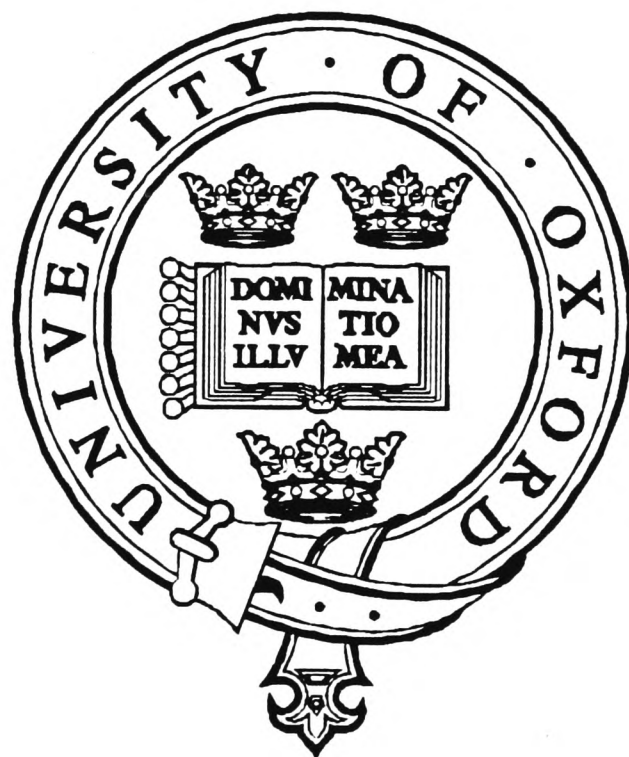


Three-Dimensional Unsteady Gas Turbine Flow Measurement

J.J.M. Batt

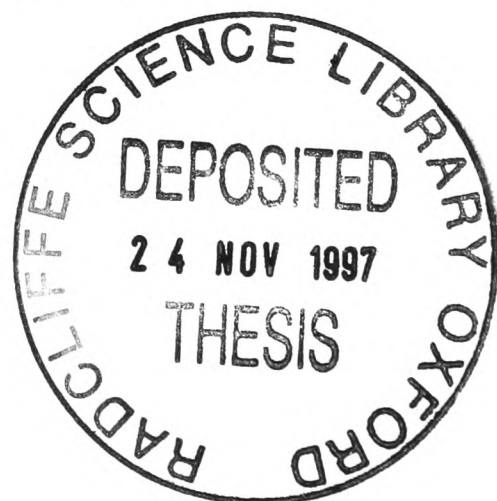
St. Catherine's College
OXFORD

*A thesis submitted in partial fulfilment of the requirements of the degree of
Doctor of Philosophy at the University of Oxford, Trinity Term 1997.*



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Abstract

The high pressure turbine stage can be considered the most important component for the efficiency and longevity of a modern gas turbine. The flow field within this stage is highly complex and is both unsteady and three-dimensional. Understanding this flow field is essential if improvements are to be made to future engine designs.

Increasingly designers are placing more emphasis on the use of Computational Fluid Dynamics (CFD) rather than experimental results. CFD methods can be more flexible and cost effective. However before these predictions can be used they must be validated against experimental data at engine conditions.

The hostile environment and complexity of flows within a gas turbine engine mean that collection of experimental data is extremely challenging. This thesis describes the development of an instrumentation technique for unsteady gas turbine flow measurement capable of resolving unsteady three-dimensional flow. The technique is based on an aerodynamic probe constructed with miniature semiconductor pressure transducers manufactured by Kulite Semiconductor Inc.

Measurements recorded using this instrumentation technique from the Oxford Rotor experiment are presented to illustrate its use, and these in turn are compared with a CFD prediction of the rotor flow-field.

This work was funded by the Engineering and Physical Sciences Research Council and Kulite Semiconductor Inc. The Oxford Rotor project is jointly funded by the Engineering and Physical Sciences Research Council (EPSRC), and Rolls-Royce Plc.

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The final completion of this thesis would not have been possible without the contributions of many people.

Foremost I would like to thank my supervisor, Roger Ainsworth, for support and guidance throughout this thesis and even before during the pioneering days of the small gas turbine project. Without Roger's influence it just would not have happened.

John Allen deserves special mention both for his magical skills in construction of the probes and teaching me the fine art of soldering. Rob Miller's enthusiasm throughout the project has been a considerable aid and those lively discussions contributed greatly. Many thanks to the rest of the Rotor team who made Osney such an enjoyable working environment: Nigel Brett, Kevin Grindrod, Roger Moss, Colin Sheldrake, and Steve Thorpe. Thanks also go to John Slater for his initial work on the aerodynamic calibration facility, and to Alistair Main and Martin Oldfield who provided considerable help with the inversion techniques.

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c	Chord
C_D	Drag coefficient
c_p	Specific heat capacity
C_M	Mach Coefficient
C_T	Total pressure coefficient
C_θ	Yaw coefficient
C_ϕ	Pitch Coefficient
F	Flow area
h	Enthalpy
l	Characteristic length
M	Mach number
m	Mass
N	Rotational speed
O	Offset voltage
P	Pressure
g	Gravity
r	Radius
S	Frontal area, Entropy, Transducer span
T	Temperature
t	Time
U	Flow velocity
V	Voltage
Z	Height
α	Temperature coefficient of resistance, Absolute flow angle
β	Relative flow angle

	γ	Ratio of specific heats
	Φ	Potential function
	ϕ	Pitch angle
	θ	Yaw angle
	η	Efficiency
	μ	Dynamic viscosity
	ρ	Fluid density
Subscripts	A	Face A
	b	Transducer bridge
	B	Face B
	C	Face C
	D	Face D
	g	Gas property
	i	Initial
	o	Supply
	p	Constant pressure
	R	Reference temperature
	t	Piston Tube
	∞	Freestream
	0	Total
	25	Value at 25°C

1. Introduction

1.0. Objective of thesis

We understand the world that surrounds us through our ability to examine and discover its secrets. Our own senses of sight, sound, touch and smell give us a starting point from which we can start our exploration. To enhance our senses we have developed tools to extend our senses; atoms may be examined through the electron microscope whilst at the other extreme the radio telescope has pushed our view out to the rest of the universe.

The field of turbomachinery is an example of where our understanding of the physical processes is important for developing and enhancing the technology. Improving turbine component efficiency is increasingly tied up with the understanding of the detailed mechanics of the flow field within the turbomachine. This understanding has come from several decades of research in which experimental measurements in turbomachines have been taken using a variety of instrumentation and visualisation techniques.

Large and expensive test rigs are required to measure turbine component efficiencies but the introduction of the use of computer simulations in the design stage has reduced the need for these experiments. These computer predictions can model many of the flow features within a turbomachine but the models they use can only be as good as our understanding of the flow mechanics. Therefore, before a code can be used it needs to be validated against experimental results. Measurement of these flows is notoriously difficult because of the harsh environment of flow within the turbomachine; gas temperatures are often well above the melting point of the metals from which the turbine blades are manufactured and the flows are highly unsteady.

The aim of the work underlying this thesis was to improve the ability with which it is possible to take detailed measurements in unsteady turbomachine flow fields. This thesis details the investigation and development that was undertaken to achieve this objective. A technique for the

measurement of three-dimensional unsteady flow fields has been developed and the calibration and application of this technique are described in detail.

This work was funded by Kulite Semiconductor Inc., manufacturers of miniature semiconductor pressure transducers, and fits within a wider programme financed by Rolls-Royce and the Engineering and Physical Sciences Research Council (EPSRC) to record detailed experimental measurements at engine conditions.

1.1. The gas turbine engine

Early turbines have their origins with water wheels in which power was extracted from running water. Today, hydroelectric schemes with modern turbines are widespread and in some countries constitute the major source of electrical power generation.

In a steam turbine, high-temperature high-pressure steam generated from a boiler-furnace or a reactor core is used as the working medium. The steam acts as an intermediate medium between the hot gases, where the heat is generated, and the turbine, where the power is generated. Steam turbines are large and efficient and are used for power generation and as marine power plants. However, the bulky equipment associated with steam generation has meant that their size has prevented use as a power source for aviation.

The gas turbine contrasts with the steam turbine in that the hot gases themselves rather than steam are used to drive the turbine. Without the need for the equipment required to generate the steam a gas turbine engine can be far more compact than its steam turbine equivalent. However the price paid for this benefit is that the requirements placed on the turbine are far more severe than for a steam turbine; the hot gases may be at temperatures higher than the melting point of the turbine materials.

The potentially high power to weight ratios that could be delivered by a gas turbine engine led to their original development for use in aviation during World War Two. It was during this period of time that the understanding of the thermodynamics and aerodynamics of the process were combined with the development of the materials that would make a working engine possible. Since the first gas

turbines were developed, they have become the dominant power plant in aviation and have found increasingly wider usage, including power generation.

1.2. Gas turbine theory

A gas turbine may be made from three components; a compressor, combustor and turbine. Air is drawn into the compressor where its total pressure is raised to provide a pressure ratio for expansion later in the turbine. Fuel is added to the compressed gas mixture in the combustion chamber before it is ignited to raise the temperature of the working fluid. The hot gases are then expanded in the turbine. The power generated by the turbine is used to drive the compressor which is attached on a common drive shaft. Any excess power may then be extracted as shaft work or as the thrust of the hot gases in an exhaust nozzle.

A schematic of the cycle is shown in figure 1.1 and may be summarised in the following stages:

1-2 Isentropic compression in the compressor.

2-3 Combustion. Fuel is mixed with the compressed air and burnt to raise the temperature prior to the turbine inlet. This increase in temperature raises the enthalpy of the gases.

3-4 Isentropic expansion in the turbine. The expansion in the turbine extracts the enthalpy from the gas which generates work.

In practice the compression and expansion will deviate from isentropic as there will be an increase in entropy resulting from loss mechanisms in the fluid. Many modern turbomachines are more sophisticated than the simple cycle described and may have multiple stages.

1.3. Axial flow turbine

A turbine stage is made up of two components. The first is a set of fixed nozzle guide vanes, also known as the stator, and the second is a row of rotating turbine blades. The purpose of the nozzle guide vanes is to impart swirl to the fluid and expand the flow. The rotor blades then remove the swirl

and further expand the flow. The change in angular momentum of the fluid during this process creates the shaft work from the turbine. Figure 1.2 is a schematic of a turbine stage.

The performance of an aerofoil is determined by the efficiency of the profile. This efficiency is usually considered relative to the idealised flow and does not incorporate factors that affect the overall thermodynamic efficiency of the turbine cycle. The flow through a turbine stage is usually idealised as being adiabatic and reversible, but the occurrence of heat transfer or aerodynamic losses will cause the component to be less efficient than for idealised flow.

Loss mechanisms within the flow reduce the work available and consequently decrease the stage efficiency. Turbine stages need to be designed for maximum efficiency and therefore understanding these loss mechanisms has been the subject of a considerable volume of research. Loss mechanisms will cause an increase in entropy in the flow but because entropy cannot be directly measured the irreversibilities are usually defined in terms of enthalpy (temperature) or total pressure changes. The four most significant sources of loss are; profile loss from the blade boundary layer; secondary flows such as the passage and horseshoe vortex; tip leakage where fluid slips through the gap between the blade tip and the outer annulus; and finally shock loss from shock waves forming in the passage.

1.4. *Cascade facilities*

Experimental measurements of flow within a gas turbine may be made within a cascade facility. The type of facility will depend on the objectives of the experimenter and may range from a small blade row to a full model turbine. The advantage of using a cascade facility over a real engine is that to take measurements it is only necessary to match the relevant non-dimensional parameters and therefore cascades allow measurements to be made outside the harsh environments encountered in a real engine. The cascades are generally simplifications of real engines to allow measurements that allow analysis of specific phenomena.

Facilities in which flow is continuous allow detailed measurements to be recorded and averaged over a period of time. However the disadvantage with continuous flow facilities is that they are expensive to run and maintain, factors that are highly significant in industrial research.

The alternative to continuous flow is the use of transient facilities in which real conditions need only be simulated for short periods of time during which detailed measurements are taken. Oxford University's two primary turbomachinery research rigs, the Cold Heat Transfer Tunnel (CHTT), Jones [1978] and the Oxford Rotor, Ainsworth et al [1988b] are transient facilities.

1.5. Instrumentation techniques

Many researchers, as part of the need for experimental results, have had to develop instrumentation techniques as a side line to their main objectives. In this work the author has had the luxury of being able to concentrate on refining an instrumentation technique. It is the intention that the results of this work may be used by others for their experimental measurements.

There are many experimental techniques for the measurement of aerodynamic flows which have been devised for various purposes ranging from the quantitative through to pure visualisation. No one instrumentation technique will be the perfect method for recording results and for every strength in a technique there is almost always a compromise. Three of the most significant techniques available for turbomachinery measurements are aerodynamic probes, hot wires and laser anemometry. The approach taken for the Oxford rotor project has been to develop a variety of techniques which when combined will give an overall picture of the experimental results. However, this author has concentrated on the development of aerodynamic probes. To see the parallel development related to instrumentation for the Oxford rotor facility for hot wires the reader is referred to Sheldrake and Ainsworth [1995].

1.5.1. Aerodynamic probes

Aerodynamic probes have been used for several decades in turbomachinery research. The solid body of a probe is inserted into a fluid flow and pressure tappings on the surface of the probe record

the local pressure on the probe surface. Careful design of the solid body allow the fluid total and static pressures to be measured from these pressure tappings and this allows the Mach number to be derived.

To measure flow direction the difference in pressure across a symmetric body may be used. In a 'null' method the body is moved until the difference is zero, indicating the flow direction. A combination probe uses multiple measurements of pressure to allow the flow direction, total pressure, static pressure and Mach number to be found from the same instrument. However in contrast to a 'null' method for measuring direction, the probe is carefully calibrated beforehand and the probe is not moved in the flow to find the flow direction.

1.5.2. Hot wire anemometry

For unsteady aerodynamic measurements the hot wire anemometry technique has found widespread use because the bandwidth is very high. The hot wire works on the principle of a thin wire placed in a flow through which a constant current flows. The heat transfer rate from the wire is a function of the air flow over the wire and therefore as the flow speed changes so does the heat transfer rate. The changing heat transfer rate is reflected by the temperature of the wire and as a result the electrical resistance of the wire. The hot wire method therefore records the voltage required to ensure that the current remains constant from which the flow speed over the wire can be derived.

The advantage of the hot wire sensor lies in that it has very high bandwidth and once the signal conditioning circuit has been purchased, the hot wires themselves are relatively inexpensive. Furthermore, the hot wire method may be extended to include direction measurement through the use of multiple hot wires.

However, hot wires are susceptible to damage by dirt particles and their sensitivity can change with the impact of particles on the wire itself. The inconvenience of replacing broken hot wires may far outweigh their low cost.

1.5.3. Laser techniques

Laser anemometry contrasts with aerodynamic probes and hot wires in that it is a truly non-intrusive technique. It works on the principle that light reflected from an object will have a Doppler shift which is a function of the objects velocity. In order to use the technique the flow needs to be seeded with particles small enough to follow the flow but large enough to reflect light from a laser. The use of laser anemometry for widespread turbomachinery measurements is restricted by the large capital expenditure required for the laser and the need to have optical access which might not be convenient in a turbomachine.

1.5.4. Semiconductor probes

Pneumatic aerodynamic probes are not suitable for unsteady flow measurement but the last two decades have seen considerable progress towards the use of fast response aerodynamic probes, based on silicon semiconductor pressure transducers. Initially these probes relied on sensors mounted in the interior of the probe and were necessarily restricted in bandwidth and therefore led to the requirement for probes with surface mounted transducers. In Oxford, following from the development of a technique for mounting miniature semiconductor pressure transducers on turbine blades, Ainsworth et al [1991], fast response probes based on this technology were devised. The aim of this new generation of aerodynamic probe was to produce an instrument capable of making the type of wide bandwidth velocity measurements (magnitude and direction) conventionally made with hot wire and laser anemometers, but in a more robust and easy to use form. The aerodynamic probe was manufactured with semiconductor sensors directly mounted on the surface of the probe rather than using pneumatic tappings to monitor Pitot side-wall pressures.

The work on probes with surface mounted sensors in Oxford has stemmed from a collaboration with Kulite Semiconductor Inc., where instead of using a standard transducer package, a means of mounting the silicon sensing chip itself directly onto a metal component was devised. This technique, which is patented, is known in Oxford and elsewhere as 'chip on' technology.

There are great advantages in using this technology for aerodynamic probe applications in that it permits a much more compact and flexible design of instrument, and since the pressure sensing element is mounted on the probe surface, high bandwidths are possible. The three-dimensional probe is shown in figures 1.3 and 1.4.

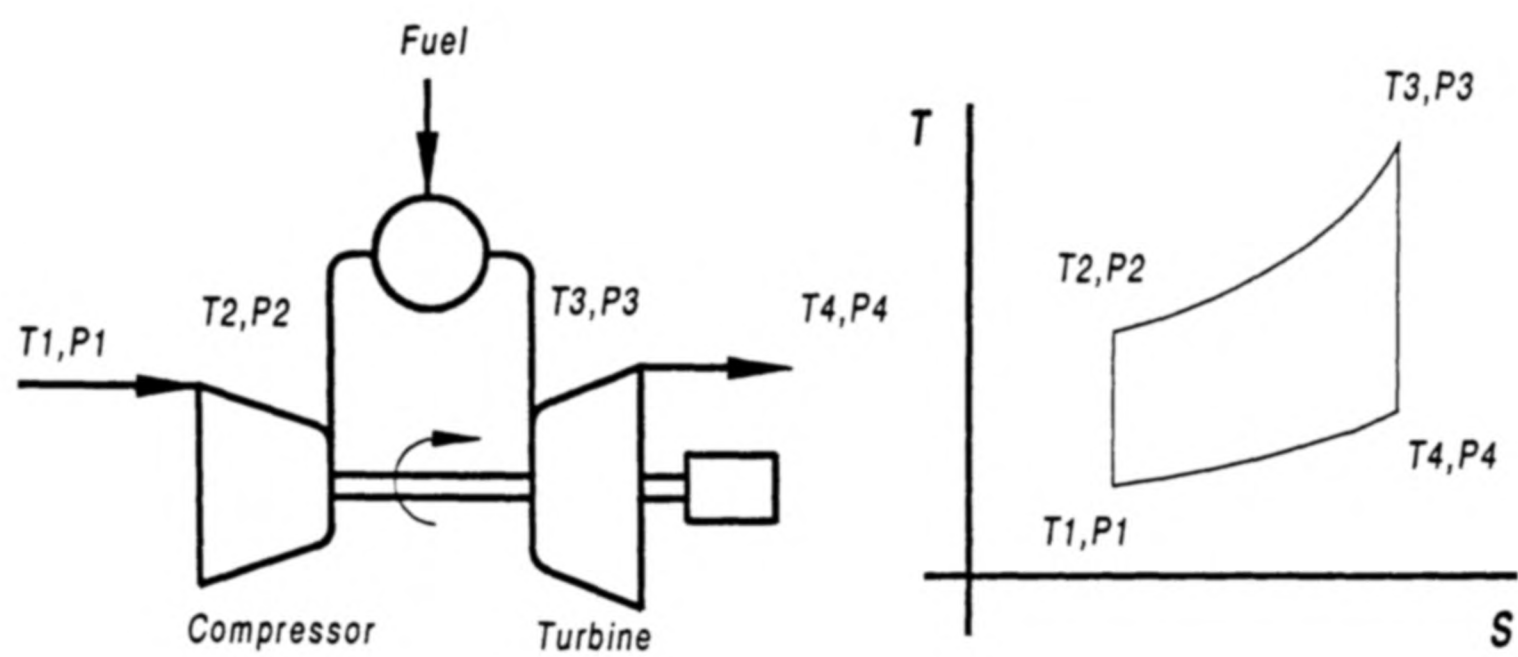


Figure 1-1 The gas turbine cycle

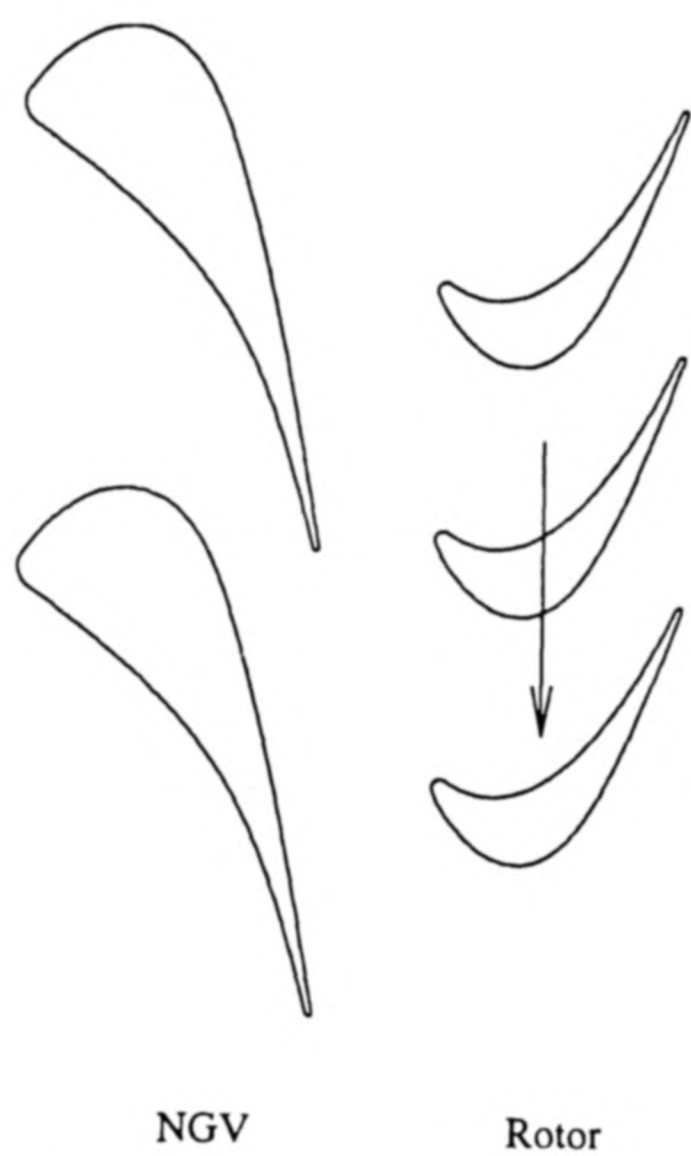


Figure 1-2 The axial turbine stage

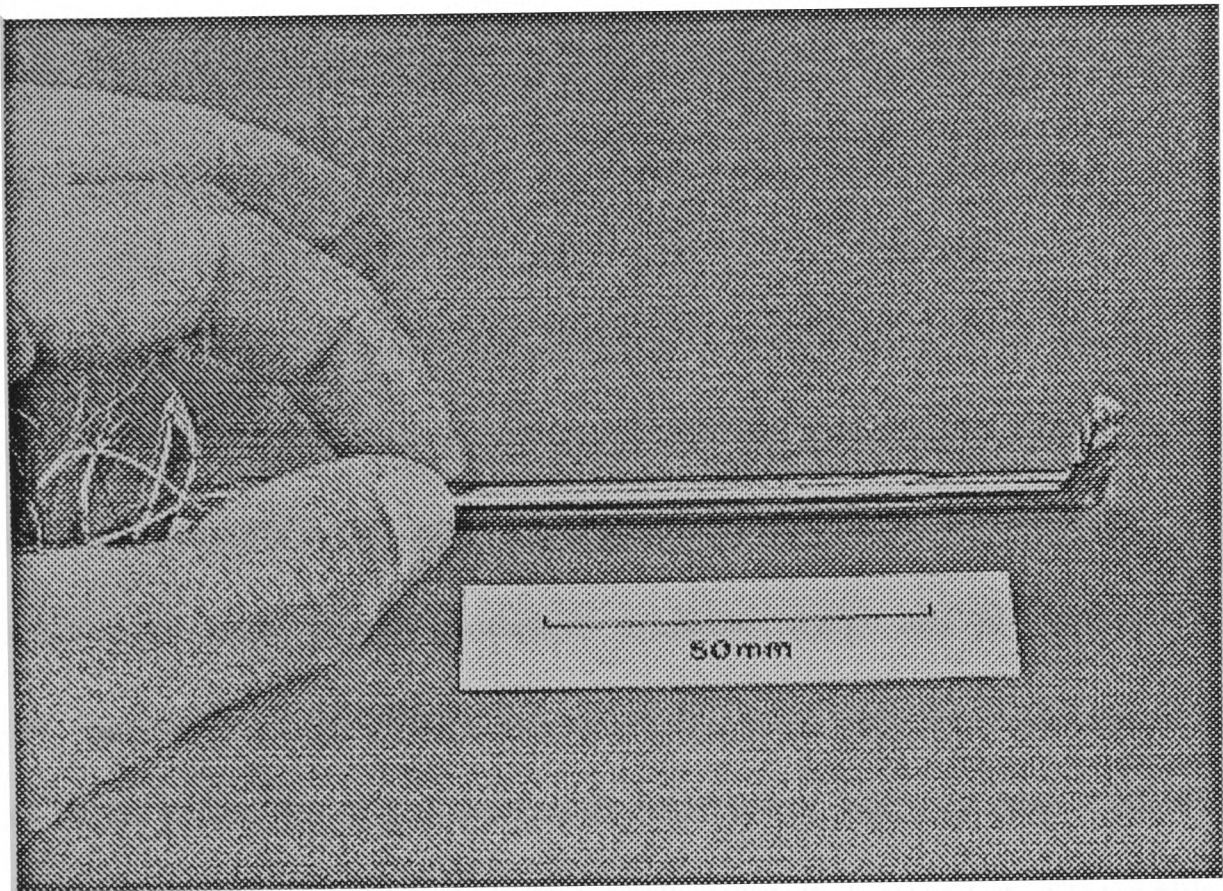


Figure 1-3 The three-dimensional semiconductor probe

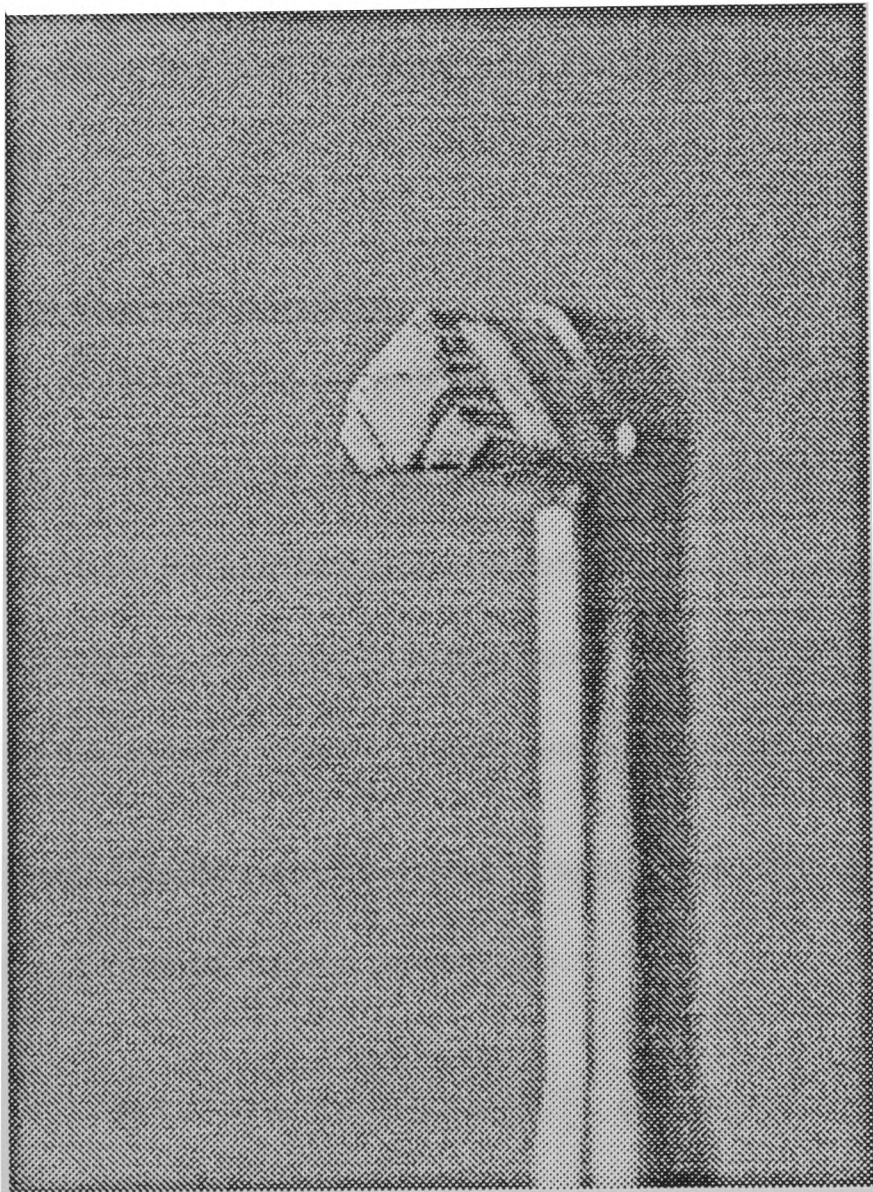


Figure 1-4 A close up of the semiconductor probe head

2. The fundamentals of aerodynamic probes

2.0. Overview

This chapter explores the fundamental operation of pressure probes for aerodynamic measurements. It starts with the basic thermodynamic relations which define the properties that are being measured and then continues by showing how an aerodynamic probe can be used for measurement. A considerable variety of aerodynamic probe types have been used over several decades of turbomachinery research. Of the geometries available, wedge and five-hole probes have been dominant. As semiconductor technology has advanced, researchers have developed semiconductor probes based on the geometries of their pneumatic counterparts. Many of these semiconductor probes have been used for measurements in turbomachines, but poor transducer steady state performance has restricted their application to unsteady measurements. In these circumstances the unsteady component from the semiconductor probe has been combined with the steady flow component from separate pneumatic measurements.

