressure ratios of 2.15 and 2.4. The discrete tones occurring at about 900 and 1400Hz correspond to the modes observed in the full scale

facility. However, the amplitudes observed for RN2 at the lower pressure ratios (similar to the operating pressure ratios of the full scale nozzle) are still at least 20dB lower than those observed in the full scale facility. The phase angles (measured across the diffuser from top to bottom) corresponding to the figure 5-32 data are shown in figure 5-33.

5.4 Boundary Layer Study of RN2

Since the discrete tones produced by RN2 had lower amplitudes than those observed in the full scale facility (although the amplitudes for RN2 were higher than those observed with RN1 installed), attempts were made to destabilize and/or thicken the jet's shear layer. The first attempt involved constructing a ridge along the upper and lower surfaces of the nozzle throat, as described in section 4.4. After a series of tests at pressure ratios ranging from 1.2 to 4.5, gaps were cut into the ridge giving it a jagged appearance. Tests were conducted with this jagged edge on the upper and lower throat surfaces at pressure ratios ranging from 2.35 to 4.5. The effect of these "upstream" boundary layer manipulations on the diverging nozzle's upper and lower surface pressure distributions is illustrated in figure 5-34 (for a pressure ratio of 2.35). As figure 5-34 shows, these alterations in the throat surface had no significant impact on diverging nozzle's boundary layer.

Figure 5-35 shows the effect of the "triangle blocks" which were distributed along the nozzle's upper and lower surfaces from the throat to the exit. This "downstream" manipulation apparently affected the surface pressure distribution more than the "upstream" manipulations described above. Figure 5-36 illustrates the surface pressure

distributions recorded after sand was glued to the diverging nozzle's upper and lower surfaces. The sand appears to have smoothed out some of the variability seen when there was no boundary layer treatment. This may be a reflection of a thicker boundary layer which weakens the compression waves before they reach the nozzle surface. Figure 5-36 also shows the results obtained when the three honeycombs (located in the plenum upstream of the nozzle) were removed. The sand was still present for this case. Removing the honeycombs appears to have had little effect on the surface pressure distribution.

Figure 5-37 displays the acoustic data observed at a pressure ratio of 2.35 for the four boundary layer manipulations (ridge, jagged edge, triangular blocks and sand), as well as the data collected with the three honeycombs removed (with and without the sand). All the spectral data for the boundary layer study was collected from a transducer located in port A of the diffuser, since the second transducer had been destroyed by this time. Surprisingly, the boundary layer manipulations generally resulted in lower amplitudes. However, removing the honeycombs resulted in a higher amplitude (150dB) at a frequency of 1250Hz. Figures 5-38 and 5-39 illustrate the results at pressure ratios of 2.8 and 4.55, respectively. Again, generally the boundary layer manipulations resulted in lower amplitudes except at frequencies higher than those corresponding to the modes of primary interest for the full scale facility.

CHAPTER 6

CONCLUSIONS

6.1 Review of Experimental Findings

Acoustic data recorded with the conical nozzle installed showed reasonable agreement with published results (figures 5-7 and 5-8), confirming geometric scaling principles employed in the facility design. The cell/diffuser had the effect of amplifying the discrete tones and shifting them to lower pressure ratios (figure 5-5). However, since the tonal frequencies exhibited the same continuous variation with operating pressure ratio as when the cell/diffuser was absent (figures 5-3 and 5-6), the higher amplitudes apparently were the result of the noise simply being confined to a small area rather than being the result of coupled resonant phenomena.

RN1 produced a highly symmetric jet (figures 5-9, 5-15 and 5-16), while RN2's jet showed some asymmetry (figure 5-10, and figures 5-17 through 5-22). When the diffuser was absent, RN2 produced a higher amplitude sound peak than RN1 at a pressure ratio of 3.2, while there was little difference between the two nozzles at pressure ratios of 2.15 and 2.3 (figure 5-24).

The discrete tone frequencies observed for RN1 and RN2 with the diffuser/cell installed were similar over the entire range of operating pressure ratios (figures 5-25, 5-26, 5-27, 5-32), indicating that the modes observed were those associated with the test cell geometry. The diffuser had the effect of increasing the number of tones observed,

particularly at the lower frequencies. However, the diffuser did not amplify the tones already present for the jet operating without the diffuser (figures 5-23, 5-24, 5-25, 5-27, 5-28, 5-29 and 5-30). This unexpected observation contrast with the results obtained for the conical nozzle, where the diffuser/cell had the effect of increasing amplitudes by about 25dB.

The discrete tone amplitudes associated with RN2 were higher than those associated with RN1 (figures 5-31 and 5-32), indicating that the jet's mean flow stability could be an important factor to its acoustic output. RN2's discrete tone amplitudes (with the test cell installed) were still about 20dB lower than those observed in the full scale facility, while RN1's were about 30dB lower those observed in the full scale facility.

The "downstream" boundary layer manipulations, especially the sand, resulted in less static pressure variability along the upper and lower surfaces of the nozzle (figures 5-35 and 5-36). Such smoothing of the static pressure variation is indicative of a thicker boundary layer which weakens the compression waves as they reflect off the nozzle surface. However, the boundary layer manipulations generally resulted in lower amplitude tones, except at frequencies higher than those corresponding to the modes observed in the full scale facility (figures 5-37 through 5-39). Hence, differences in the boundary layers of the scaled model and the real facility were not critical to the lower amplitudes recorded for the model. Removing the three honeycombs upstream of the nozzle did result in a higher amplitude peak (150dB) at 1250Hz for a pressure ratio of 2.35. This indicates that a less uniform and, thus, less stable flow could increase the jet's acoustic output, which is consistent with the comparison of RN1 and RN2 reported above.

The most important findings of the investigation reported herein can be summarized as follows:

- (1) Acoustical data collected for the conical nozzle showed agreement with published data, thereby confirming scaling principles employed in the facility design.
- (2) When the conical nozzle was installed, the presence of the diffuser/cell resulted in amplitudes approximately 25dB higher than those produced by the jet alone.
- (3) When RN1 or RN2 was installed, the presence of the diffuser/cell resulted in more observable tones, particularly at the lower frequencies.
- (4) When RN1 or RN2 was installed, the tones observed with the diffuser/cell absent were not significantly amplified by the presence of the diffuser/cell.
- (5) The acoustic amplitudes observed for RN2, while the diffuser/cell was installed, were higher (by about 10-15dB) than those observed for RN1. However, the amplitudes produced by RN2 (with the diffuser installed) were still about 20dB lower than the high amplitude tones observed at the full scale facility.
- (6) Efforts to thicken RN2's boundary layer generally resulted in lower amplitudes.
- (7) Removing the three honeycombs located in the plenum immediately upstream of the nozzle resulted in a relatively high amplitude peak (150dB) at 1250Hz for a pressure ratio of 2.35.

Findings (5) and (7) indicate that, while flow uniformity and, hence, mean flow stability may be important factors in jet noise, it is likely that there are other reasons why our model did not fully reproduce the phenomena observed in the full scale facility. Finding (6) dispels the notion that a thinner boundary layer in the subscale model was responsible for the lower amplitudes.

6.2 Recommendations

The real engine produces a very hot jet. As stated in chapter 4, consideration of the temperature influence on the jet's acoustic output was not included in the subscale investigation due to the prohibitive cost of producing a hot jet. However, it is clear, from the discussion in section 3.7, that a temperature gradient in the jet's shear layer could have a significant impact on the jet's acoustic output and the acoustic interaction between the jet and the diffuser. Hence, a subscale investigation using a hot jet is needed for a more complete simulation of the full scale facility.

Additionally, since the natural acoustic frequencies of the test cell geometry depend on the size of the test cell, it would be useful to run experiments on the same geometry but with different scaling factors. Such an examination of scaling influences would also provide information on the impact of test cell size on sound wave attenuation.

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LIST OF REFERENCES

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APPENDICES

APPENDIX A

Figures

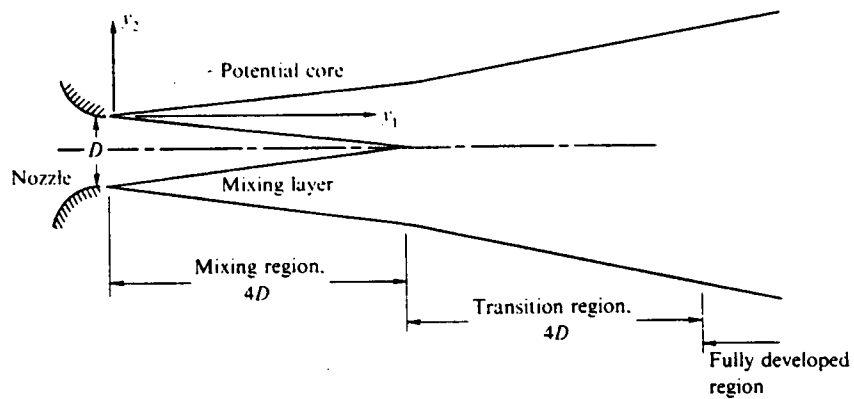


Figure 1-1. Flow Structure of a Subsonic Jet

Source: Goldstein, M.E., Aeroacoustics, pg. 88, McGraw-Hill (1976)

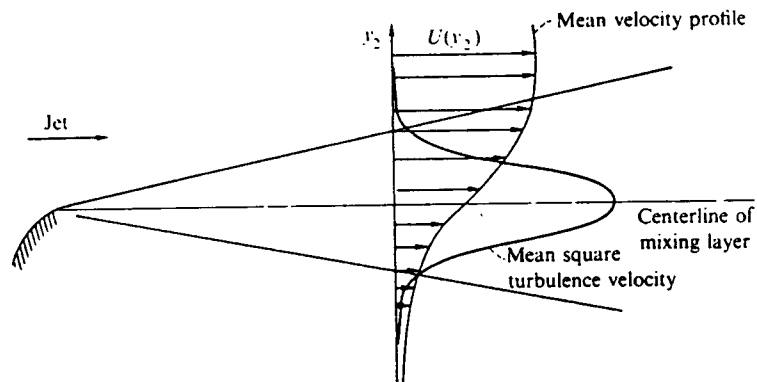


Figure 1-2. Mean Velocity and Mean Square Turbulence Velocity profiles

Source: Goldstein, M.E., Aeroacoustics, pg. 90, McGraw-Hill (1976)

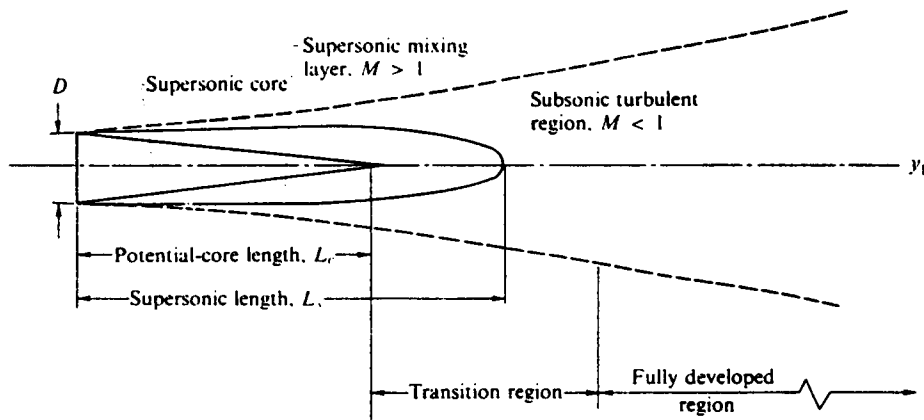


Figure 1-3. Flow Structure of a Shock-Free Supersonic Jet

Source: Goldstein, M.E., Aeroacoustics, pg. 102, McGraw-Hill (1976)

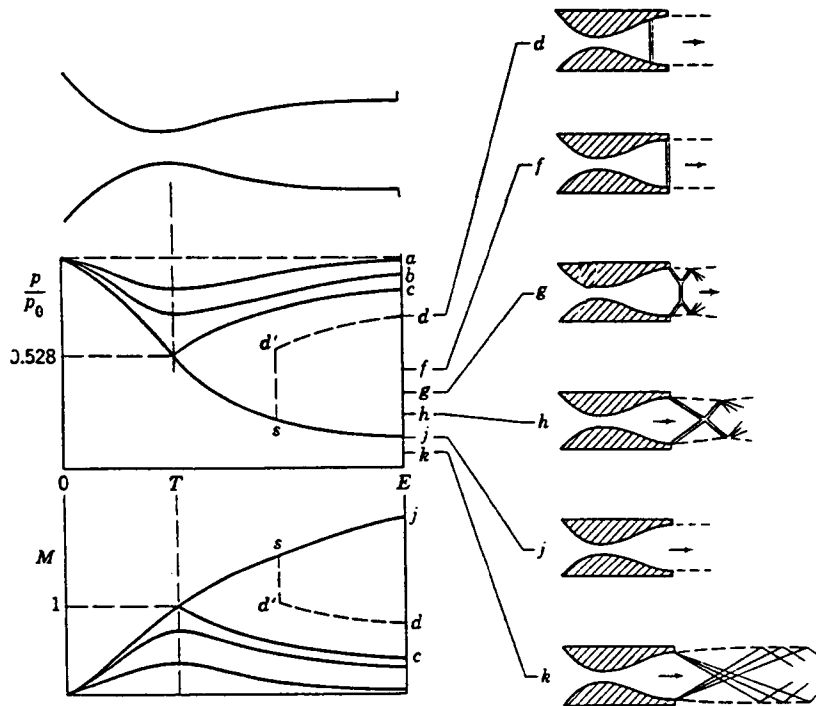


Figure 1-4. Local P/P_t vs Axial Distance and the Resulting Wave Configurations

Source: Liepmann, H.W. and Roshko, A., Elements of Gas Dynamics, pg. 127, Wiley (1957).

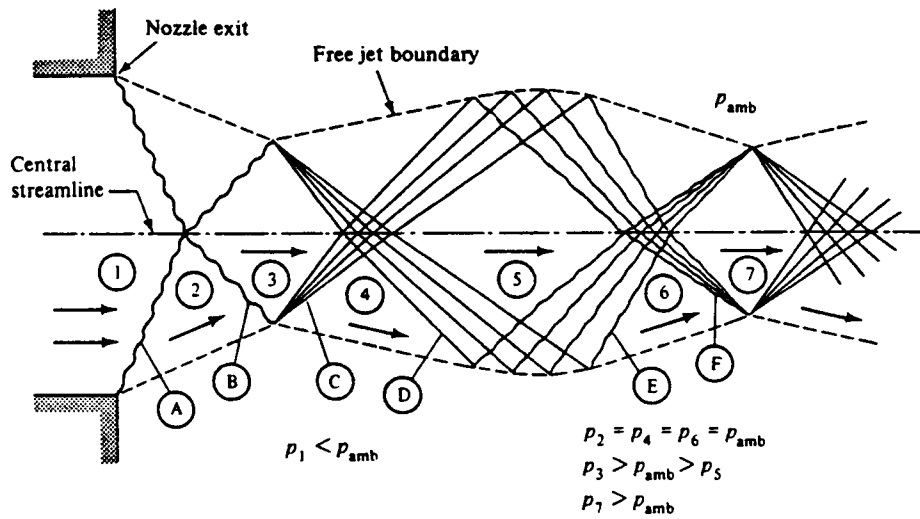


Figure 1-5. Overexpanded Nozzle

Source: Zucker, R.D., Fundamentals of Gas Dynamics, pg. 219, Matrix (1977).

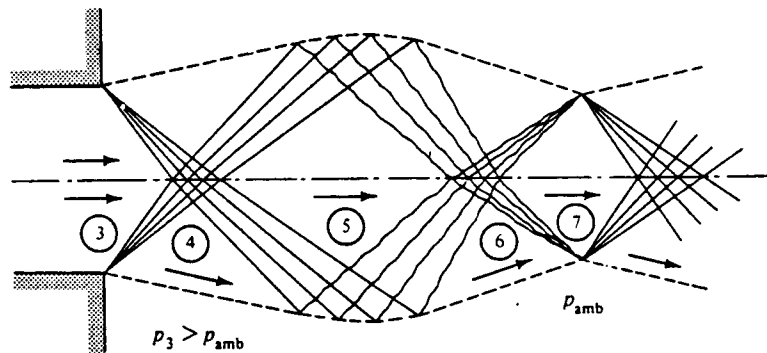


Figure 1-6. Underexpanded Nozzle

Source: Zucker, R.D., Fundamentals of Gas Dynamics, pg. 220, Matrix (1977).

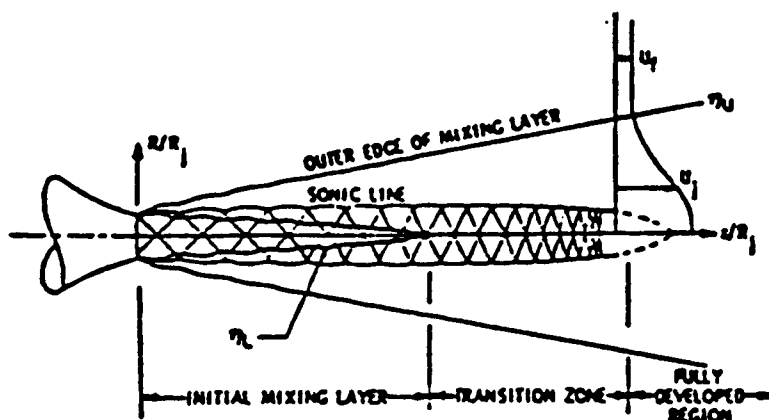


Figure 1-7. Shock Containing Supersonic Jet Plume

Source: Seiner, J.M., "Advances in High Speed Jet Aeroacoustics," AIAA Paper No. 84-2275, pg. 30, AIAA/NASA 9th Aeroacoustics Conference, Williamsburg, Virginia (October 15-17, 1984).

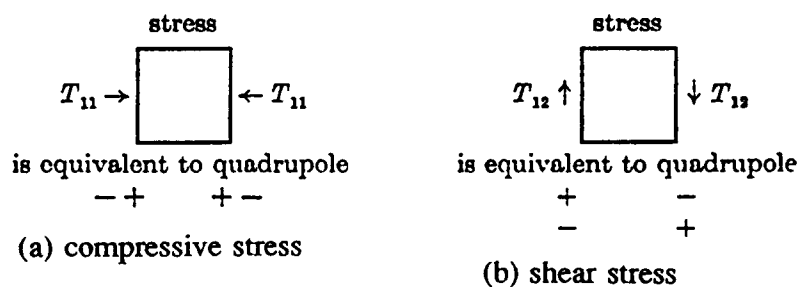


Figure 2-1. Fluid Elements Under Stress

Source: Lighthill, M.J., "Sound Generated Aerodynamically," The Bakerian Lecture, Royal Society of London, Proceedings, Series A, Vol. 267, pg. 154 (1962).

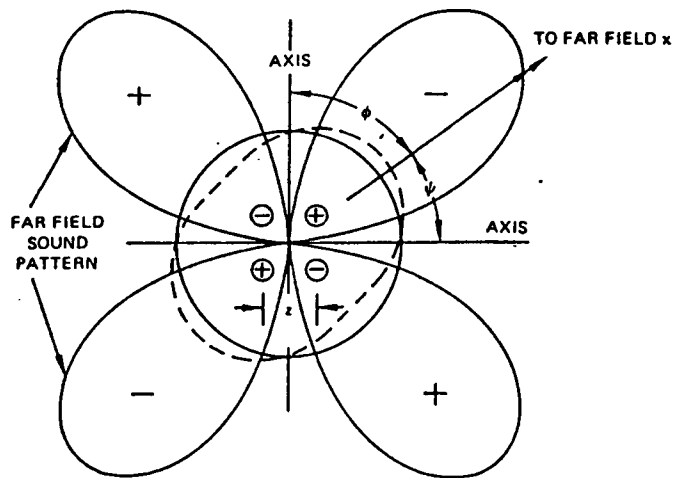


Figure 2-2. Construction of a Lateral Quadrupole from Equal and Opposite Dipoles a Small Distance z Apart

Source: Powell, A., "Jet Noise," Noise-Con 75 Proceedings, pp. 41, National Conference on Noise Control Engineering, Washington, DC (1975).

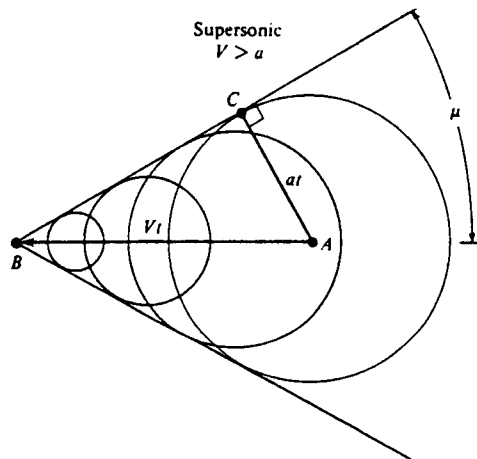


Figure 3-1. Coalescence of Sound Waves into a Mach Wave for a Source Moving at Supersonic Speed

Source: Anderson, J.D., Modern Compressible Flow, pg. 86, McGraw-Hill (1982)

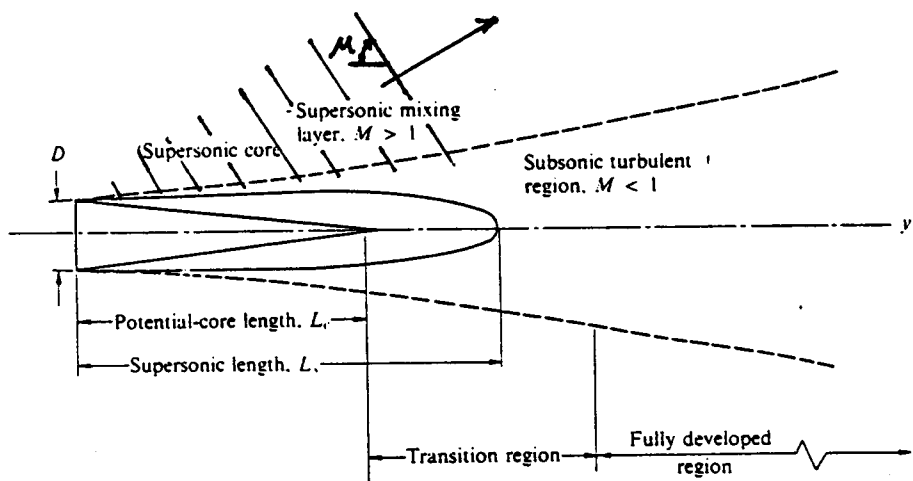


Figure 3-2. Eddy Mach Wave Radiation from the Shear Layer of a Supersonic Jet

Adapted from Goldstein, M.E., Aeroacoustics, pg. 102, McGraw-Hill (1976)