2.1. Aerodynamic probes

An aerodynamic probe works on the principle of inserting a physical object into a flow and taking measurements on the surface of the probe, usually temperature and pressure. The flow properties are then derived from these surface measurements. The thermodynamic relations can be used to convert properties measured by the probe to measurements of interest and in the following section the relationships between temperature, pressure, velocity and enthalpy will be shown.

Steady Flow Energy Equation may be derived by applying the first law of thermodynamics to steady flow of fluid through a control volume, equation 2.1.

$$\dot{Q} - \dot{W} = \dot{m} \left[(h_2 - h_1) + \frac{1}{2}(U_2^2 - U_1^2) + g(Z_2 - Z_1) \right]$$

Equation 2.1

In the context of aerodynamic probes there will be no shaft work and the flow should be adiabatic and therefore the Steady Flow Energy Equation reduces to the form given in equation 2.2:

$$(h_2 - h_1) + \frac{1}{2}(U_2^2 - U_1^2) + g(Z_2 - Z_1) = 0$$

Equation 2.2

If changes in height can be ignored this relationship may be expressed in terms of a total enthalpy, h_o :

$$h_o = h + \frac{U^2}{2}$$

Equation 2.3

In compressible flow, provided it can be assumed that the medium will behave as an ideal gas, the enthalpy, h , will be a function of temperature only. Therefore the enthalpy term in equation 2.3 may be replaced by the product of the specific heat capacity and temperature to give the relationship between total and static temperature, equation 2.4.

$$T_o = T + \frac{U^2}{2c_p}$$

Equation 2.4

This equation may be expressed in terms of the Mach number and the ratio of specific heats to give the compressible form of the adiabatic Steady Flow Energy Equation:

$$\frac{T_o}{T} = 1 + \frac{(\gamma - 1)}{2} M^2$$

Equation 2.5

For an adiabatic and reversible flow process on an ideal gas the total and static pressures are related by:

$$\frac{P_0}{P} = \left(\frac{T_0}{T} \right)^{\left(\frac{\gamma}{\gamma-1} \right)}$$

Equation 2.6

Therefore the adiabatic Steady Flow Energy Equation may be expressed in terms of the total and static pressure provided the pressure is recovered reversibly, equation 2.7.

$$\frac{P_0}{P} = \left(1 + \frac{(\gamma - 1)}{2} M^2 \right)^{\frac{\gamma}{\gamma-1}}$$

Equation 2.7

If the fluid is no longer a compressible gas but of constant density instead, it is possible to return to the Steady Flow Energy Equation and reconsider the enthalpy term. In the absence of friction a fluid will behave reversibly. The first law of thermodynamics expressed in terms of enthalpy for a reversible process gives the relationship in equation 2.8:

$$dh = dq + \frac{dP}{\rho}$$

Equation 2.8

The steady flow energy equation, 2.1, when expressed in differential form may be written as follows:

$$dq - dw = dh + udu + gdz$$

Equation 2.9

Which upon substitution of equation 2.8, assuming no shaft work and a constant density may be integrated and to give a relationship which matches the Bernoulli equation, equation 2.10.

$$\frac{P}{\rho} + \frac{1}{2}U^2 + gZ = k$$

Equation 2.10

Where the changes in the height may be ignored, the Bernoulli equation reduces to its simplest form giving the relationship between total and static pressure for incompressible flow, equation 2.11.

$$P_0 = P + \frac{1}{2} \rho U^2$$

Equation 2.11

Possibly the simplest type of aerodynamic probe is the Pitot pressure probe which when inserted into a flow-field creates a stagnation point. The stagnation point causes the dynamic pressure to be recovered, and, provided the process is adiabatic and reversible, the pressure at the stagnation point will be the total pressure. A suitably placed pressure tapping on the Pitot probe will therefore measure the total pressure in the flow.

On the Pitot probe a second pressure tapping on the probe is located normal to the flow direction to ensure that no velocity recovery occurs and thereby measure the flow static pressure, provided there is no curvature of the streamlines. From the relationship given in equation 2.11 the flow velocity may be measured from the static and total pressure measurements for a given flow density. Henry Pitot described such a probe for measuring flow velocity in 1732 which became known as the Pitot-static probe. A modern version of this type of probe is shown in figure 2.1.

For compressible flow the Pitot probe may also be used, but this time to find the Mach number using equation 2.7, rather than the flow velocity. To find the flow velocity the temperature has to be measured independently.

2.2. Direction measurement

In order to measure the flow direction using an aerodynamic probe, it is necessary to have symmetrically oriented pressure tapings sensitive to the flow direction. The probe can then be used in two fashions.

The first is the null method, in which the probe is rotated in the flow until the pressures on either face are equalised indicating the flow direction has been found. This approach relies purely on

the symmetry of the probe and requires no analytical modelling or calibration. The disadvantage with the null method is that it relies on geometric perfection of the probe and, if the flow is varying with time, it requires continuous realignment of the probe which may not be practical.

The alternative method for measuring flow direction relies on fixing the probe position and measuring the pressure difference between the two side faces. If the relationship between the pressure difference and the angle of yaw of the flow to the probe is known the flow direction may be determined. The probe therefore requires a calibration on yaw angle to pressure difference before it is used.

A cylinder in a flow lends itself to easy analysis because the potential flow solution is available. Taking the potential flow solution for a cylinder of radius, a , equation 2.12, it is possible to derive the velocity over the surface of the cylinder as in equation 2.13:

$$\Phi = U_{\infty} \left(r - \frac{a^2}{r} \right) \sin \theta$$

Equation 2.12

$$u_{\theta} = \frac{\partial \Phi}{\partial r} \bigg|_{r=a} = 2U_{\infty} \sin \theta$$

Equation 2.13

Upon substitution of this velocity in the Bernoulli equation the yaw coefficient for two pressure tappings equally spaced at angle θ to the axis and the rotated by angle δ may be shown to be as in equation 2.14

$$C_{\theta} = \frac{\Delta P}{\frac{1}{2} \rho U_{\infty}^2} = 4 \sin^2(\theta + \delta) - 4 \sin^2(\theta - \delta)$$

Equation 2.14

In figure 2.2 the yaw coefficient has been plotted for pressure tapping angles of 15° , 30° and 45° over a range of incidence angles from -30° to $+30^\circ$. With the pressure tapping located at 45° the probe is at its most sensitive as can be expected from differentiation of the $\sin^2$ function.

As this analysis is based on a potential flow solution it ignores the effects of the separation. Separation will occur around the cylinder as the adverse pressure gradient pulls the boundary layer from the cylinder body. At this point the cylinder pressure will deviate from the theoretical. This means that in practice a theoretical calibration is inappropriate and a probe will need to be experimentally calibrated. However the potential flow solution provides a good first analysis and gives the foundation for the definition of a yaw coefficient independent of flow speed.

2.3. The wedge probe

Whilst the cylindrical probe lends itself to easy analysis, a more practical probe based on the geometry of the wedge shape has found widespread application in turbomachinery, figure 2.3. The front face, A, is used for measurement of the total pressure, whilst the two side faces, B and C, serve the dual purpose of measuring the static pressure and yaw pressure from any misalignment of the probe. For two-dimensional flow measurement the wedge probe has become dominant in turbomachinery because it allows the yaw pressure tapings to be located closer than on the corresponding cylindrical probes and is therefore less likely to give false readings in sharp pressure gradients, Bryer and Pankhurst [1971]. The measurement of the total pressure by the front face is, however, at a different radial location from two side faces and therefore care must be taken when using in flows with a radial pressure gradient.

By defining yaw pressure as the difference between the two side face pressures a measure of the yaw angle may be achieved. The yaw pressure will be zero when the probe has no incidence to the flow and will increase or decrease with change in incidence.

$$\text{Yaw Pressure} = P_A - P_B$$

Equation 2.15

However, the yaw pressure as defined in equation 2.15 will also be dependent on the total pressure of the experiment and the dynamic pressure. It is therefore necessary for the yaw pressure to be normalised into a non-dimensional pressure coefficient. Using the true dynamic pressure would

achieve this aim but this would require an independent measurement of flow speed to be made. Fortunately a perfectly satisfactory solution is to non-dimensionalise the coefficient using the probe dynamic pressure, equation 2.16. In this case it is assumed that the front face gives an estimate of the total pressure, and the average of the two side faces is a measure of the static pressure. Whilst each of the side faces will be sensitive to the yaw angle the average of the side face pressures should give a measure independent of yaw, although in practice the sensitivity is not completely eliminated.

$$\text{Probe Dynamic Pressure} = P_A - \left(\frac{P_B + P_C}{2} \right)$$

Equation 2.16

Therefore normalising the yaw pressure by the probe dynamic pressure gives the conventional yaw coefficient:

$$C_\theta = \frac{P_C - P_B}{P_A - \left(\frac{P_B + P_C}{2} \right)}$$

Equation 2.17

Figure 2.4 is taken from Ainsworth et al [1995] and shows the yaw coefficient as defined in equation 2.17 varied over yaw angle for a range of Mach numbers. At higher Mach numbers compressibility affects the probe behaviour and the plots at higher Mach numbers start to diverge.

2.4. Pneumatic probes

There are a wide variety of designs of probe that can be used for flow direction and velocity measurement. A large number of shapes and geometries have been adopted over the past few decades, the choice depending on the specific needs of the research and the construction techniques available. Three-hole probes, which include wedge geometries, give two-dimensional flow measurement and five-hole probes enable three-dimensional measurements.

Early researchers such as Schulz et al [1952] designed their probes in the shape of claws and with this arrangement it was possible to make two-dimensional flow measurements, figure 2.5. Wedge

probes were later developed because they were easier to construct and gave better resolution in flows with sharp gradients. A pneumatic wedge probe constructed by Morris [1961] is also illustrated in figure 2.5.

The spherical geometry is suitable for modelling with a potential flow solution. Wright [1969] compared the experimental with the theoretical calibration but found that in practice all probes need to be calibrated because manufacture of the probe causes deviation from the theoretical model. These deviations include the influence of the stem, the size of the sensing holes and constructional errors since miniature probes often push the limits of precision manufacture.

Five-hole probes are attractive because they are relatively simple to construct and provide a uniform sensitivity to flow direction. Many probes have been constructed from steel tubing, where one tube faces forward and the others have chamfered sides to allow flow direction to be resolved. Variants of this construction technique include solid conical probe heads into which the pressure tapping is machined.

A direction probe need not be calibrated for velocity if it is to be used over a small range of velocities because the coefficients are normalised against the dynamic pressure. However, compressibility effects influence the calibration at higher Mach numbers. Amongst the first probes to be calibrated for both Mach number and direction were the probes of Krause and Dudzinski [1969]. Their probes were calibrated over a range of Mach numbers from 0.3 to 0.9. All measurements were manually recorded from water manometers and therefore the data density was restricted. Today, the use of electronic pressure transducers combined with automated data logging have transformed the calibration of aerodynamic probes.

As the probe head size is reduced the frequency response of the pressure tappings starts to become a limiting factor. Depolt et al [1990] found that when the probe head diameter was reduced below 2mm the size of the pressure tappings caused the response time of the probe to be unacceptably small.

2.4.1. Four-hole probes

Shepherd [1981] developed the first four-hole probe for measuring flow velocity in three-dimensions. The probe had a prismatic tip shaped similar to a frustum of a pyramid, with three side holes equally spaced around a central hole, figure 2.6. The advantage over conventional five-hole probes was a simpler probe with a smaller head and the avoidance of redundant information. The choice of pyramidal probe over a conical head was aimed to ensure minimal Reynolds number sensitivity by positive location of the flow separation. Shepherd's calibration was undertaken at one velocity of 38ms (Reynolds number based on tip width 8×10^3) and a zoning system was used in the calibration to allow measurements up to $\pm 50^\circ$. This zoning system divided the calibration into six zones depending on the order of magnitude of the three side pressure tapings before a third order polynomial was used to fit the data in each zone.

Hooper and Musgrove [1991] extended the calibration of the Shepherd probe to Mach 0.3. They reduced the zoning system from six to three zones to reduce the problem of matching the calibration surfaces at the zone boundaries. This boundary mismatch was found to introduce large errors when calibration points were sparsely distributed at the boundaries. This probe was used for low speed velocity measurements in industrial swirlers. Musgrove and Hooper [1993] improved the frequency response of the probe to 1.5kHz by incorporating the transducers in probe stem to reduce the distance between the transducer and the pressure tapping. This enabled them to extend their industrial swirler measurements to include turbulence.

The advantages gained by reducing the measurement ports from five to four were rather lost by the increase in complexity of applying the calibration data. Sitaram and Treaster [1985] proposed using the four-hole probe in a similar fashion to the five-hole probe by defining the yaw and pitch coefficients in terms of the face pressures, resolving the faces into the yaw and pitch directions. Using this approach the static pressure is firstly estimated from the average of the pressures from the three side faces:

$$P_M = \frac{(P_B + P_C + P_D)}{3}$$

Equation 2.18

The nomenclature for the faces referred to in equation 2.18 is shown in figure 2.7. This estimate for the static pressure, P_M , is then used in the numerator of the yaw coefficient, equation 2.19. The result is a yaw coefficient which is similar to that used for wedge probes.

$$C_\theta = \left(\frac{P_C - P_B}{P_A - P_M} \right)$$

Equation 2.19

The pitch coefficient is generated by assuming that averaging the pressures of the two side faces B and C will remove the effect of yaw. The difference between the top face, D, and the average of the other two side faces will give a pitch pressure. The pitch coefficient is therefore defined as in equation 2.20.

$$C_\phi = \left(\frac{P_D - \left(\frac{P_B + P_C}{2} \right)}{P_A - P_M} \right)$$

Equation 2.20

By applying the coefficients in this way the benefits of a four-hole probe could be achieved without the complexity of the zoning system. In their paper Sitaram and Treaster go on further to show the sensitivities of a conventional five hole and four-hole probe were very similar.

Cherrett et al [1992] compared the performance of three different types of four-hole probe. The first two probes were manufactured with a pyramid geometry head similar to Shepherd [1981], although one of the probes heads was mounted on a sting to take it forward of the probe stem. The third probe was a wedge probe, but with a fourth pressure tapping located on the head to measure pressure changes in the pitch direction. They found their wedge probe to be less sensitive than the

pyramid geometry probe. Furthermore for the wedge probe, the yaw and pitch coefficients had greater cross sensitivity to pitch and yaw than the pyramid probe. When used for experimental measurements of flow behind a compressor stage they found qualitative agreement between the wedge and the sting mounted probe to be good. The pyramid probe without the sting was found to be less able to resolve the detail of the exit flow field seen by the other two probes. This was considered to be related to the influence of the close proximity of the stem to the probe head.

2.4.2. Seven-hole probes

Flow at high angles will separate from the probe body. Pressure ports in the region of flow separation are insensitive to changes in flow angle and therefore standard five-hole probes cannot be used at high angles of attack. To allow measurement of flow at higher angles of incidence, seven hole probes have been developed, Everett et al [1983]. To use this type of probe, the calibration data is divided up into zones so that the appropriate combination of pressure tapings can be selected for optimum measurement sensitivity. Using this method it was possible to make flow measurements up to angular ranges of $\pm 75^\circ$.

2.4.3. Reynolds number sensitivity

The Reynolds number of the calibration can influence the performance of an aerodynamic probe. For dimensional similarity between a probe during its calibration and a probe used in an experiment, the Reynolds number would need to be included. However to calibrate a probe for direction, Mach number and Reynolds number creates considerable difficulties in finding a calibration facility capable of simulation of all combinations of conditions, and gives a large increase in the number of calibration points needed.

It is therefore advantageous to design an aerodynamic probe so that its aerodynamic calibration characteristic shows minimal Reynolds number sensitivity. Krause and Dudzinski [1969] investigated the influence of Reynolds number on their calibrations over the range of 2×10^3 to 4×10^5 and their calibration data showed a change of 3° in measured pitch at the lowest Reynolds numbers and little influence at the highest.

Dominy and Hodson [1992] undertook a comprehensive examination of the influence of Reynolds number on both conical and prismatic designs of five-holed probes. They highlighted two major Reynolds number effects. The first Reynolds number effect is the separation of the boundary layer from the probe side faces when a probe is at incidence. The position of the separation point is sensitive to Reynolds number and causes the calibration characteristic to vary with Reynolds number.

Dominy and Hodson found that prismatic designs of probes were less sensitive to Reynolds number effects than conical or spherical probes. The advantage of the prismatic face is that the sharp edge fixes the point of separation, whereas on a conical or spherical face the position of separation is not pre-determined. By elimination of the movement of the point of separation about the probe surface, the influence of the Reynolds number on the calibration is minimised.

The second Reynolds number effect, termed 'hole effect', is the result of localised flow patterns around the pressure tappings. These flow patterns, which were demonstrated by flow visualisation, are of a similar magnitude to flow around the whole probe. They found that the choice of hole geometry can reduce this 'hole effect'. When the hole is designed to have pressure tapping perpendicular to the probe surface the Reynolds number sensitivity was less than where the pressure tapping was forward facing.

2.5. Probe blockage

Introducing a probe into a wind tunnel will influence the flow surrounding it because the probe reduces the cross sectional area of the tunnel. The significance of the magnitude of this change will be a function of the size and shape of the probe. This effect is known as probe blockage.

It can be argued that a free jet calibration facility is less affected by probe blockage than an equivalent closed jet facility because the free boundary is free to adjust to any disturbance. However, Wyler [1975], investigated the blockage effects of cylindrical probes in both open and free jet calibration and found that probe blockage in open jets was of an equal magnitude to that of closed

tunnels but in the opposite direction. Wyler derived an expression, equation 2.21, relating the change in local pressure as a result of probe blockage to the Mach number, probe drag and the nozzle area.

$$\frac{\delta P}{P} = -\frac{\gamma}{2} \left(\frac{M^2}{1 - M^2} \right) C_D \left(\frac{S}{F} \right)$$

Equation 2.21

Wyler used a polynomial fit for the drag of a cylindrical probe to give estimates of the percentage error in pressure as a function of Mach number and the probe size as shown in figure 2.8.

Broichhausen and Franson[1984] reported on a series of experiments on a two-dimensional wedge probe, calibrated in a variety of calibration facilities. These experiments were to measure the differences in the calibrations from different facilities, both open jets and closed tunnels. The results showed a maximum deviation of 3.5 degrees in the angle calibration and but found a 6% deviation in the Mach number.

The blockage effect, as predicted by Wyler, will occur locally to the source of the blockage. The blockage decreases with distance from the probe in a linear manner and Wyler quotes a reduction to 40% of its original value for a distance of one half tunnel diameter. This is why sharp needle type probes can be used with minimal blockage effect. One of the advantages of the conical and pyramid geometries over the wedge and cylinder geometries is that it is possible to move the sensor head forward from the stem using a sting and thereby reduce the blockage effect.

2.6. Semiconductor probes

Amongst the first applications of semiconductor pressure transducers to aerodynamic probes was the work of Senoo et al [1973], figure 2.9. They took a conventional cobra probe and connected each of the pressure tubes to a semiconductor incorporated in the stem of the probe. This probe was used to make two-dimensional measurements of the flow from the exit of a pump impeller. Although the bandwidth of the transducers was up to 40 kHz, the frequency response of the probe was found to

only be up to 1.5kHz. This was because the volume between the pressure port and the pressure transducer influences the transmitted pressure signal, restricting the bandwidth.

The first move towards surface mounted transducers was started by Kerrebrock et al [1980] who developed a spherical probe with five flush mounted silicon pressure sensors for three-dimensional velocity and pressure measurements. With the transducer mounted flush with the surface, the full bandwidth of the transducer was available for measurement. The semiconductor transducers were of an early design and suffered from thermal drift to as much as ten percent of full scale. This temperature sensitivity was very restrictive and prevented the probes use for steady state measurements. In order to generate a calibration, a pneumatic model of twice the scale of the semiconductor probe was used and the data from this calibration applied to the semiconductor probe measurements.

From the experience gained with the spherical probe, Epstein [1985] developed a four port cylindrical probe in which to overcome the temperature problems, a cooling mechanism was incorporated into the probe head. This probe is shown in figure 2.10. The cylindrical probe allowed good spatial resolution in the circumferential direction at the cost of poorer spatial resolution in the radial direction. As turbomachines generally have greater circumferential gradients than radial gradients this gave a satisfactory compromise and allowed a smaller size than the equivalent spherical probe.

Rather than mounting multiple transducers on one probe head Shreeve and Neuhoft [1984] obtained two-dimensional wake measurements using a dual probe digital sampling technique. Their probes were simple impact probes incorporating a single semiconductor transducer. To resolve the flow the probes were rotated through a series of yaw angles. The measurements were then combined as if it had been recorded by a four transducer probe. This technique, however, necessitates sequential data acquisition and ensemble averaging to pre-process the data. This may not be appropriate for unsteady flows.

Elmendorf and Kauke [1985] designed a wedge probe incorporating commercially available semiconductor transducers for use in supersonic flows. The first probe was designed with a front face transducer without any protective shielding but was destroyed in the harsh environment behind a compressor stage. Temperature effects were again a problem and so the time averaged data was measured pneumatically with external transducers. Since the probe was used in supersonic flows the yaw calibration was derived theoretically assuming the shock expansion equations were valid.

Bubeck and Wachter [1987] attempted to address the problem of transducer drift using pneumatic tappings to measure the steady component of the signal combined with flush mounted transducers for the unsteady component. Their design was based on the geometry of a wedge probe with fourth face to allow three-dimensional measurement. The probe had semiconductor transducers only on the side faces and none on the front face. Whilst this gave them the advantage of a smaller probe and reduced risk of damage to the front transducer, it required measurement at two different positions to give the four independent pressure values required to resolve the flow.

To overcome the same problem of temperature drift affecting steady flow measurements, Ruck and Stetter [1990] decided to use two similar probes, the first for steady component and the second for unsteady component. For the unsteady measurement, the pressure transducer was not mounted flush with the probe surface but within the probe head figure 2.11, behind a small pressure tapping. For the steady measurements the transducers were mounted externally. Mounting the transducer in the probe head prevented damage to the transducer but this had the disadvantage of creating a resonant cavity caused by the finite volume formed in front of the diaphragm. The transfer characteristic of the cavity had to be determined and used to correct all measured experimental results. This additional complexity was clearly undesirable.

Custom transducers were manufactured by Kulite to allow Cook [1989] to construct a 30° fast response wedge probe. This probe used an active temperature compensation technique in which the temperature behaviour of each transducer was recorded on an Erasable Programmable Read Only

Memory (EPROM). The data on the EPROM was used to correct temperature changes during the experiment by feedback to the transducer amplifier zero offset and the transducer current source.

Ainsworth et al [1991] developed a technique whereby a semiconductor transducer could be mounted directly on the surface of metal turbine blades. When applied to the construction of fast response probes it was found to provide a flexible method for manufacturing probe geometries of both two-dimensional wedge, Ainsworth et al [1992], and three-dimensional pyramid types, Ainsworth and Batt [1993b]. The first measurements with probes constructed with these techniques, which were of flow in a high pressure turbine, were published in 1994, Ainsworth et al [1994].

Cherrett et al [1994] used the same probe design as Cook [1989] for unsteady yaw measurements in an axial compressor. Rather than the active correction technique used by Cook, a passive temperature correction technique based on the current drawn by the transducer, was used. Comparison of the time averaged semiconductor probe measurements was favourable with pneumatic measurements taken in the same facility.

2.7. Dynamic effects

Aerodynamic probes are calibrated in steady state calibration facilities to find their characteristics and then introduced into unsteady flows in turbomachines for measurement. Relatively little research has been done to find if the unsteady environment affects the steady state calibration.

Senoo et al [1973] considered the validity of applying static calibrations in dynamic flows by undertaking their experiments at a variety of flow speed and probe incidences. They compared the results when plotted in non-dimensional form and corrected for offset angles. As they found that their experimental results were principally the same for all speeds and probe incidences, they concluded that the results were uninfluenced by dynamic effects over the range of their experiments and that the static calibration was applicable in the dynamic case.

Kovaszny et al [1981] attempted to quantify the magnitude of inertial effects on a probe and studied the instantaneous pressure distribution around a sphere in unsteady flow. They developed an

unsteady potential flow solution by combining the potential function for flow around a sphere with a two-dimensional disturbance to give the pressure on a sphere in unsteady flow in terms of the instantaneous dynamic pressure and the time derivative of the velocity. They undertook experimental work in which a 6.35mm diameter sphere was subjected to an unsteady flow to allow comparison of the experimental results with their unsteady potential flow theory. Their experimental technique was restricted by the available pressure instrumentation which consisted of a remote microphone whose phase lag was corrected through estimates from a hot wire probe. From these results they concluded that the inertial component could contribute up to 10-20% of the measured pressure.

Recent work published by Humm et al [1994] has suggested that there may be significant errors associated if static calibrations are used for recording unsteady flows. Experiments were carried out in which a variety of geometries of probes were oscillated while moving in a water channel. A comparison of the static calibration with the dynamic calibration indicated a considerable disagreement between the two calibrations for a range of probe geometries. They proposed that causes of probe calibration disagreements included dynamic stall, inertia effects, dynamic boundary layers, vortex shedding and hysteresis.

It is difficult to quantify the magnitude and significance of these dynamic effects. In chapter 7 aerodynamic probe measurements in an unsteady rotor turbine will be shown to have good agreement with computational predictions of the unsteady flow. For these measurements emphasis has been placed on ensuring that the bandwidth of the probe does not restrict the measurements, but no further consideration for dynamic effects has been taken.

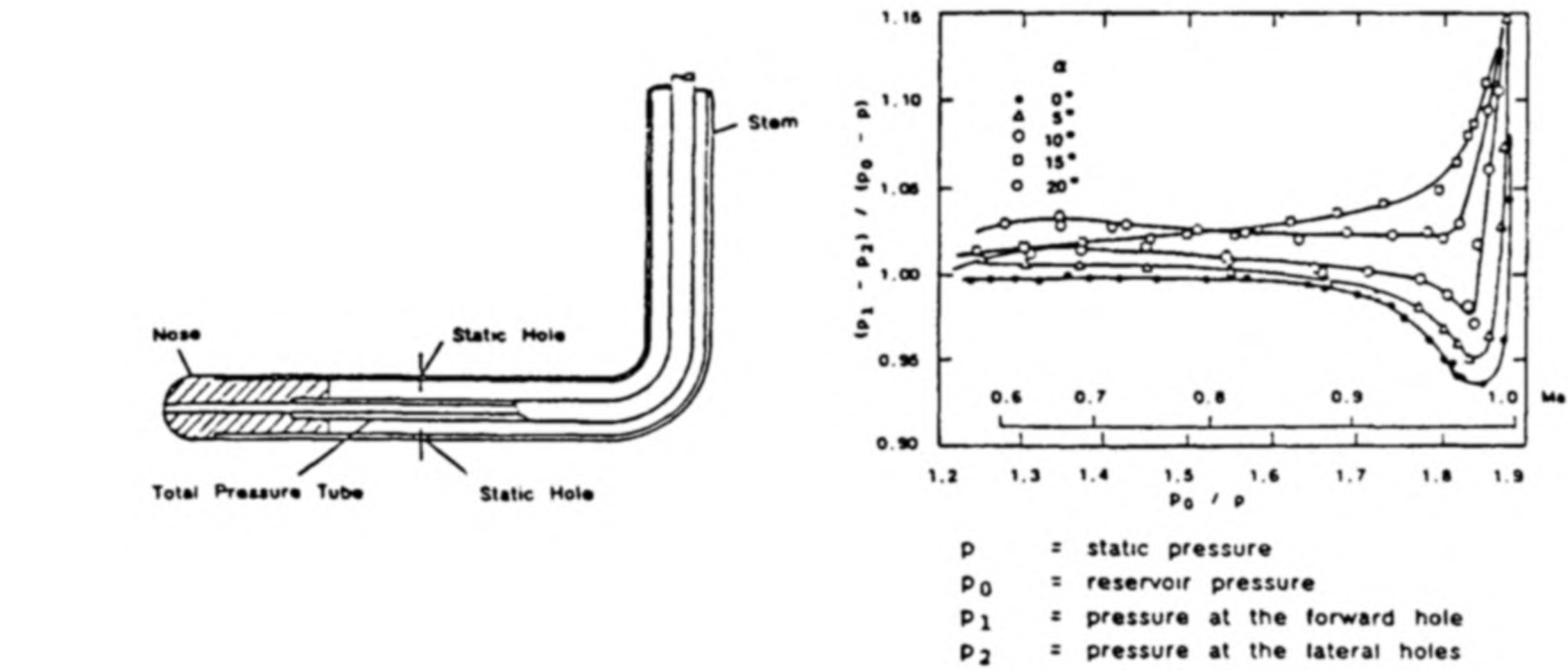


Figure 2.1 A Pitot-Static probe

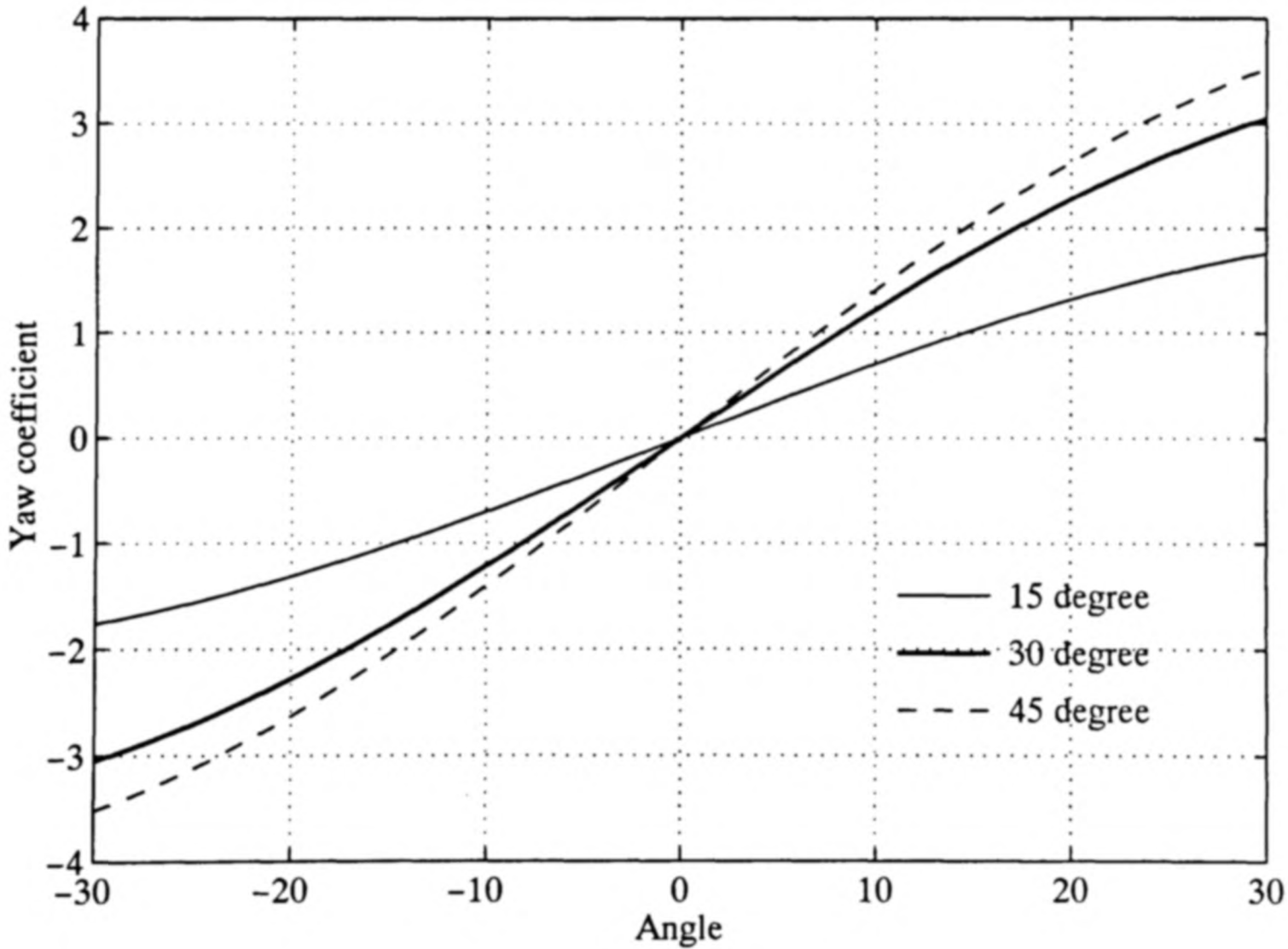


Figure 2.2 Potential flow prediction of yaw calibration

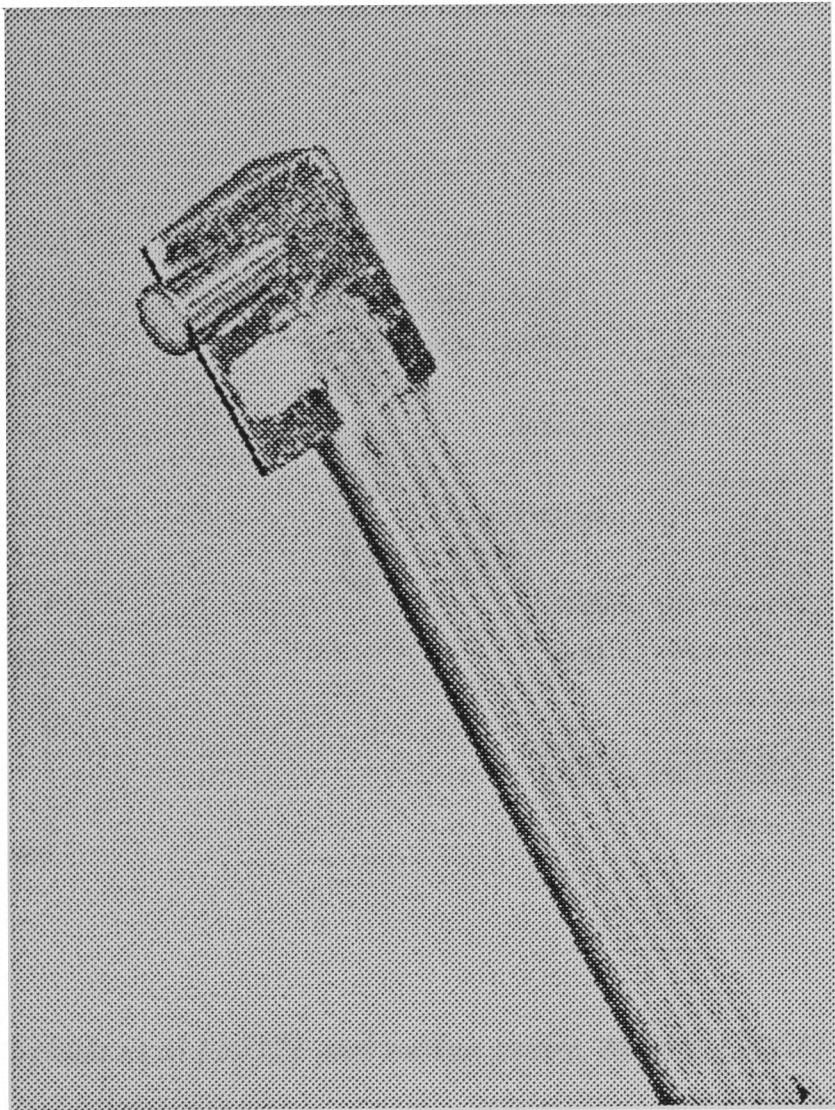


Figure 2.3 Semiconductor wedge probe

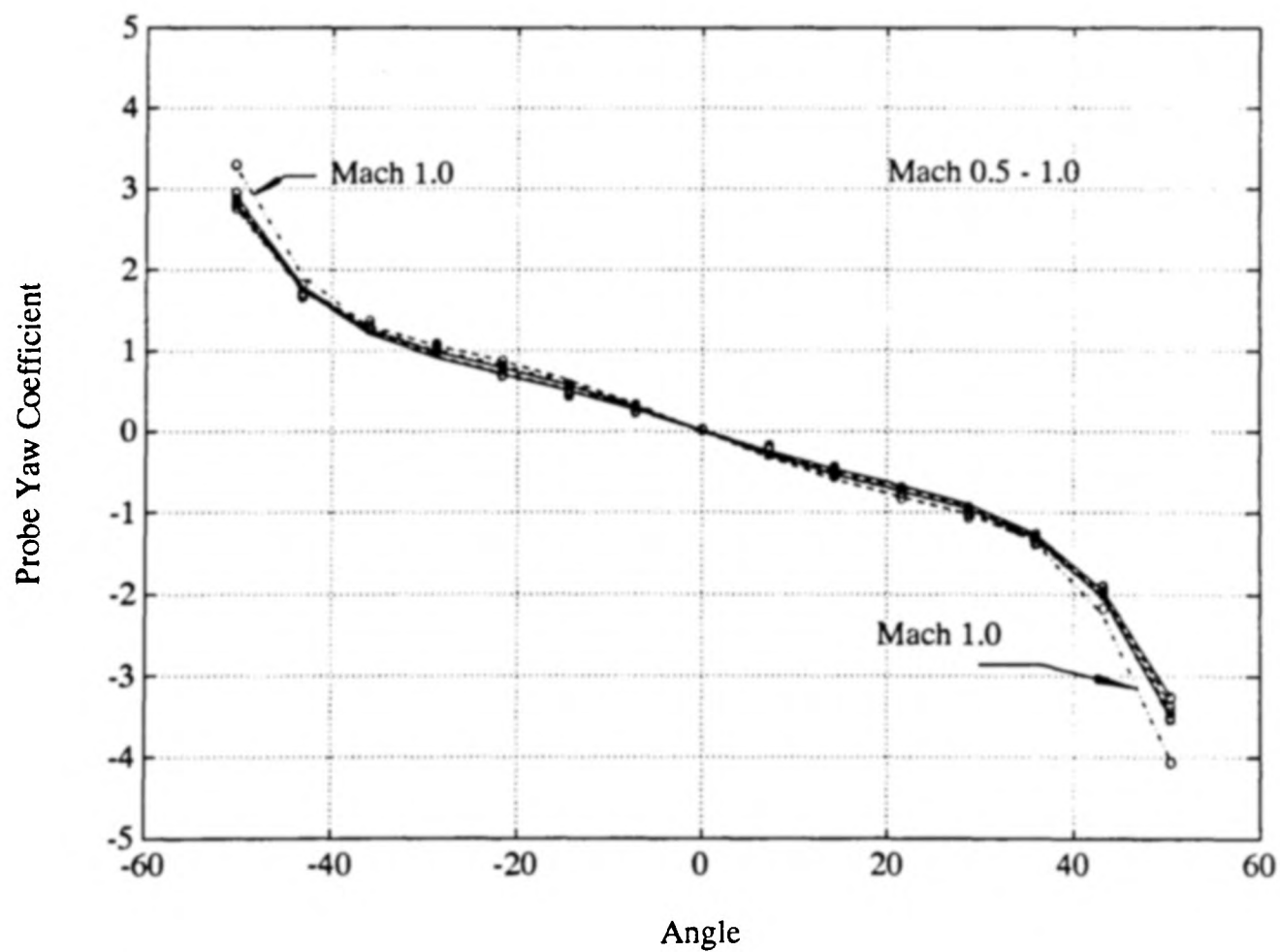


Figure 2.4 Yaw coefficient, Ainsworth et al[1994]

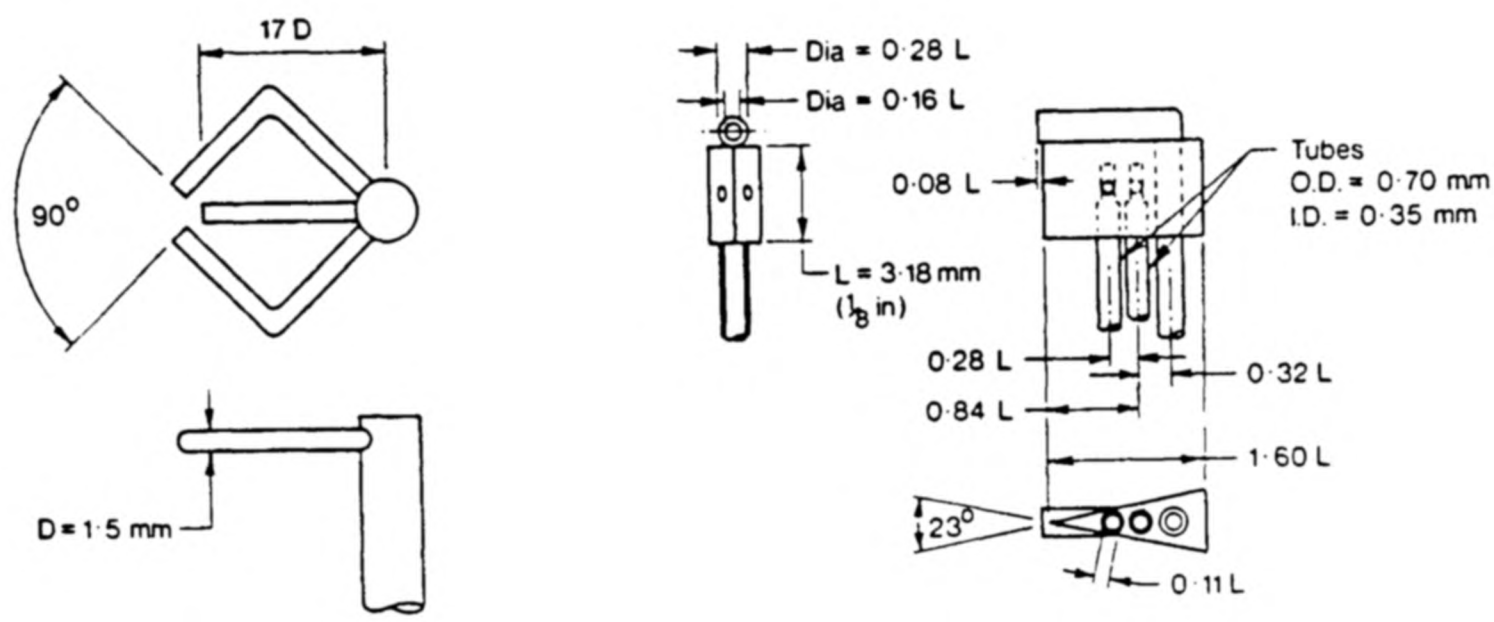


Figure 2.5 Claw probe, Schulz et al [1952] and wedge probe, Morris [1961]

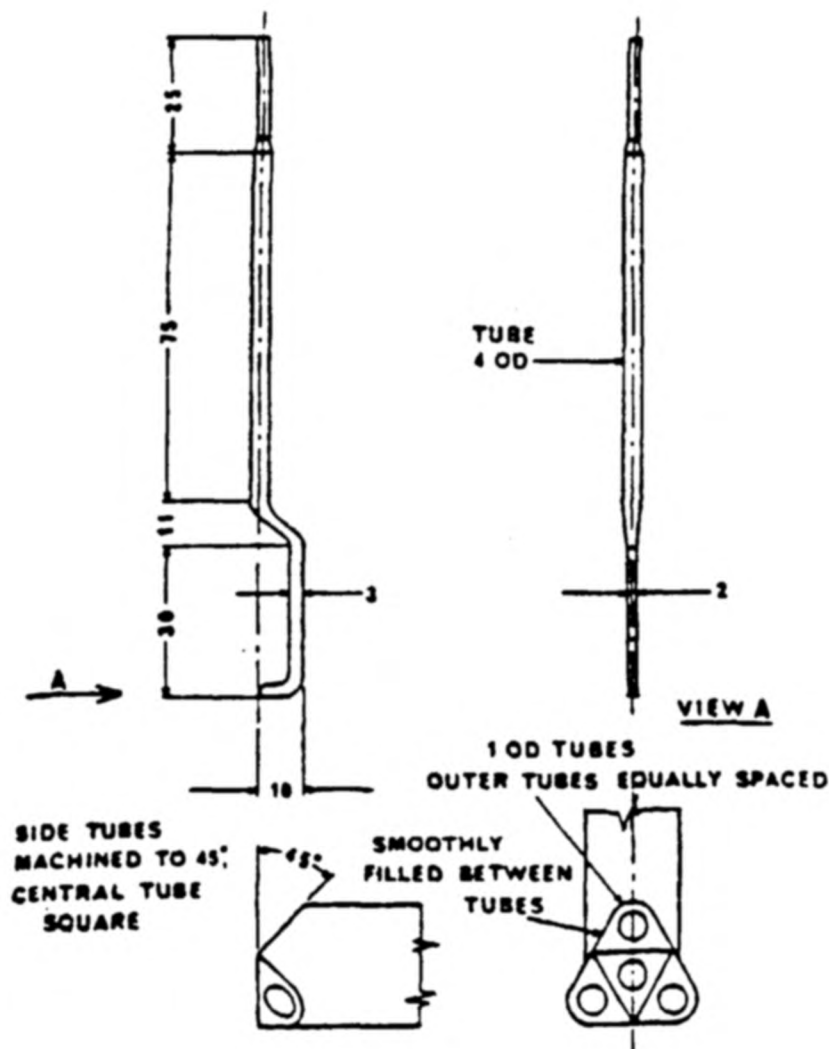


Figure 2.6 Pyramid probe, Shepherd [1981]

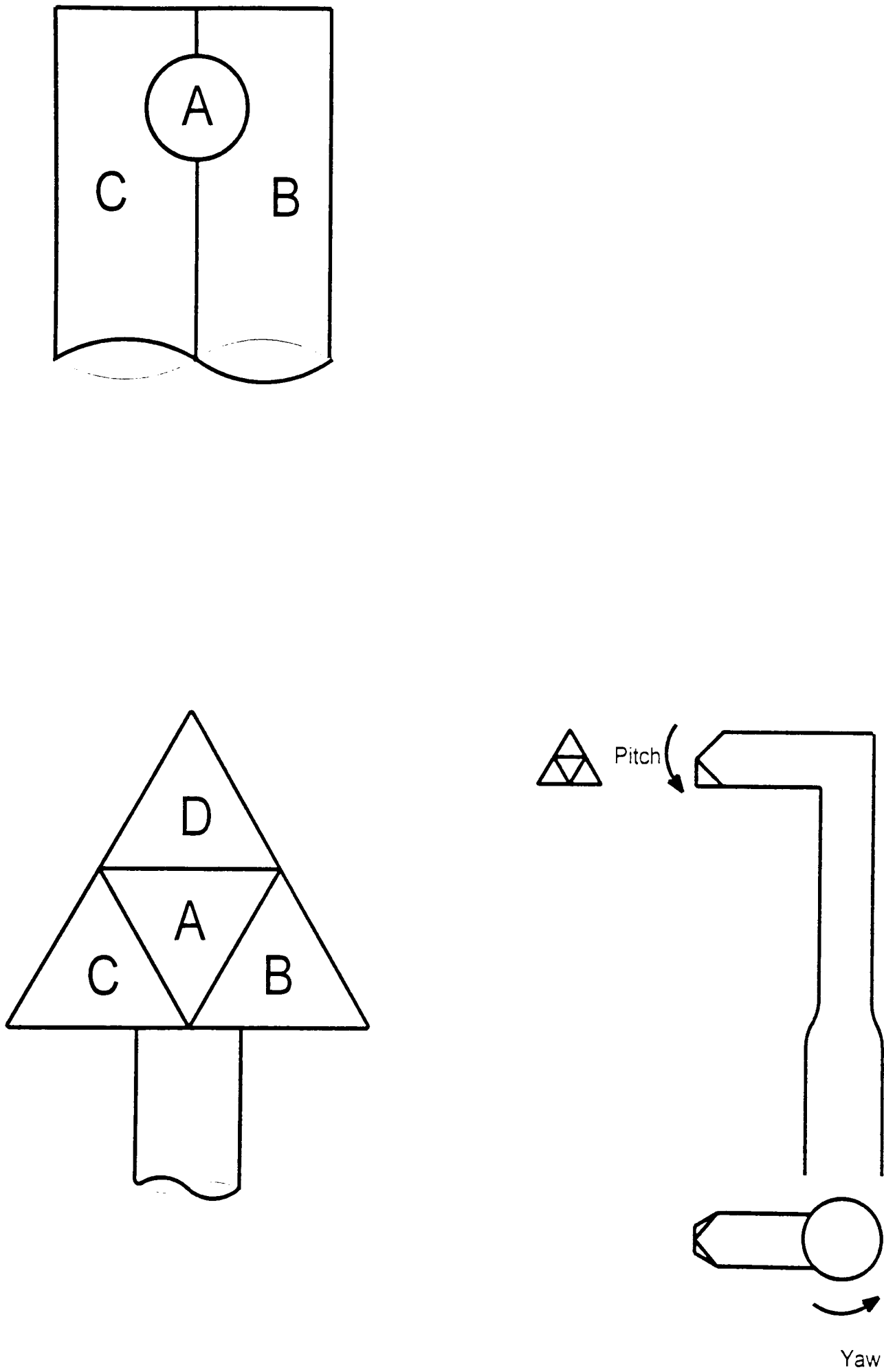


Figure 2.7 Wedge and pyramid probe face nomenclature

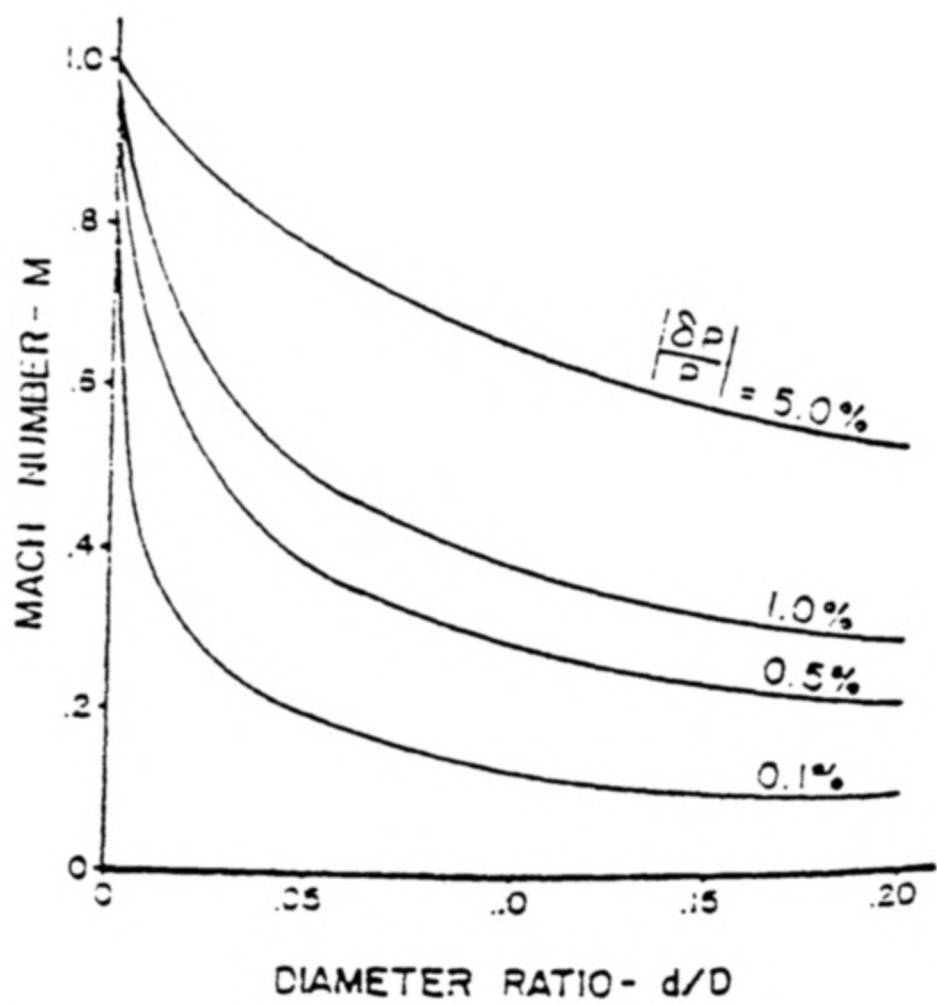


Figure 2.8 Probe blockage, Wyler [1975]

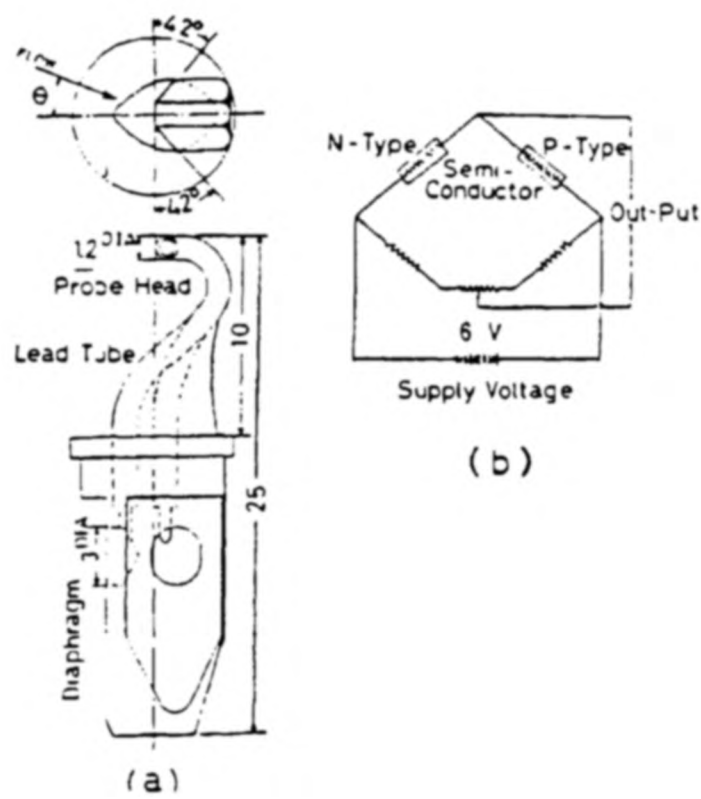


Figure 2.9 Semiconductor cobra probe, Senoo et al [1973]

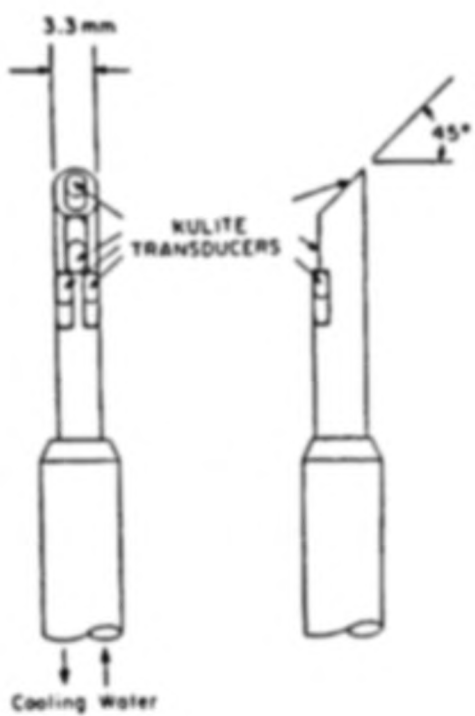


Figure 2.10 Cylindrical probe, Epstein [1985]

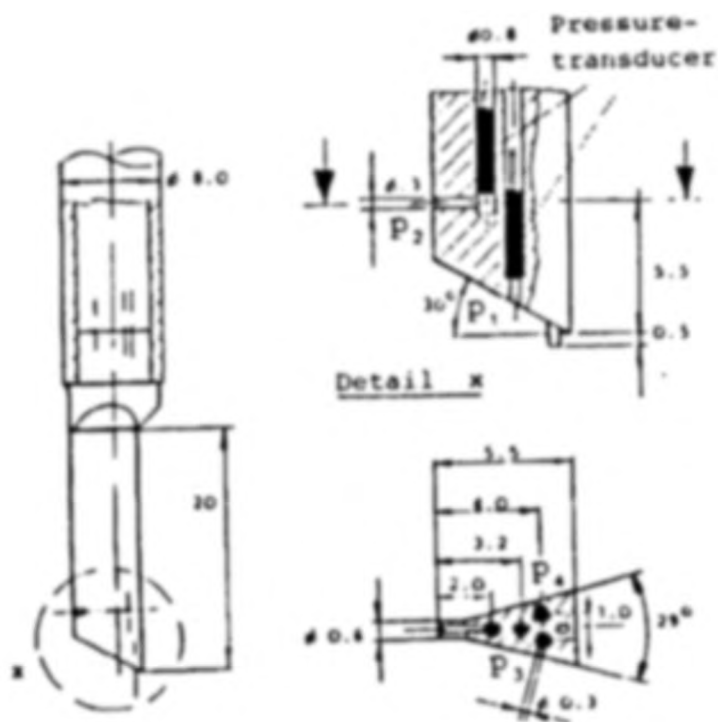


Figure 2.11 Four port wedge probe, Ruck and Stetter [1990]

3. Electrical calibration

3.0. Overview

The use of semiconductor based pressure instrumentation in turbomachinery has suffered from poor stability of the transducers, particularly as a result of temperature sensitivity. In this programme of work, semiconductor transducers are used to provide the fast response times required for unsteady flow measurements. Using a novel mounting technique developed in Oxford the transducers are mounted directly on a metal surface which gives considerable flexibility to their location. This chapter details the electrical calibration which is required for the transducers when they are mounted in this fashion. The transducers are calibrated for temperature sensitivity in addition to pressure sensitivity so that unwanted temperature effects may be removed from the measurements. A bridge circuit is used to allow temperature variation of the transducer to be monitored. This provides the means for temperature correction of the transducer's output.

3.1. *The semiconductor pressure transducer*

The semiconductor strain gauge pressure transducer, figure 3.1, is a highly robust linear pressure sensor. The transducer consists of a silicon diaphragm which is bonded to a supporting pillar, figure 3.2. The underside of the diaphragm is micro etched creating an evacuated cavity between it and the supporting pillar, whilst a Wheatstone resistor bridge is diffused onto the upper side. When the diaphragm deflects under pressure, the strain in the resistors causes them to change in value. This change in resistance causes a difference between either side of the bridge and therefore provides an output voltage. By careful matching and positioning of the resistors, good linearity with little hysteresis can be achieved. These have been successfully used in a variety of applications and have proved to be particularly suitable for use in turbomachinery research.

Care must be taken to avoid the effects of changes in output from the semiconductor transducer due to anything other than pressure since these would clearly cause errors. The potential reasons for errors include non linearity, pressure hysteresis, drift, and thermal hysteresis, Greenwood

[1983]. Three decades of experience in gauge design have allowed commercial manufacturers to reduce their influence.

In practice, the main source of error with the current designs of transducers is caused by the influence of temperature on the output voltage. The transducer designer attempts to minimise any temperature variation by careful matching of the temperature coefficients of resistance of the bridge elements. Additional temperature compensation can be applied with external circuitry to the transducer bridge and in commercial transducers, this hardware temperature compensation is included in the packaging of the transducer.

The environment in the turbomachine can contain a wide range of conditions and the temperature effects on transducers have been restrictive to the use of semiconductor pressure transducers. Kerrebrock et al [1980] suffered transducer drift and the probe could only be used for unsteady measurements. Epstein [1985] incorporated a cooling system in the probe head to remove the heat dissipated from integral transducers. Bubeck and Wachter [1987], and Ruck and Stetter [1990] both chose to superimpose pneumatic pressure measurements to their unsteady measurements from their semiconductor probes. Even using an active temperature compensation technique, Cook [1989], the transducer output still suffered from a long term drift.

3.2. Surface mounted transducers

The technique of mounting a semiconductor pressure transducer directly on a surface was developed at Oxford University in association with Kulite Semiconductors to allow detailed unsteady pressure measurements to be taken over aerofoil surfaces, Ainsworth et al [1991]. It had been found that the packaging associated with an off the shelf transducer interfered with the profile of the aerofoil and did not allow sufficient measurement point density. Therefore the standard packaging for the chip, incorporating any compensation hardware, was removed to minimise the transducer's size and the semiconductor chip mounted on the aerofoil. A compensation circuit external to the transducer was used to avoid a potential problem with temperature effecting the transducer output.