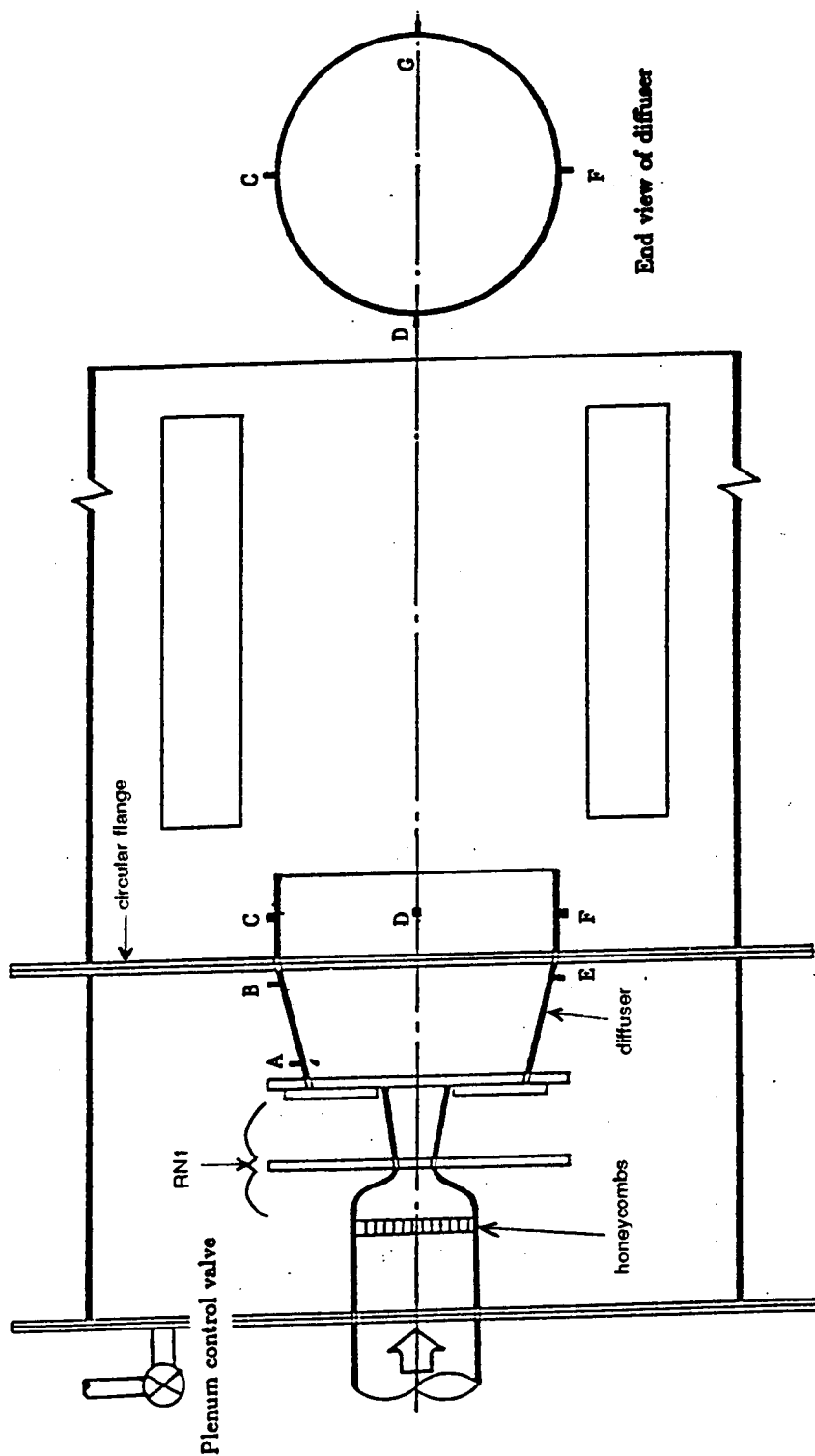
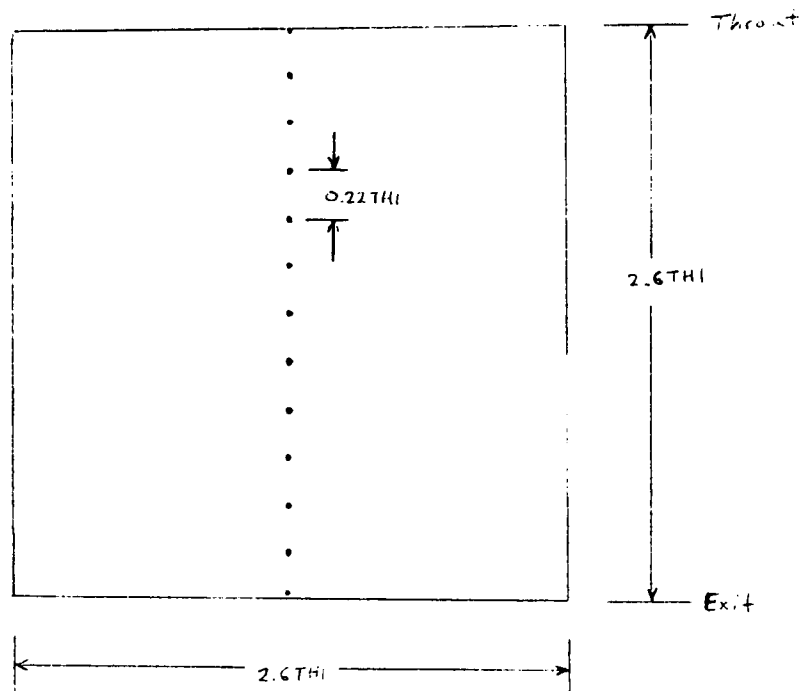
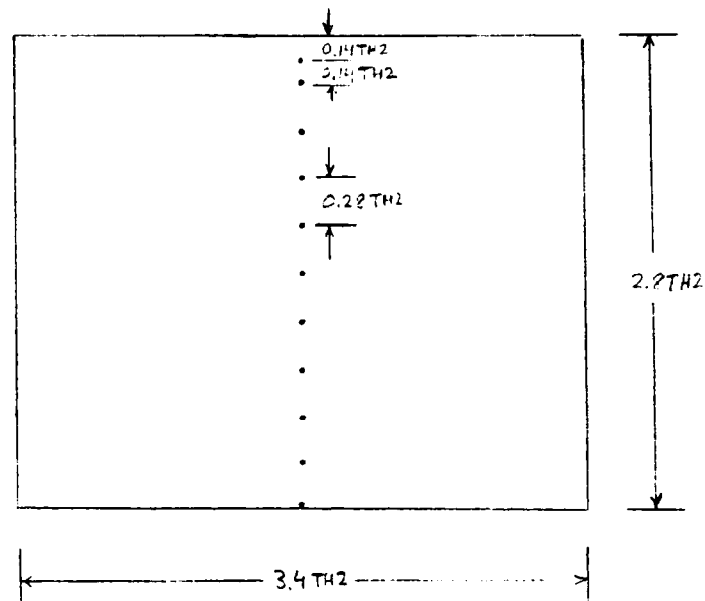


Figure 4-1. Schematic of Subscale Facility

Adapted from Caruthers, J.E. and Vakili, A.D., "Investigation of Resonant Phenomena in an Engine Test Cell," Final Report on Project No. DD22EW, Sverdrup Technology (November 1989)



(a) Ports on Upper/Lower Surfaces of RN1



(b) Ports on Upper/Lower Surfaces of RN2

Figure 4-2. Location of Static Pressure Ports for RN1 and RN2

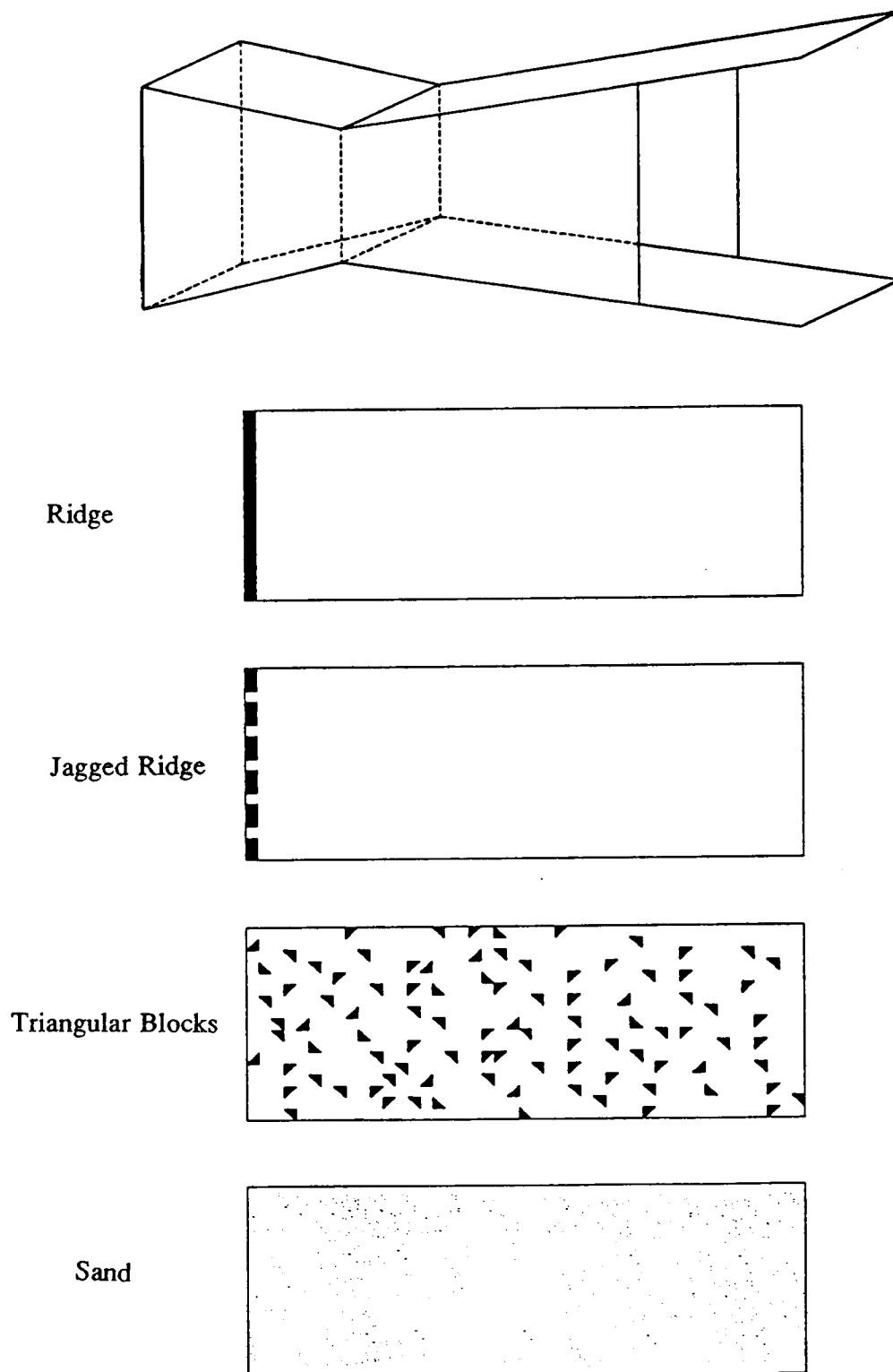


Figure 4-3. Efforts to Thicken RN2's Boundary Layer

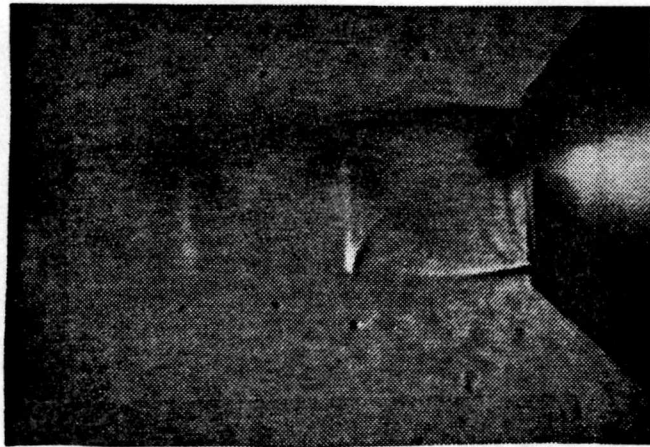


Figure 5-1. Conical Nozzle Schlieren
(Mach No. 1.3)

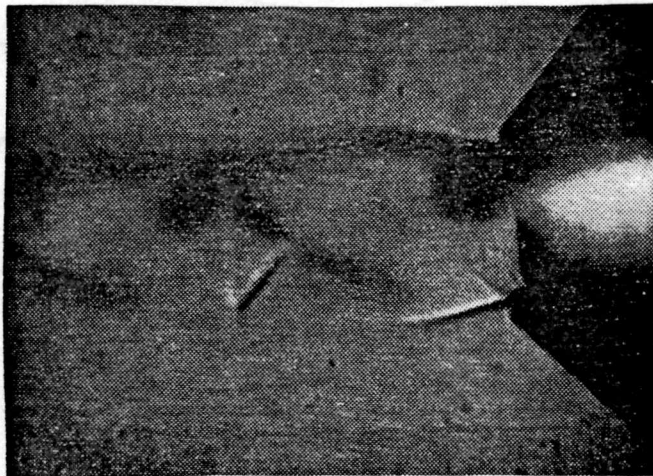


Figure 5-2. Conical Nozzle Schlieren
(Mach No. 1.6)

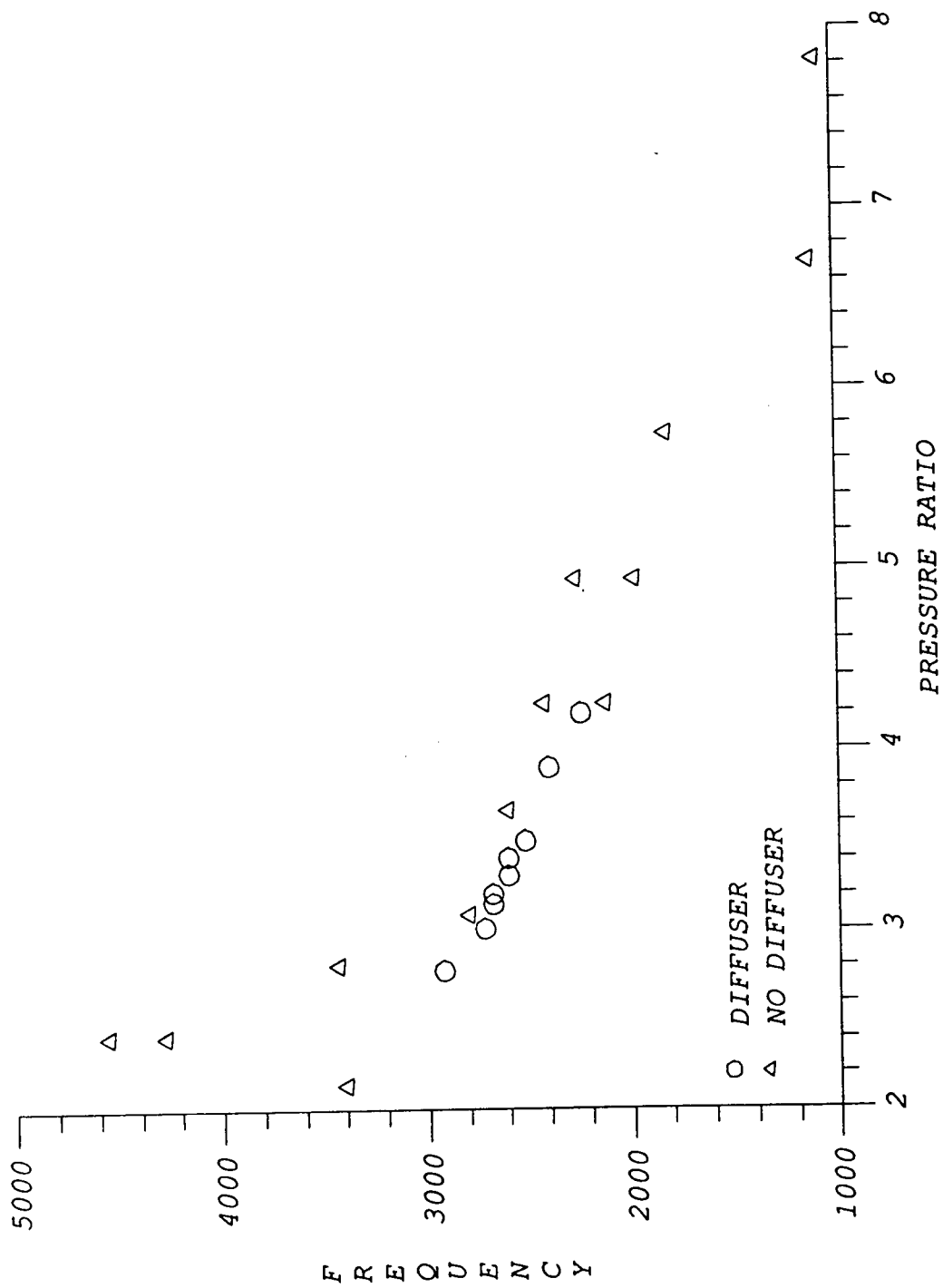


Figure 5-3. Discrete Tone Freq vs Operating PR, Conical Nozzle

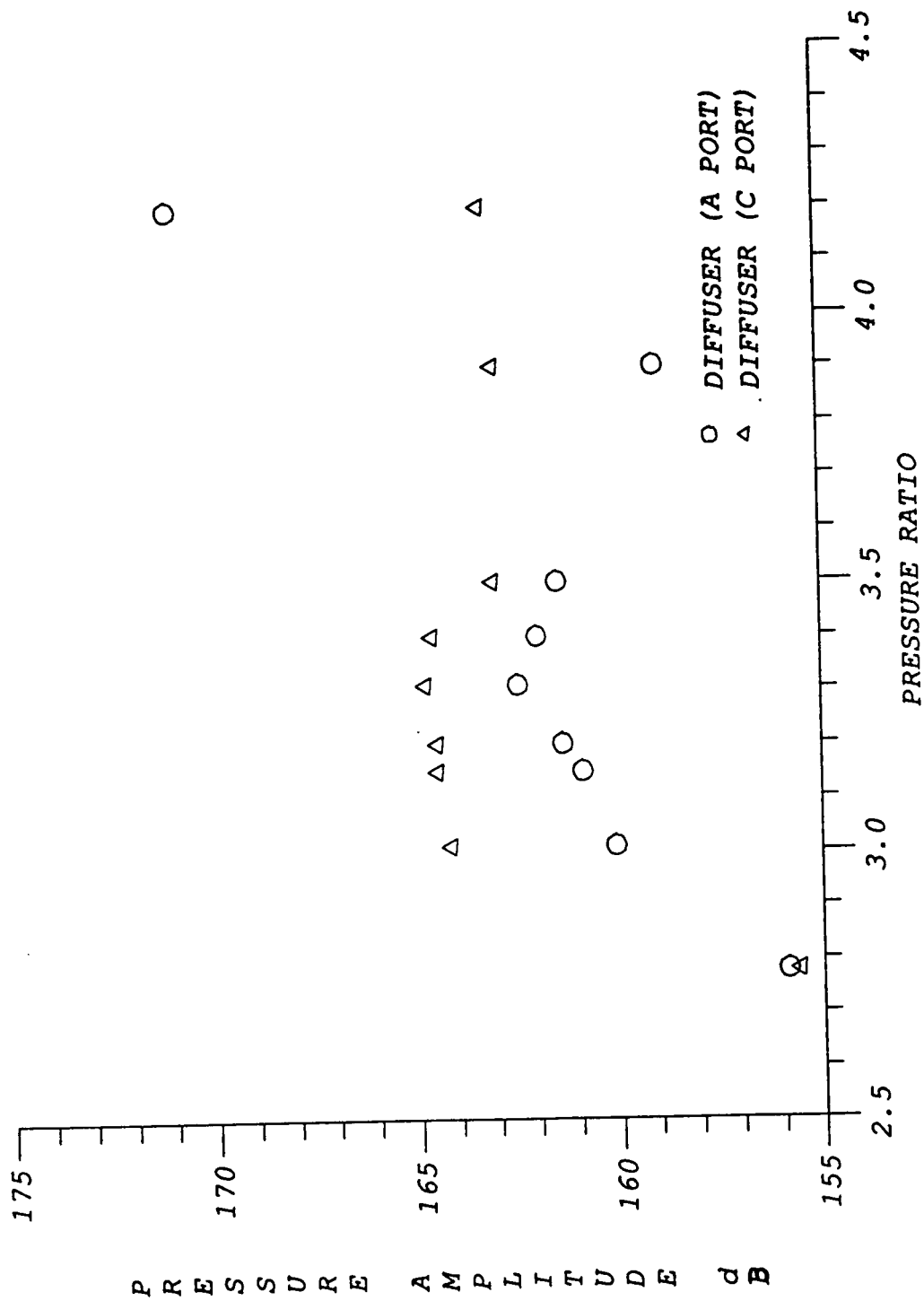


Figure 5-4. Discrete Tone Amp vs Operating PR, Conical Nozzle & Diffuser

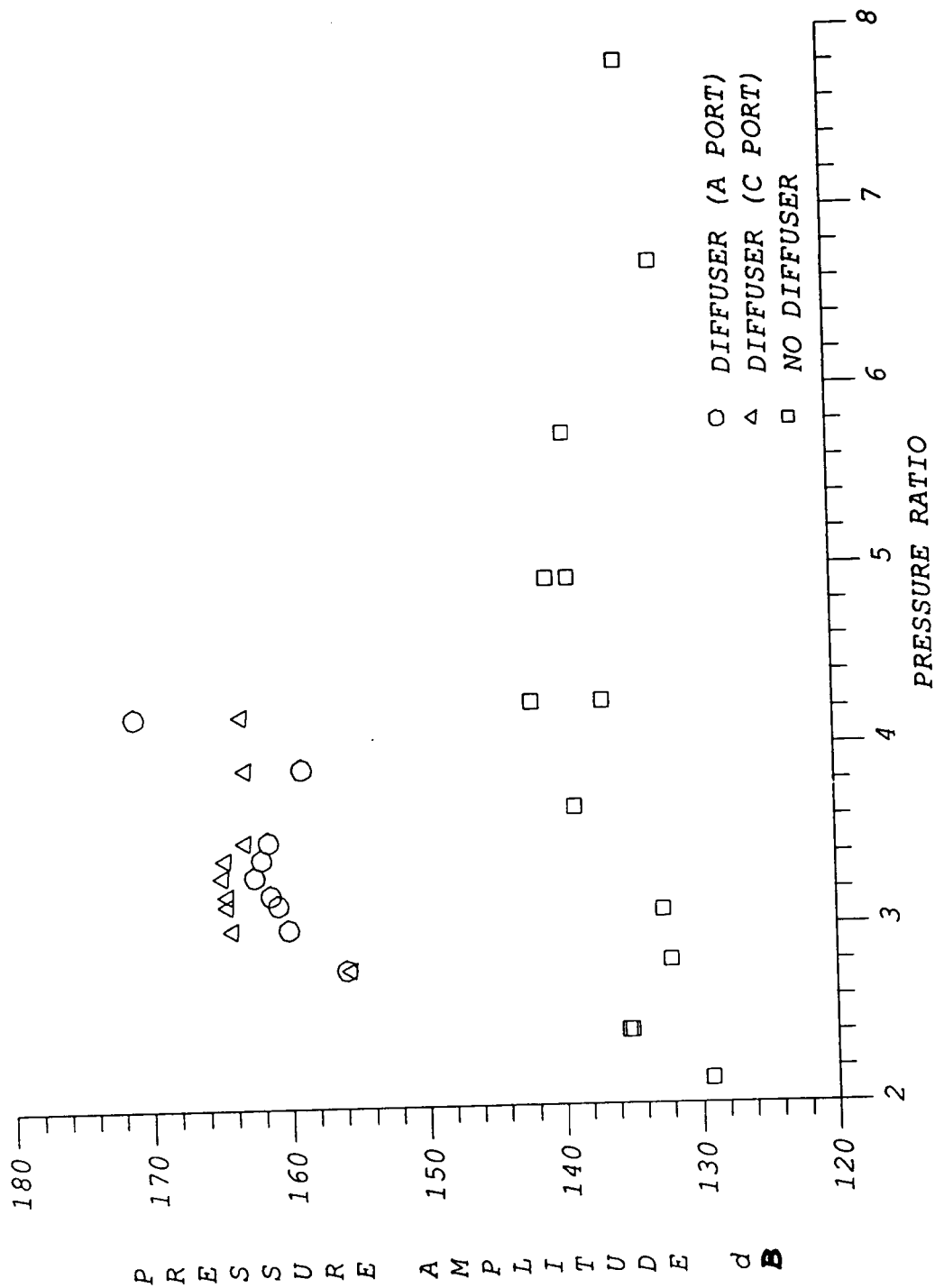


Figure 5-5. Discrete Tone Amp vs Operating PR, Conical Nozzle

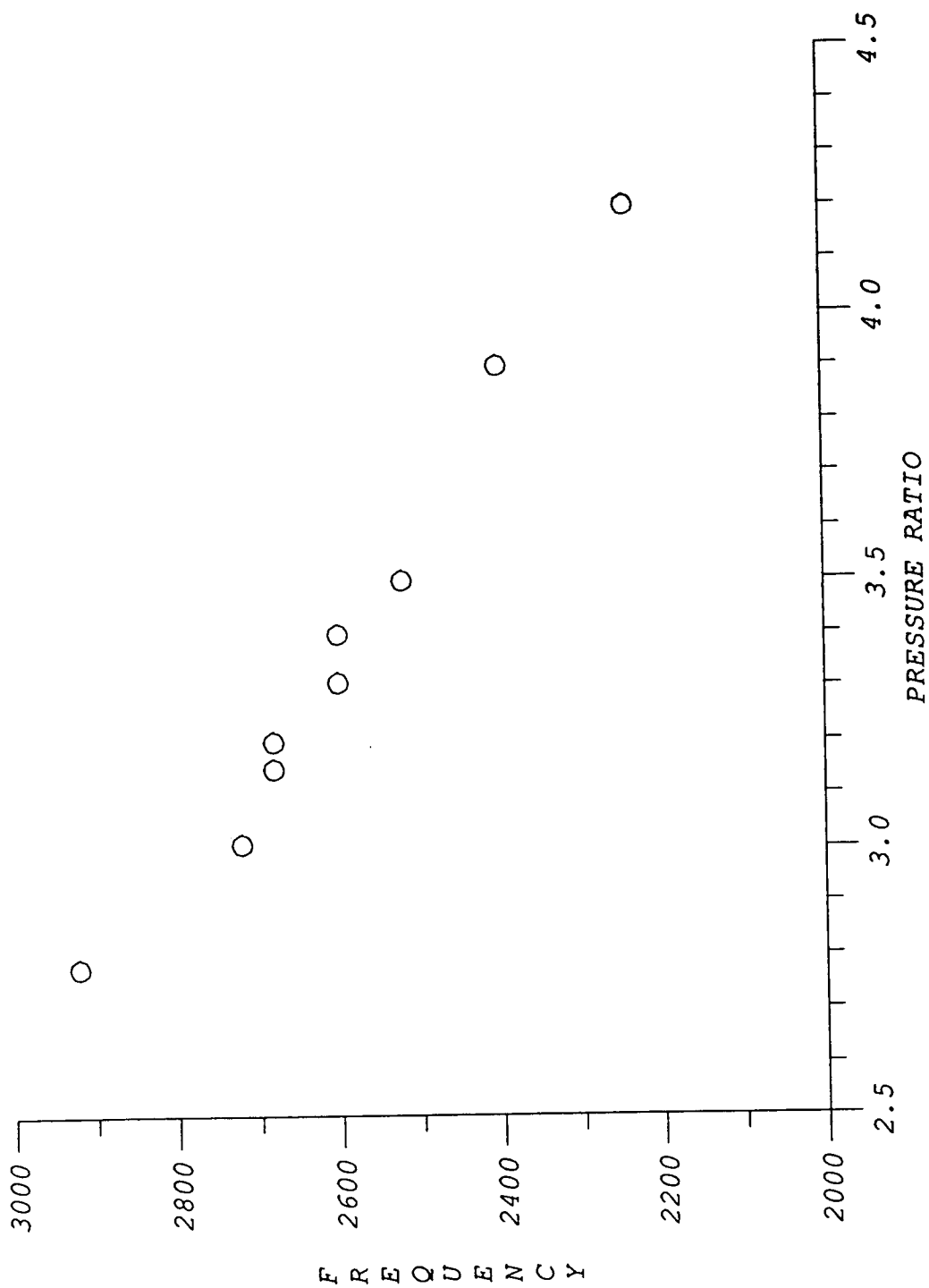


Figure 5-6. Discrete Tone Freq vs Operating PR, Conical Nozzle & Diffuser

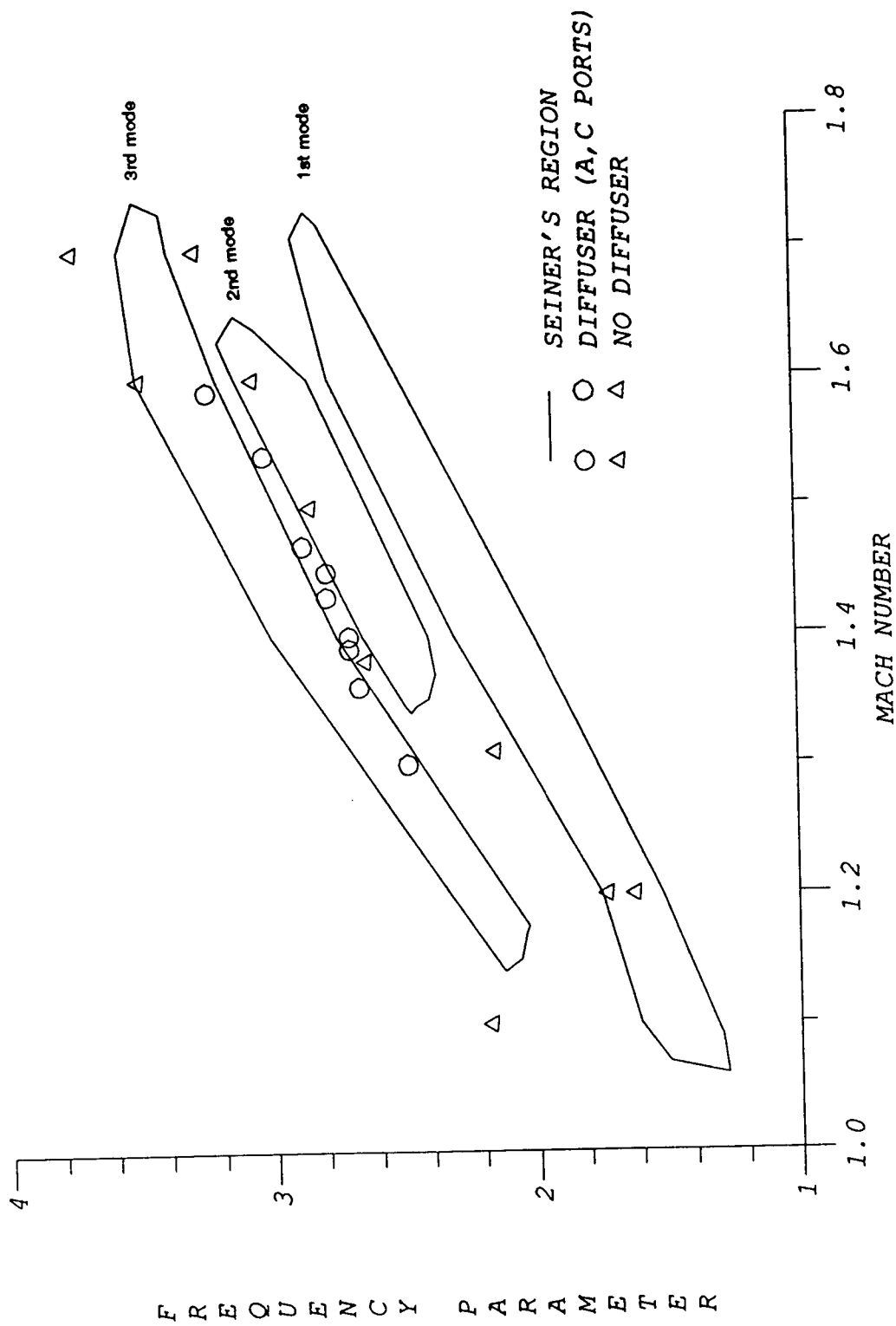


Figure 5-7. D) Comparison with Seiner's Data, Conical Nozzle

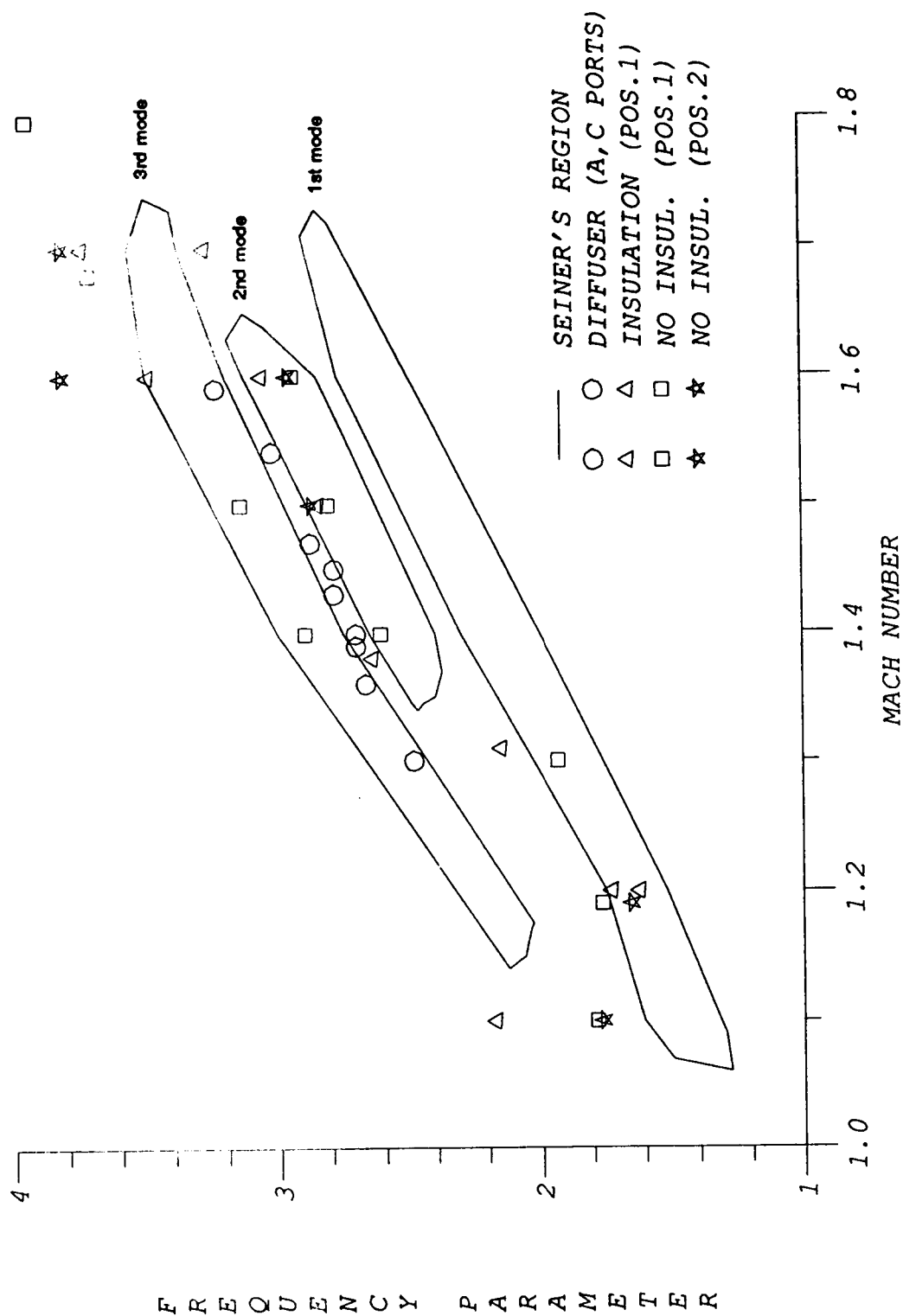


Figure 5-8. II) Comparison with Seiner's Data, Conical Nozzle

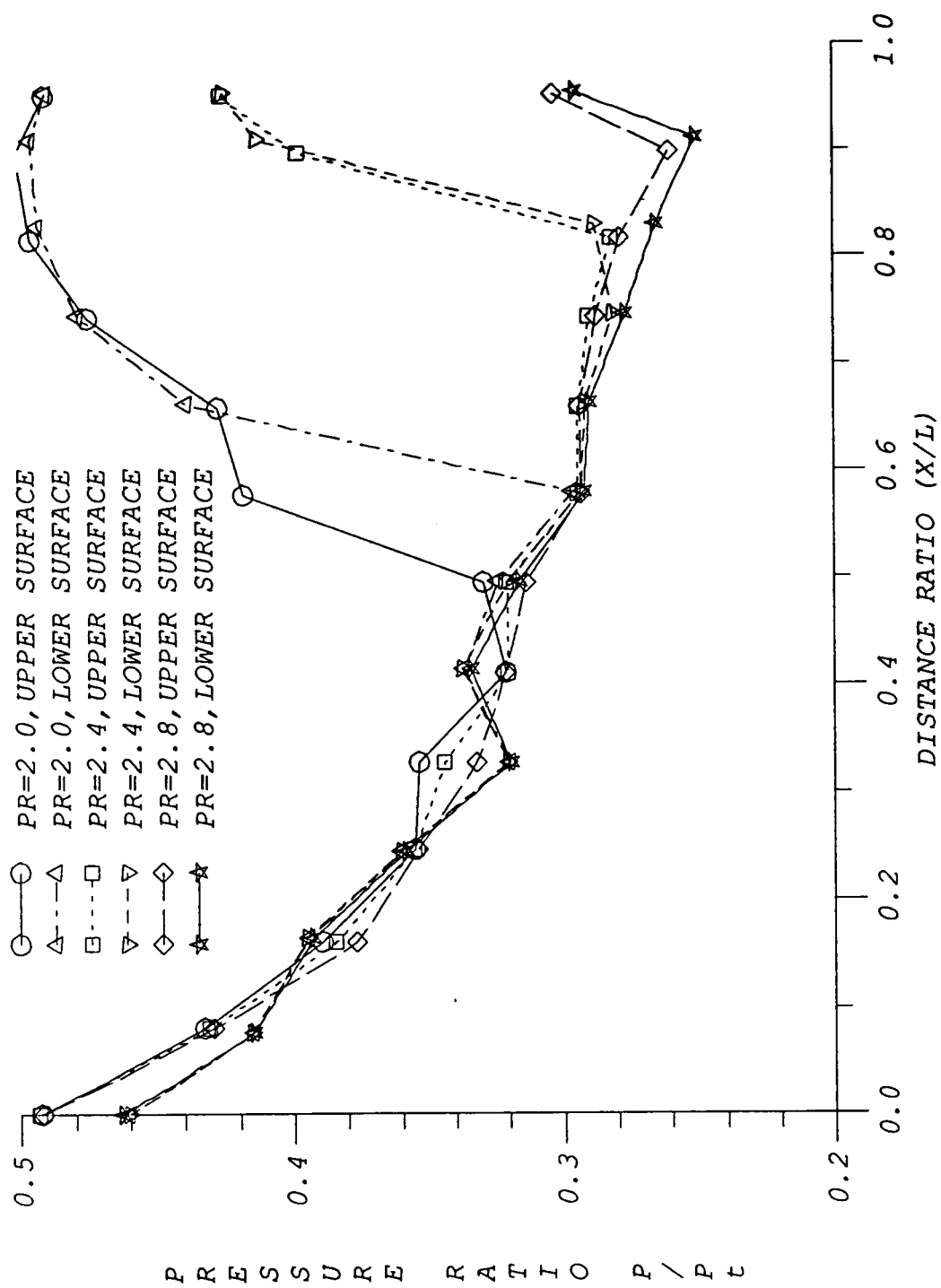


Figure 5-9. Local P/P_t vs X/L for RN1 (AR=1.2)

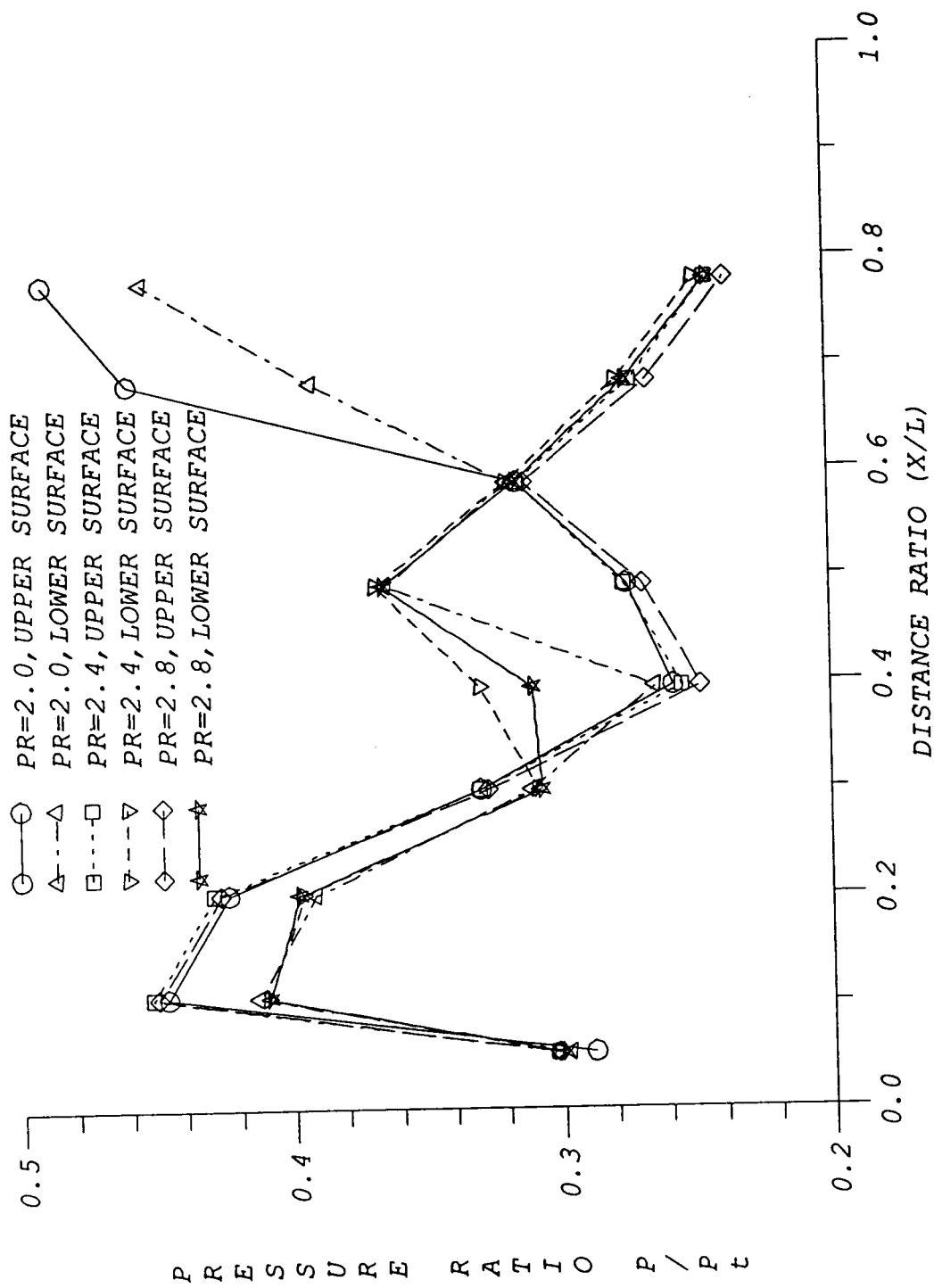


Figure 5-10. Local P/P_t vs X/L for RN2 (AR=1.2)

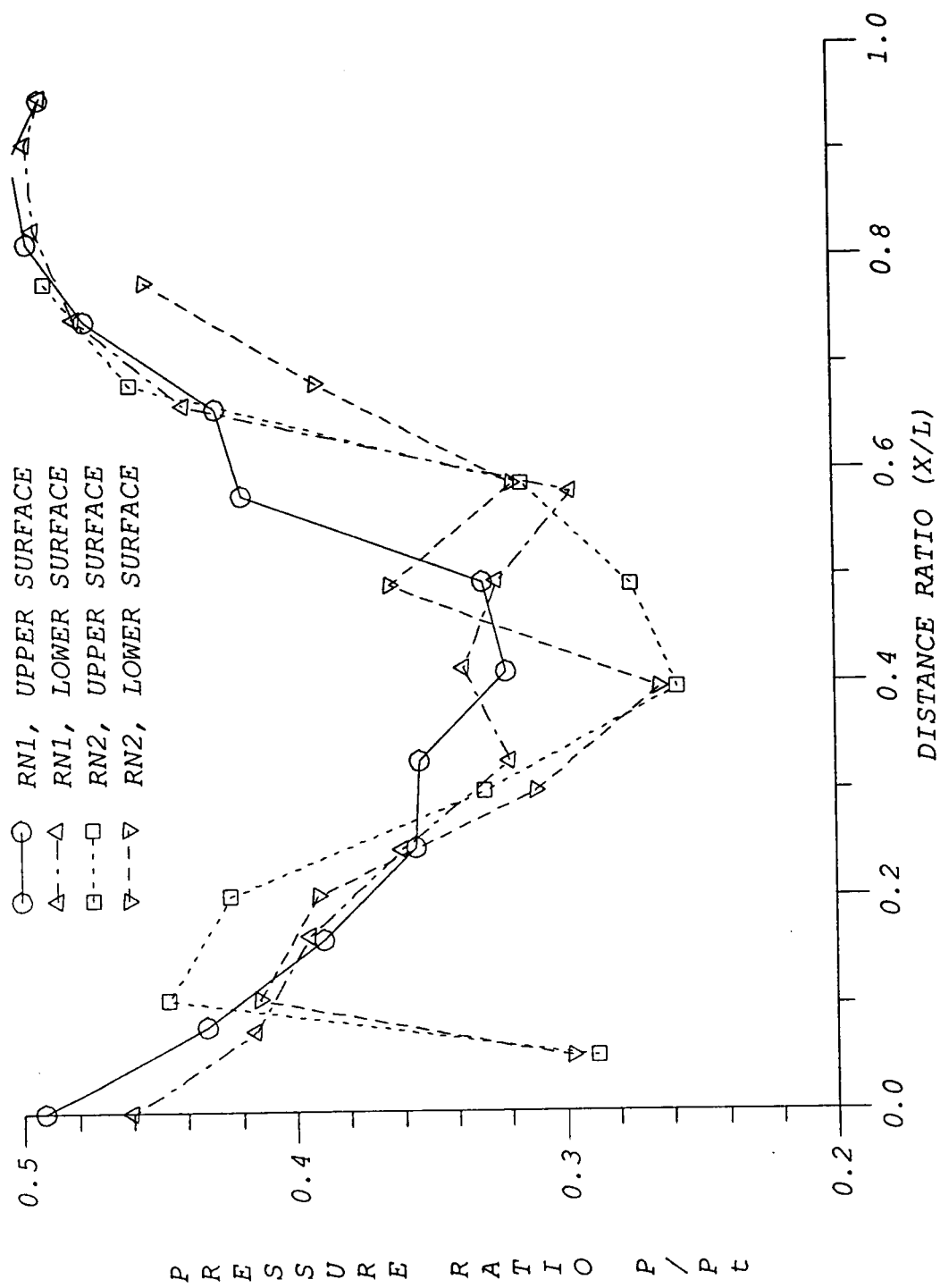


Figure 5-11. Local P/P_t vs X/L for RN1 and RN2 (AR=1.2, PR=2.0)

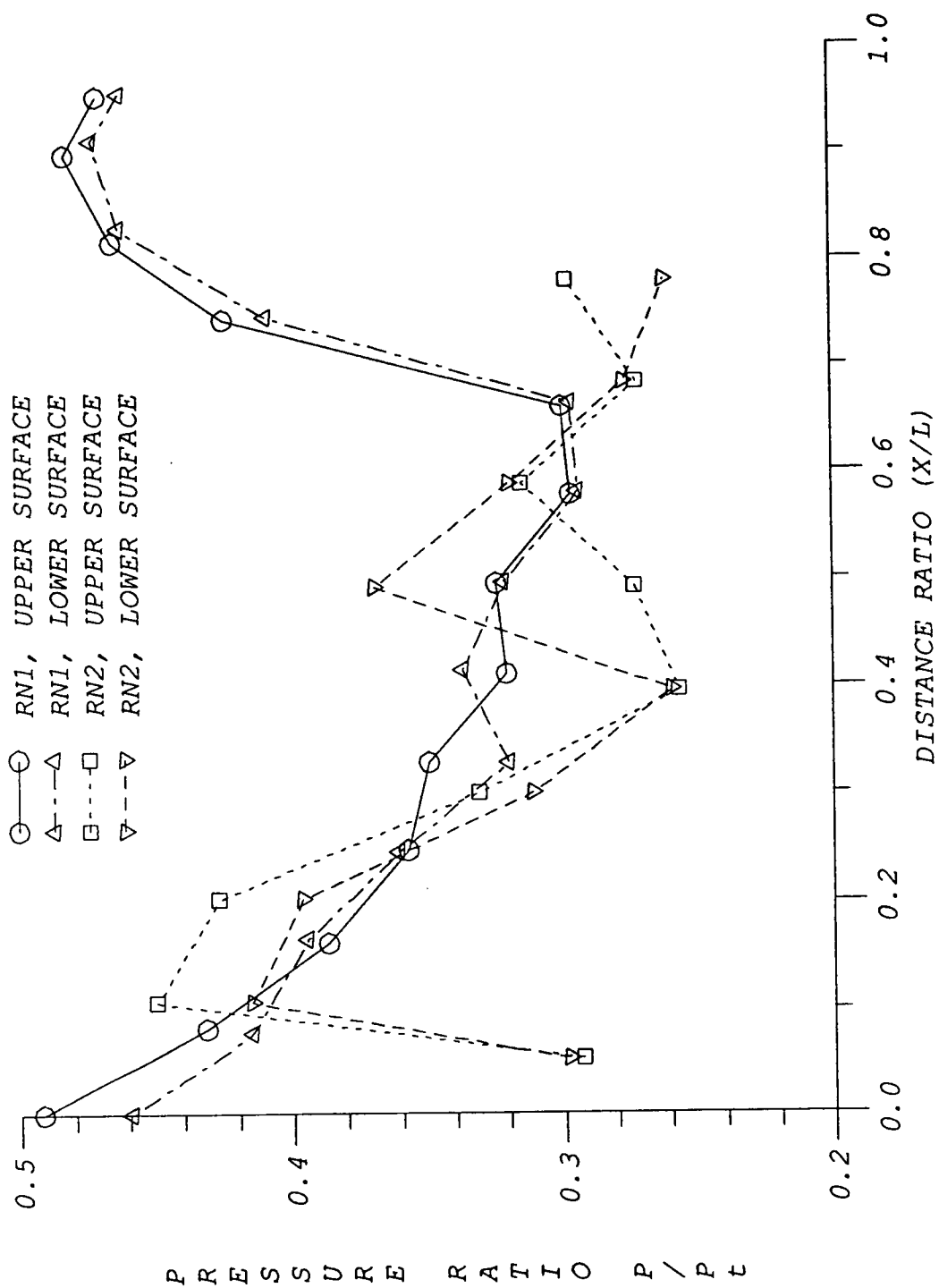


Figure 5-12. Local P/P_t vs X/L for RN1 and RN2 (AR=1.2, PR=2.15)