The method by which a transducer is mounted is very important for its temperature sensitivity because thermal stresses between the mounting surface and the transducer chip can cause considerable measurement errors. To mount the transducer, a small slot is spark eroded into the surface of the blade and the blade surface covered with an insulating enamel, figure 3.3. The insulating enamel allows gold tracks to be laid on the surface and this provides a method of getting electrical connection to the semiconductor sensor, figure 3.4.

The slot provides the location for the transducer which is then secured by an adhesive. The adhesive is carefully chosen so that the semiconductor chip is secured but the thermal stresses are not transmitted between bonding surface and the chip. A ball bonded joint provides the connection between the transducer and the gold track. Finally, a thin layer of elastomer is placed over the chip face to protect the delicate connections and the transducer from impact. The elastomer layer also provides an amount of damping to the transducer diaphragm which in its natural state is underdamped. Careful choice of the elastomer thickness prevents a sharp resonance that would otherwise occur at the diaphragm's natural frequency (400kHz) while still providing sufficient bandwidth for measurements, Ainsworth and Allen [1990].

Further details of the surface mounting technique and temperature compensation are given in Ainsworth et al [1991], and an example of a turbine blade instrumented with surface mounted pressure transducers is shown in figure 3.5.

An aerodynamic probe needs to be robust, small and requires multiple pressure measurements within a confined area. The surface mounted transducer technique gives considerable flexibility to where a transducer can be positioned and therefore provides an ideal technique for the construction of aerodynamic probes. Therefore, by taking conventional pneumatic probe geometries and using the surface mounting technique developed for aerofoils, a semiconductor probe can be constructed.

3.3. *Temperature compensation*

The simplest hardware compensation scheme may be achieved with the use of a resistor in series with the bridge, known as a span resistor, to correct overall span variation of the transducer. This scheme works only if the bridge output is decreasing with temperature. At the same time as any output drop with temperature, the overall bridge resistance will be increasing. The voltage distribution with the series resistor will therefore act to increase voltage across the bridge, increasing the bridge output and thereby compensating for the span variation. Careful selection of the value of the series resistor will minimise any temperature variation.

In wide bandwidth applications, the data processing overheads are relatively high and there is therefore a requirement to ensure any correction scheme is not over complicated. For the surface mounted transducers, correction is undertaken by using the semiconductor resistance-temperature characteristic to measure the chip temperature and then applying a pre-determined calibration for that temperature.

The resistance of doped silicon increases with temperature because of the decreased mobility of the charge carriers and is a direct function of temperature. By using a circuit in which the current through the bridge can be measured, the resistance of the bridge may be found and from this the temperature derived.

Practical implementation of this technique involves taking the raw measurements from the transducer and applying corrections to the measured voltage so that it always appears that the experiment was done at 25°C. By making the corrections to the measured voltage rather than to the sensor calibration, it was possible to isolate any software that which further processed the results from the need to consider temperature effects.

3.4. *Derivation*

The circuit configuration shown in figure 3.6 is used to power each transducer. The voltage supply to this circuit is regulated using a voltage regulator circuit shown in figure 3.7. Initially, the

temperature sense voltage, V_{sense} , is trimmed to zero and the transducer behaves as if it had been powered up directly to the voltage supply. It is only when the temperature changes that the external circuit has an effect.

Figure 3.8 shows the relationship between transducer voltage, pressure and temperature for a typical Kulite transducer. The high degree of linearity associated with the Kulite transducer is clear and the pressure may be modelled by a linear equation:

$$V_{25} = S_{25}P + O_{25}$$

Equation 3.1

In this equation the reference span, S_{25} , is the sensor span in volts per Pascal at 25°C and the offset, O_{25} , is the transducer voltage output referenced to absolute zero pressure at 25°C.

The influence of temperature on the transducer output may be shown by plotting the variation of the transducers span and offset with temperature as shown in figure 3.9 and 3.10. The linear behaviour of the pressure and temperature sensitivity allows changes in the transducer output to be considered incrementally. This allows linear superposition of any correction to the output. The variation may be summarised by defining two coefficients, the fractional span sensitivity and the offset sensitivity:

$$\text{Fractional span sensitivity} = \frac{1}{S_{25}} \cdot \frac{dS}{dT}$$

Equation 3.2

$$\text{Offset sensitivity} = \left[\frac{dO}{dT} \right]$$

Equation 3.3

Note the sign of fractional span and offset sensitivity will be semiconductor chip dependent and may not necessarily be of the same sign as shown in the plots here.

The bridge in series with the span resistor acts as a potential divider. A span resistor with a very low temperature coefficient of resistance is chosen so that its resistance is virtually independent of temperature. Furthermore, the span resistor is external to the transducer so that it is isolated from any influences on the temperature of the transducer from its environment.

In contrast the bridge resistance will vary with temperature and any small changes in the bridge resistance will be accurately measured by the temperature sense voltage, V_{sense} . This voltage change is therefore a measure of the transducer temperature.

The variation of bridge resistance with temperature of the bridge is normally linear and therefore the relationship between the temperature sense voltage and temperature may be derived by considering a first order increase in resistance of the bridge with temperature.

$$R_b = R_{b25} \times (1 + \alpha \Delta T)$$

Equation 3.4

At 25°C the span resistance, R_s , is adjusted so the temperature sense voltage, V_{sense} , is zero and therefore the span resistance will have the same value as the initial bridge resistance, R_{b25} . Any change in the bridge resistance will result in the voltage across the span resistor changing to:

$$V_{span} = \frac{R_s}{R_s + R_{b25}(1 + \alpha \Delta T)} \times V_o$$

Equation 3.5

This gives rise to a temperature sense voltage, V_{sense} :

$$V_{sense} = V_{span_i} - V_{span} = V_o \times \left[\frac{R_s}{R_s + R_{b25}} - \frac{R_s}{R_s + R_{b25}(1 + \alpha \Delta T)} \right]$$

Equation 3.6

Rearranging for T using the relationship that $R_s = R_{b25}$:

$$\Delta T = \frac{1}{\alpha} \times \left[\frac{4V_{sense}}{V_o - 2V_{sense}} \right]$$

Equation 3.7

Since $V_{sense} \ll V_o$ for small temperature changes, this may be approximated as :

$$\Delta T = \frac{1}{\alpha} \times \left[\frac{4V_{sense}}{V_o} \right]$$

Equation 3.8

The temperature sense voltage therefore gives a convenient method of measuring any temperature change on the transducer.

The voltage from the transducer at a temperature ΔT above 25°C may be written as:

$$V_{25+\Delta T} = \left(S_{25} + \frac{dS}{dT} \Delta T \right) P + \left(O_{25} + \frac{dO}{dT} \Delta T \right)$$

Equation 3.9

From equations 3.9 and 3.1, it may be shown that the voltage that would have been measured if the experiment had been undertaken at 25°C is given by the following expression.

$$V_{25} = \frac{V_{25+\Delta T} + O_{25} \cdot \left(\frac{1}{S_{25}} \cdot \frac{dS}{dT} \right) \cdot \Delta T - \frac{dO}{dT} \cdot \Delta T}{\left[1 + \left(\frac{1}{S_{25}} \cdot \frac{dS}{dT} \right) \cdot \Delta T \right]}$$

Equation 3.10

This equation may be simplified if the transducer offset, O_{25} is near zero and the second order terms are ignored to give:

$$V_{25} = V_{25+\Delta T} \cdot \left[1 - \left(\frac{1}{S_{25}} \cdot \frac{dS}{dT} \right) \cdot \Delta T \right] - \frac{dO}{dT} \cdot \Delta T$$

Equation 3.11

3.5. *Electrical calibration*

To apply this temperature compensation technique, each transducer is calibrated to find its individual electrical calibration properties. The transducers are calibrated over a wide range of pressures and temperatures using an automated calibration facility originally developed by D. Layfield. A Druck DPI501 digital pressure transducer, linked to an environmental chamber is controlled via the IEEE bus interface by a dedicated computer. This is used to run the transducer through a standardised cycle. The pressure sensing element for the DPI501 is a vibrating cylinder pressure transducer manufactured by Schlumberger measuring to an accuracy of 0.007% full scale.

During the calibration the transducer is cycled from 200mB to 3500mB in five steps. This pressure cycle is repeated over temperatures from 5°C to 40°C, returning three times to 25°C to test for hysteresis and repeatability. Allowing for generous settling time, this whole test cycle lasts approximately 14 hours. This length of test cycle also serves to establish the integrity of each transducer and any unreliable transducer can be rejected.

3.6. *Reference conditions*

The values of the temperature sensitivities given are specified at reference conditions of 25°C and the offset sensitivity has been referenced to zero pressure. However it may not always be convenient to reference the use of the transducers to these conditions. It was found that for aerodynamic calibration, it is more convenient to record the initial conditions at 1 bar rather than evacuating the entire calibration tunnel. In contrast the evacuated tunnel before an experiment on the Oxford rotor facility provides a convenient time to record the reference pressure and temperature, but the ambient temperature was generally found to be considerably below 25°C.

This was overcome by adjusting the sensitivity values found from electrical calibration to the reference temperature recorded before the experiment. The four steps required for these adjustments are as follows:

STEP 1 Measure ambient temperature T , pressure P , and V_0 from the transducer.

STEP 2 The span at the reference temperature may be estimated from the expression given in equation 3.12.

$$S_R = S_{25} \times \left[1 + (T - 25) \times \left(\frac{1}{S_{25}} \cdot \frac{dS}{dT} \right) \right]$$

Equation 3.12

However this expression assumes that the voltage across the bridge is the unchanged from when the temperature sense voltage was trimmed to zero at 25°C during calibration. In practice the bridge voltage will differ by the value of temperature sense voltage recorded during calibration at the equivalent of the ambient temperature, T . Therefore the transducer span was calibrated in situ before each experiment to find the new reference span S_R .

STEP 3 The fractional span sensitivity at this reference temperature was found from the value specified at 25°C.

$$\frac{1}{S_R} \cdot \frac{dS}{dT} = \frac{S_{25}}{S_R} \times \left(\frac{1}{S_{25}} \cdot \frac{dS}{dT} \right)$$

Equation 3.13

STEP 4 Finally the zero offset O_R , for the reference conditions was calculated

$$O_R = V_R - S_R P$$

Equation 3.14

3.7. Temperature correction

Variations resulting from temperature changes during the experiment were corrected by measuring the temperature sense voltage, V_{sense} , during the experiments and then applying corrections based on the prior temperature calibrations.

STEP 1 The reference conditions for the experiment were recorded by trimming the temperature sense voltage, V_{sense} , to zero and measuring the temperature T , pressure P , and transducer voltage, V_R .

STEP 2 The transducer pressure sensitivity was calibrated in situ at this temperature. The reference values for fractional span sensitivity and zero offset were then found using equations 3.13 and 3.14.

STEP 3 The experimental measurements were recorded.

STEP 4 From the temperature sense voltage measured over the experiment, the temperature changes were derived using equation 3.7. During aerodynamic calibration this correction was displayed on line to aid the diagnosis of faults.

STEP 5 The voltage that would have been measured if the transducer had recorded the data at the reference temperature, V_R , was found by allowing for the change in offset and sensor span during the experiment, as in equation 3.15.

$$V_R = \frac{V_{R+\Delta T} + O_R \cdot \left(\frac{1}{S_R} \cdot \frac{dS}{dT} \right) \cdot \Delta T - \frac{dO}{dT} \cdot \Delta T}{\left[1 + \left(\frac{1}{S_R} \cdot \frac{dS}{dT} \right) \cdot \Delta T \right]}$$

Equation 3.15

STEP 6 With the output voltage from the chip corrected for temperature variation the reference span and offset can be used to find the true pressure, equation 3.14.

$$P = \frac{1}{S_R} \times [V_R - O_o]$$

Equation 3.16

3.8. Transducer performance

The figures in table 3.1 give an overall guide to the performance of the Kulite transducers when they are mounted on a metal surface using the ‘chip on’ technique. The figures in this table are the values found for one of the transducers on the three-dimensional probe described later in this thesis and are typical of the other transducers.

Supply voltage		5 V
Resistance		1.1 kΩ
Power		0.023 W
Span	S_{25}	$1.567 \times 10^{-7} \text{ V Pa}^{-1}$
Span sensitivity	$\frac{1}{S_{25}} \cdot \frac{dS}{dT}$	$5.66 \times 10^{-2} \% \text{ C}^{-1}$
Offset sensitivity	$\left[\frac{dO}{dT} \right]$	$2.49 \times 10^{-2} \text{ mV C}^{-1}$
Temperature coefficient of resistance	α	$1.80 \times 10^{-3} \text{ C}^{-1}$
Hysteresis		0.185% FS
Span repeatability		0.11%
Temperature error	before temperature compensation:	1.46% FS
	after temperature compensation:	0.073% FS
Overall error	before temperature compensation:	1.48% FS
	after temperature compensation:	0.226% FS

Table 3.1

There main sources of error have been divided into three components, namely hysteresis, variation in sensitivity and temperature effects. The temperature effects include both the span variation

and offset variation, which have been derived assuming a 20° change in temperature. Full scale for the transducer has been taken to be 3.5×10^5 Pa to reflect the operating range of the Oxford rotor experiment. Assuming that temperature error, hysteresis and sensitivity variation are independent an overall error can be derived by finding the root of the mean square of the individual error components, in this case 1.48%. The major component of this error is a result of temperature variation which can be removed by the temperature correction techniques described earlier.

The values for the span and offset sensitivities have been both found to be repeatable to 5%. This figure may sound high but reflects the difficulty of measuring and controlling the temperature of the transducer during calibration. The transducer temperature will be above its surroundings because of ohmic heating. However the error in the sensitivities has a second order effect on the overall error and by correcting the temperature effects the overall error drops to 0.226%. This is clearly a considerable improvement on the uncorrected value. In absolute terms the pressure error from the transducer has been reduced from 5170 Pa to 790 Pa. The influence of this error on the performance of an aerodynamic probe will be seen in chapter 5.