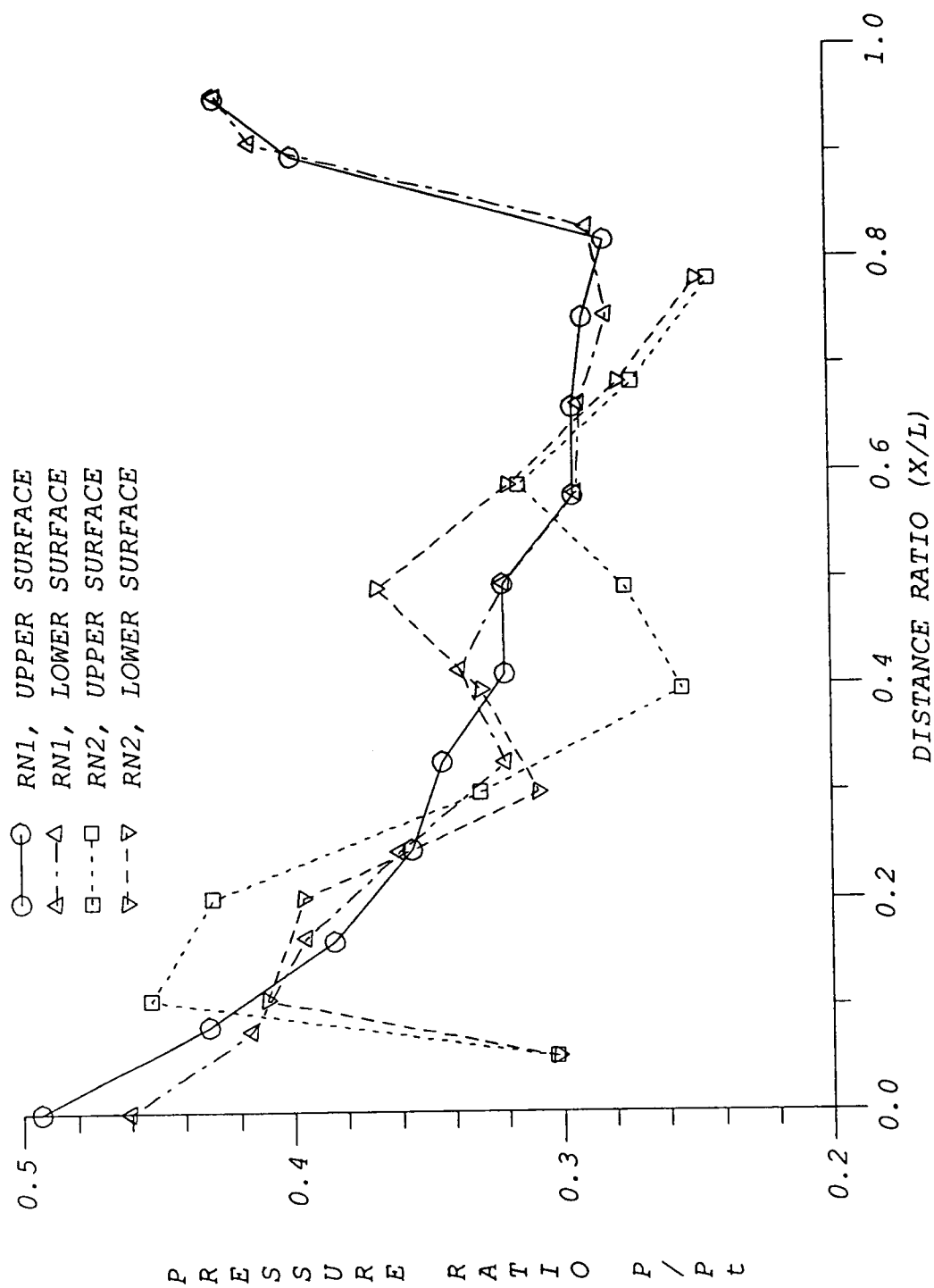


Figure 5-13. Local P/P_t vs X/L for RN1 and RN2 ($AR=1.2$, $PR=2.4$)

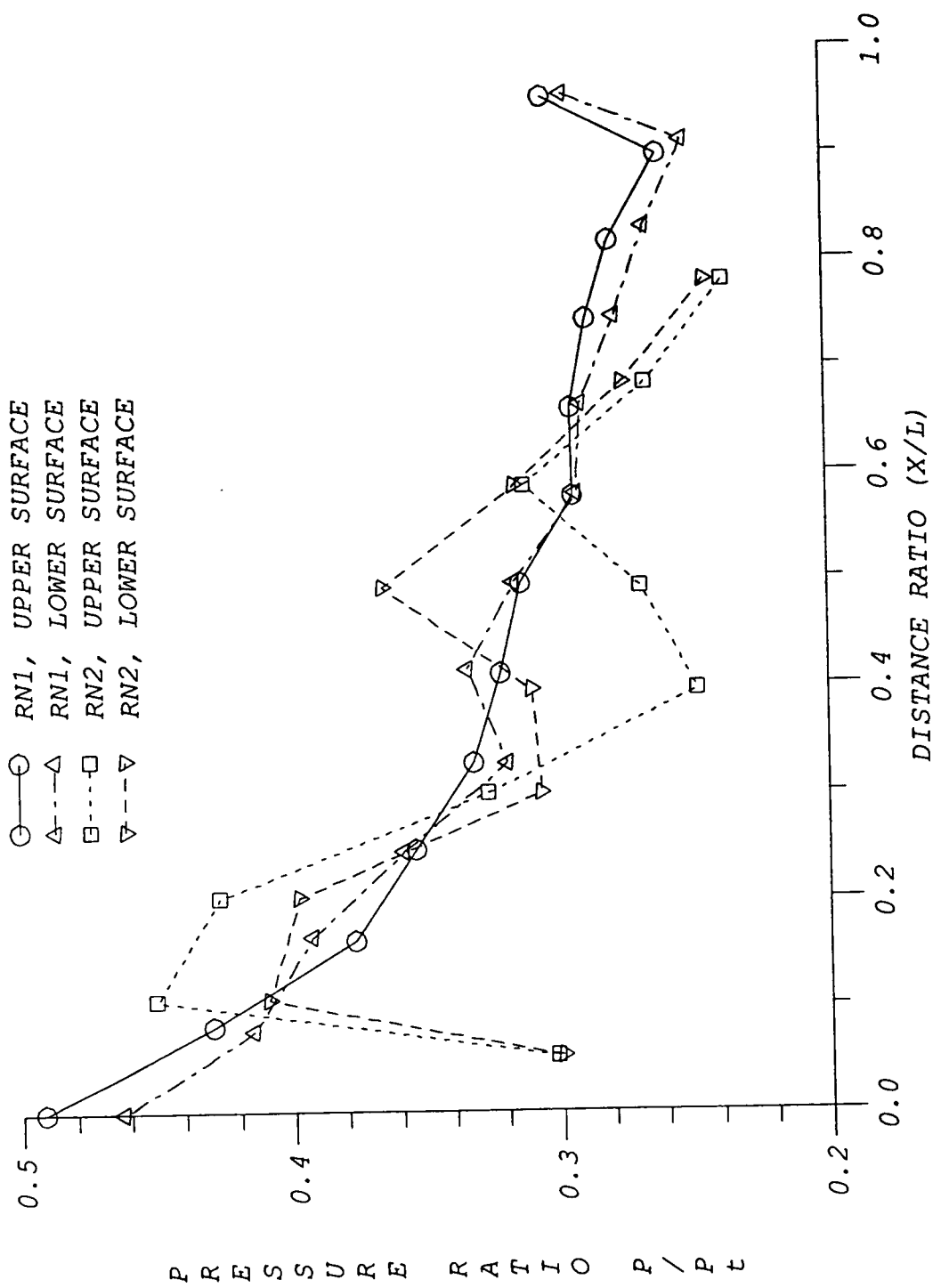


Figure 5-14. Local P/P_t vs X/L for RN1 and RN2 ($AR=1.2$, $PR=2.8$)

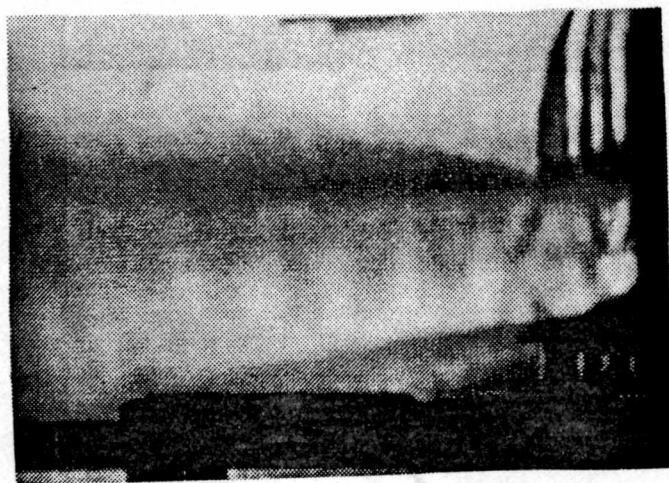


Figure 5-15. RN1 Schlieren (PR=2.15)

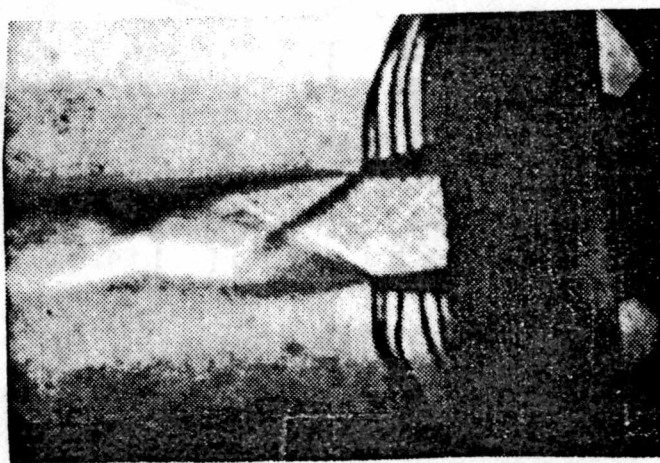


Figure 5-16. RN1 Schlieren (PR=3.18)

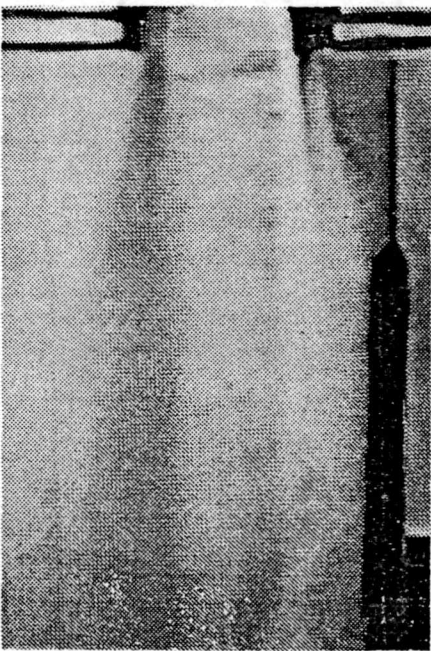


Figure 5-17. RN2 Schlieren (PR=2.0)

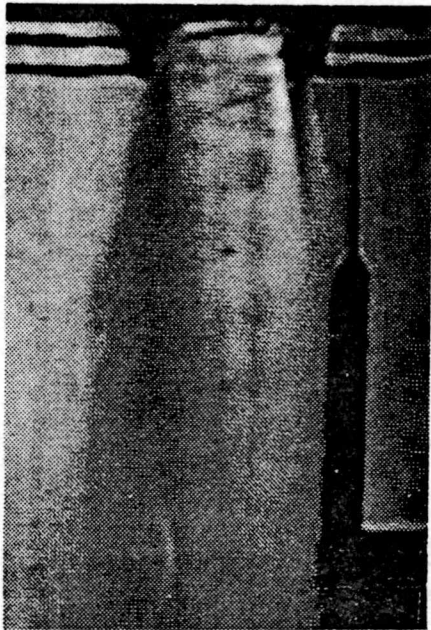


Figure 5-18. RN2 Schlieren (PR=2.15)



Figure 5-19. RN2 Schlieren (PR=2.3)

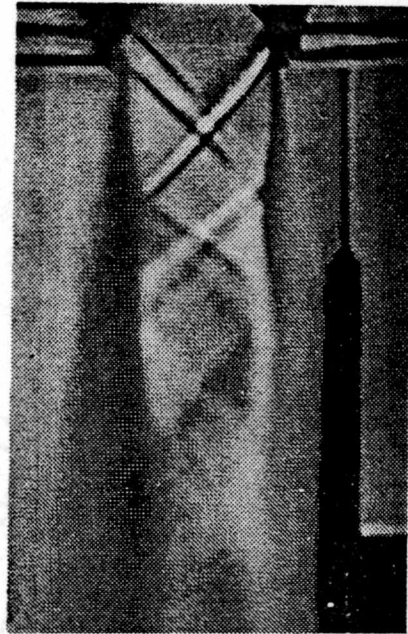


Figure 5-20. RN2 Schlieren (PR=3.5)

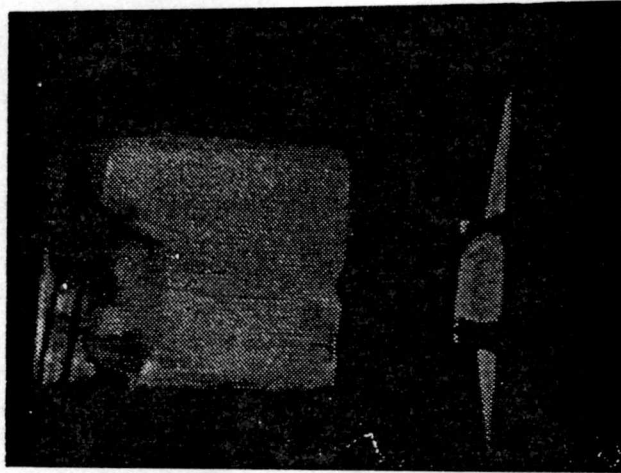


Figure 5-21. RN2 & Diffuser (PR=2.15)

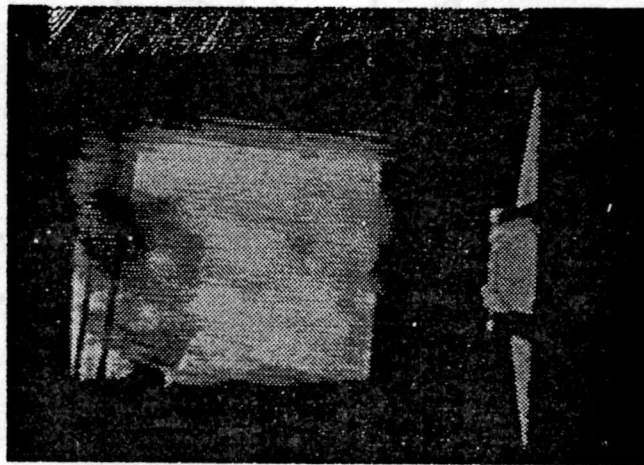


Figure 5-22. RN2 & Diffuser (PR=2.5)

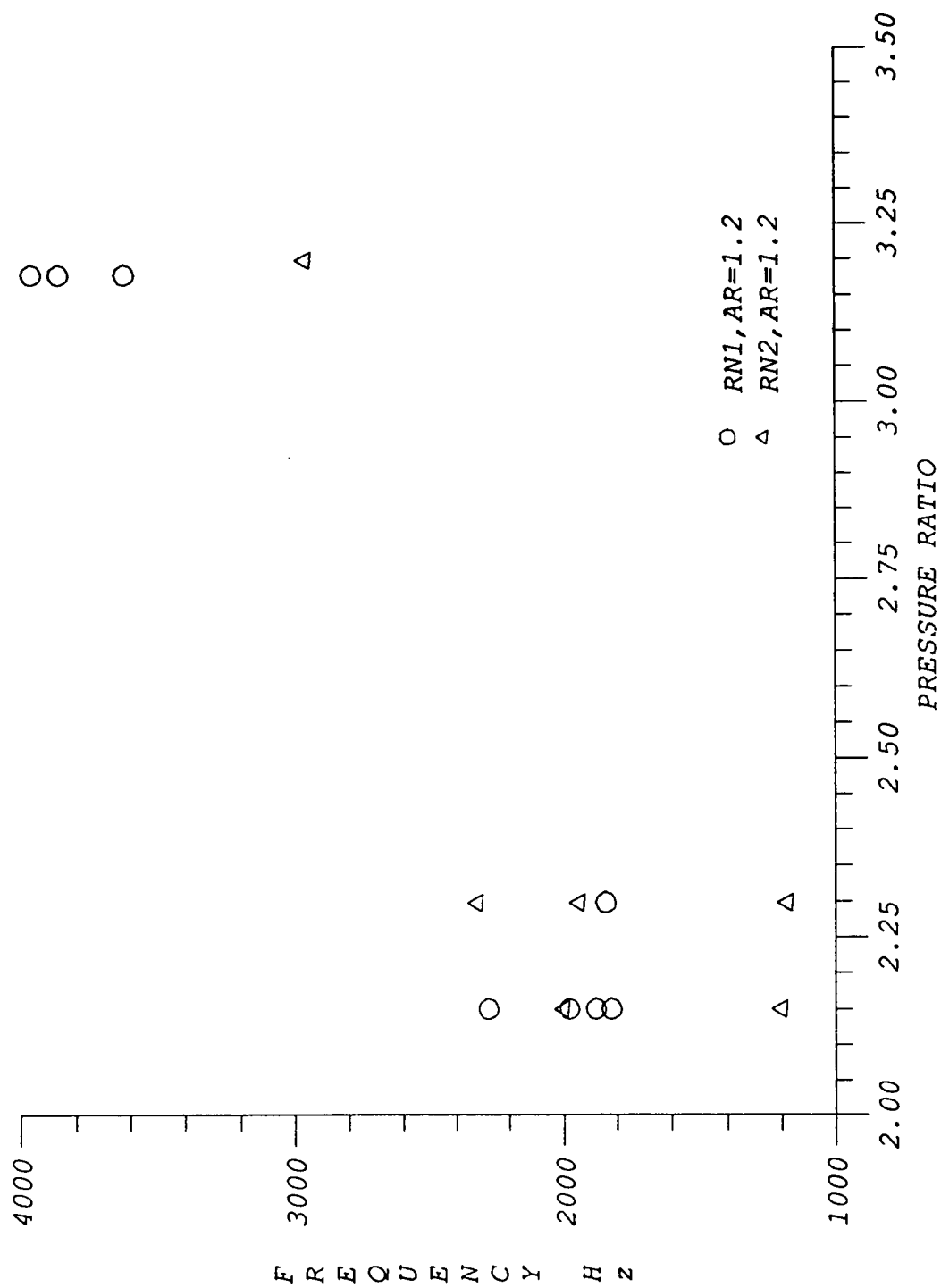


Figure 5-23. Discrete Tone Freq vs Operating PR, RN1 & RN2, No Diffuser

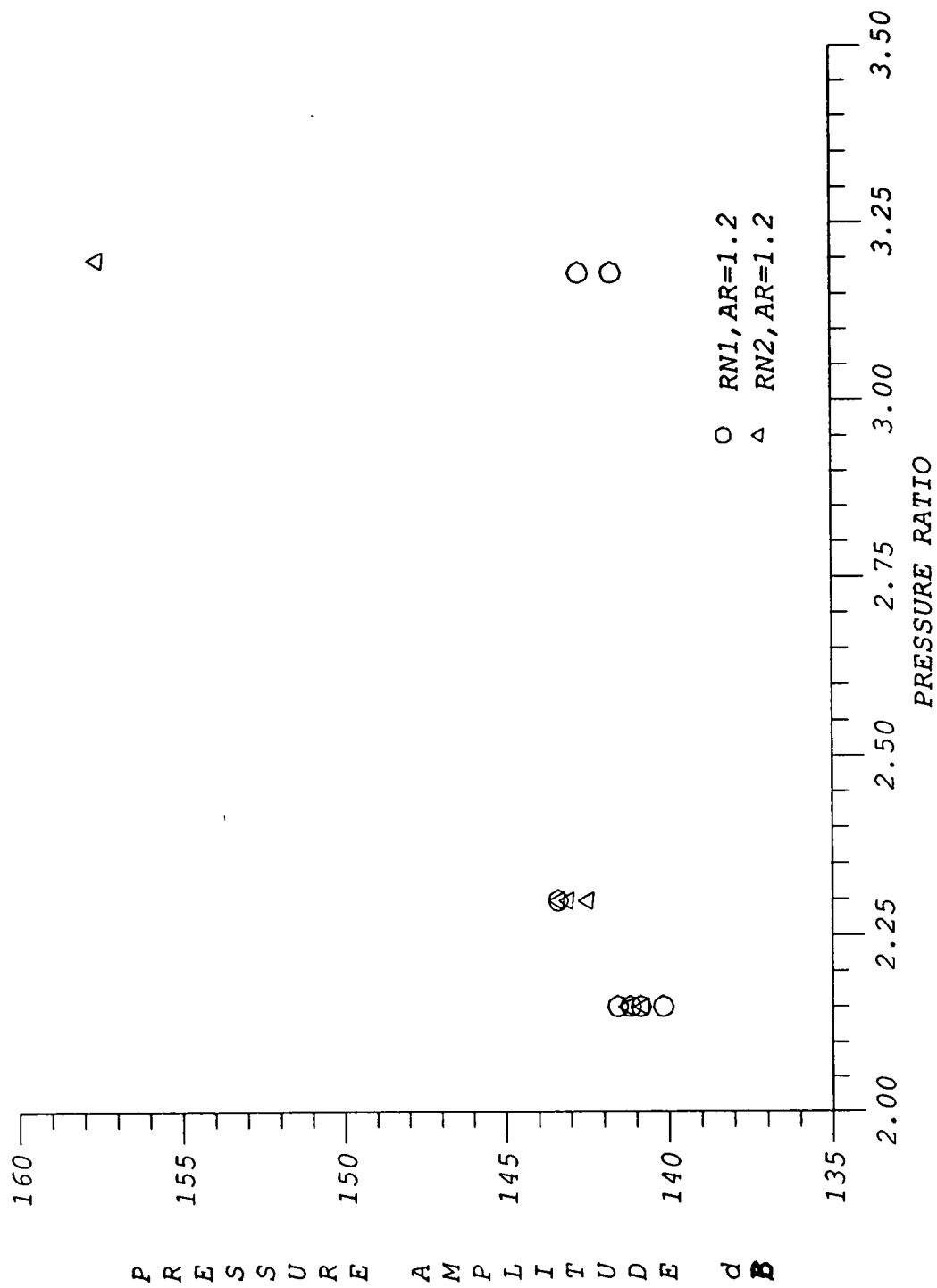


Figure 5-24. Discrete Tone Amp vs Operating PR, RN1 & RN2, No Diffuser

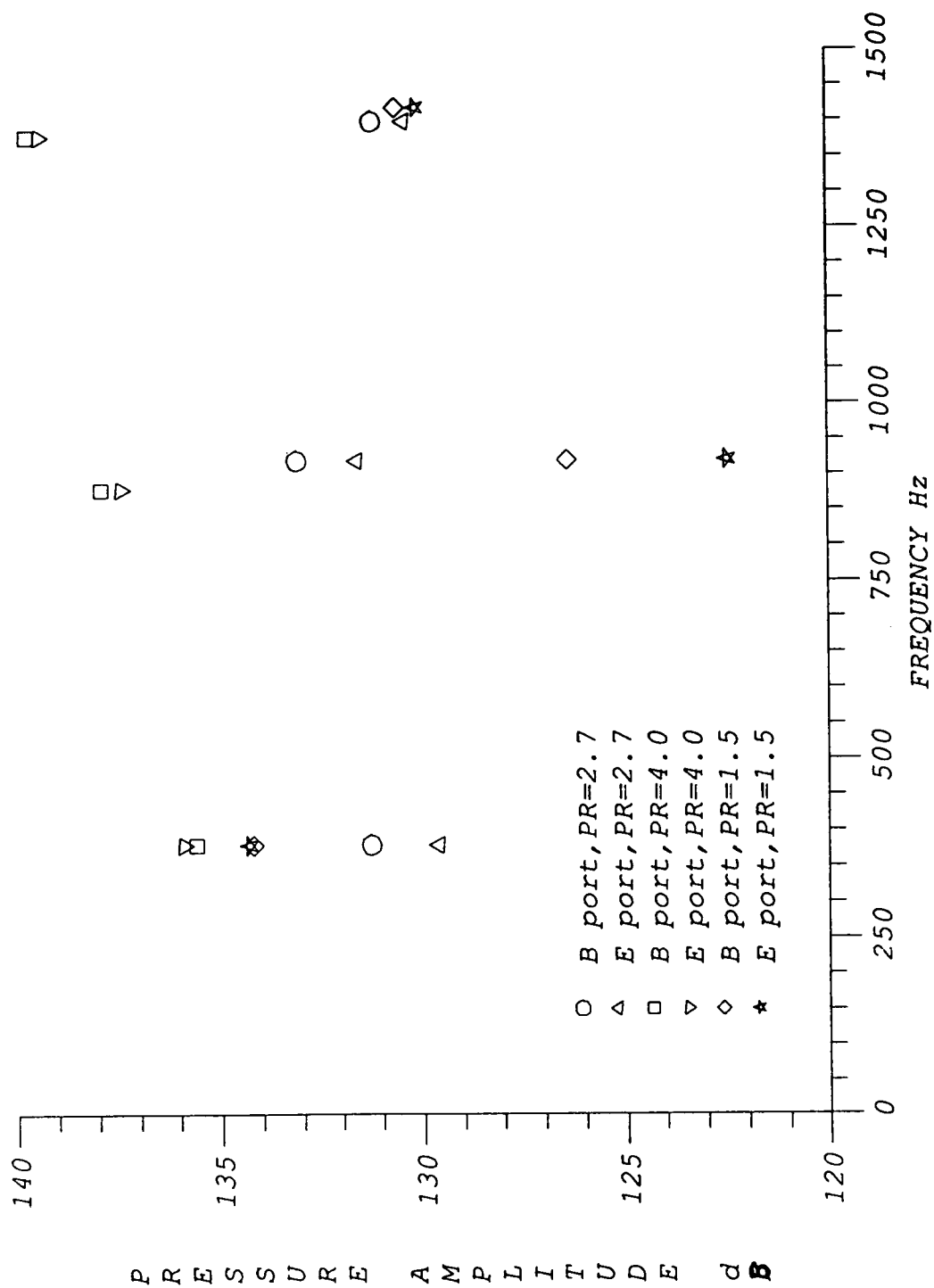


Figure 5-25. Spectral Peaks for RN1 (AR=1.2)

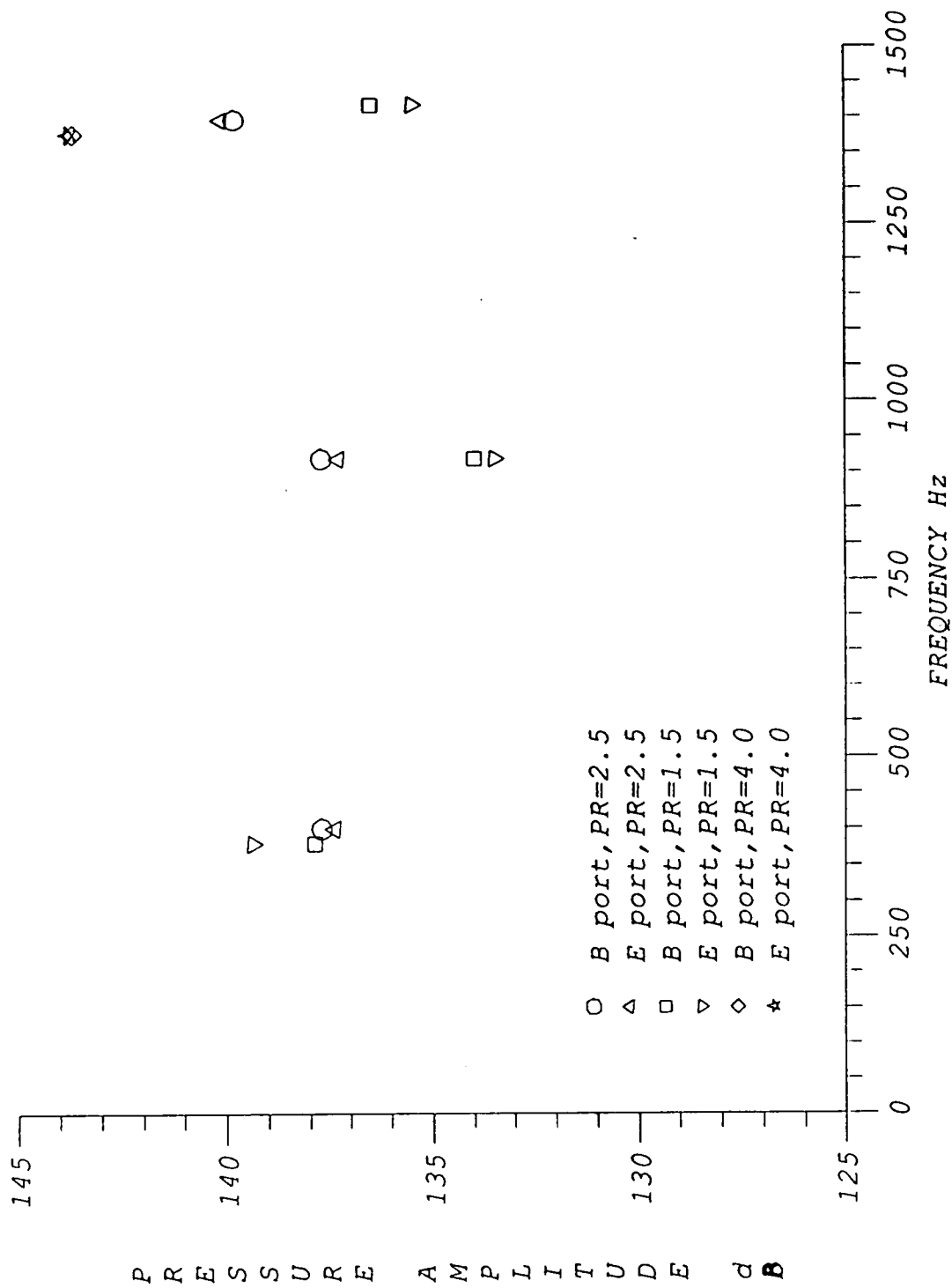


Figure 5-26. Spectral Peaks for RN1 (AR=1.5)

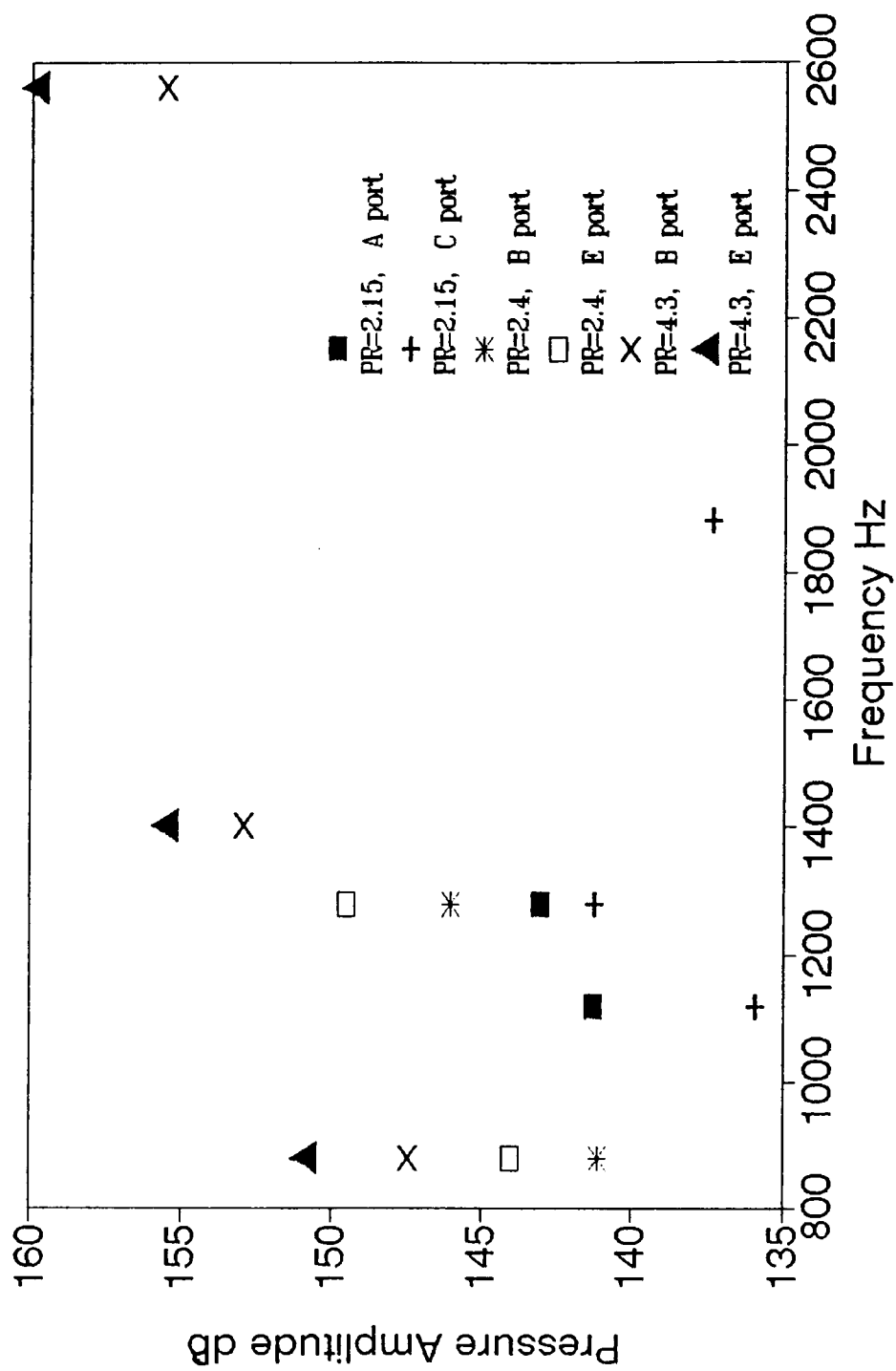


Figure 5-27. Spectral Peaks for RN2 (AR=1.2)

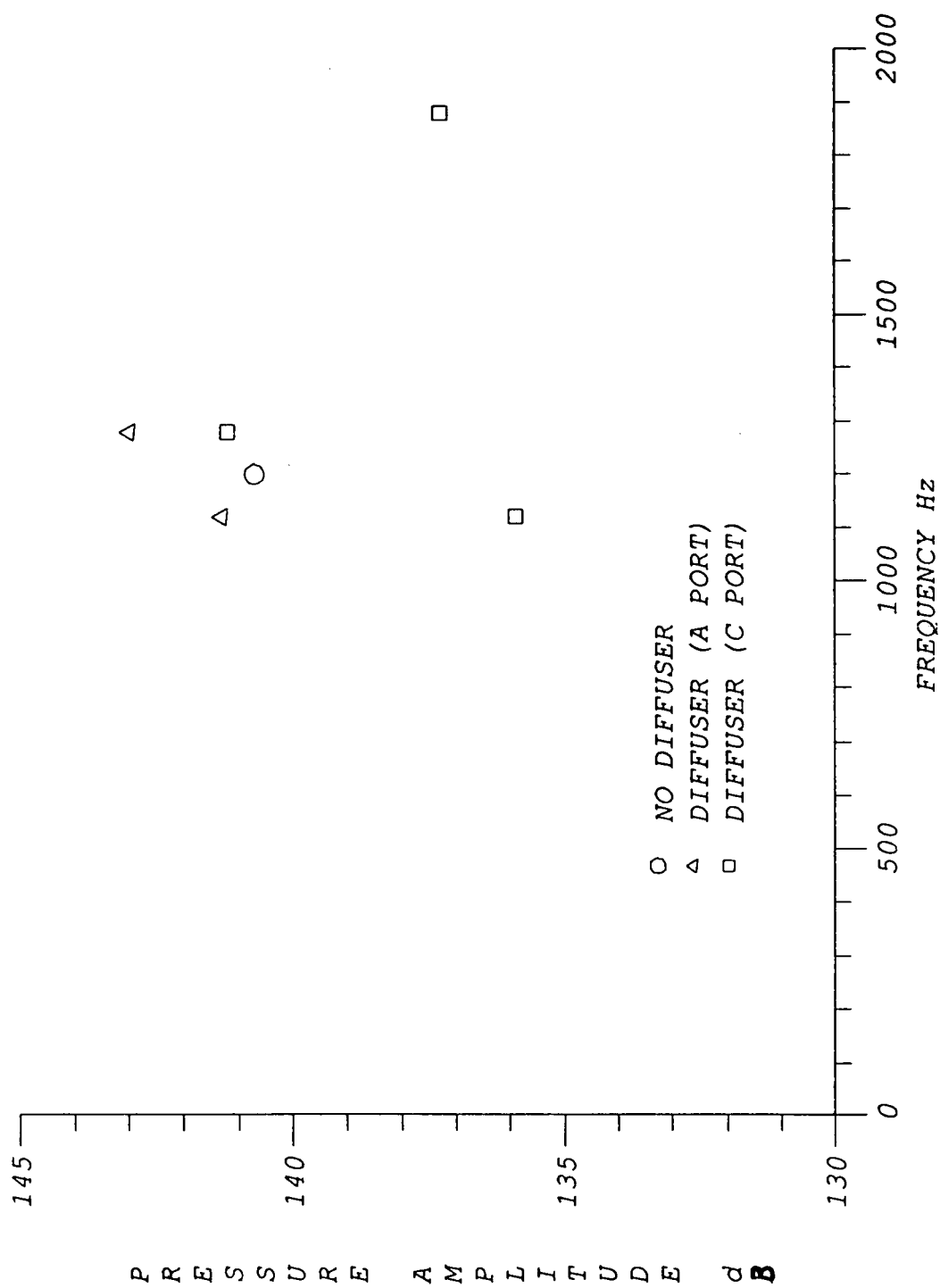


Figure 5-28. Effect of Diffuser on RN2 (AR=1.2, PR=2.15)

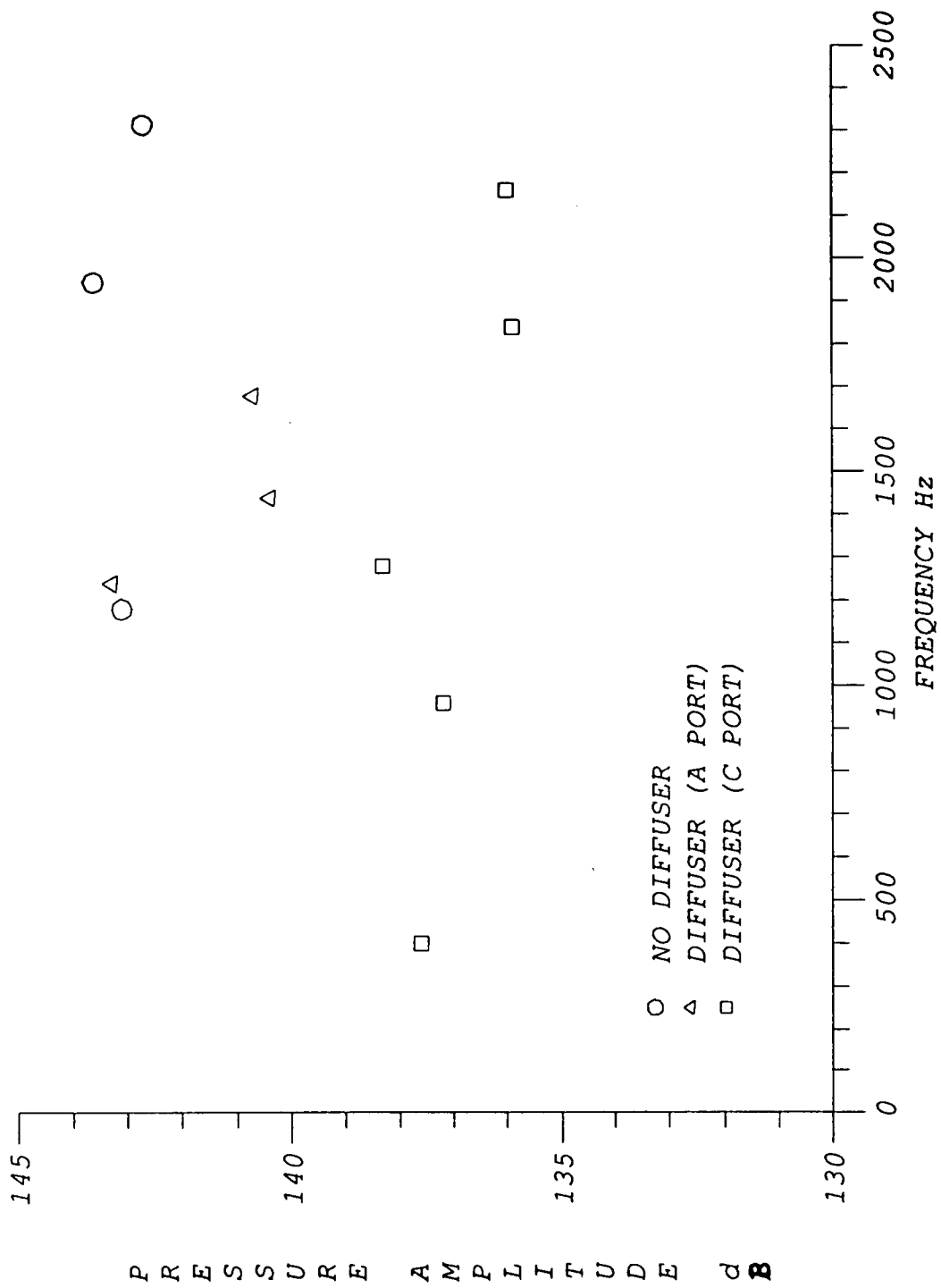


Figure 5-29. Effect of Diffuser on RN2 (AR=1.2, PR=2.3)

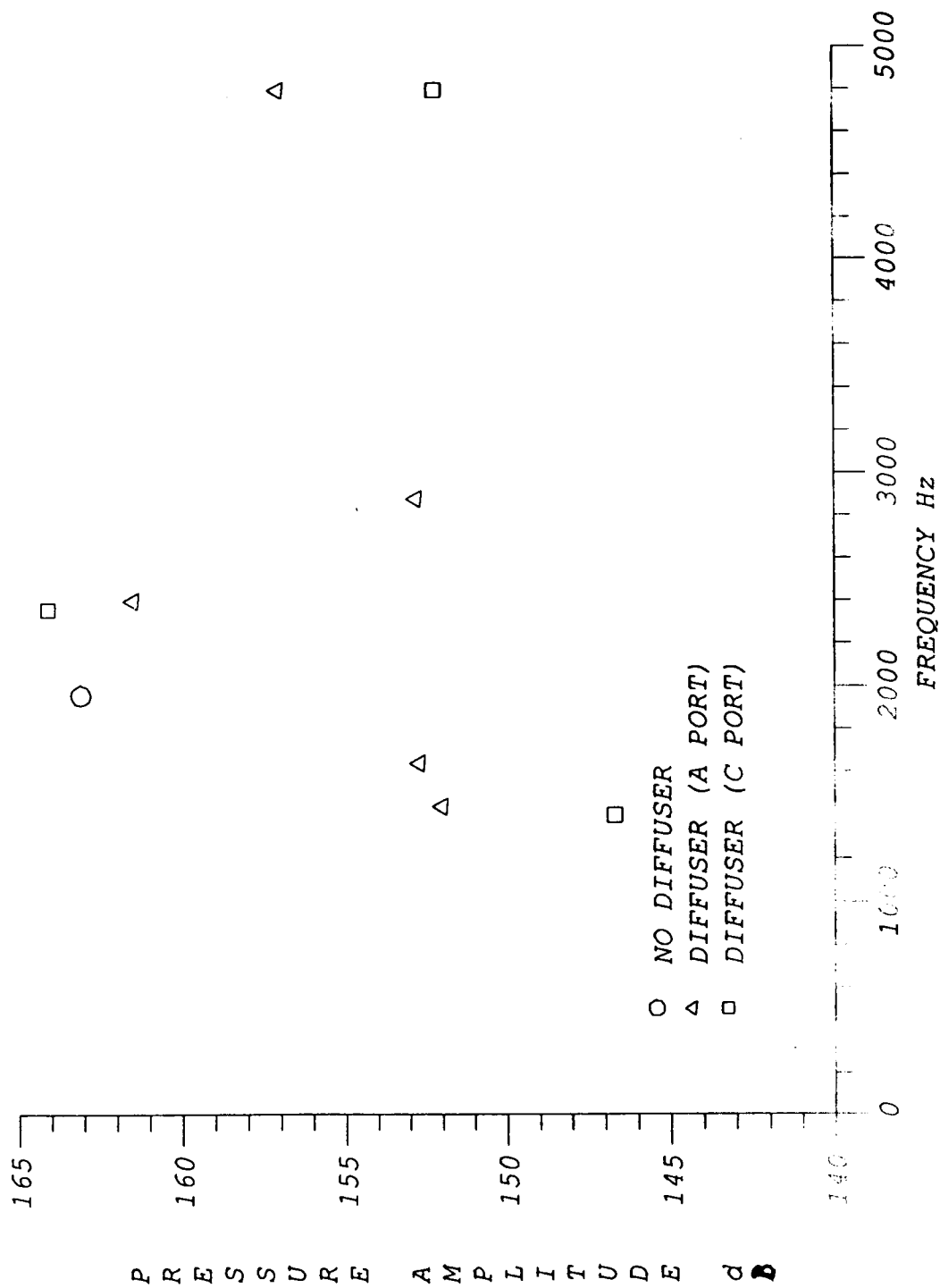


Figure 5-30. Effect of Diffuser on RN2 (AR=1.2, PR=5.1)