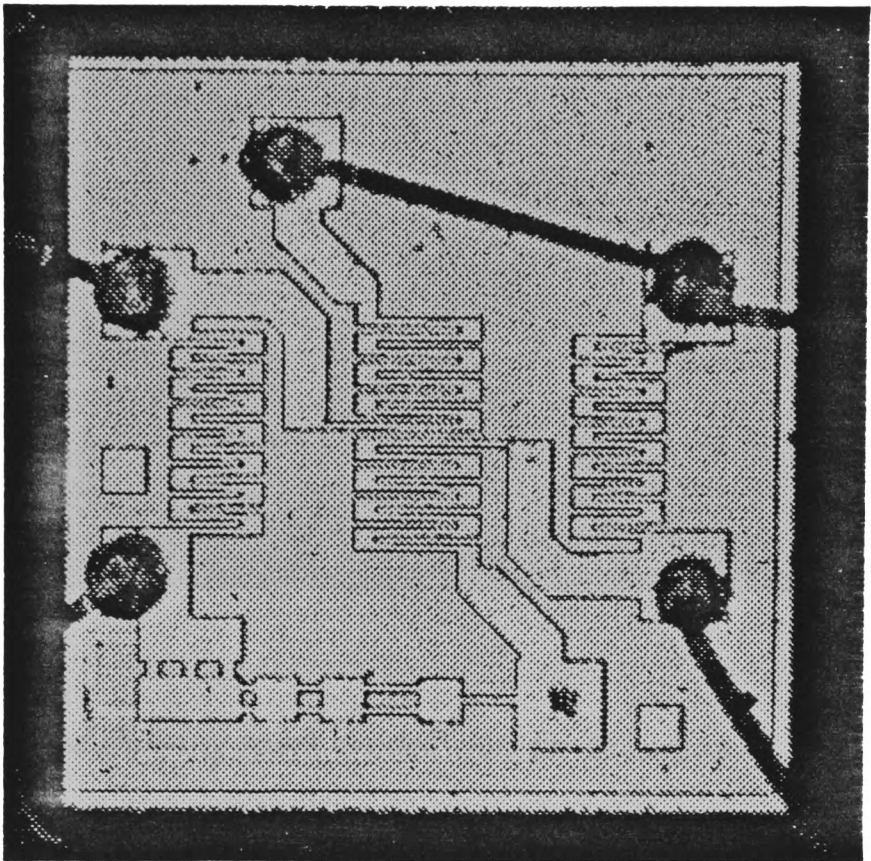


Figure 3.1 Semiconductor pressure transducer

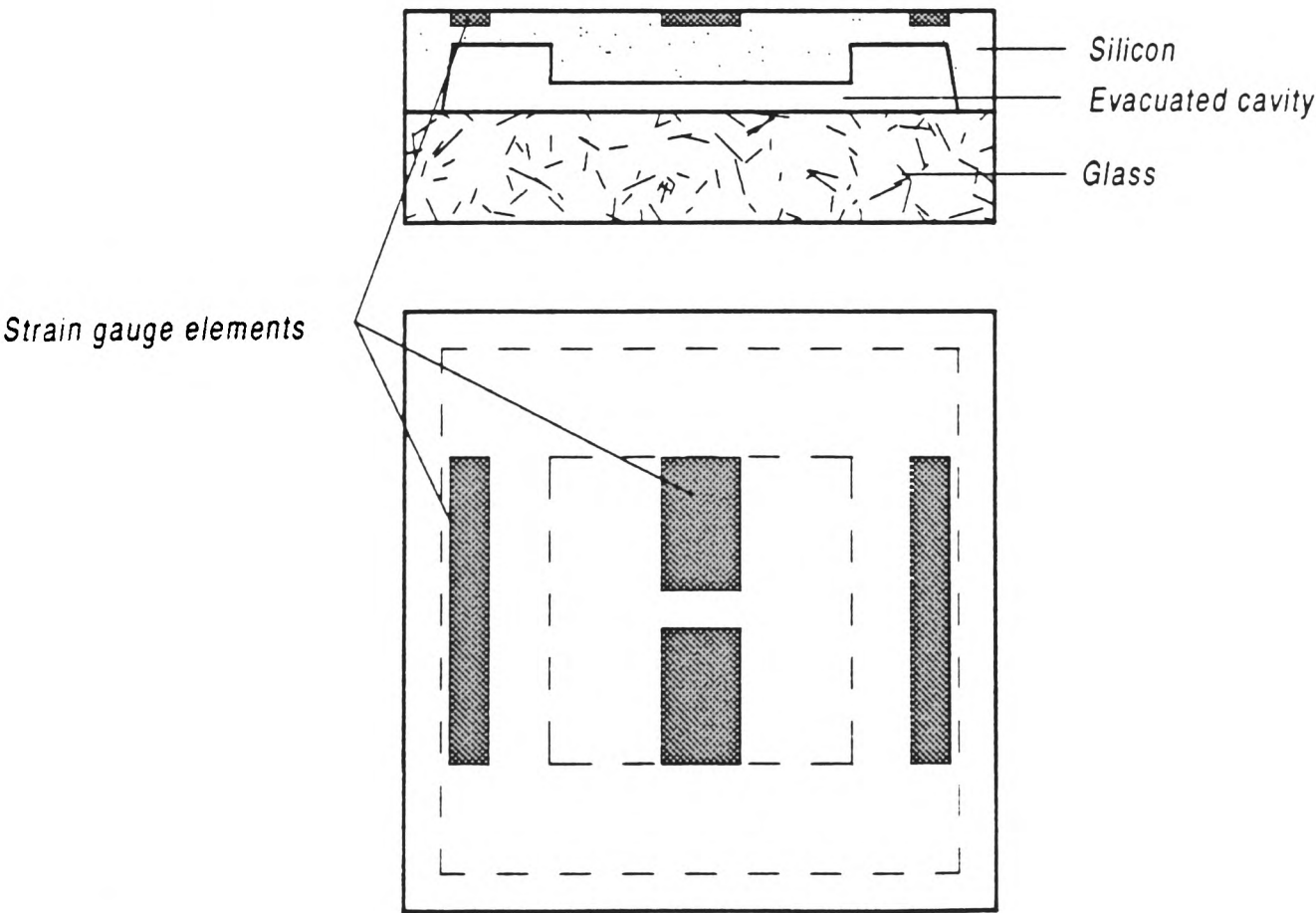


Figure 3.2 Transducer schematic

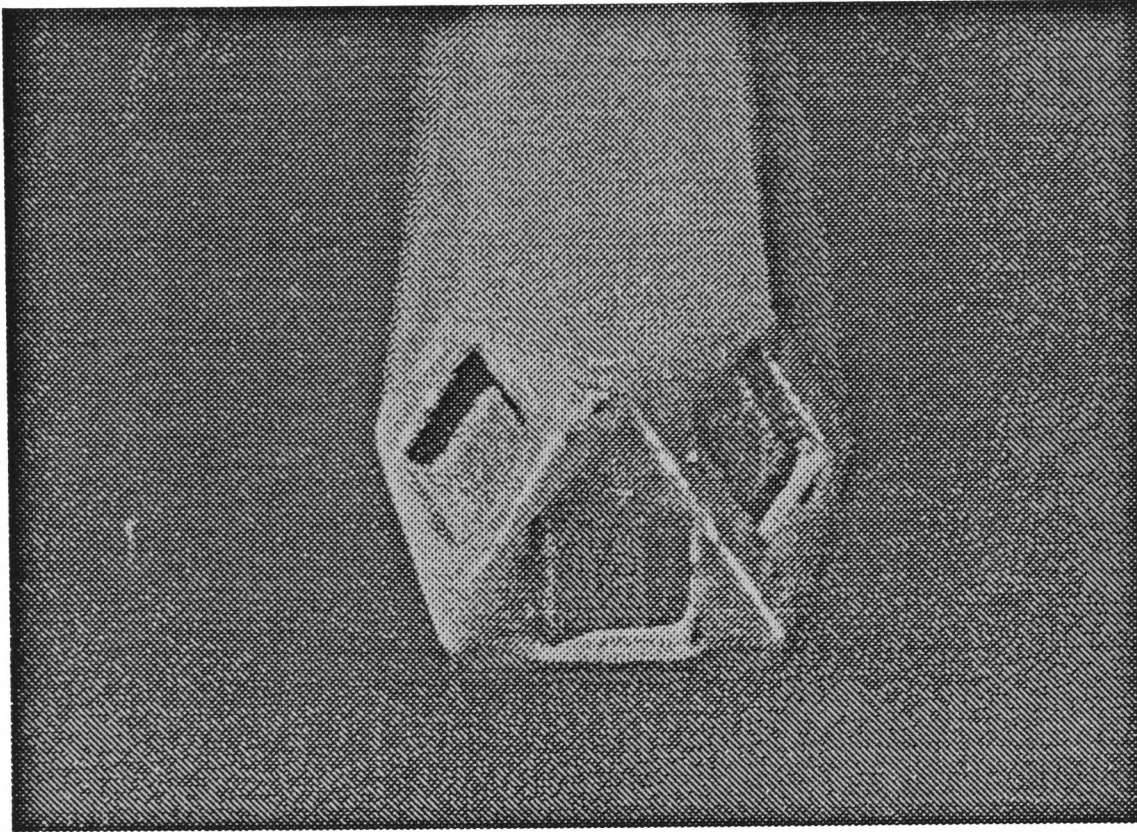


Figure 3.3 Blank probe head showing spark eroded slots for transducer

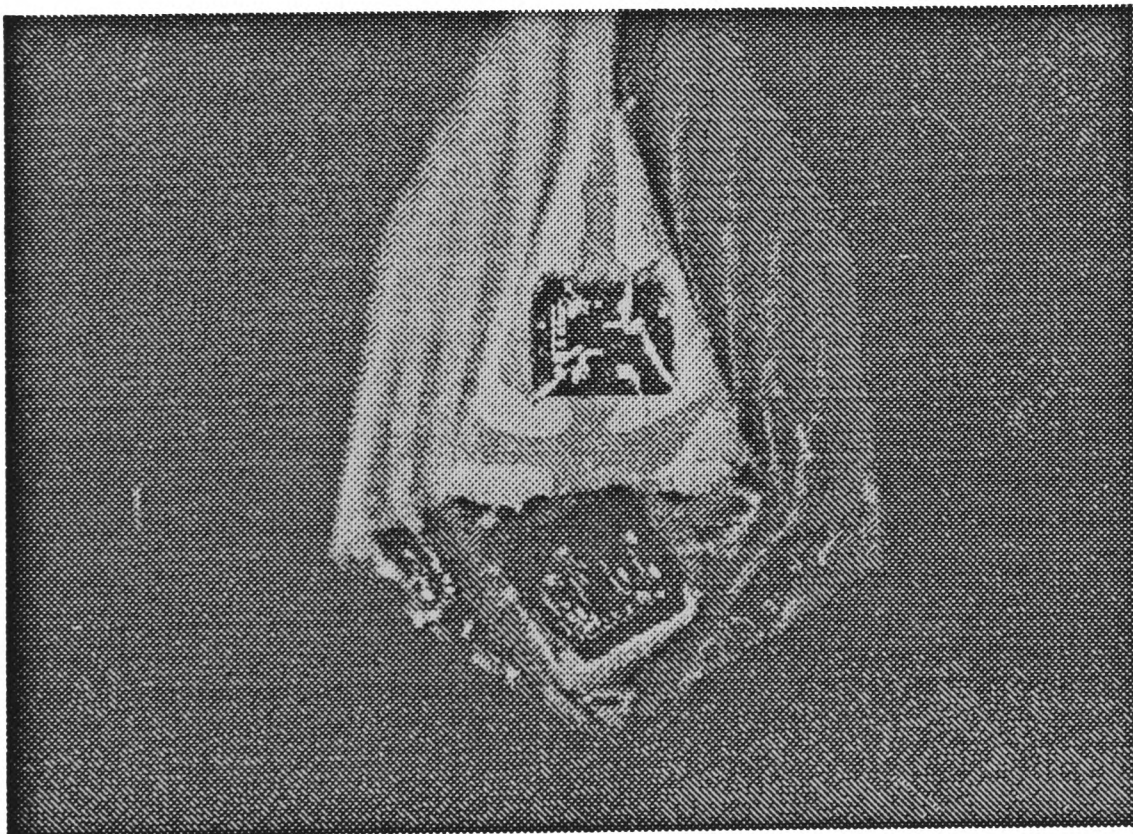


Figure 3.4 Probe head with transducer mounted

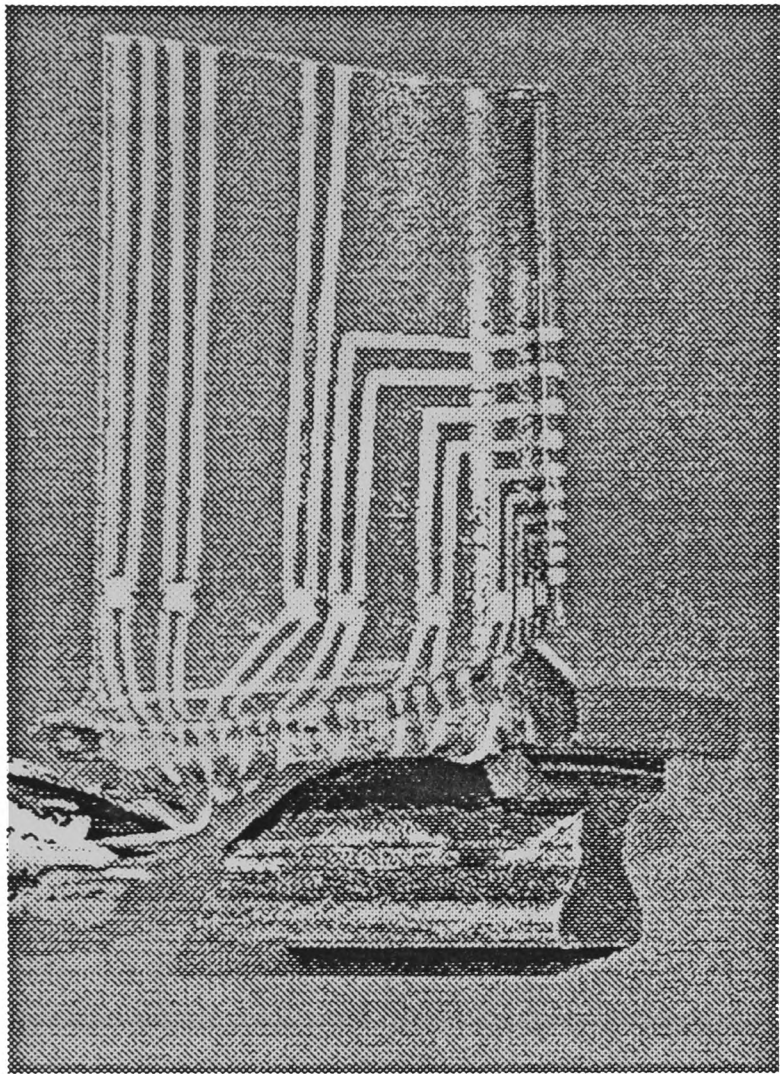


Figure 3.5 Instrumented turbine blade

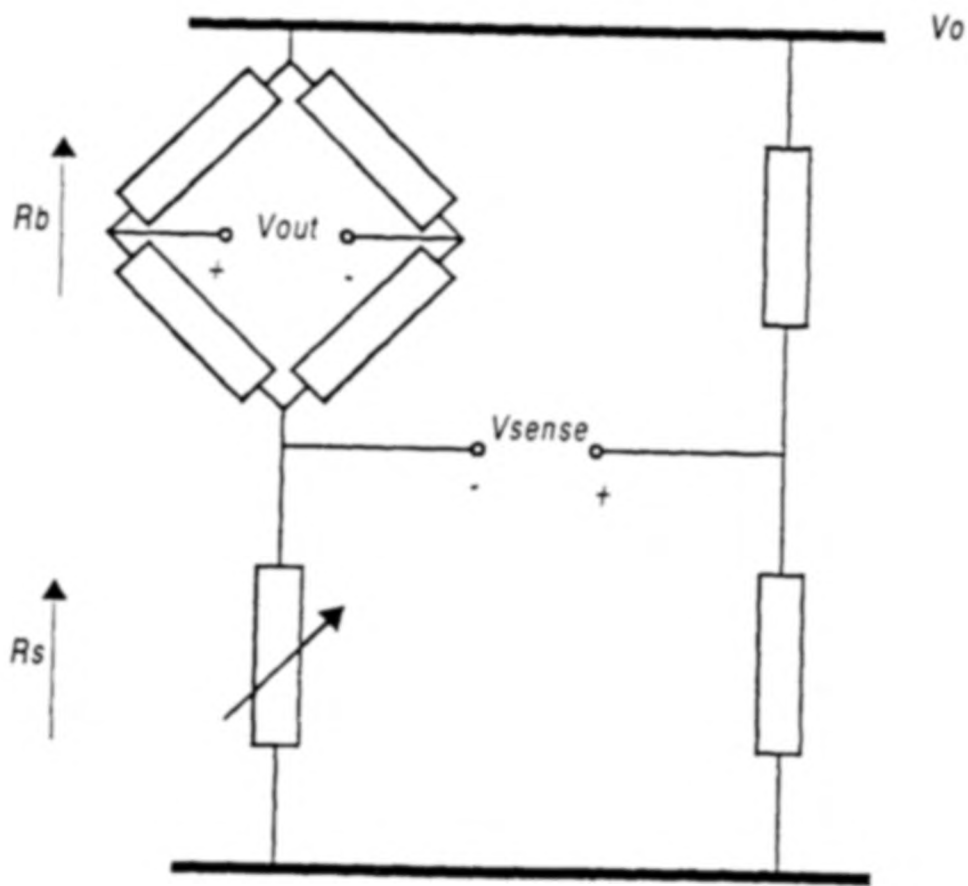


Figure 3.6 Temperature compensation circuit

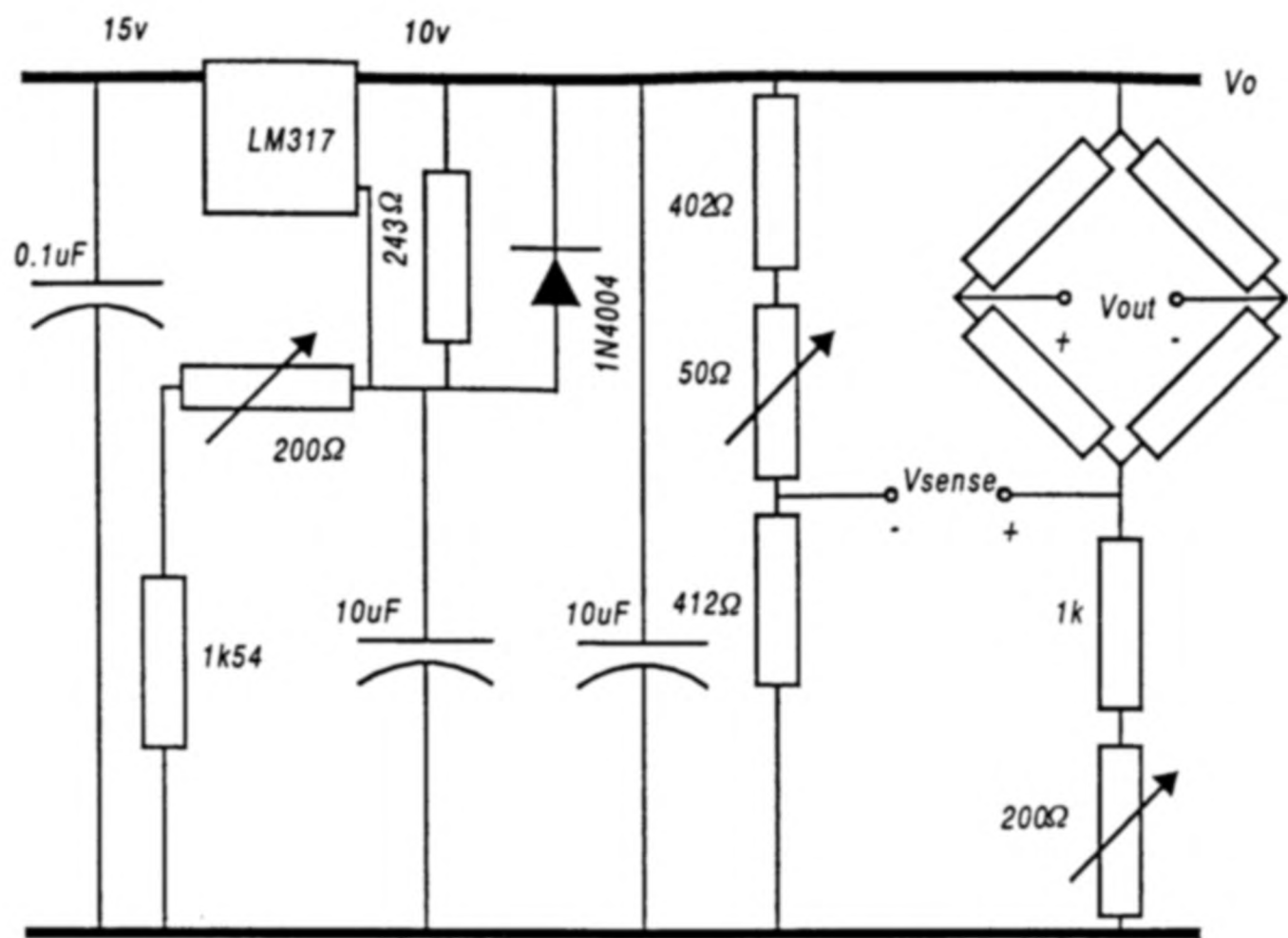


Figure 3.7 Voltage regulatory and temperature compensation circuit

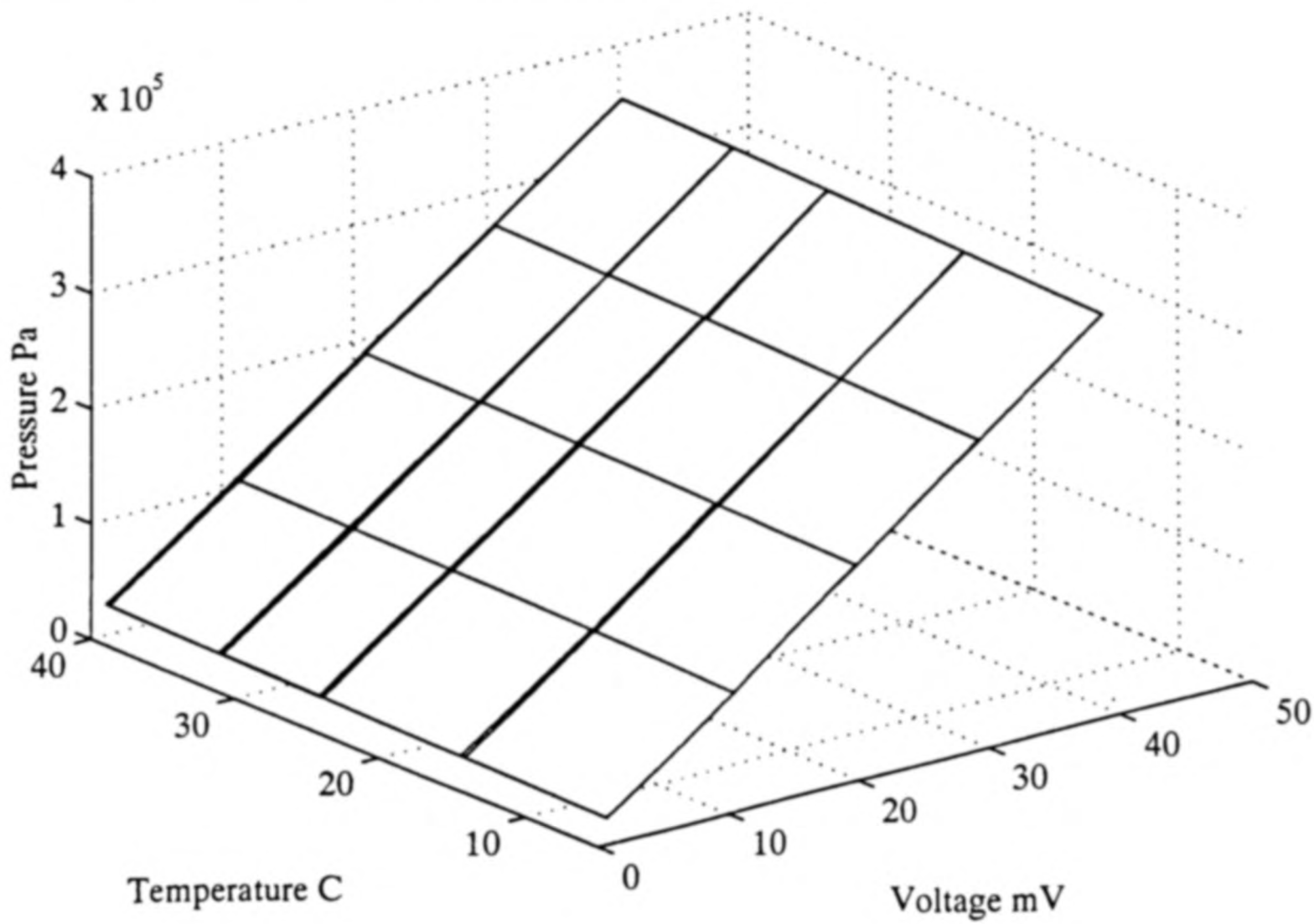


Figure 3.8 Variation of transducer output with pressure and temperature

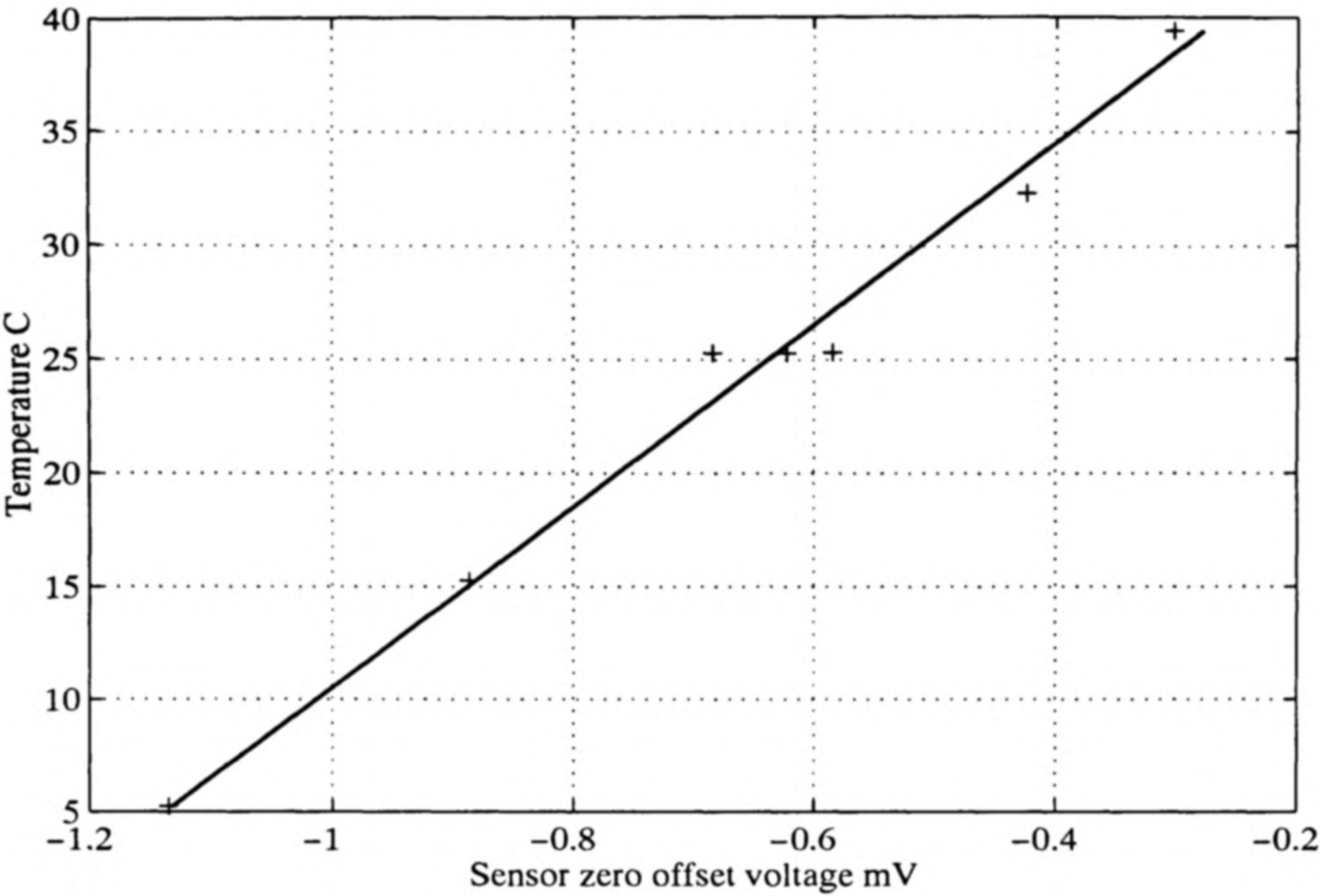


Figure 3.9 Variation of transducer offset with temperature

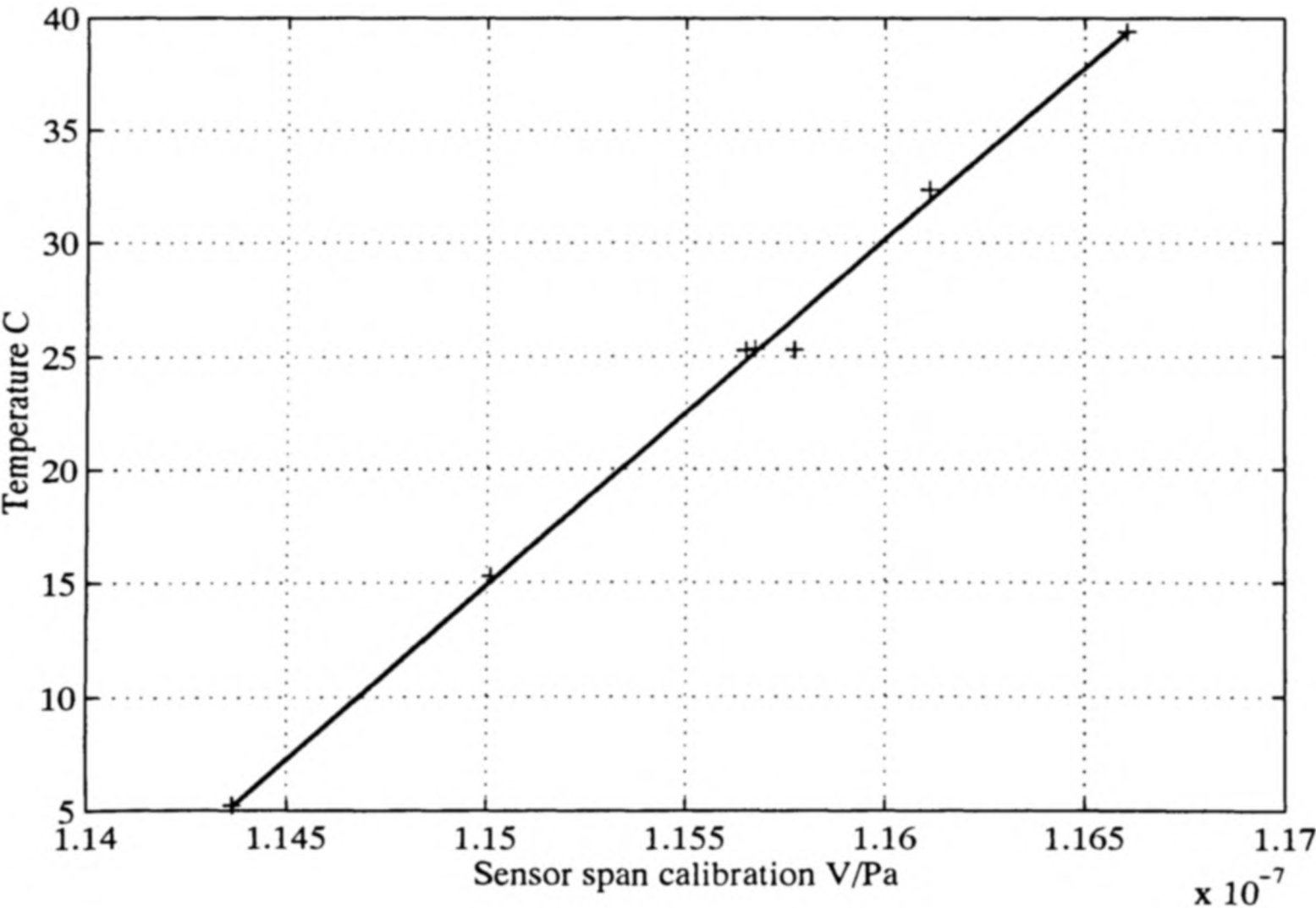


Figure 3.10 Variation of transducer span with temperature

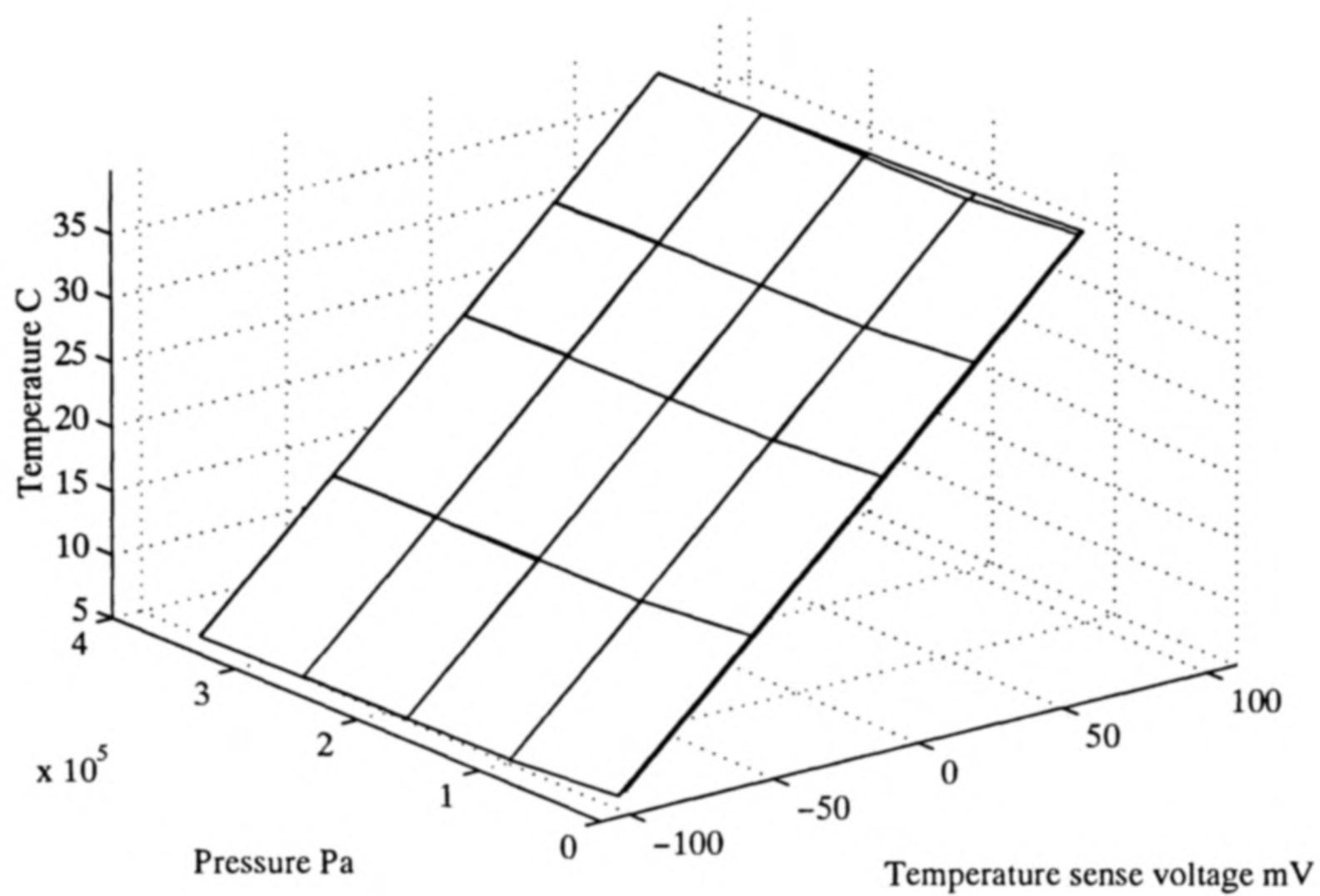


Figure 3.11 Variation of transducer temperature sense voltage with temperature and pressure

4. Aerodynamic calibration

4.0. Overview

This chapter details the techniques that were used to generate the calibration for the aerodynamic probes manufactured during the course of this research. A free jet aerodynamic calibration facility is described and the techniques for creating the calibration data for experimental measurements are outlined. Semiconductor probes are more difficult to calibrate than pneumatic probes because the semiconductor transducer is located in the probe head and therefore the transducers can be affected by temperature changes in the flow.

It is believed that this is the first example of where a three-dimensional semiconductor probe which is capable of both steady and unsteady flow measurements has been fully calibrated.

4.1. Probe calibration

Fast response probes which use high response semiconductor transducers are more difficult to calibrate than their pneumatic counterparts because of the thermal and temporal effects on the transducer output. Early researchers have attempted to overcome this by either calibration of pneumatic versions of their semiconductor probe, Kerrebrock et al [1980], or by superimposing the unsteady component from the semiconductor with the steady component of pneumatic measurements, Bubeck and Wachter [1987]. Even contemporary workers in the field of fast response probes carry out the full calibration of pneumatic versions of their probes and only check a few discrete points to ensure that the semiconductor probe calibration agrees, Gossweiler et al [1994].

Practical experience has shown that small manufacturing differences between probes lead to changes, albeit small, in the detail of the calibration between apparently identical probes. Furthermore, Dominy and Hodson [1992] highlighted the influence of the hole of the pressure tapping on the calibration of aerodynamic probes. When semiconductor transducers are mounted on the surface of the probe there are no pressure tappings and comparison with a pneumatic probe will not be accurate.

The approach taken to probe calibration in this thesis is different. The full aerodynamic calibration is carried out on the actual semiconductor probe in a process whereby the probe is calibrated in only a short period of time. This has the advantage that the transducer calibration may be recorded as close as possible to when the results are taken. By calibration in this way it is possible to avoid the need to make the assumptions taken by other researchers over superimposing calibrations from different probes.

However before this step can be taken a number of obstacles had to be overcome. These included finding a semiconductor technology robust and reliable enough to withstand the rigours of calibration. Furthermore the combination of calibration technique and semiconductor technology had to be reliable enough to not suffer from the temperature and drift problems that had been the bane of earlier semiconductor probe researchers.

4.2. Three-dimensional semiconductor probe design

Whilst five-holed probes have dominated three-dimensional pressure probe measurements a perfectly satisfactory probe design is possible using only four holes. The choice of a four holed probe geometry for a semiconductor probe has considerable advantages in that it allows a smaller probe head to be used and reduces the number of data channels to be recorded, important for unsteady measurements.

The first three-dimensional semiconductor aerodynamic probe to be calibrated was a design based on a frustum of a truncated pyramid geometry, as first used by Shepherd [1981]. The four faces of the truncated pyramid were used as the pressure faces onto which the semiconductor transducers were located. The choice of a prismatic geometry was to ensure that Reynolds number sensitivity was minimised. A probe head with a semi-angle of 45° was selected to give the optimum directional sensitivity. The probe was instrumented using Kulite transducers using the techniques outlined in chapter 3. A similar probe design is pictured in figures 1.3 and 1.4.

The support stem was constructed as a separate item from the head so that different designs of stems could be interchanged for the same probe head if necessary. The stem serves the dual purpose of holding the probe head in the flow and providing a passage for the signal and power wires from the probe. The affect of the stem on the probe performance has been the subject of considerable recent research, Smout and Ivey [1996] or Truckenmuller et al [1996]. Earlier CFD modelling by Depolt and Koschel [1991] found that the influence of the stem on a probe head was considerably reduced if the probe head was mounted forward of the stem by a distance five times the stem diameter. This figure was used as the guide for the final configuration of the probe head and stem.

4.3. *The calibration tunnel*

In order to use an aerodynamic probe for experimental measurement a calibration is required to convert measured parameters, usually pressures, to desired results such as Mach number, flow direction or total pressure. Whilst it is possible to generate theoretical calibrations for simple geometries the performance of real probes deviates from the ideal. This deviation of experimental results from the theoretical is well documented, for example Wright [1969]. Therefore, before any aerodynamic probe may be utilised for measurement it must at first undergo extensive calibration in a known flow-field.

The calibration tunnel in Oxford was commissioned as part of an EPSRC-Kulite programme of work on semiconductor probes and enabled the probe calibration to be undertaken without interfering with the workload on any other facilities. It was designed by J.T.D. Slater and the full details are given in his thesis, Slater [1993].

The initial semiconductor probes tested were wedge type probes which need to be calibrated for their yaw response, the first results being published in 1992, Ainsworth et al [1992a]. The type of probes to be calibrated in the facility has been extended further as part of the work for this thesis to allow the calibration of full three-dimensional probes in which both pitch and yaw of the probe are considered. It has proved to be a valuable asset and has been used to calibrate aerodynamic probes for

use outside the University; a number of probes have been calibrated for DRA Pyestock and it is hoped to extend this work elsewhere.

In order to use an aerodynamic probe to determine flow velocity (Mach number and angle), it must first be calibrated in a flow at dimensionally similar conditions. For the compressible flows of interest here, the relevant non-dimensional parameters are Reynolds number and Mach number. One of the design objectives of the aerodynamic probes developed for use in the Oxford rotor facility, has been to produce a probe with minimum sensitivity to Reynolds number, simplifying the calibration procedure. For a probe insensitive to Reynolds number, it is sufficient to determine the probe's aerodynamic coefficients as functions of pitch and yaw angle, and flow Mach number.

The calibration tunnel operates as a free jet discharging into a pressurised chamber. The jet Mach number is set by adjusting the pressure ratio between upstream of the nozzle and the downstream chamber. The tunnel absolute pressure gives control over the Reynolds number. By adjusting the position of the probe in the jet the flow direction may also be set. The main parameters for the tunnels performance are given in table 4.1

Tunnel type	Free jet
Mach number range	0.2-0.95
Free stream turbulence	0.15%
Contraction ratio	12.96:1
Flow duration	>120 seconds
Nominal total pressure	3×10^5 Pa

Table 4-1 Calibration tunnel characteristics

The air for the calibration is provided by the laboratory reservoir tank which has a volume of approximately 31m^3 and is compressed to 28 Bar. This tank supplies a variety of experiments

throughout the laboratory. With exclusive access to the air the calibration tunnel is capable of running for approximately 4 minutes of continuous flow, although the total pressure of the main tank would drop considerably. In practice short duration experiments lasting approximately 45 seconds gave a good compromise between air usage, total pressure drop and calibration duration. Using short experiments for the calibration of the probe allowed electrical calibration of the transducers, in particular recording the transducer offset, to be undertaken close to when the measurements were being taken. These short experiments allowed sufficient time to acquire data at 85 different probe positions.

The calibration facility layout is shown in figure 4.1. The flow enters the tunnel from the main reservoir tank via a pipe on the lower right of the figure, passes through a series of control valves and turns through 180° before entering the main working section and accelerating through the nozzle.

Probes are positioned one diameter downstream of the nozzle using a stepper motor traverse mechanism controlled from a computer. The mechanism was designed to yaw the probe to $\pm 90^\circ$ and to $\pm 30^\circ$ in pitch. Using a circular track the mechanism is designed to rotate about the front face of the probe whilst maintaining a constant position relative to the nozzle.

The exit nozzle flow was measured by Slater [1993] and by Sheldrake [1995]. Slater reported a total pressure loss of 0.342% at a Mach number of 0.55 and found a uniform total pressure profile at the nozzle exit. Sheldrake traversed the nozzle with a fine needle probe to measure the static pressure and found it uniform across the centre. Sheldrake also showed that the static pressure at the centre of the nozzle is the same as the static pressure measured outside the flow provided the Mach number is below 1. This allows the measurement of the Mach number whilst a probe is being calibrated.

There is a considerable degree of skill to be developed before the tunnel conditions can be efficiently and accurately set. The throttle valve controls are non-linear; there is hysteresis on the electronic actuator of the upstream valve and backlash on the downstream valve. The tunnel operator needs to develop a good understanding of how the tunnel functions otherwise the resource of air available can be wasted before the desired conditions for the experiment are achieved. The compressed

air used by the tunnel comes from a reservoir which is pressurised from by two compressors. The compressors take several hours to compress the air in the reservoir tank - the calibration tunnel can empty the tank in several minutes. As the tunnel shares the air reservoir with other experiments in the laboratory it is important that air is not wasted.

4.4. Calibration software

The tunnel operates with two dedicated computers. The first is used to measure the flow conditions, display the readings and control the probe position. The second computer has a high specification data acquisition board used to record the output from the probe transducers during the experiment. On this computer a suite of programs written in MATLAB, MathWorks[1991], are used to process the experimental results.

With the separation of the data analysis from the rig control program it is possible to commence setting up the next experiment whilst the previous experimental results are still being processed.

4.4.1. Calibration control

The calibration is controlled using a program written in Quick Basic called CPTMP. The software was written in Quick Basic for ease of interfacing with external equipment via the IEEE-488 interface using a library of vendor supplied functions. This program is based in core modules developed by J.T.D. Slater to which a number of enhancements have been added to facilitate aerodynamic probe calibration. The functionality of the calibration software can be broken up into four components as illustrated in the process flow diagram in figure 4.2.

- Electrical calibration

The software has been written to allow in situ calibration of the rig and probe transducers. Isolation valves at each end of the calibration tunnel allow the calibration chamber to be sealed and either evacuated or pressurised. A Druck DPI 501 digital pressure controller is used to set the pressure inside the calibration chamber. The Druck can be controlled via the IEEE-488 interface and by

initialising the Druck's operation to remote control, the calibration software can send the appropriate instructions to set the pressure chamber at intervals over the full scale range of the transducer. The data is automatically logged as the pressure is varied and stored in the computer's memory until the end of the calibration. Regression of the data recorded allows the pressure sensitivity of each channel to be calculated and then saved to a calibration file for later use.