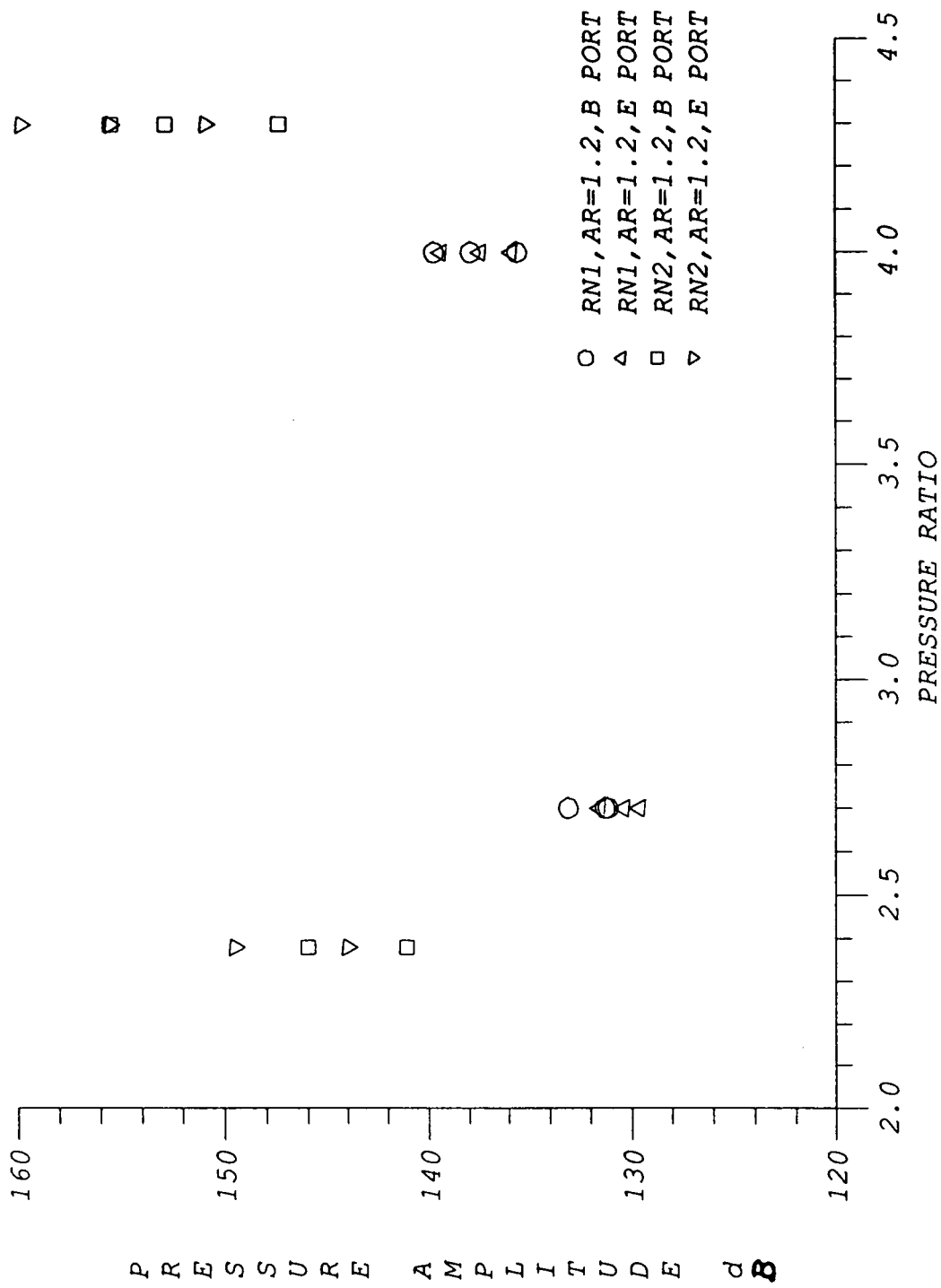


Figure 5-31. Discrete Tone Amp vs Operating PR, RN1 & RN2, with Diffuser

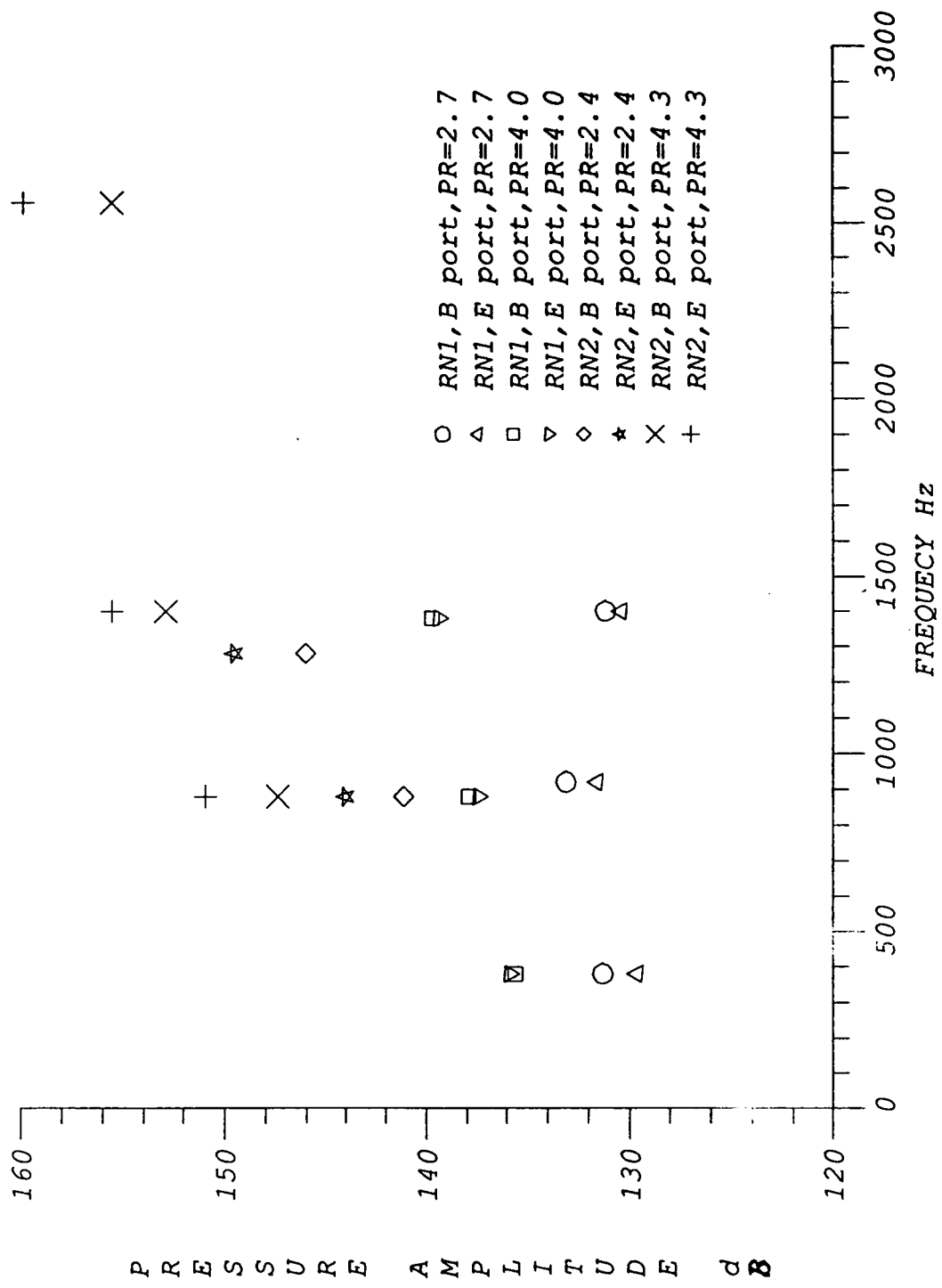


Figure 5-32. Comparison of Spectral Peaks for RN1 & RN2 (AR=1.2)

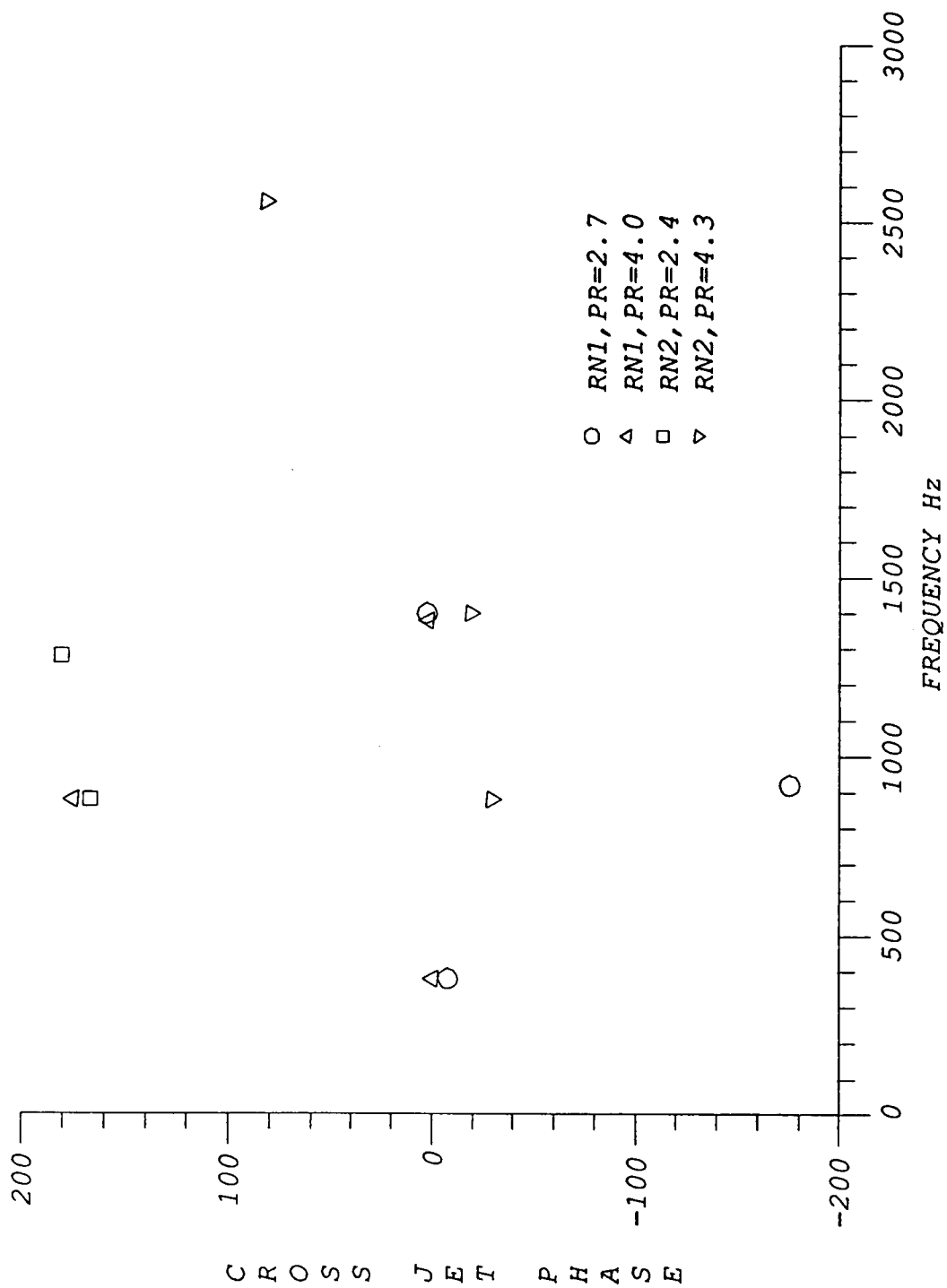


Figure 5-33. Cross Jet Phase Angles for RN1 & RN2 (AR=1.2)

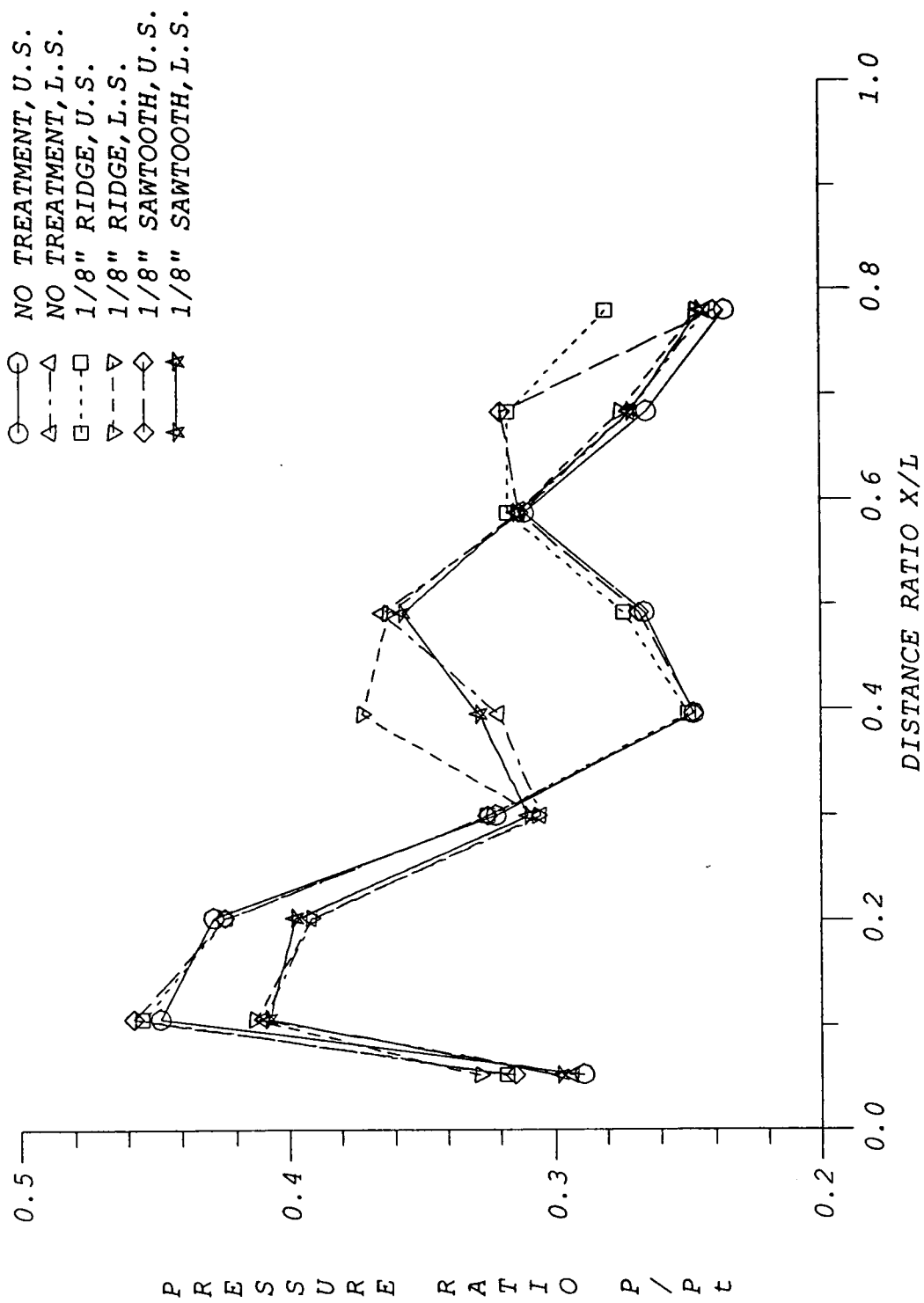


Figure 5-34. Effect of Upstream Boundary Layer Treatments (AR=1.2, PR=2.35)

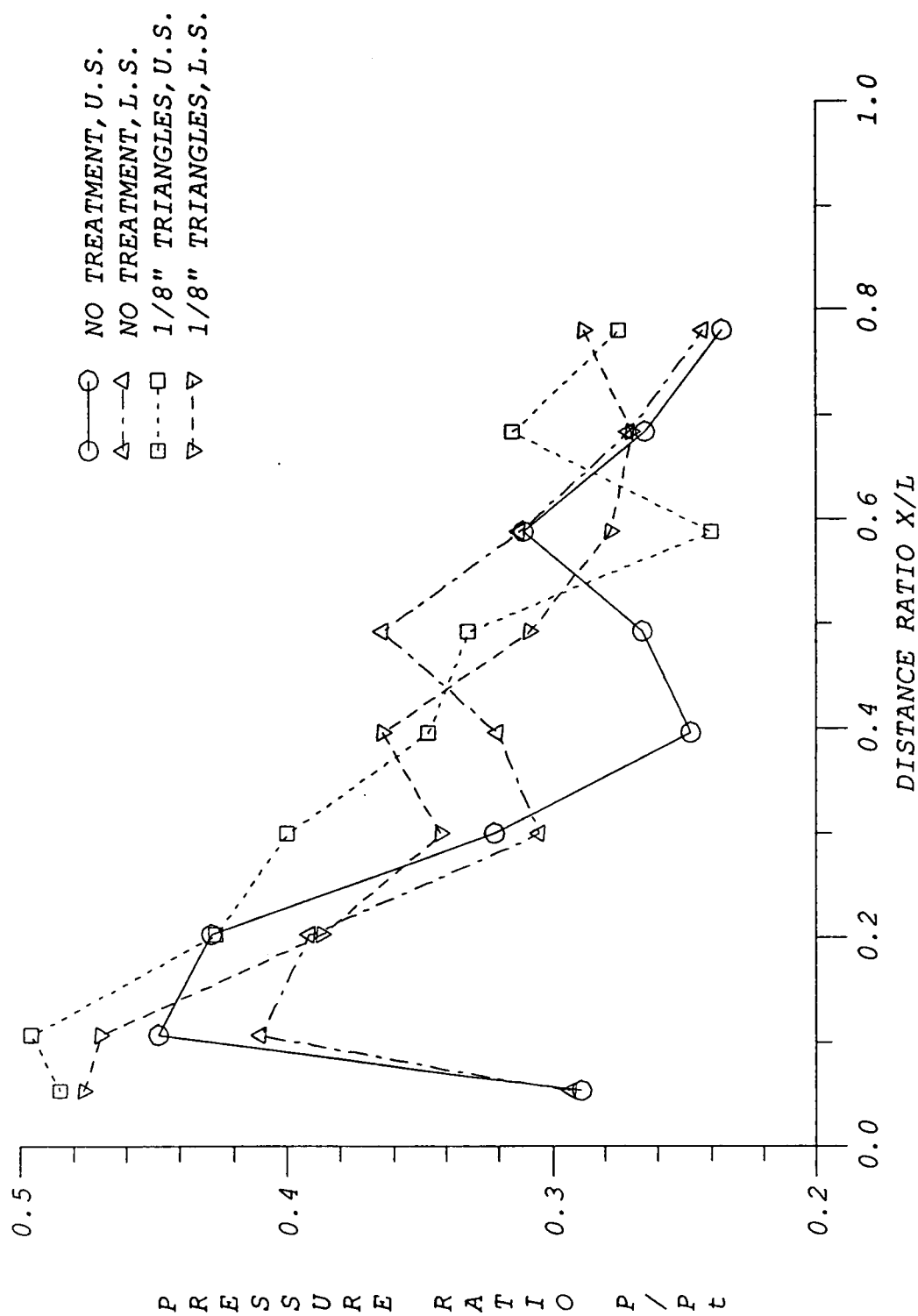


Figure 5-35. D) Effect of Downstream Boundary Layer Treatments (AR=1.2, PR=2.35)

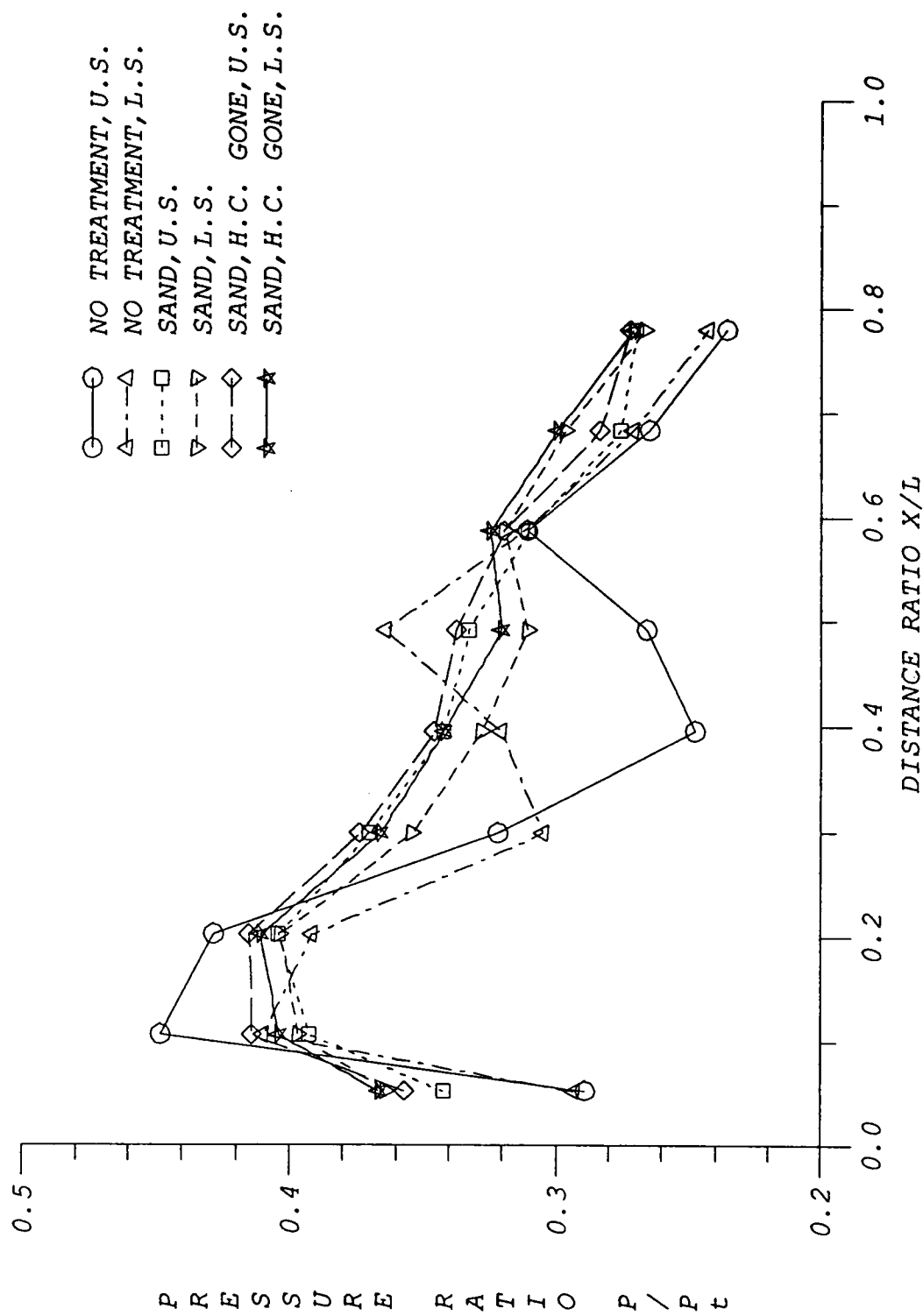


Figure 5-36. II) Effect of Downstream Boundary Layer Treatments ($AR=1.2$, $PR=2.35$)

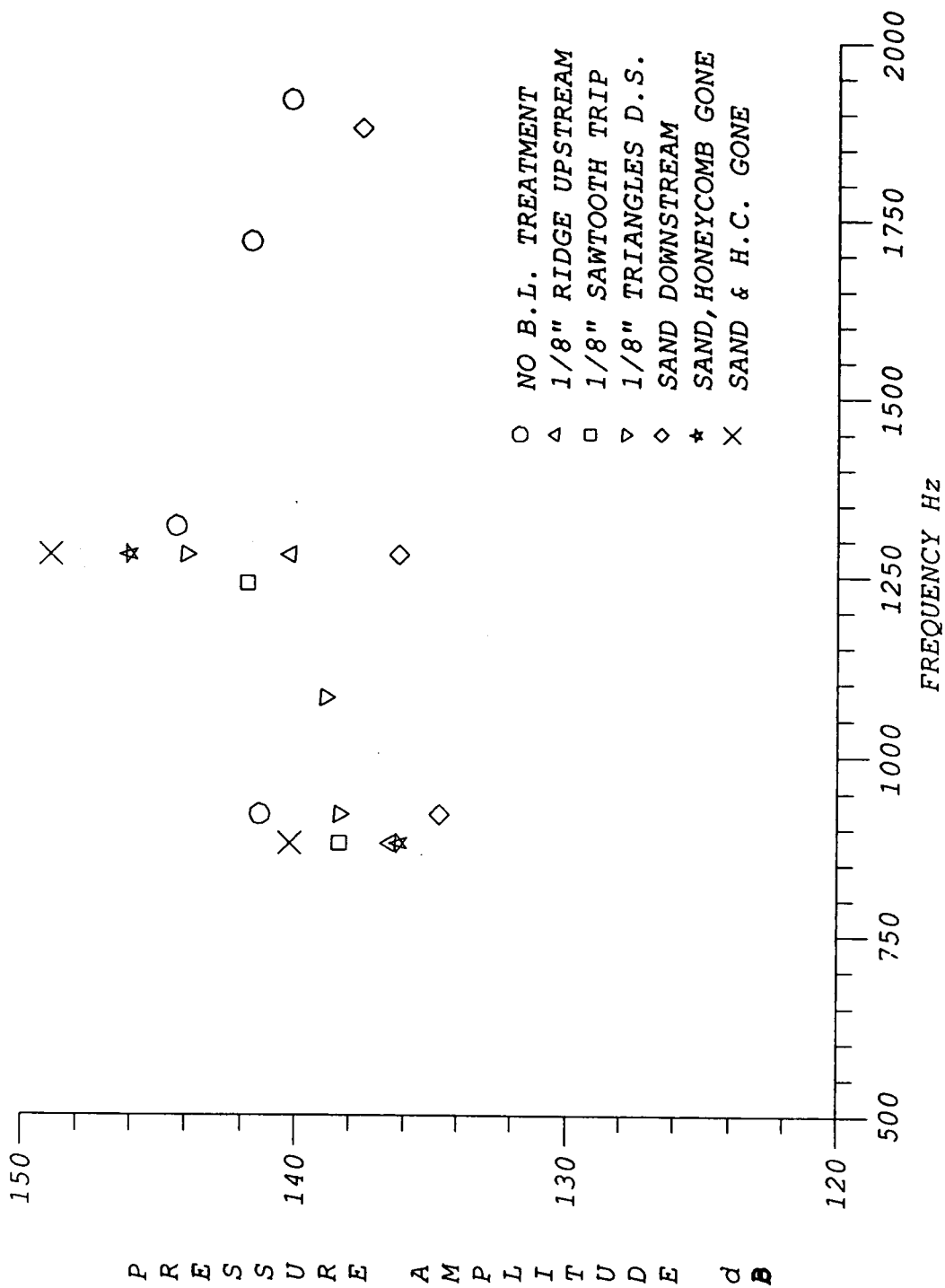


Figure 5-37. Boundary Layer Effects on Spectral Peaks (AR=1.2, PR=2.35)