- Initial set up

The position of the probe is set by a pair of stepper motors acting in the yaw and pitch axis respectively. Each time the probe needs to be moved the instructions are passed over the IEEE-488 interface to a Bentham 23-301 stepper motor control system which moves the stepper motor appropriately. In the yaw axis the probe position may be adjusted in steps of 1.8° to a repeatability claimed by the manufacturer of $\pm 0.1^\circ$. In the pitch axis the same stepper motor is used but the positioning mechanism is geared through a circular track. This combination gives a rotation of 0.061° in pitch for each step on the motor.

Micro-switches are used to find the zero position of the probe in pitch and yaw. The micro-switches are positioned so that when the probe has been rotated to the minimum pitch angle the switch closes providing a reference point for all other pitch positions. The process for the yaw direction is the same. Once this reference point has been found the probe can be rotated to the initial position for the experiment, which is usually zero incidence in pitch and yaw so that the alignment of the probe can be checked. The program gives the option for fine adjustments about the start position if necessary.

Once the probe position has been set the data from the probe is logged and the raw voltages displayed continuously. When the values are satisfactory the zero values for the experiment are recorded at the press of the space bar on the keyboard.

- Pre-traverse

The calibration file which contains the pressure sensitivity of each channel is loaded during the pre-traverse stage. This calibration file also contains the values of the temperature sensitivities measured during the temperature calibrations. A second file called the configuration file contains the combination of positions the probe is to be moved to during the course of the experiment. The details in the configuration file include the number of points to be recorded, selection of yaw and, or a pitch traverse, choice of angular range and the direction of the traverse. This gives flexibility to alter the type of experiment without editing the software.

The data from the probe and the rig is logged and by applying the calibration the pressures can be found. At this stage the temperature corrections to the probe output are also calculated so that the probe output can be corrected in real time. The Mach number in the rig is calculated from the total and static pressures.

Whilst the probe position can be set by the computer, the tunnel operator needs to set the flow Mach number and total pressure. This is achieved by the operator finding appropriate combinations of the upstream and downstream throttle values. The Mach number and pressures from the various channels are continuously displayed to give the operator the necessary feedback required to set the flow conditions and show that the probe is working correctly.

- Aerodynamic calibration

Once the tunnel operator is satisfied with the conditions in the tunnel the aerodynamic calibration routine can begin at the press of the space bar on the keyboard. The probe is moved to the positions detailed in the configuration file and the data logged at each position. The data is stored in an array in the computer's memory. The high specification data acquisition board in the second computer is connected to the trigger channel and simultaneously records voltages from the probe transducers at each position.

When the calibration is complete the electrical calibrations are applied to the data and the results saved, including records of the positions and Mach numbers. The temperature sense voltage is also saved so that it can be applied to the results recorded on the high specification data acquisition boards.

4.4.2. Data processing

The data processing computer is installed with a high specification 16 bit A/D board manufactured by Digitimer Ltd. Whilst the first computer records data from all channels, the Digitimer board is only used for measurements from the probe under calibration. The Digitimer board is triggered to record data each time the probe has been moved to a new position. The data is recorded directly onto the A/D board's 2Mb of memory and downloaded to the computer after the experiment.

The Digitimer board is built around a transputer and the system is capable of sampling up to 400kHz for fast response measurements but in calibration is usually operated at 50kHz. At each position 1000 data points are recorded which acts to reduce the aerodynamic and signal noise. The signal is amplified close to the signal source to both minimise the effect of interference on the signal and give the A/D improved voltage resolution. The AD524 differential input instrumentation amplifier was chosen for this application because of its wide bandwidth (~100 kHz), low noise and its ability to remove any offset voltage that might saturate the A/D board.

Quick Basic has some restrictions, the most serious for this work is the limit on the maximum data array size available. Whilst Quick Basic is perfectly satisfactory for running the tunnel control and data logging, it would be restrictive to attempt to process the calibration data within Quick Basic.

MATLAB is a flexible data processing application which provides a powerful numerical and graphical tool set and was therefore chosen to be the tool for the core calibration data processing. A suite of programs written in MATLAB were used to take the raw data recorded during the experiment and generate the full calibration results. When the probe is used in an experiment the data is again processed in MATLAB.

4.5. Calibration techniques

The method through which the aerodynamic probes were calibrated has evolved over the course of this research. The first probes to be calibrated were yaw probes based on the wedge geometry. These types of probe are easier to calibrate than a three-dimensional probe purely because the number of combinations of positions required for the calibration of a two-dimensional probe is significantly less. As the calibration of the probe becomes more complex (and longer) the precautions required to ensure the probe successfully completes the calibration process become more important.

4.5.1. Minimising the calibration time

The method by which the probes have been calibrated has been developed to minimise the time taken for the probe to be calibrated. Whilst this gives immediate cost and productivity advantages there are a number of other real benefits by taking this approach. By reducing the length of the calibration, external influences on the semiconductor transducer can be minimised. In chapter 3 a technique for correcting for temperature effects was described, but this is best complemented by avoiding unnecessary temperature changes on the probe. During the calibration the air flow acts to cool the probe. By reducing the calibration time the variation in temperature during the experiment will be reduced.

A second reason to reduce the aerodynamic calibration time is to increase the survivability of the probe. There are unfortunately a variety of potential failures of the probe which can occur during calibration. Whilst the probes are designed to avoid these problems the rigours of the calibration are such that the longer the calibration lasts the higher the probability that one of these failure modes can occur. The two most likely problems during calibration are particle damage from the flow or loss of wiring.

4.5.2. Early and reliable fault diagnosis

To help ensure that a probe has been correctly installed it is important to have early and reliable fault diagnosis. Regular monitoring of the transducer resistance allows problems with the probe to be quickly diagnosed.

The in situ electrical calibrations provides a health check on the transducers. As part of the regression to find the transducer sensitivity the standard deviation of the result also gives an indication of the channel noise level. The noise level on a channel starts to increase if any of the connections start to deteriorate. This can happen when a connection wire is working loose. On line display of the probe's output gives a visual check that the probe is operating satisfactorily.

Finally as soon as the results have been saved they are passed to MATLAB where pre-written processing software can very rapidly calculate the experimental results and display them graphically. These graphical results provide the means to identify faults during calibration and give a very reliable way of ensuring integrity of the calibration data.

4.5.3. Robust installation

The most fragile component of an aerodynamic probe is surprisingly not the transducer but the connections to the transducers and the wiring to the probe. The elastomer layer protects the ball bond joint between the transducer and the gold strip on the probe head, but the connection between the electrical wire and the gold strip is harder to protect. Early probe designs had this connection external to the probe and a great deal of care had to be taken in their installation. The aerodynamic forces during calibration are such that they can easily break the connection of the wire to the probe if not secure. Similarly the soldered connection between the wire and the external junction experiences aerodynamic forces which require the wires to be securely attached. Experience showed that wiring failure could be avoided if the wires were securely to clamped and all exposed wiring covered with tape.

4.6. Calibration results

The calibration tunnel cannot be run continuously and therefore the number of results per experiment are limited. To calibrate a probe over a wide range, the calibration has to be built up piecemeal, with individual calibrations, each at different conditions, combined to generate a full

aerodynamic calibration. For a reliable calibration the flow conditions in the calibration chamber must be controllable and consistent, but more importantly the results generated must also be repeatable.

In the following section the results of the 45° semiconductor probe are used to demonstrate how the complete calibration data set is generated. This probe has been calibrated over a range of approximately $\pm 25^\circ$ in pitch and yaw, at intervals between Mach 0.25 to Mach 0.9. The full set of data corresponds to 2550 data points.

The processing of the results can be broken up into the following key stages:

1. Recording the tunnel conditions
2. Application of pressure and temperature calibration
3. Generation of calibration coefficients for the experiment
4. Assimilation of calibration data for different conditions
5. Inversion of calibration data

Stages 1-4 are discussed in the remainder of this chapter, whilst details of inversion of the data can found in chapter 5.

4.6.1. Tunnel Conditions

The tunnel conditions are recorded by the rig computer over the course of the experiment. The disadvantage of using a fixed reservoir as the supply of compressed gas is that as the reservoir level drops, the pressure available for the experiment will fall. This fall in pressure can be seen in both the total pressure and the static pressure as plotted in figure 4.3. However the probes are designed to operate independently of the total pressure; the important non-dimensional parameters in this experiment are the Mach and Reynolds number. These are plotted in figures 4.4 and 4.5. In the first plot the Mach number is constant even though the total pressure drops. This gives the uniform conditions required for calibration. The small drop in the Reynolds number during this experiment is

acceptable because the Reynolds number effects in this region are small and the prismatic probe design is generally accepted to have good Reynolds number insensitivity.

4.6.2. Temperature variation

During the calibration the temperature of the probe will vary. There are two competing factors at play. The first is the heat dissipated by each of the semiconductor transducers; a simple calculation shows that a $1\text{K}\Omega$ transducer operating at 5V is dissipating 25mW . With four transducers located in close proximity the temperature of the probe head is well above ambient temperatures. Measurements have shown that the probe head may be at temperatures of up to 40°C . The temperature reaches an equilibrium when the ohmic heating equals the heat convection from the probe surface and conduction through the probe stem.

In figure 4.6 the change in temperature of one of the probe's transducers has been plotted. This temperature has been derived from the temperature sense voltage as calculated from equation 3.8. Before the experiment, the temperature of the probe sits in equilibrium with its surroundings. It takes a finite amount of time to get the tunnel operating at the desired flow conditions. During this period the probe temperature falls. Therefore when the calibration data starts to be recorded there is a temperature difference between when the probe initial readings and when data is first acquired.

During the experiment there is a further drop in the temperature of the probe as the expanding air flow continues to cool the probe head, however this temperature drop is not as dramatic as the initial temperature change.

It has been found that even a gentle flow of air over the probe dramatically changes the balance of equilibrium because the convection transfer coefficient will be raised considerably by the moving air. This change in temperature due to the change in equilibrium between the convection and power dissipated by the transducers has been found to be minimised if the probe is never allowed to reach a temperature equilibrium with its surroundings but kept cooled by a light bleed flow of air over the probe. This improves the quality of the calibration by reducing the temperature correction.

4.6.3. Raw transducer output

During each experiment the probe is moved through 85 different combinations of pitch and yaw. As the probe is moved through these position combinations each of the transducers will record pressures corresponding to how the flow incidence falls on each of the individual transducers.

The plot in figure 4.7 shows the data from the 17 positions recorded during a calibration as the probe yaw is varied. The total face, A, shows a variation in the measured pressure which corresponds to the face's sensitivity to misalignment to the flow. When the yaw angle is zero this face almost fully recovers the total head. As the incidence of the probe increases the degree to which the total pressure is recovered falls.

The second plot shows the sensitivity of face B to the yaw angle. This face exhibits an almost linear sensitivity to the variation in yaw angle. The face pressure is lowest when the face is rotated away from the flow and greatest when rotated towards the flow. The probe head is symmetrical in the yaw plane and therefore it is not surprising that the third face, C, has a pressure plot that is almost the mirror image of face B.

Finally the fourth face D shows rather different behaviour. In the design of the probe, face D is intended to help measure the pitch angle. This face therefore shows little sensitivity to the variation in yaw angle and its measured pressure remains fairly constant.

The plot in figure 4.7 contained 17 of the 85 data points recorded during a calibration experiment. The full 85 points from the transducer outputs during the experiment are plotted in figure 4.8, but this time against a time axis. In this plot both the yaw angle and pitch angle have been varied; the pitch angle has been changed each time the range of yaw angles has been traversed. The probe traverses through 17 positions in yaw ($+28.8^\circ$ to -28.8° in steps of 3.6°) for each of 5 positions in pitch ($+7^\circ$ to -7° in steps of 3.5°). Each point plotted on this graph is the result of the averaging of 1000 samples so that any noise and flow turbulence can be removed.

4.6.4. Derived coefficients

A review of the literature shows that many different methods have been used for reducing the data from a probe calibration. The method adopted here is to use the yaw and pitch coefficients as first defined by Sitaram and Treaster [1985] for four port probes, equations 2.19 and 2.20. As already highlighted in chapter 2, these coefficients may be considered to be made up from a numerator which is a function of the direction to be measured, and the denominator which estimates the dynamic head. In this way the calibration coefficients will be non-dimensional and they can therefore be applied in a wide range of conditions.

To extend the measurements of the probe to include Mach number, a Mach coefficient is needed. The Mach coefficient needs to be a parameter that is strongly dependent on the Mach number. The obvious choice might be the ratio of the pseudo static pressure, P_M to the pseudo total, P_A , however a better choice is given in equation 4.1 where the Mach coefficient has been defined as a function that makes its value linear with Mach number. Defining the function in this way has advantages when inverting the data where linear interpolation is used, particularly at higher Mach numbers.

$$C_M = \sqrt{\left(\frac{2}{\gamma - 1} \left[\left(\frac{P_M}{P_A} \right)^{\frac{1-\gamma}{\gamma}} - 1 \right] \right)}$$

Equation 4.1

The yaw, pitch and Mach coefficients can be considered to be primary coefficients because they are all that is needed to find the yaw angle, pitch angle and Mach number of the probe. A total pressure coefficient can also be defined to allow calculation of the total pressure but the use of this coefficient differs from the other coefficients because the total pressure coefficient cannot be used directly; the flow Mach number and direction have been found first using the other primary coefficients.

$$C_T = \left(\frac{P_A - P_T}{P_A - P_M} \right)$$

Equation 4.2

The yaw coefficient from the calibration data plotted earlier is shown in figure 4.9. In the first plot the yaw coefficient from the data taken at zero pitch angle is plotted on its own. In the second plot the yaw coefficients at each of the different pitch angles have been superimposed. The influence of pitch angle on the yaw coefficient is small, a sign of a good probe, but it can be seen that at higher angles of yaw the influence of the pitch angle is greater.

In figure 4.10 the pitch coefficient has been plotted for the experiment, each of the five lines corresponding to a constant pitch angle. When the pitch is zero the influence of the yaw angle on the pitch coefficient can again be seen to be small. However as the pitch angle increases the pitch coefficient is increasingly influenced by the yaw angle.

With all the coefficients the cross dependency of the coefficient is greater at the higher incidences. The main reason for this lies with the denominator of the coefficients; the probe pseudo total and static pressures are used to normalise the coefficient but the measured values are affected by the probes incidence and therefore become increasingly worse estimates of the true total and static pressures as the incidence increases.

The variation of the Mach coefficient during the experiment is plotted in figure 4.11. As the Mach number is held constant during the experiment a large variation is not expected. The plot illustrates the influence of incidence on the Mach coefficient.

A more informative plot of the Mach coefficient is shown in figure 4.12. This plot combines the data from a series of calibrations to show the variation of Mach coefficient with Mach number at zero incidence.

The total pressure coefficient is used in an experiment to find the total pressure only after the flow direction and Mach number have been found. Since the total pressure is in a second step, the total

pressure coefficient is considered a secondary coefficient. The total pressure coefficient, also plotted in figure 4.13, never reaches zero because the front face pressure does not fully recover the total pressure. This is because even with the probe at zero incidence the stagnation point may not be directly in front of the front face of the probe.

The data considered up to now has been obtained during one calibration experiment at a single Mach number, 17 yaw positions and 5 pitch positions. To complete the calibration this needs to be combined with data from other experiments. The calibration at this Mach number is repeated two more times with different combinations of pitch angles to give a grid of 17 yaw angles and 15 pitch angles.

In figure 4.14 the calibration data is plotted to illustrate the three separate experimental results. The graphs show lines of constant yaw angle and pitch angle on ordinates of yaw and pitch coefficients. Plotting the data in this way allows the three sets of results to be combined to show the full calibration at one Mach number as in figure 4.15.

4.6.5. Repeatability

In figure 4.16 the same method of plotting the experimental data as in figure 4.15 is used to illustrate the significance of the temperature correction on the results. In this plot the results are shown twice, the first plot incorporates the temperature correction whilst the second plot does not.

In a similar fashion figure 4.17 plots the repeatability of the experimental results. This shows the same experimental results recorded in two different experiments superimposed. Whilst not perfectly repeatable, the worst case error is approximately 0.5° , and this for a semiconductor probe is satisfactory. The techniques adopted by Kerrebrock et al [1980], Bubeck and Wachter [1987], or Gossweiler et al [1994] to overcome the problems with calibration of their semiconductor probes are therefore not necessary or even desirable.

It is noticeable that the performance of the probe is always at its worst at the higher angles of attack. This is again the result of the denominator used in the coefficients. As the incidence of the probe increases the values of the pseudo static and pseudo total pressure become poorer estimates of

their true value. As these values tend to converge on themselves the pseudo dynamic head tends to zero causing small perturbations in the value of the numerator of the coefficient to become magnified.

4.6.6. Complete calibration

To complete the calibration the data obtained at the one Mach number needs to be combined with the values at other Mach numbers. In figure 4.18 the results from several calibration experiments have been combined to construct part of the calibration space. Completing this process of combining the calibration data, generates the full calibration space shown in figure 4.19. In this plot five planes of data recorded at constant Mach number are plotted, each plane has lines of constant yaw and pitch angle plotted on it. The influence of Mach number cannot be ignored since although the calibration surfaces look superficially similar, there are some distinct scaling effects and a shift of the whole surface pitch coefficient with Mach number. This last effect is a result of stem causing the local pressure on the probe head to rise on the stem side. The stem effect increases with Mach number causing the shift in pitch coefficient. Another effect of increasing the Mach number is the corners of the Mach surfaces become increasingly bent over at higher incidence. When the probe is at incidence the pseudo total and static pressure become worse estimates for the true total and static pressure. The effect of this is for the Mach coefficient to be less linear with Mach number when the probe is at incidence, creating the bent over corners of the calibration surfaces.

When the calibration space has been created the probe is ready for use in an experiment for the measurement of flow. In chapter 5 the methods by which the data can be inverted are discussed, whilst chapters 6 and 7 give practical application of the calibrated probe in an experiment.

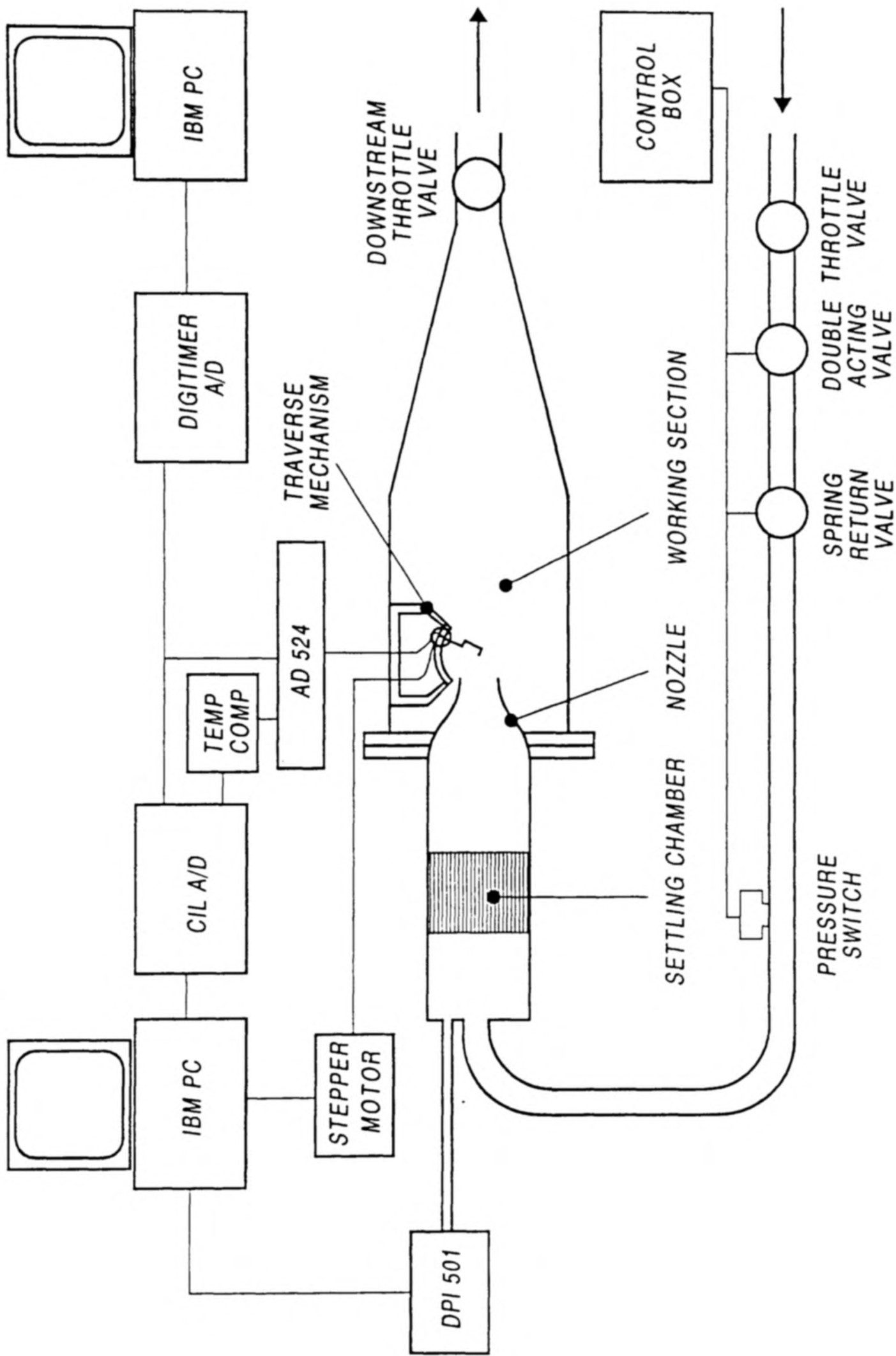


Figure 4.1 Schematic of the calibration tunnel

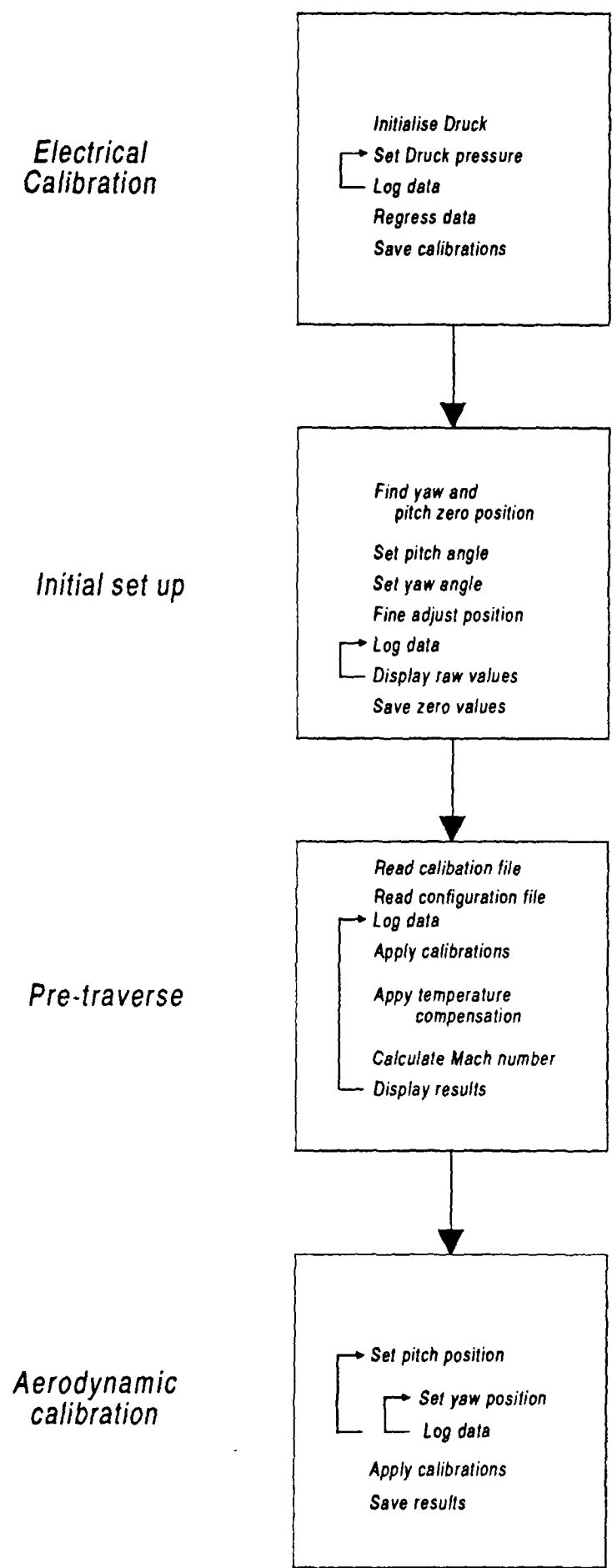


Figure 4.2 Process flow for calibration rig control software

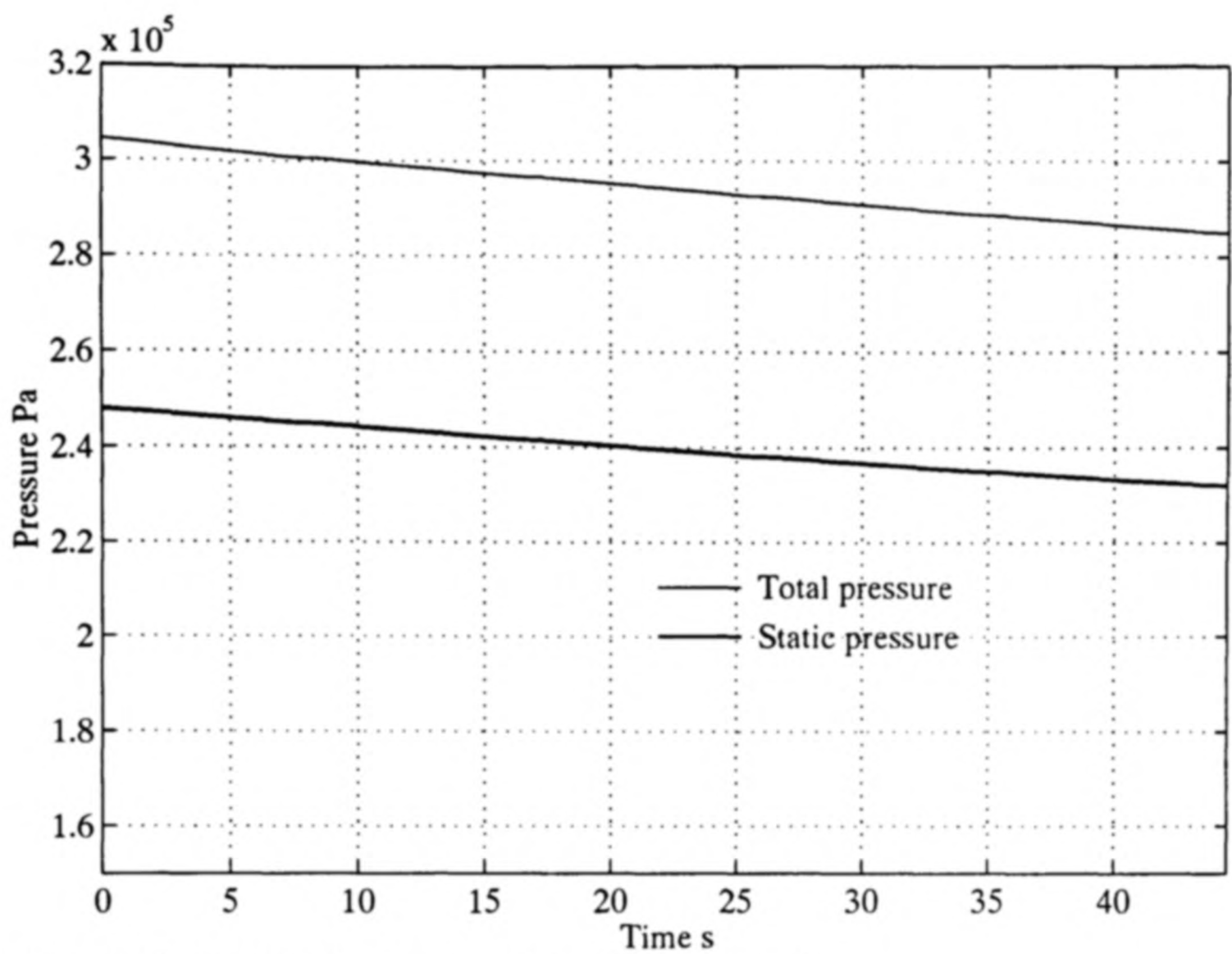


Figure 4.3 Total and static pressure variation during calibration

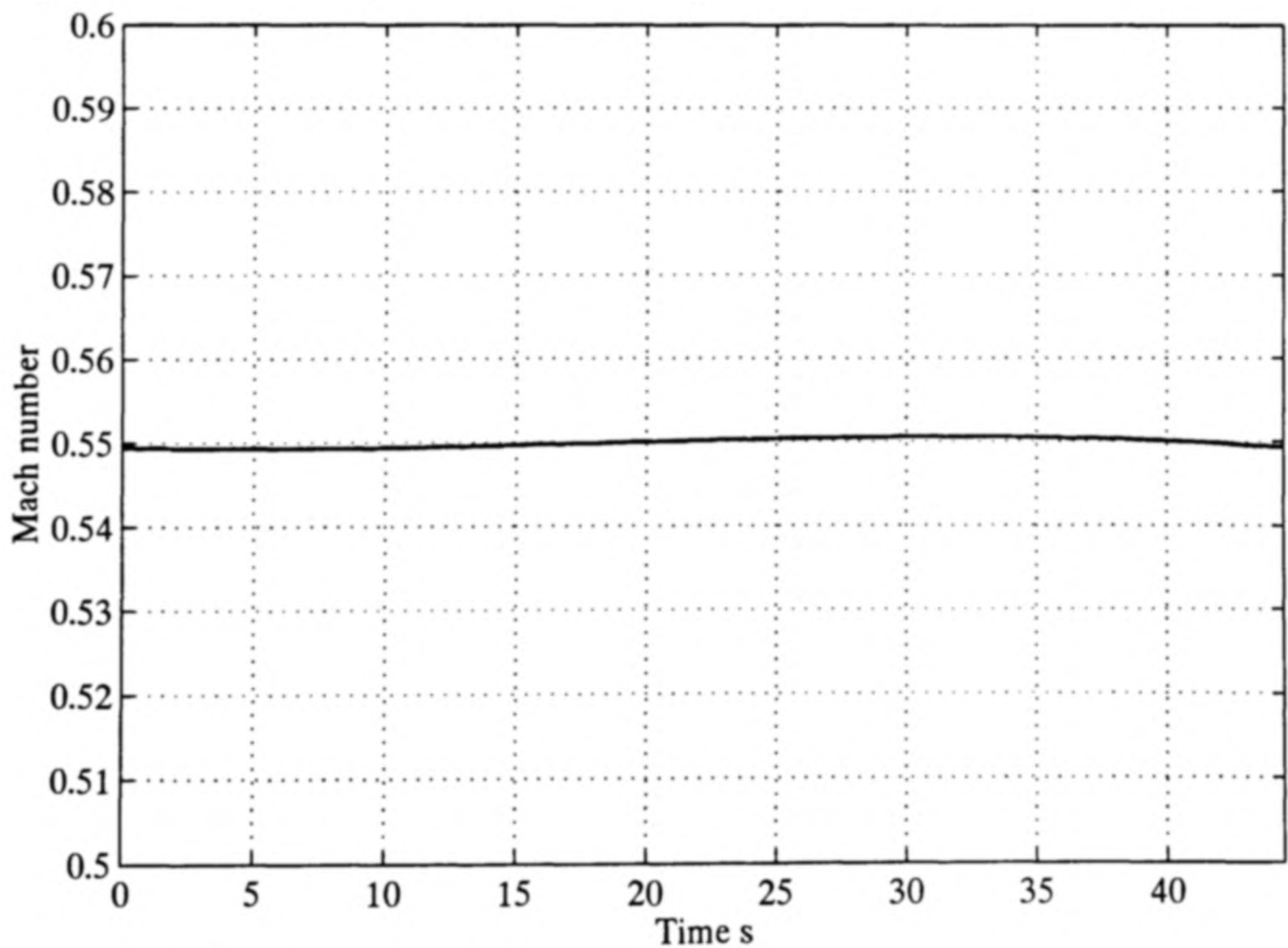


Figure 4.4 Mach number variation during calibration

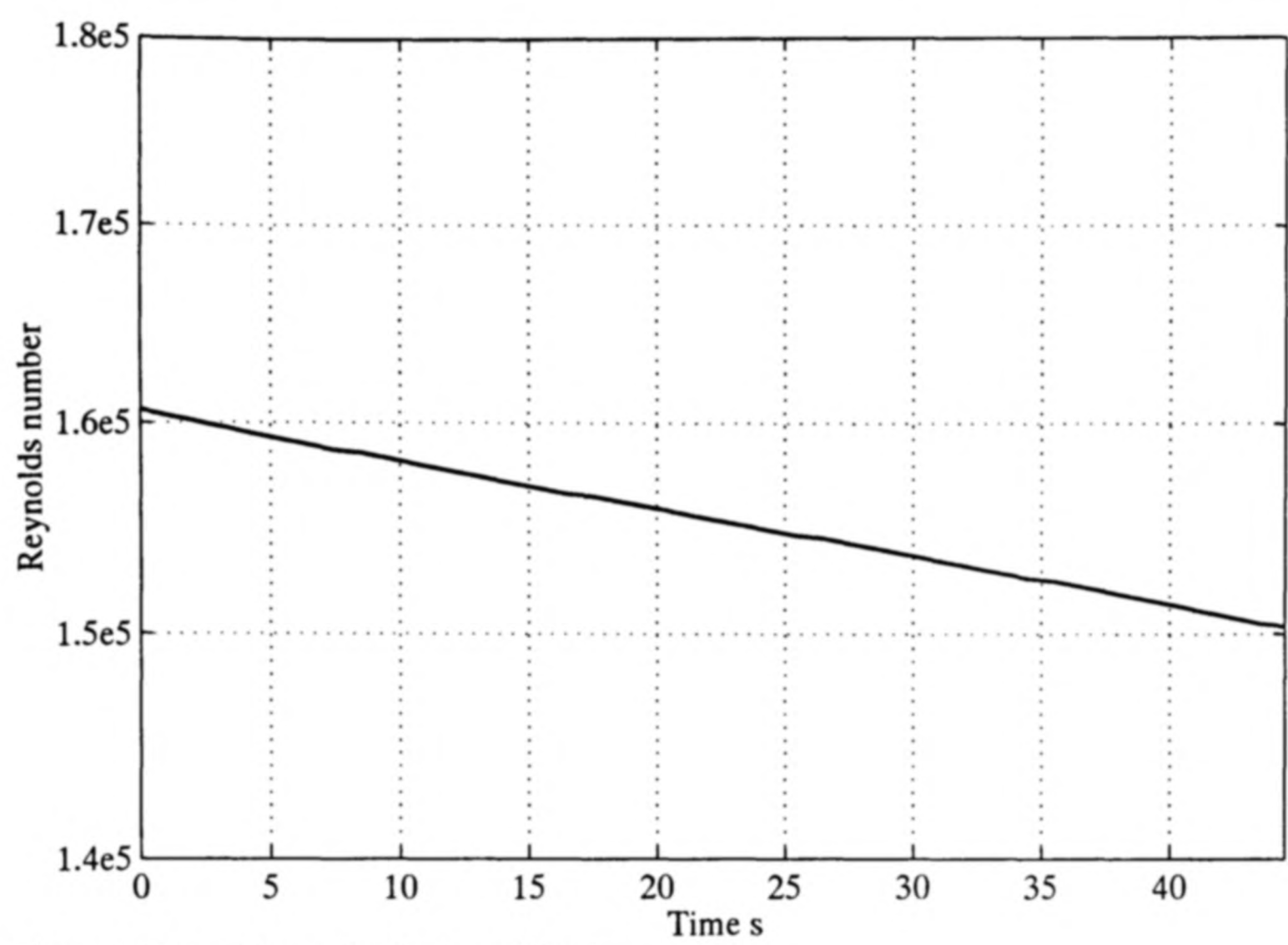


Figure 4.5 Reynolds number during calibration

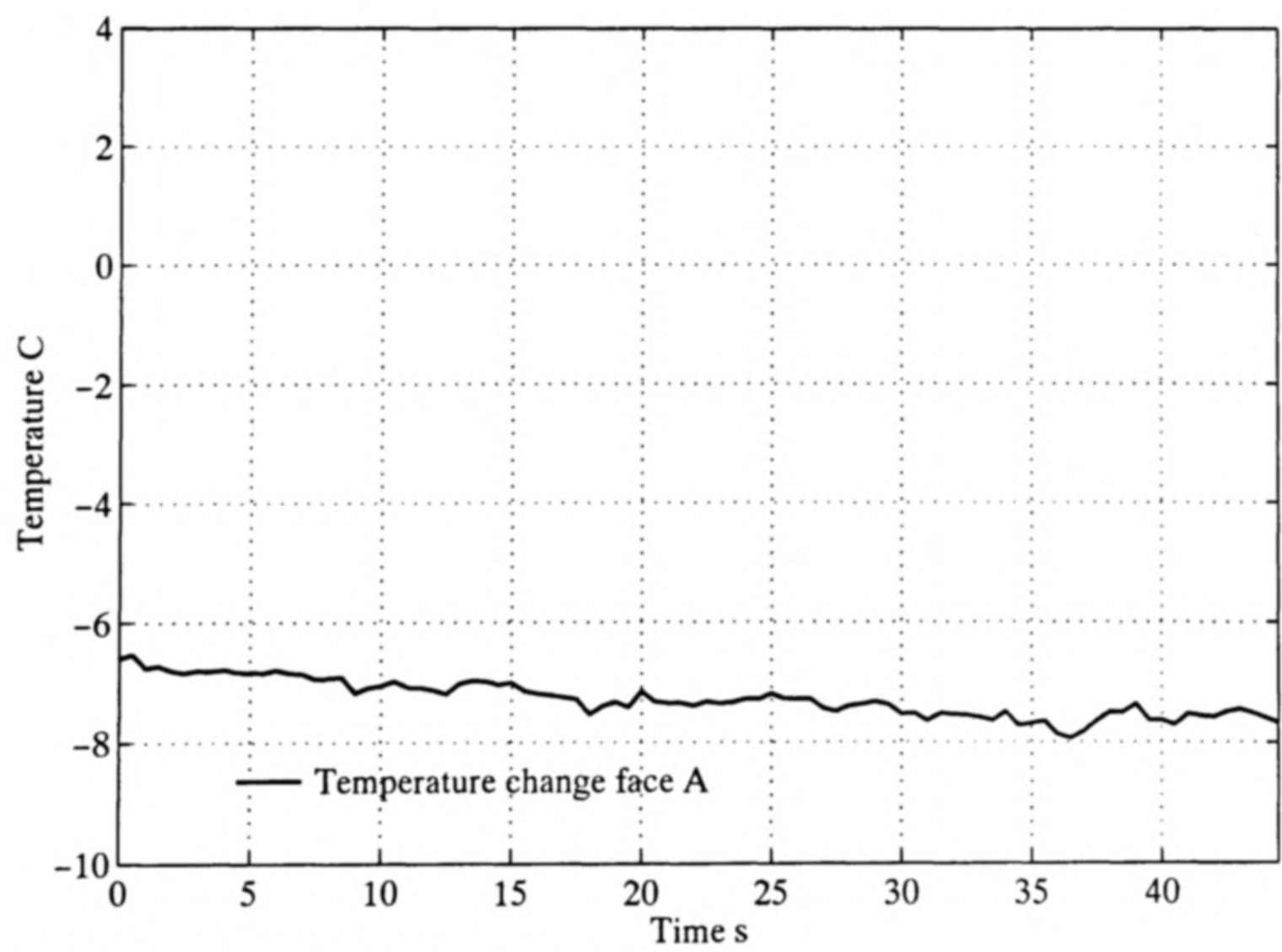


Figure 4.6 Temperature of the probe head during calibration

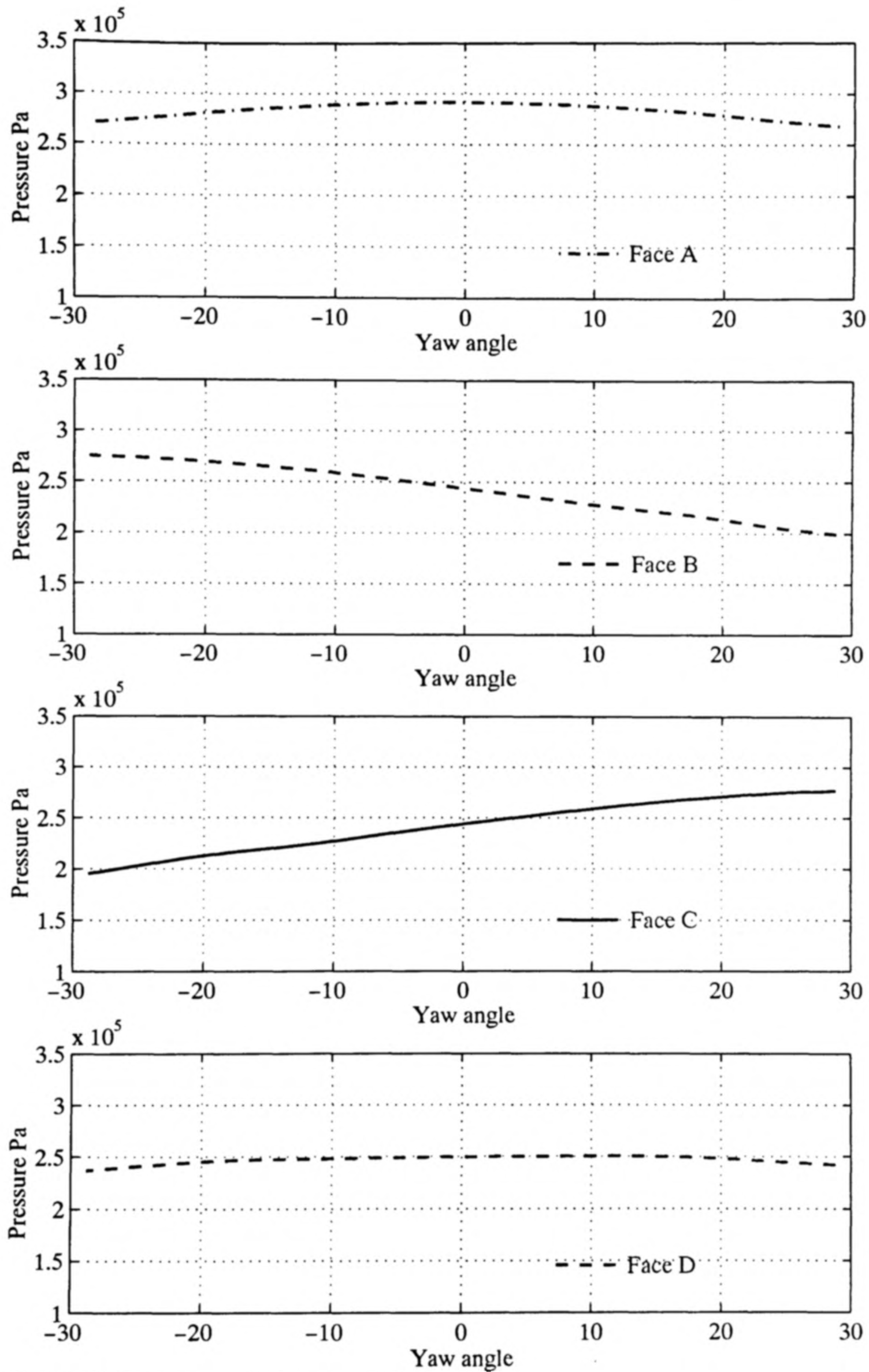


Figure 4.7 Face pressures during traverse in yaw angle

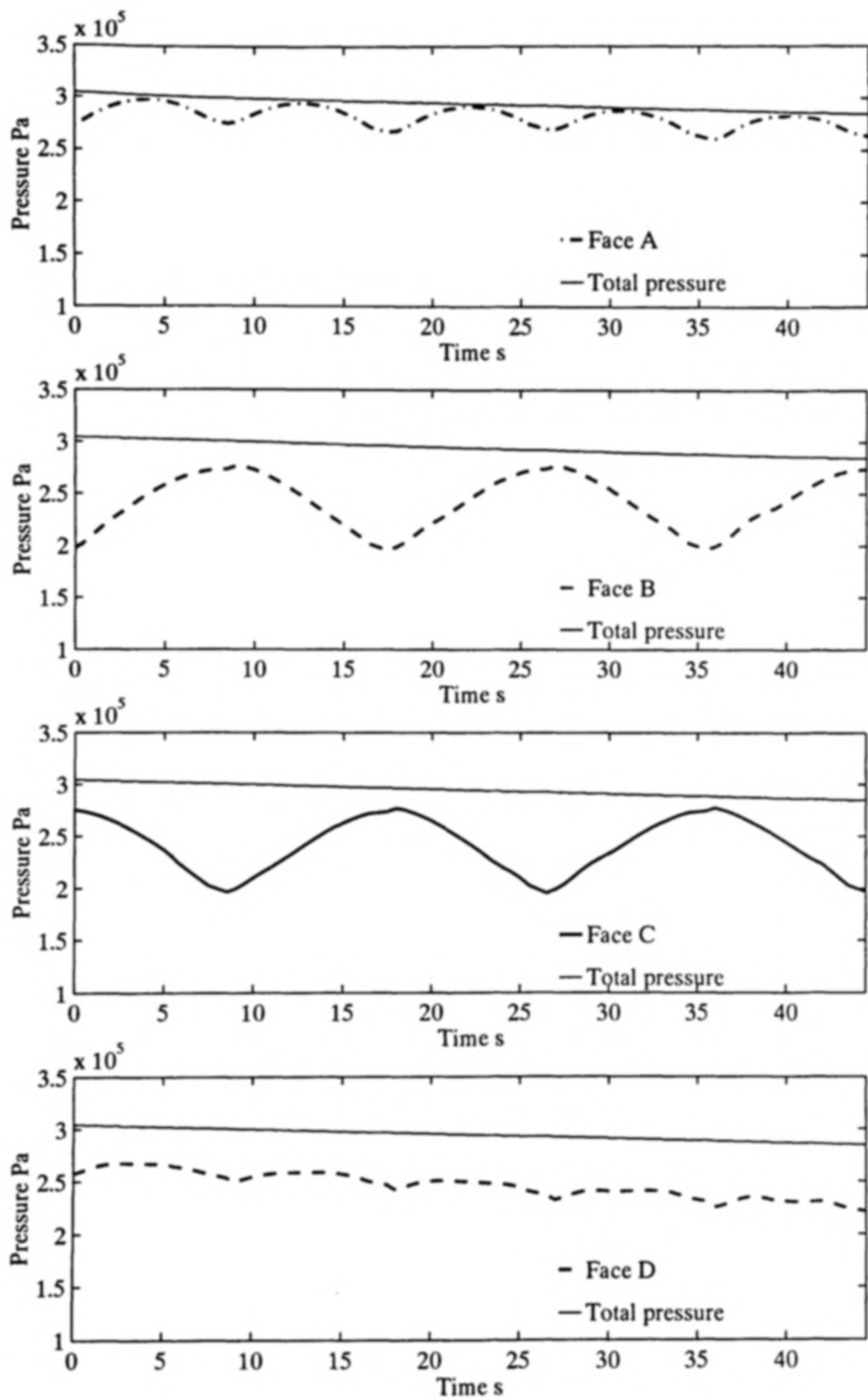


Figure 4.8 Face pressure variation over calibration

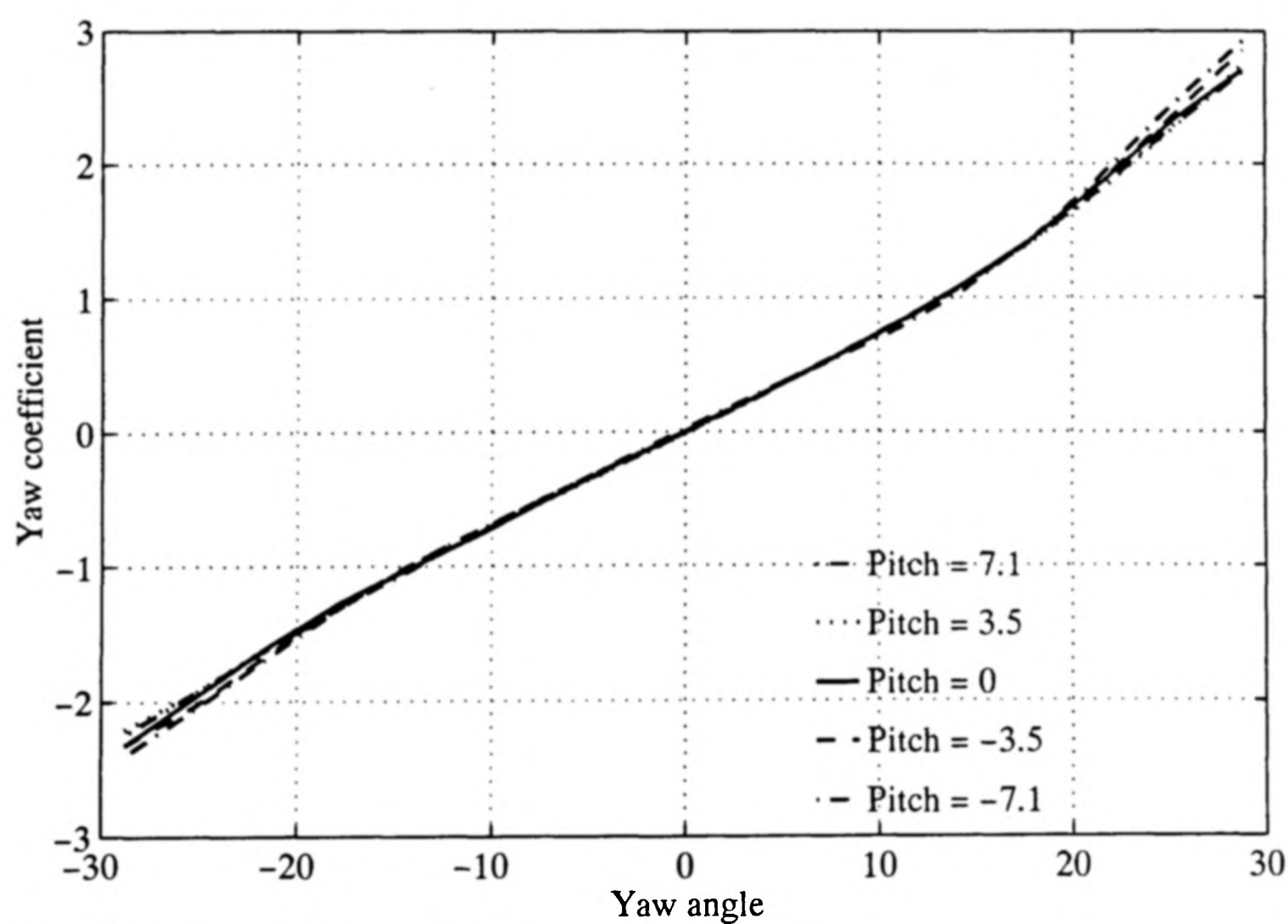
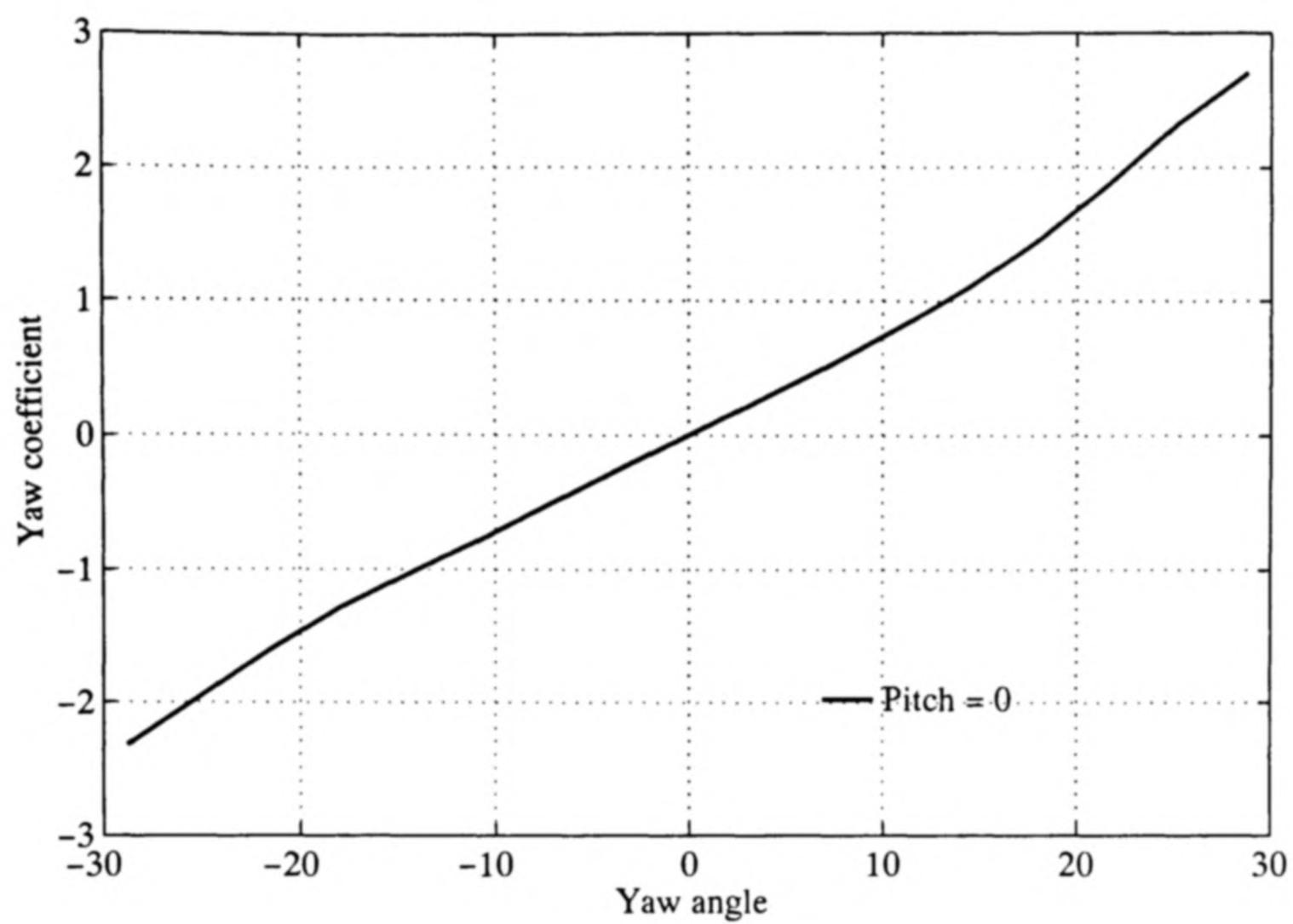