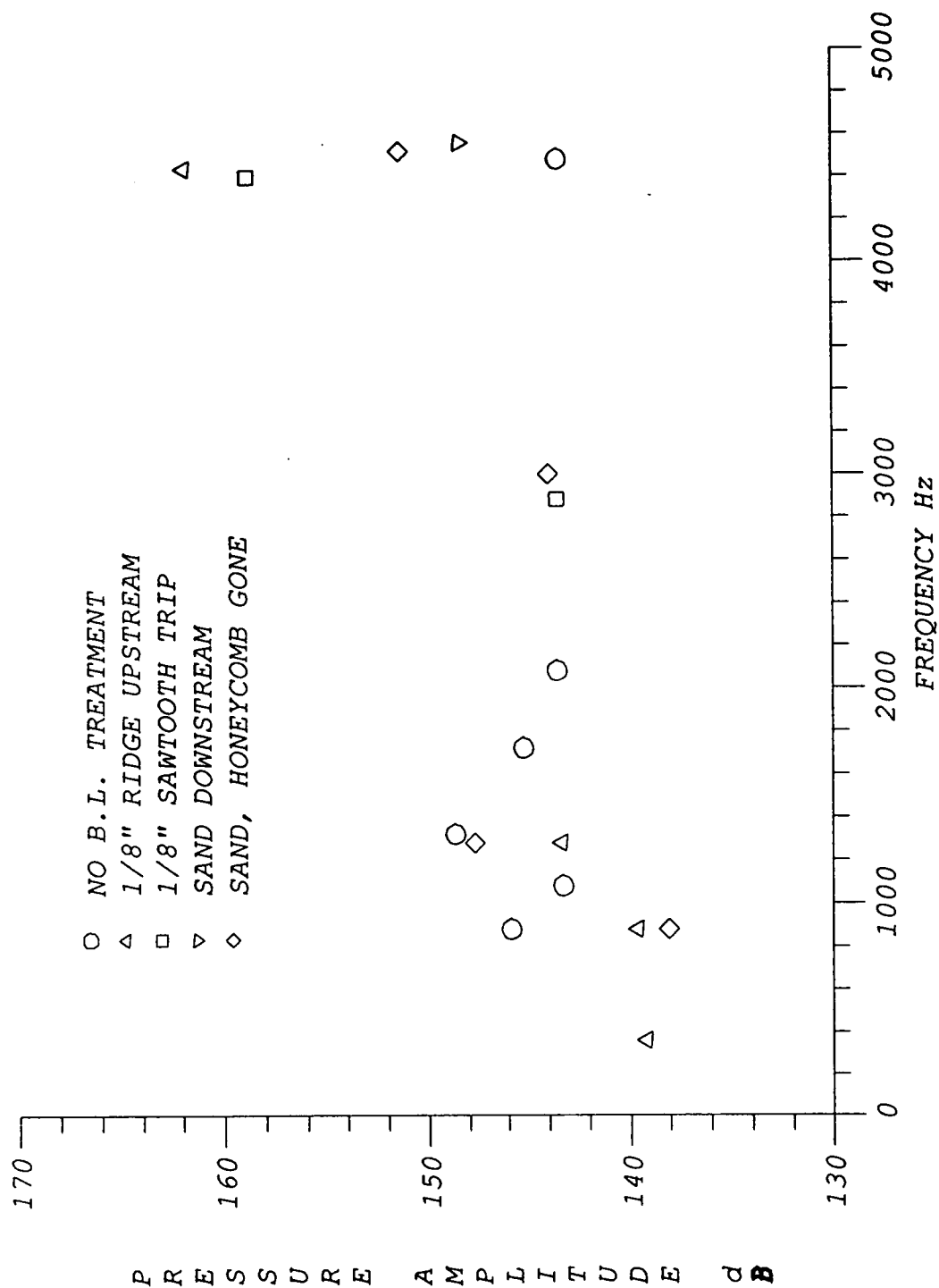


Figure 5-38. Boundary Layer Effects on Spectral Peaks (AR=1.2, PR=2.8)

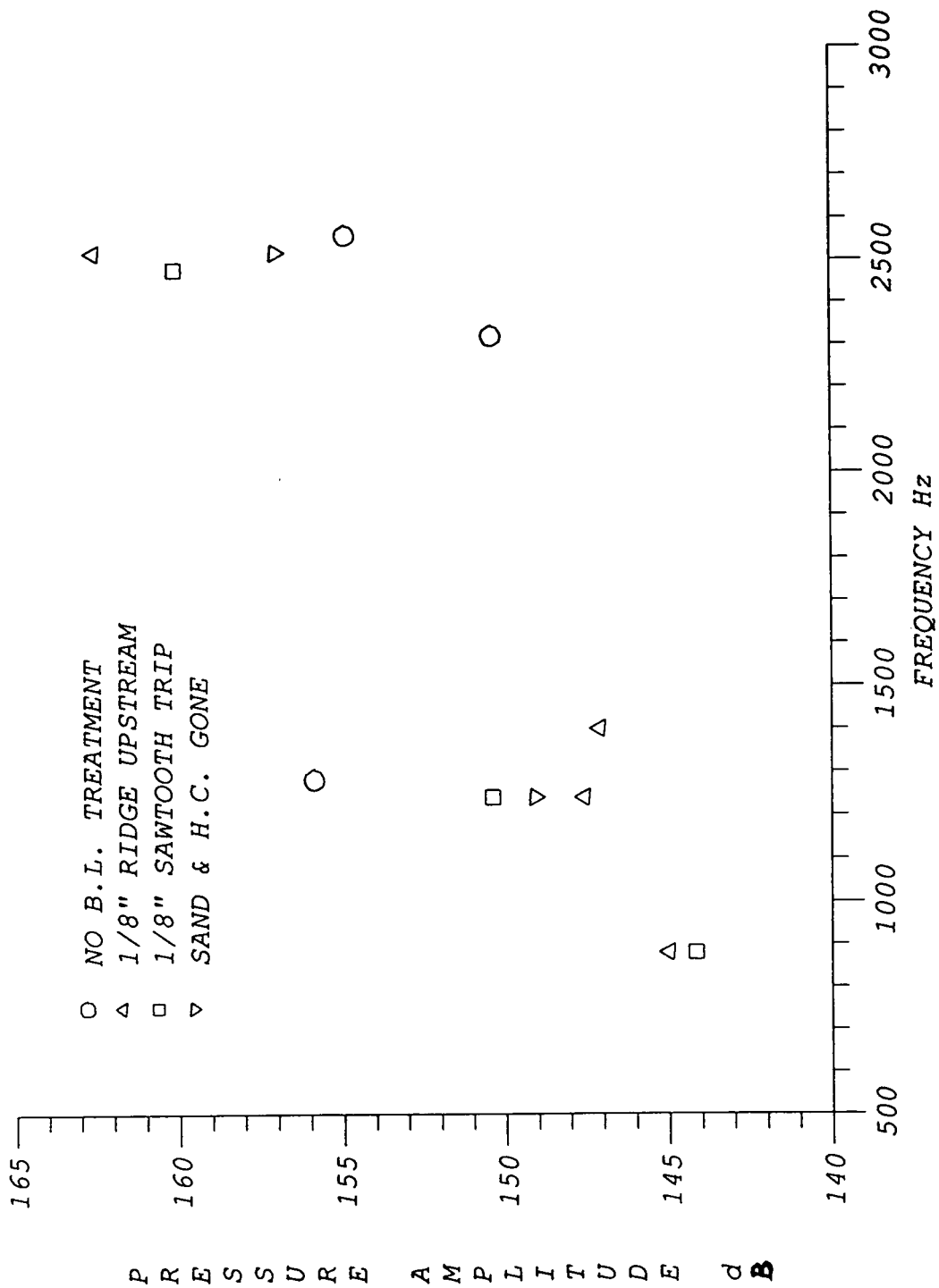
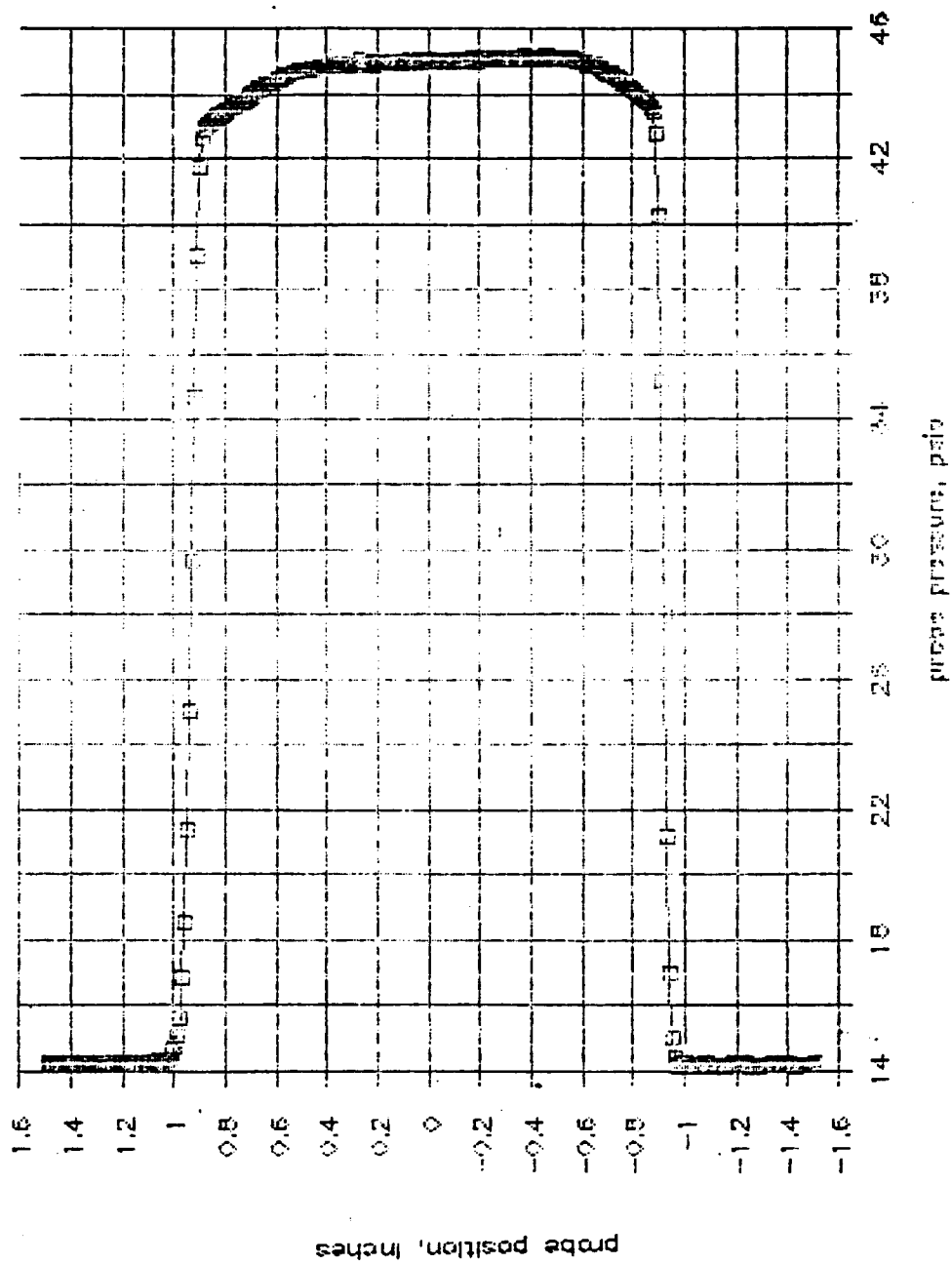
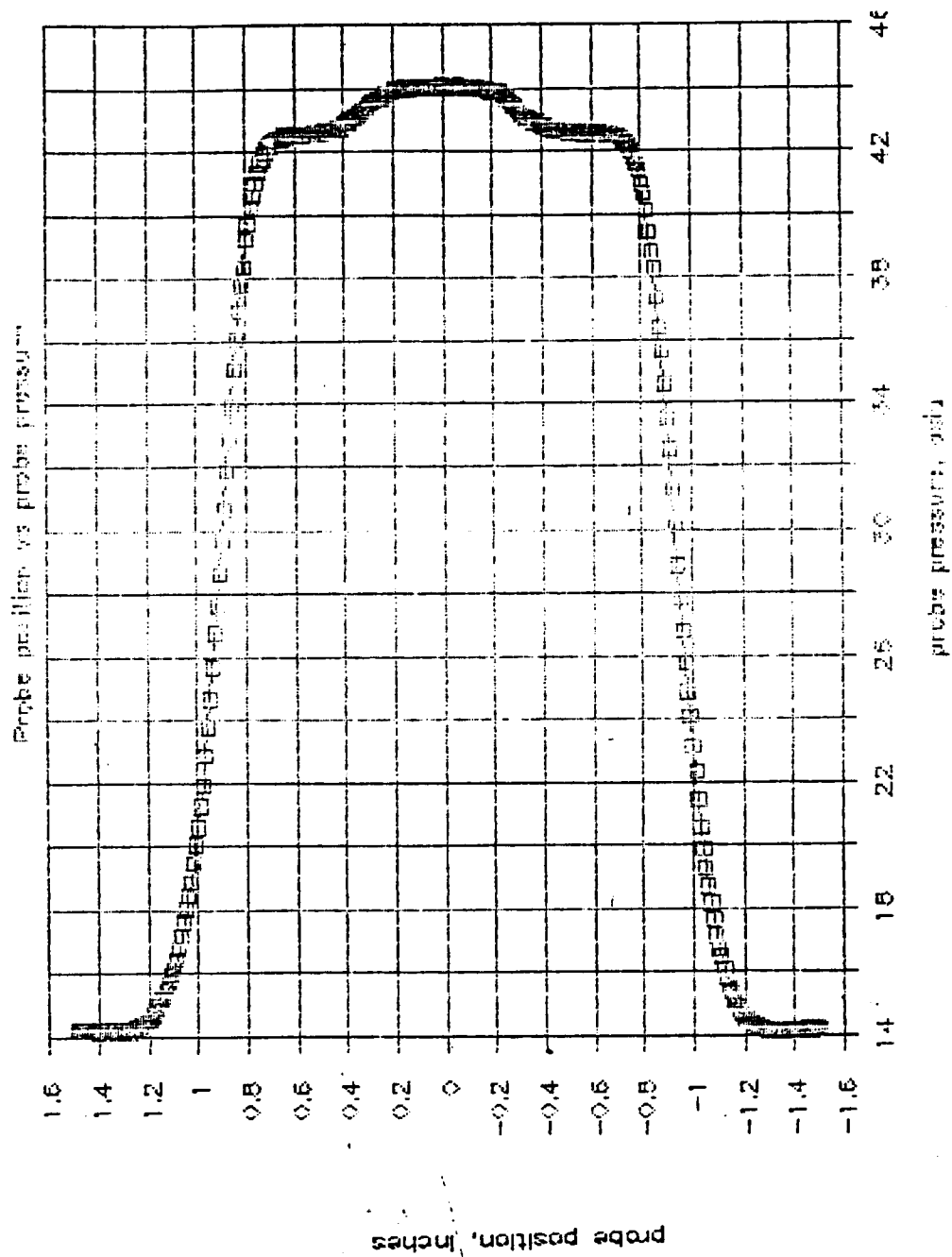


Figure 5-39. Boundary Layer Effects on Spectral Peaks (AR=1.2, PR=4.55)

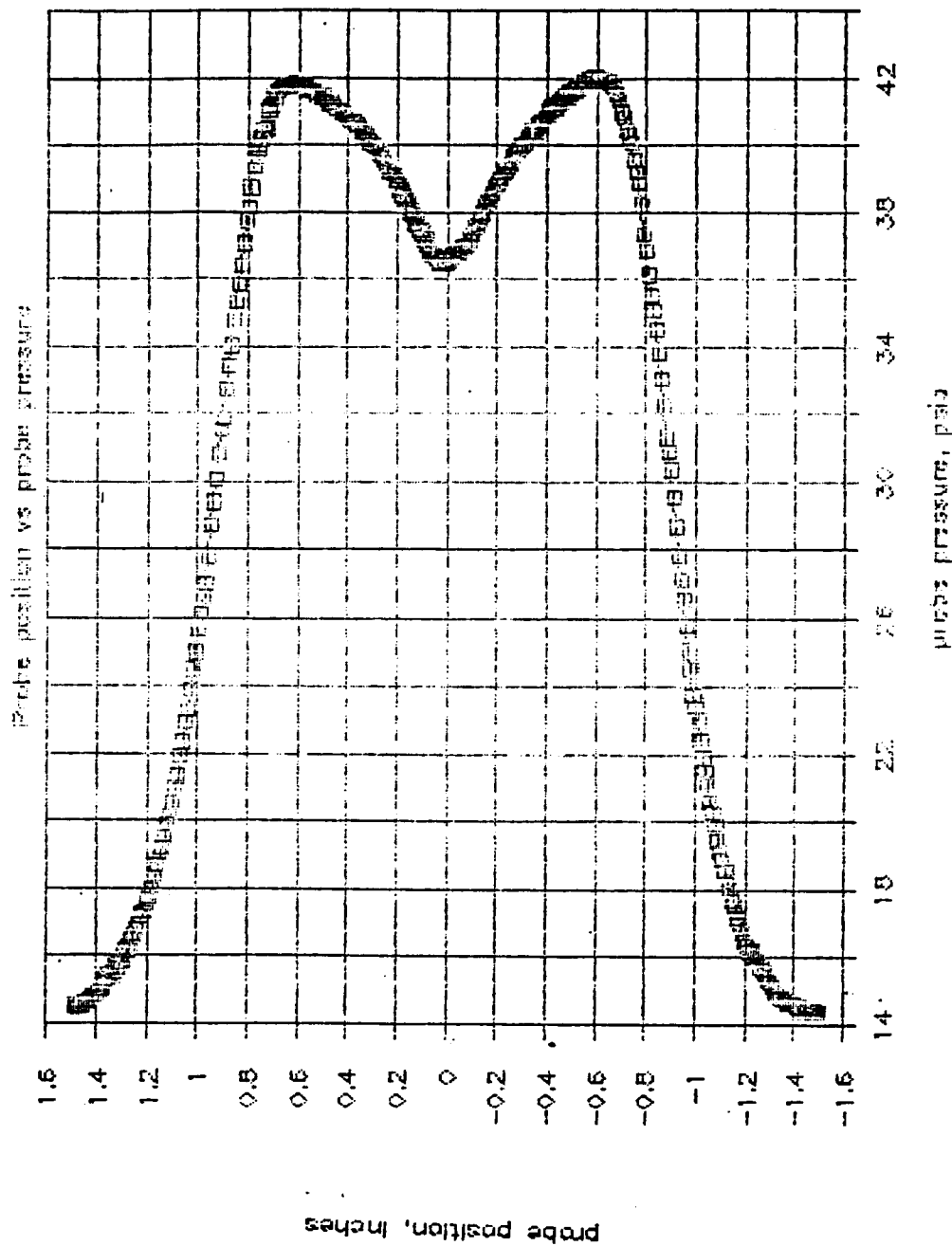
APPENDIX B
Pitot Probe Traverses



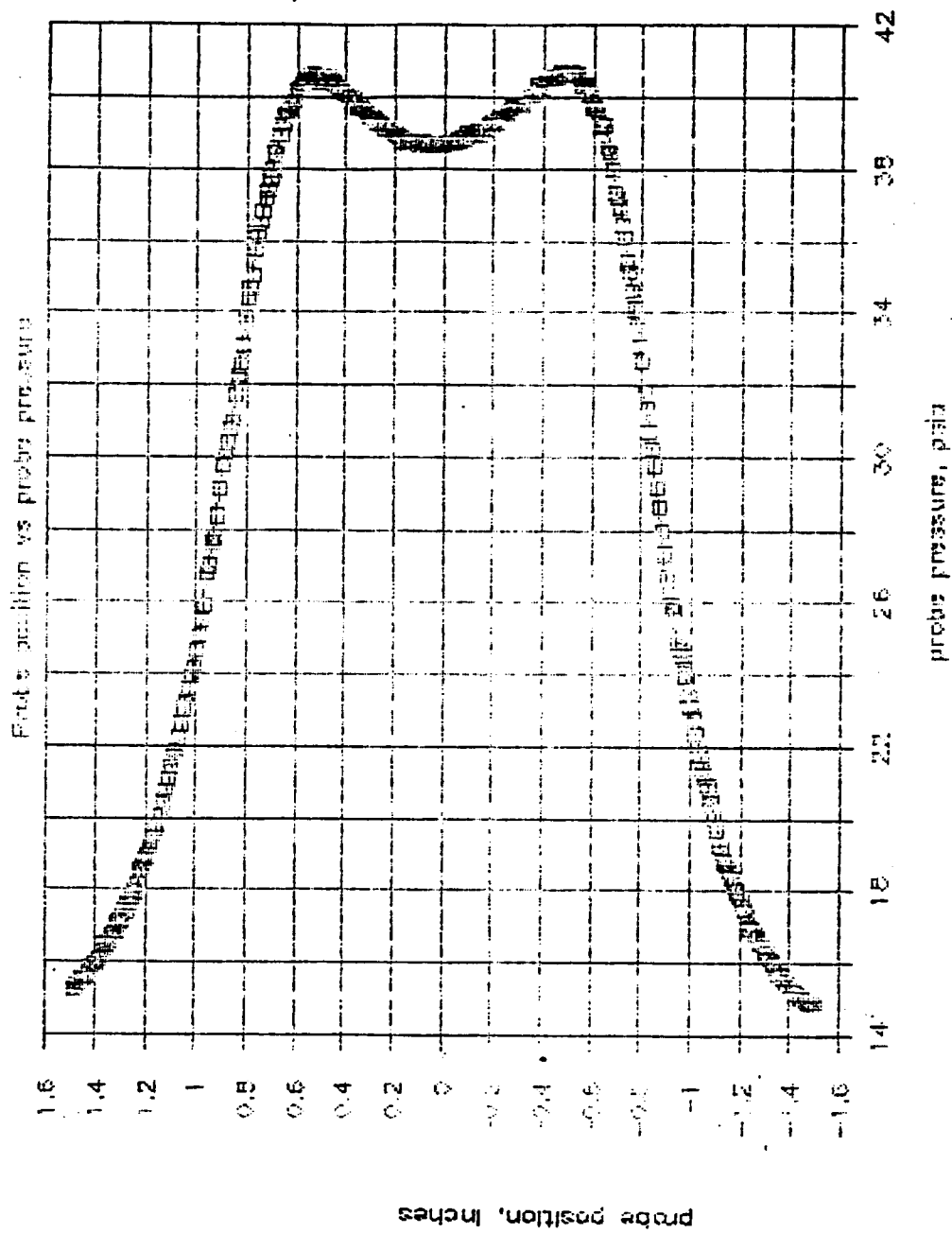
B-1. Conical Nozzle, Pt Vertical Traverse at 0.14D Downstream of Nozzle Exit
(Mach No. 1.4)



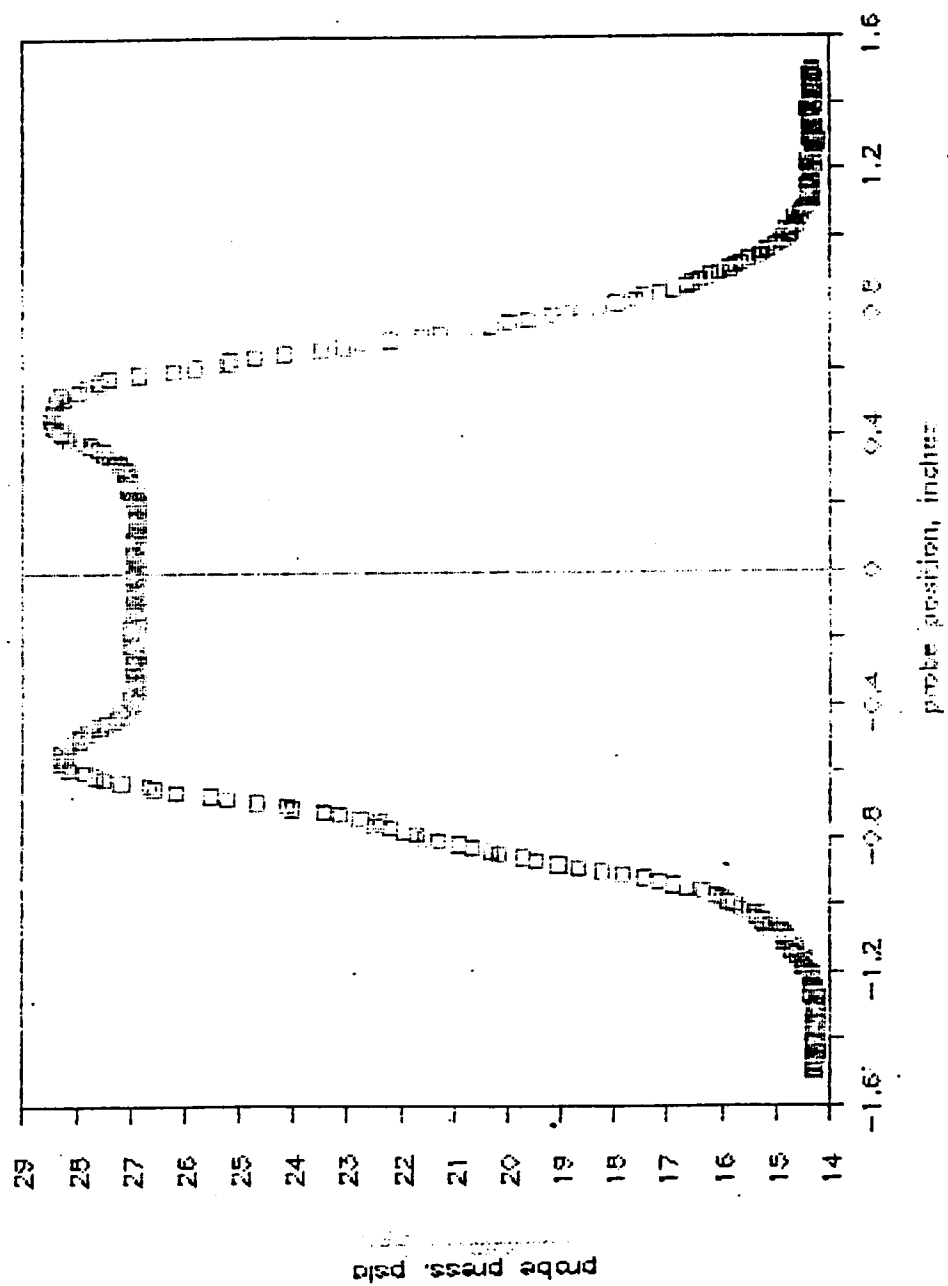
B-2. Conical Nozzle, Pt Vertical Traverse at 1.14D Downstream of Nozzle Exit
(Mach No. 1.4)



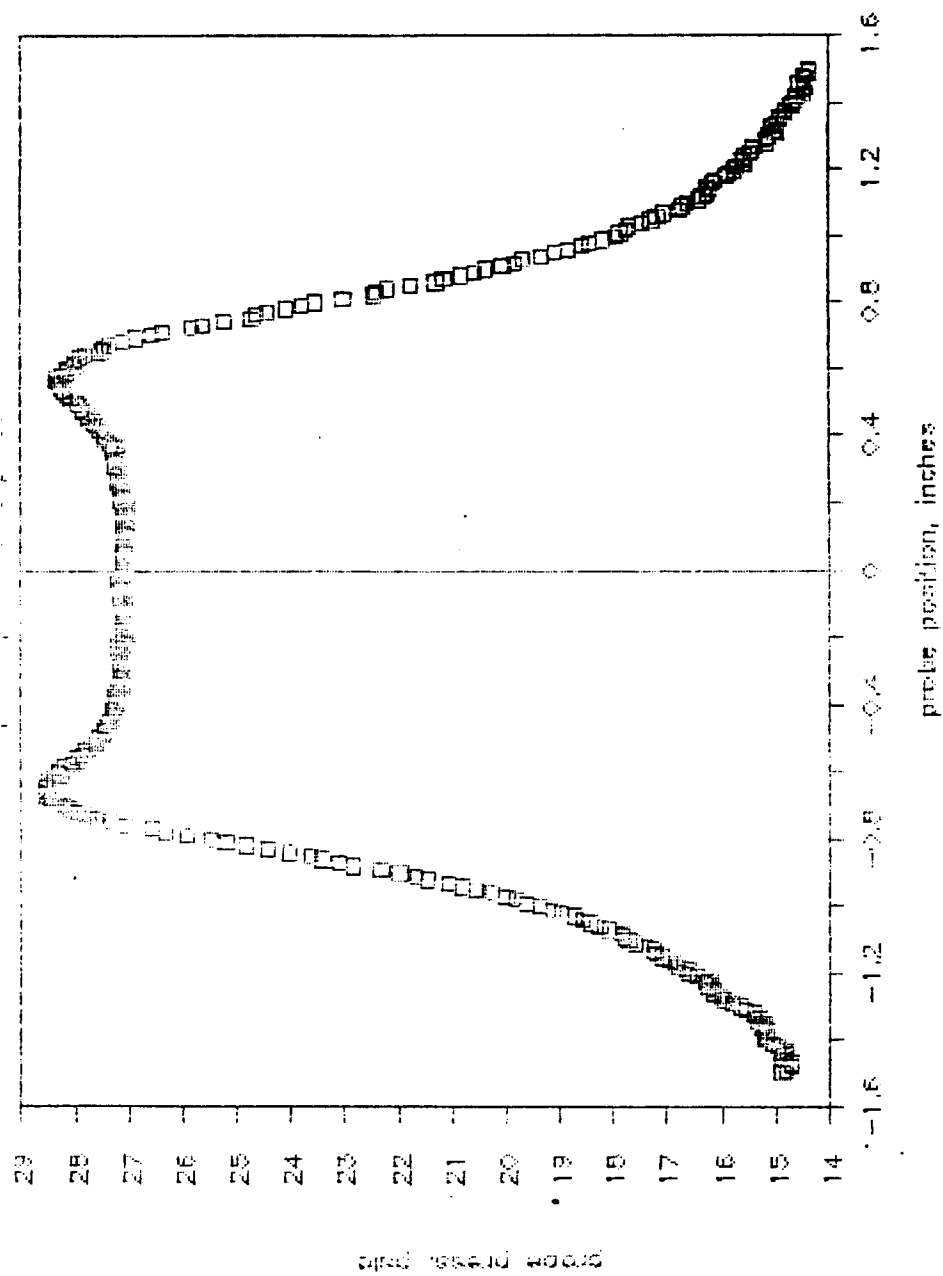
B-3. Conical Nozzle, Pt Vertical Traverse at 2.14D Downstream of Nozzle Exit
(Mach No. 1.4)



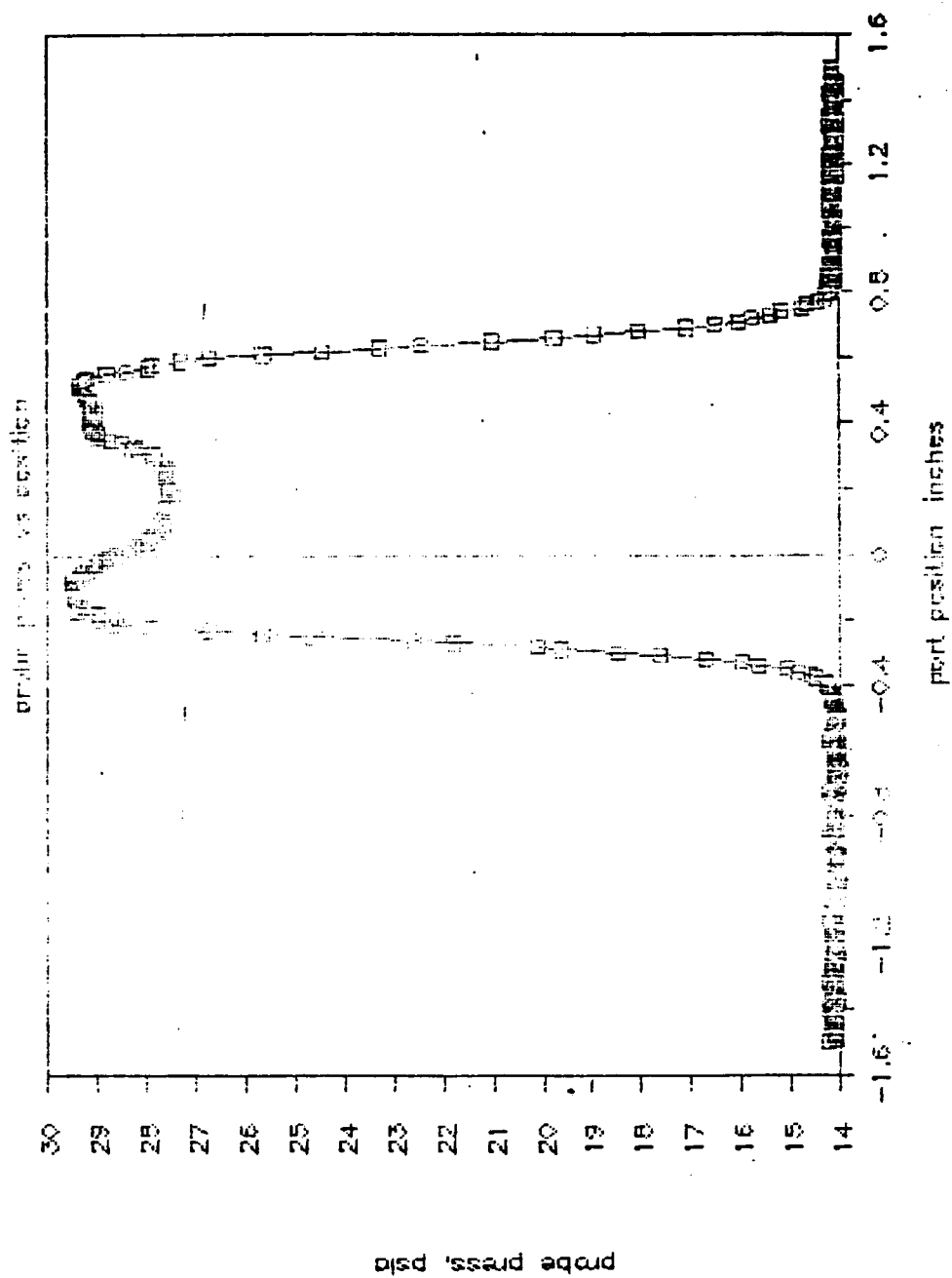
B-4. Conical Nozzle, Pt Vertical Traverse at 3.14D Downstream of Nozzle Exit
(Mach No. 1.4)



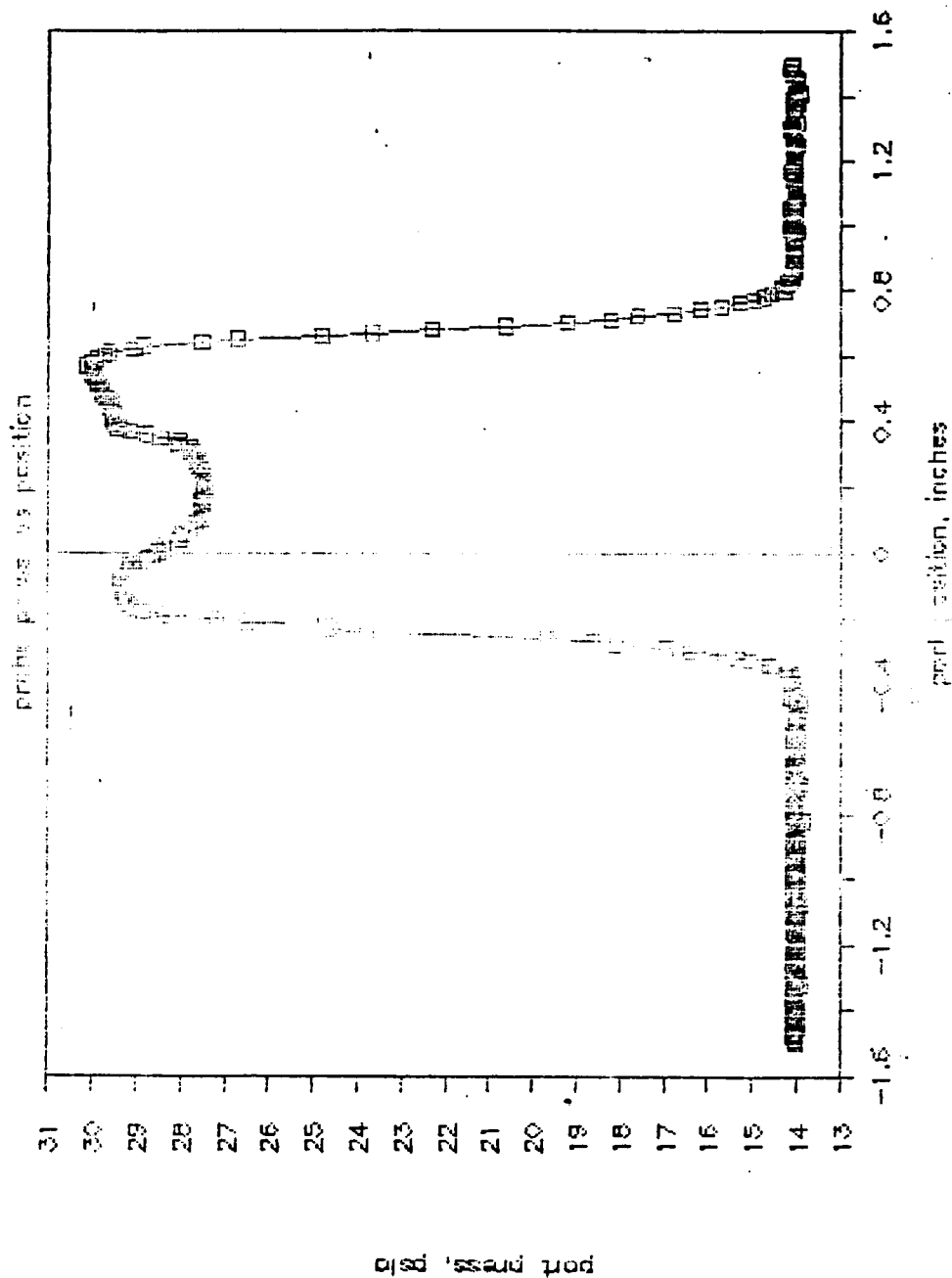
B-5. RN1, Pt Vertical Traverse at 1.32TH1 Downstream of Nozzle Exit
(AR=1.2, PR=2.15)



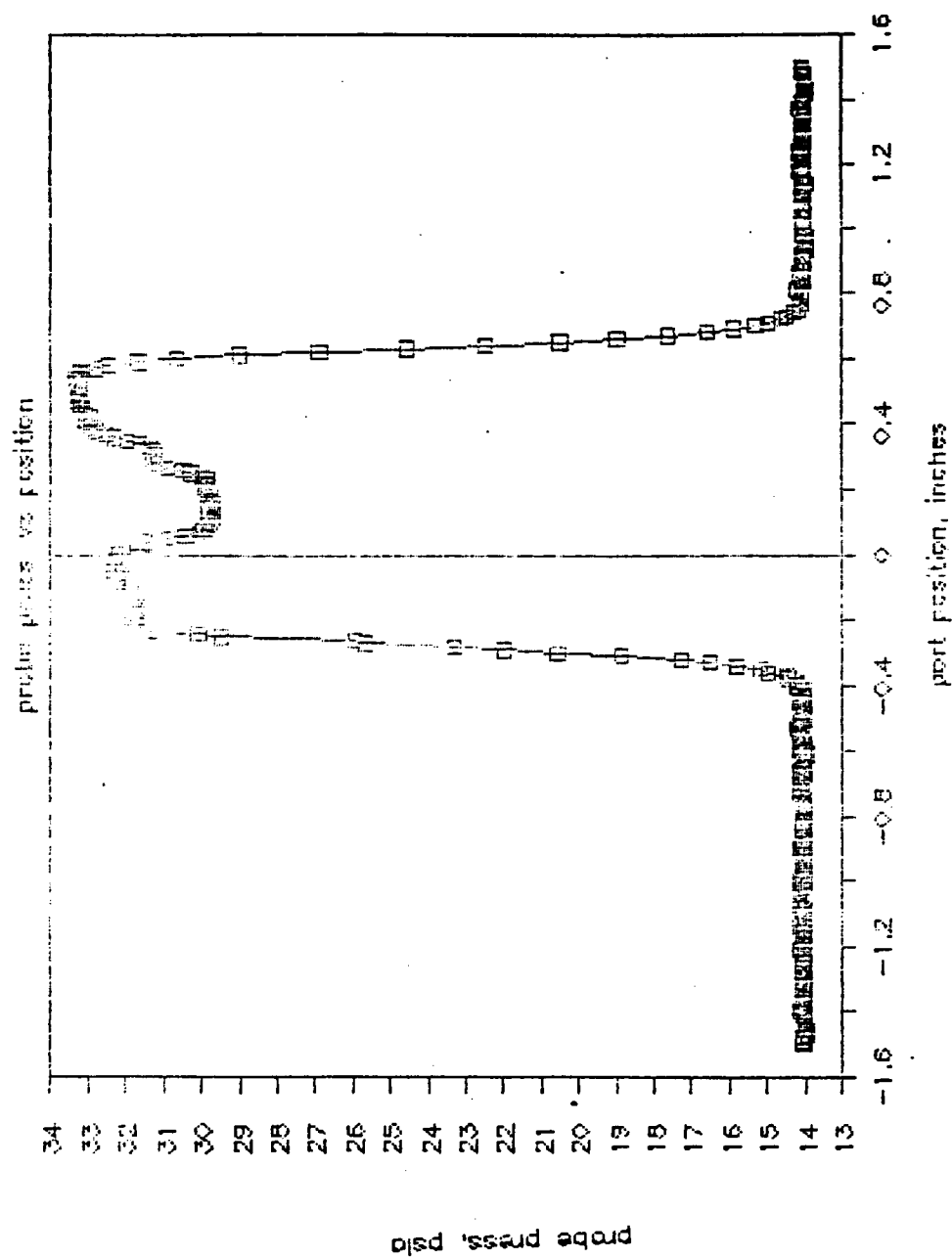
B-6. RN1, Pt Vertical Traverse at 2.63TH1 Downstream of Nozzle Exit
(AR=1.2, PR=2.15)



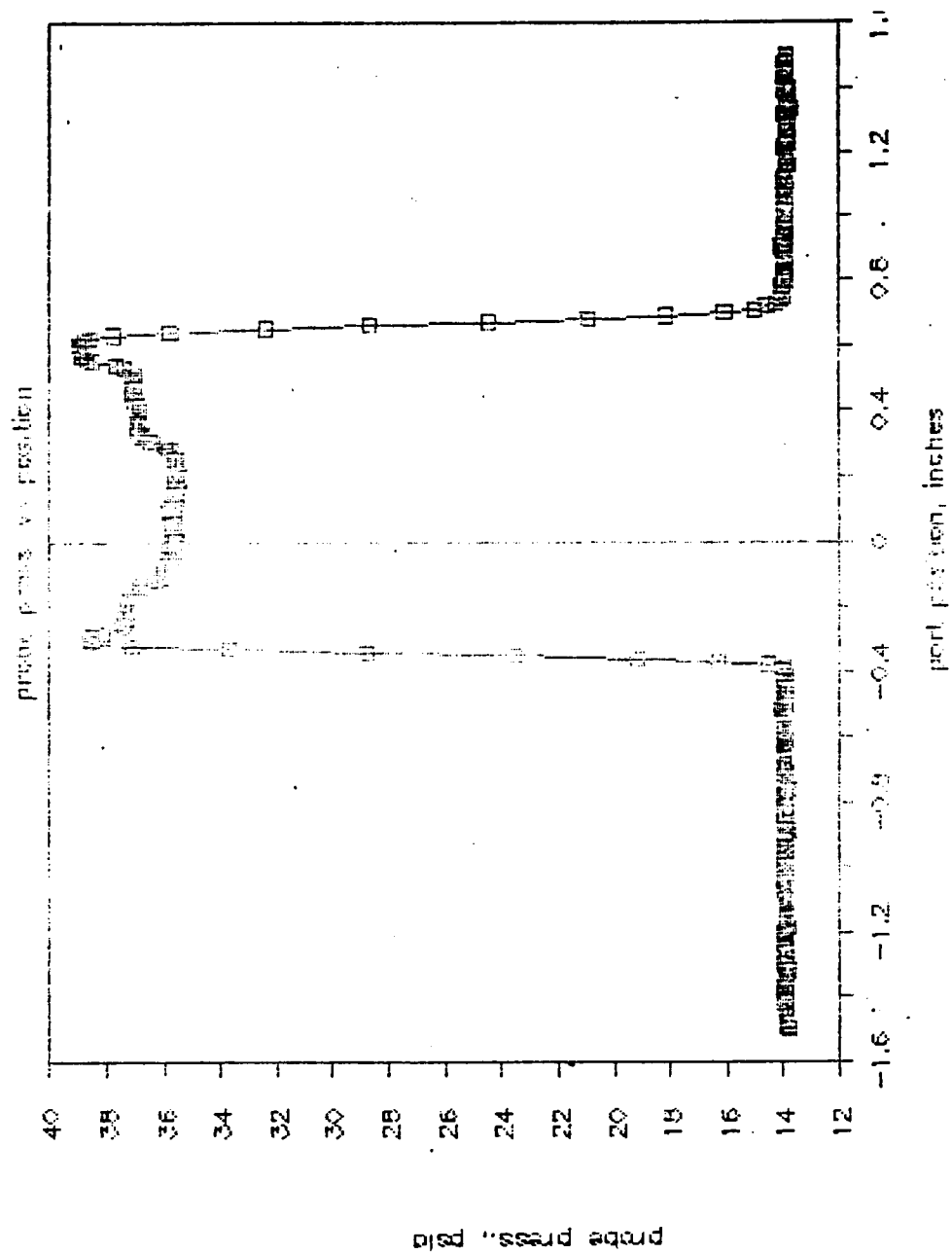
B-7. RN2, Pt Vertical Traverse at 1.12TH2 Downstream of Nozzle Exit
(AR=1.2, PR=2.0)



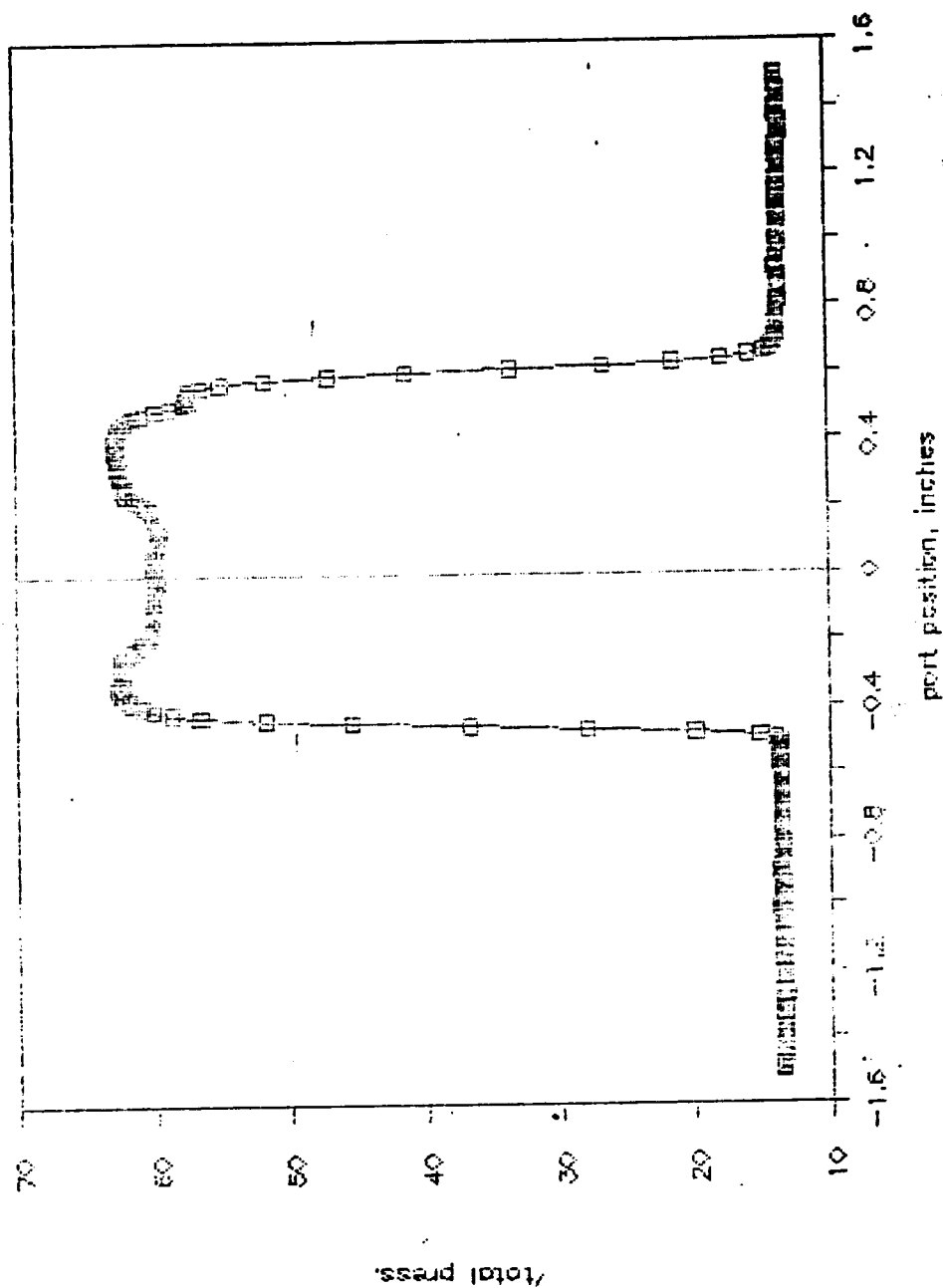
B-8. RN2, Pt Vertical Traverse at 1.12TH2 Downstream of Nozzle Exit
(AR=1.2, PR=2.15)



B-9. RN2, Pt Vertical Traverse at 1.12TH2 Downstream of Nozzle Exit
(AR=1.2, PR=2.3)



B-10. RN2, Pt Vertical Traverse at the Nozzle Exit (AR=1.2, PR=2.8)



B-11. RN2, Pt Vertical Traverse at the Nozzle Exit (AR=1.2, PR=4.75)

VITA

Paul Steven Cooke was born in Murfreesboro, Tennessee on August 7, 1966. After attending elementary schools in Georgia, Alabama, New Jersey and Tennessee, he graduated from Grundy County High School (Tracy City, Tennessee) in May 1984. The following September he entered Tennessee Technological University and in March, 1989 received the degree of Bachelor of Science in Mechanical Engineering. In May, 1989 he entered the University of Tennessee Space Institute in Tullahoma and in August, 1992 received a Master of Science degree in Mechanical Engineering.

Since July, 1991, he has been employed as an engineer for the U.S. Department of Energy in Morgantown, West Virginia.