Figure 4.9 Yaw coefficient variation with yaw and pitch angle

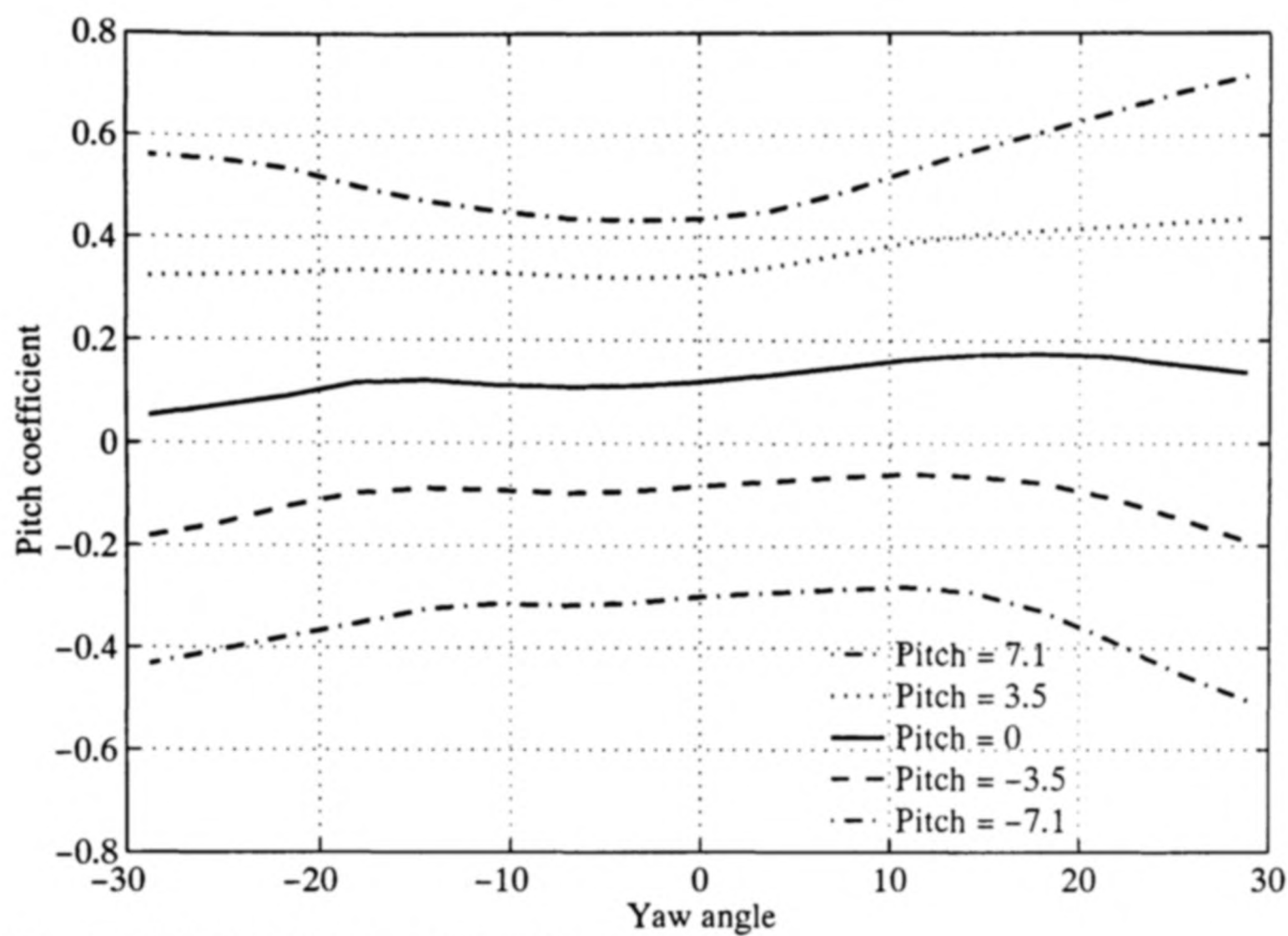


Figure 4.10 Pitch coefficient variation with yaw angle

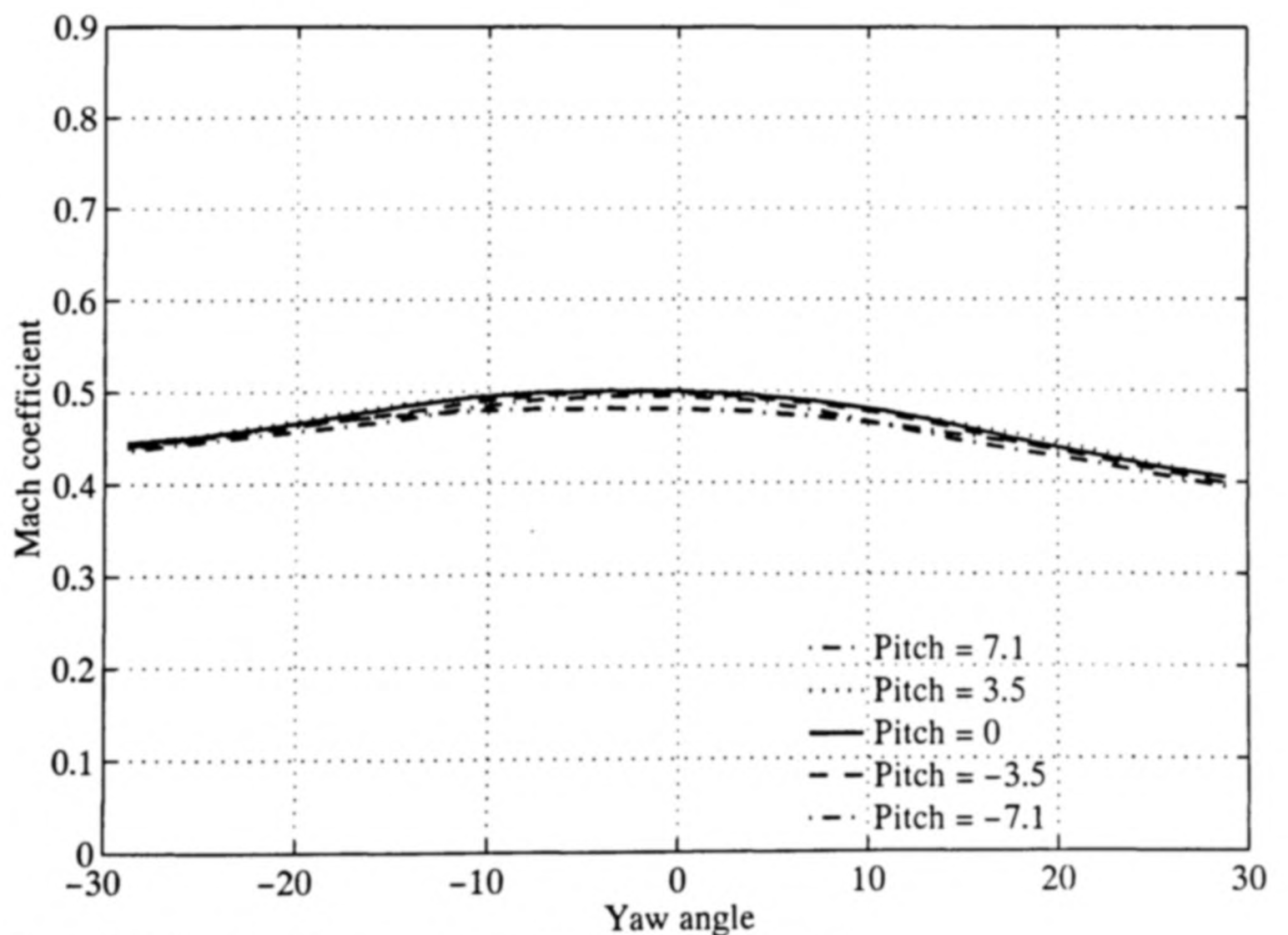


Figure 4.11 Mach coefficient variation with yaw and pitch angle

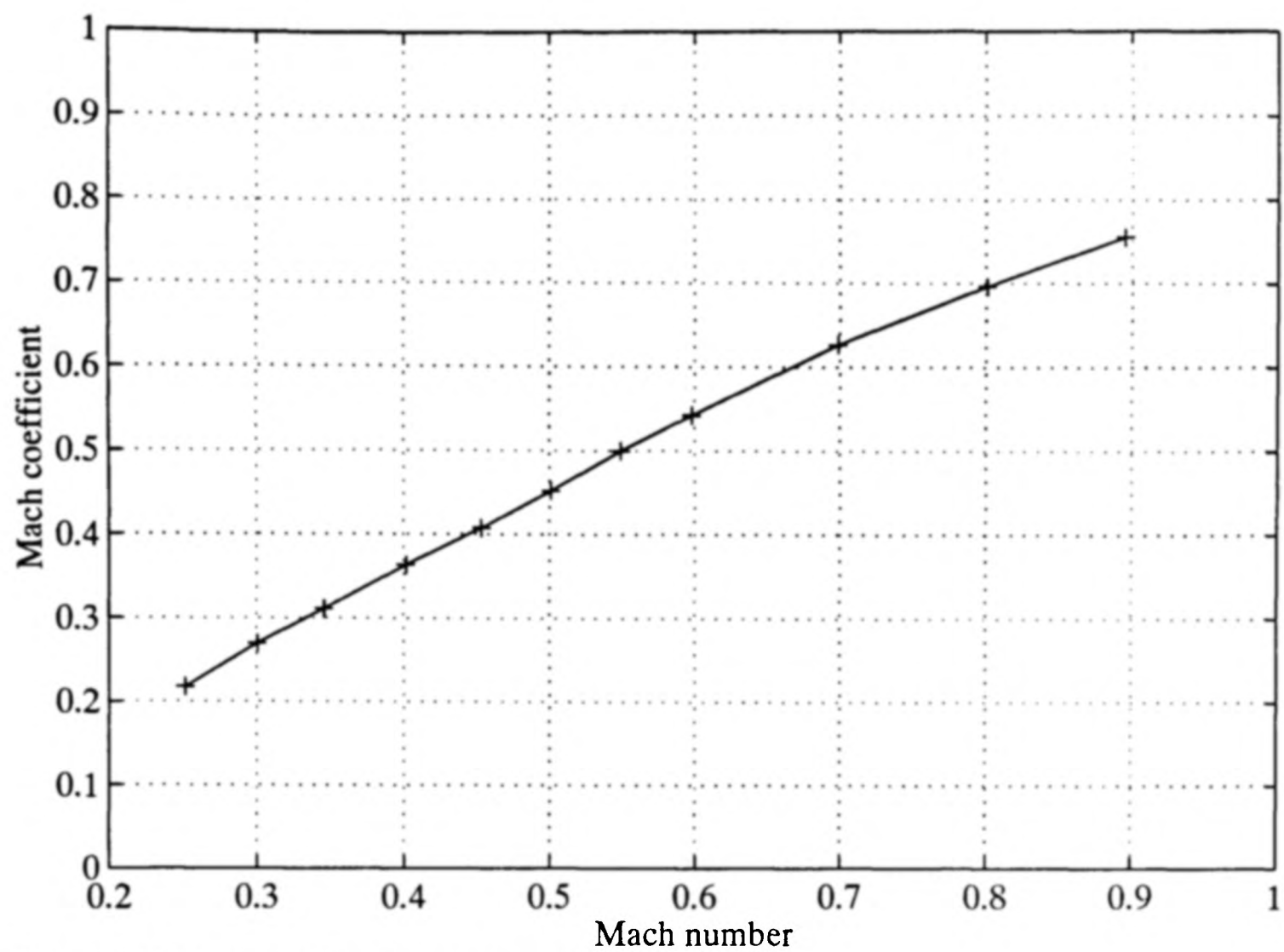


Figure 4.12 Mach coefficient variation with Mach number

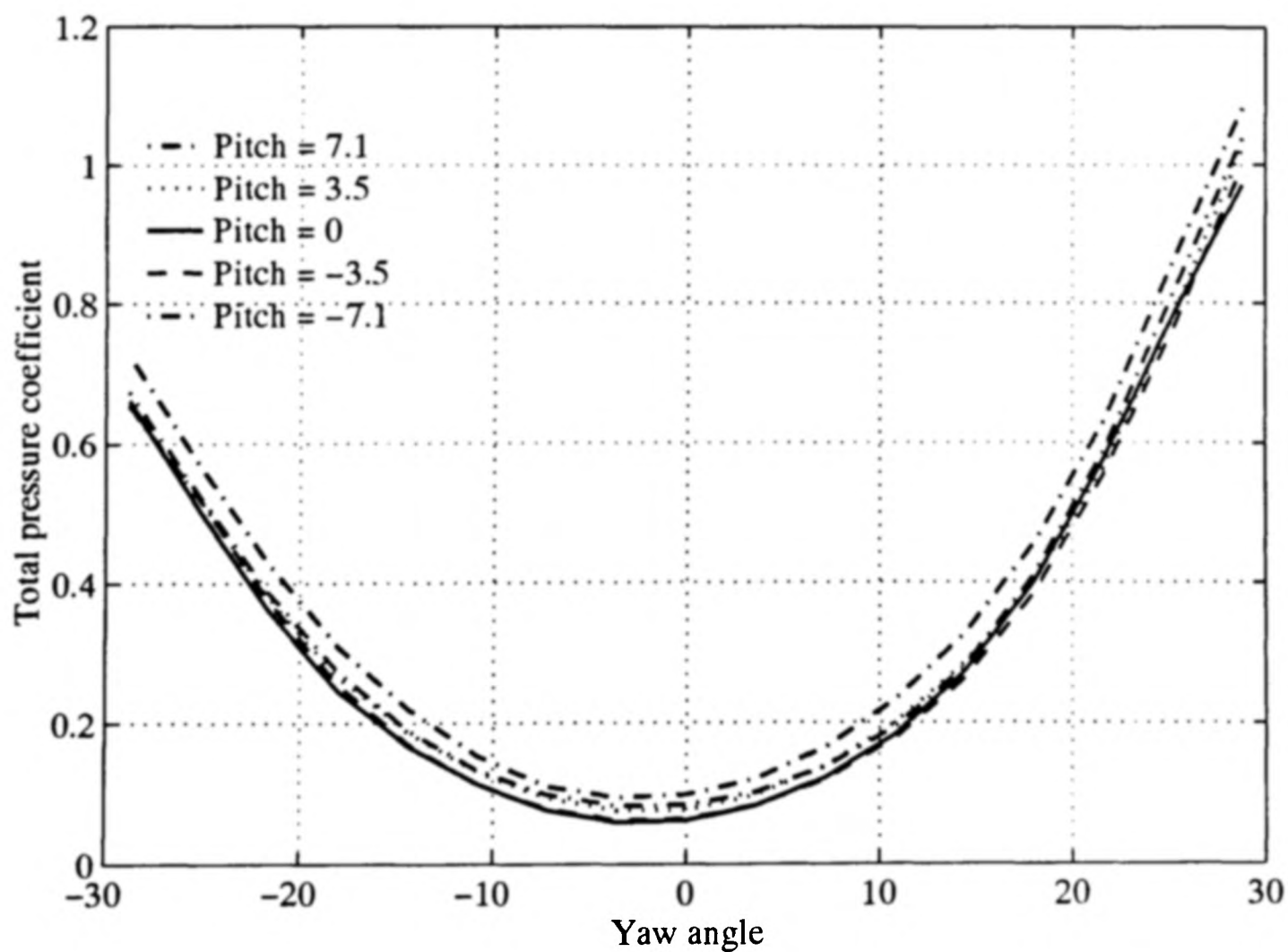


Figure 4.13 Total pressure coefficient variation with yaw and pitch angle

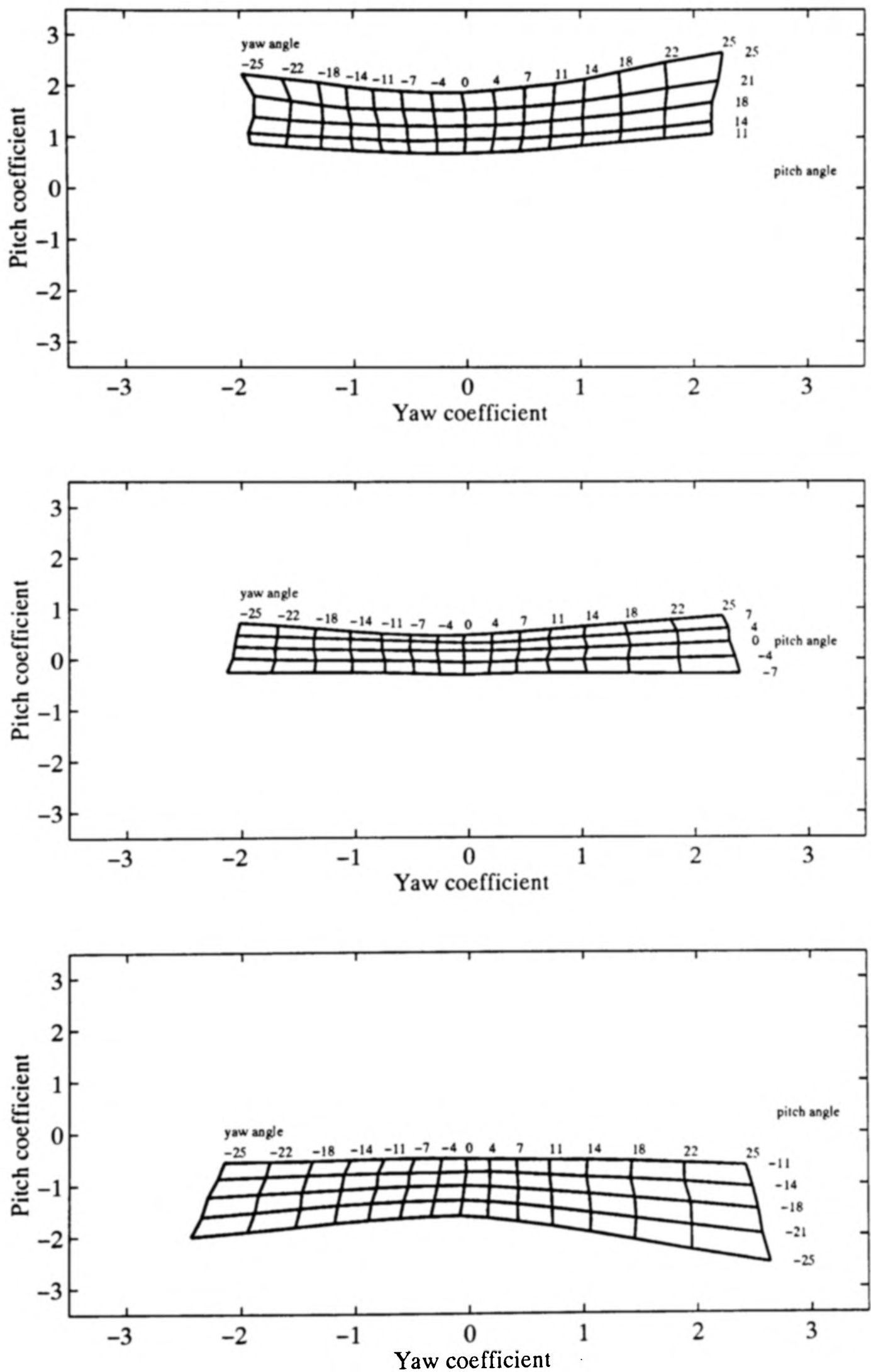


Figure 4.14 Three calibrations results at different combinations of pitch angle

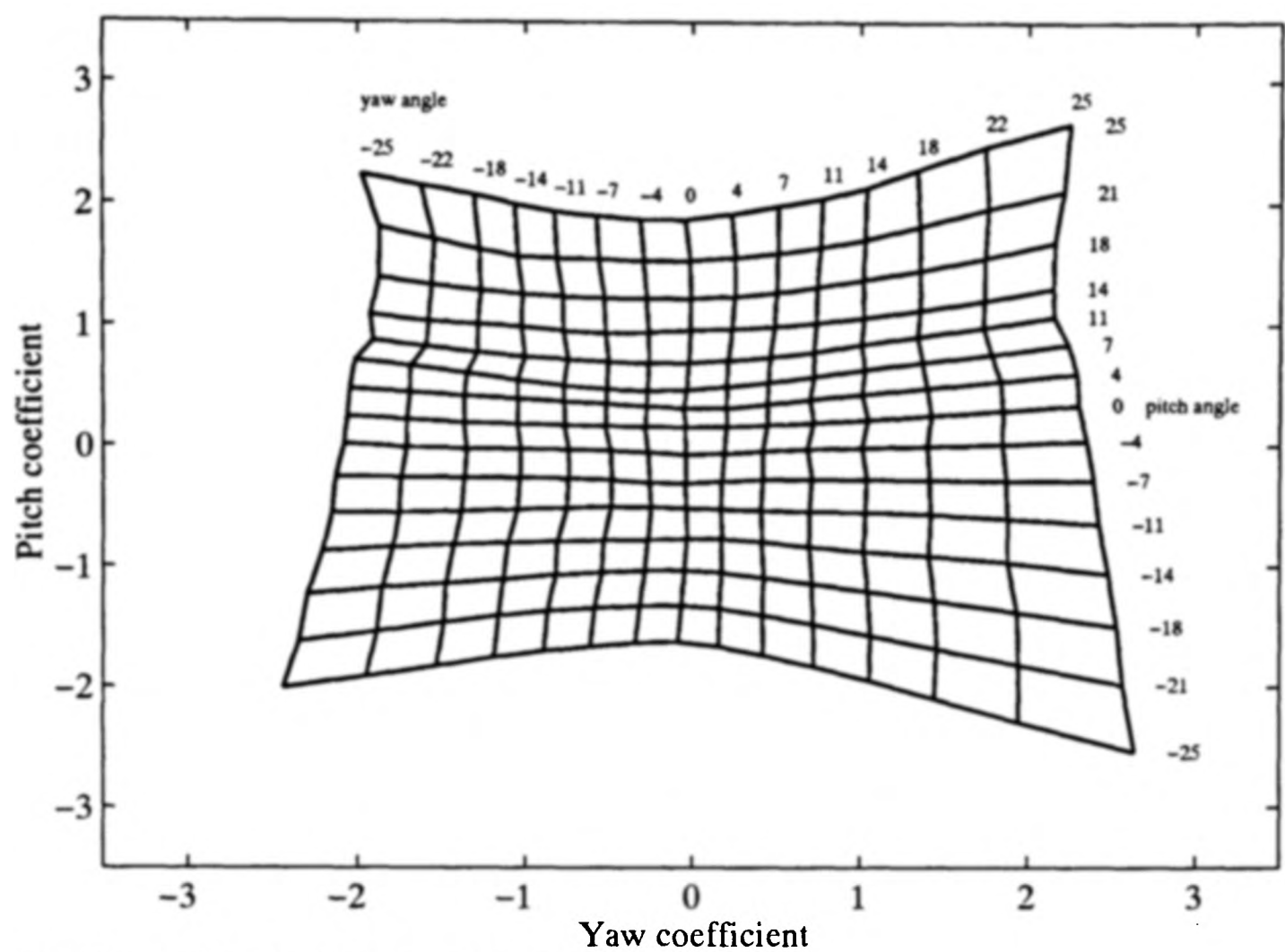


Figure 4.15 The combined calibration plane, Mach 0.55

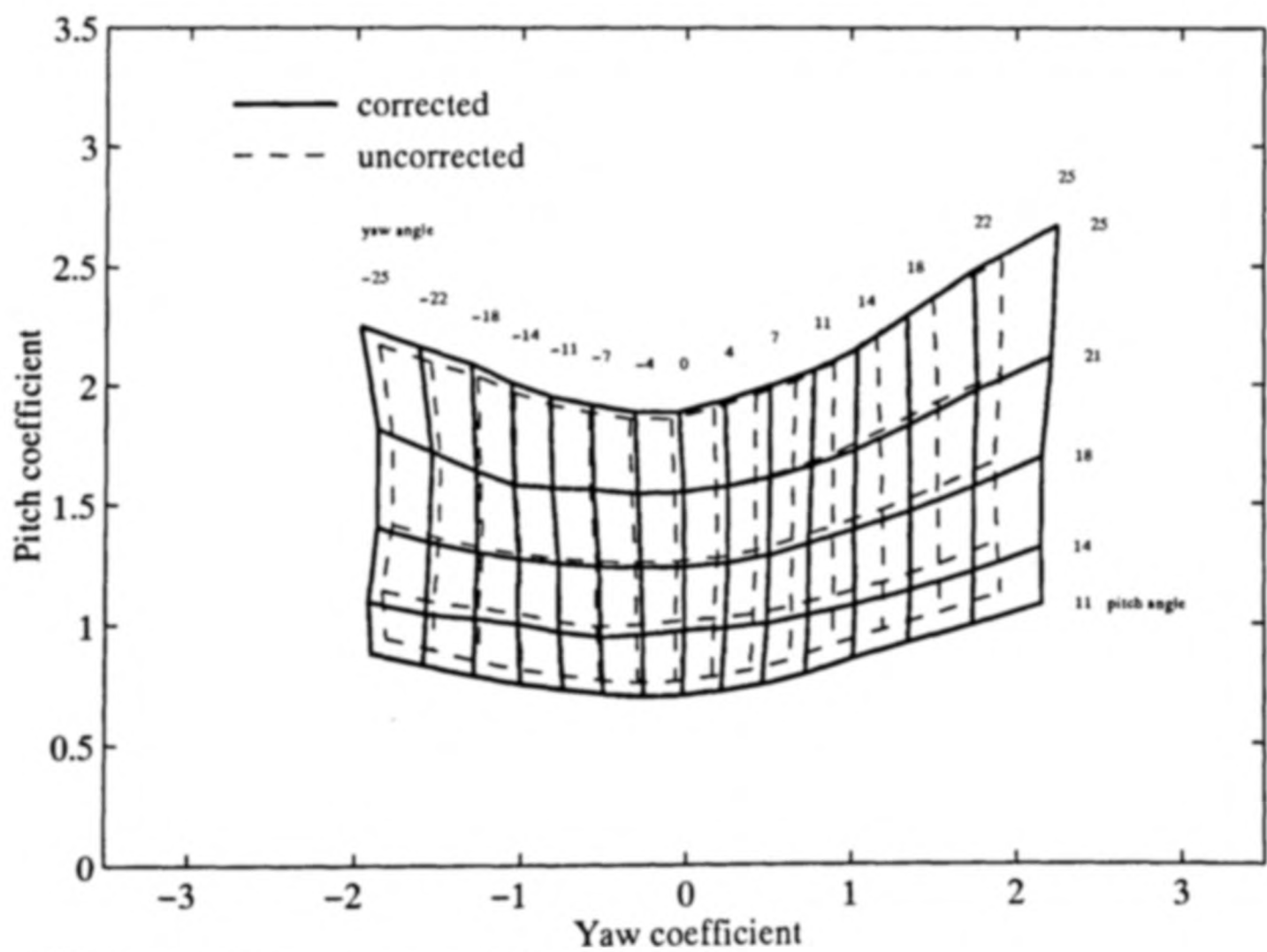


Figure 4.16 Influence of temperature correction

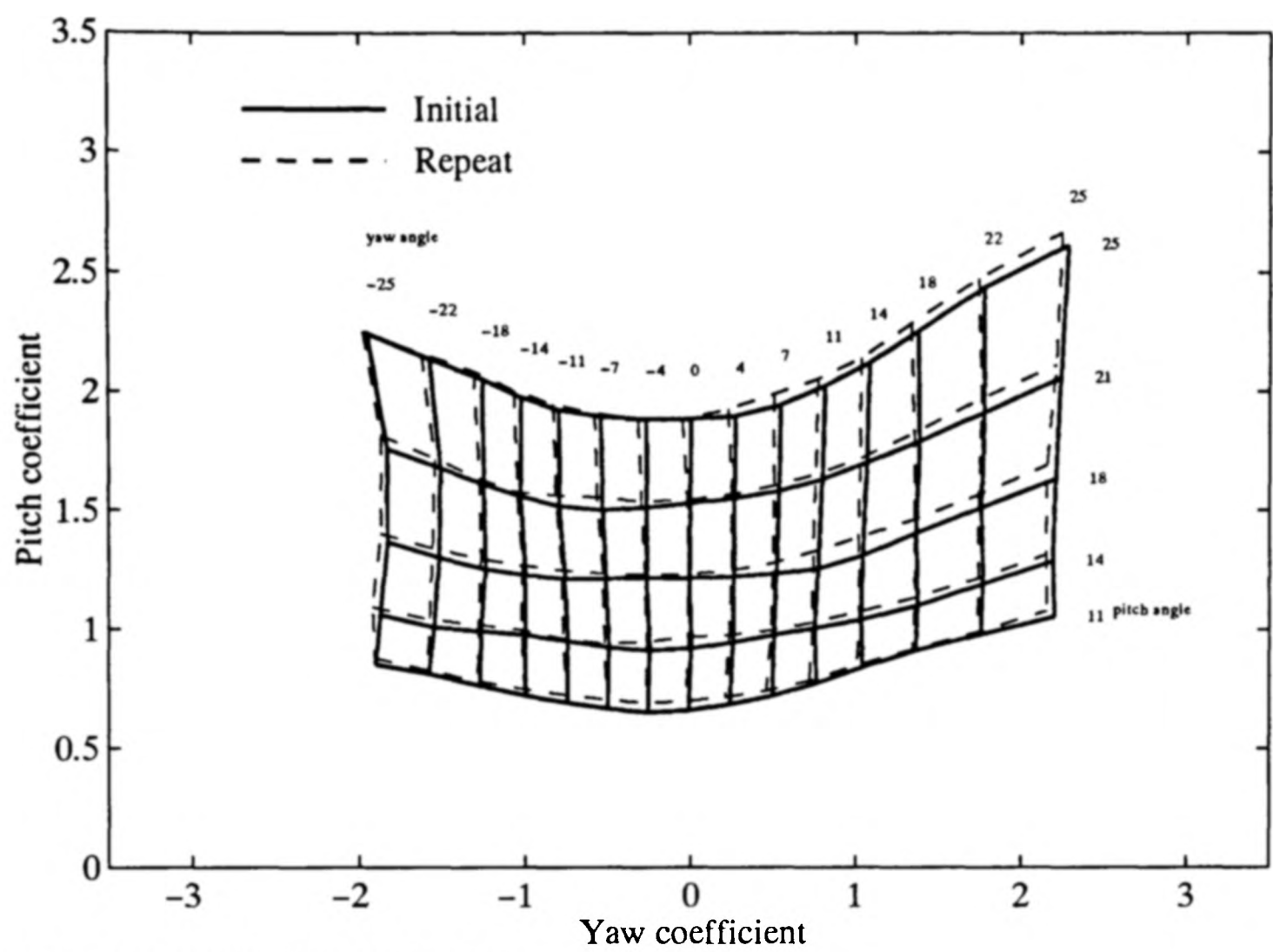


Figure 4.17 Repeatability of the calibration

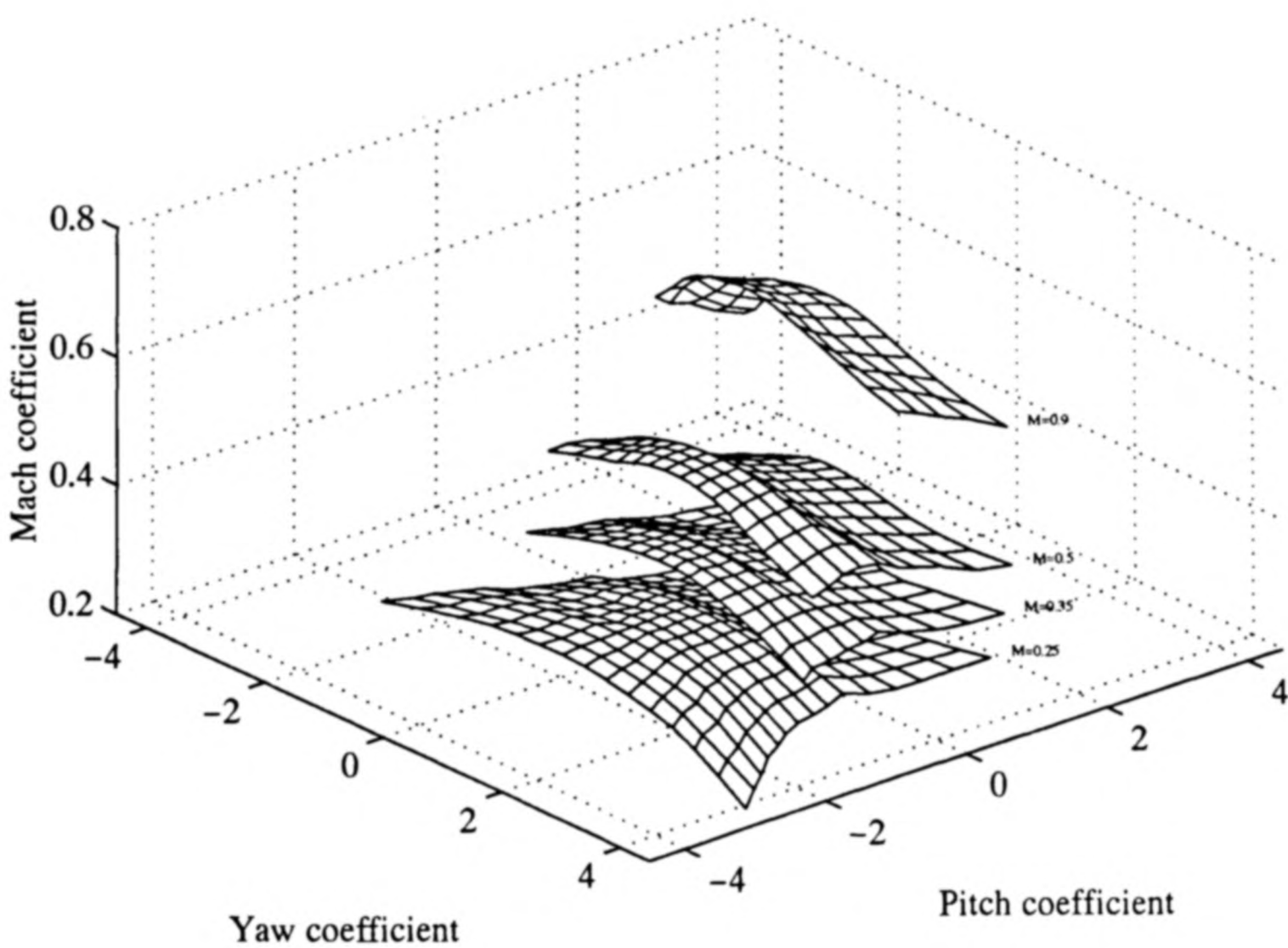


Figure 4.18 Partially constructed calibration space

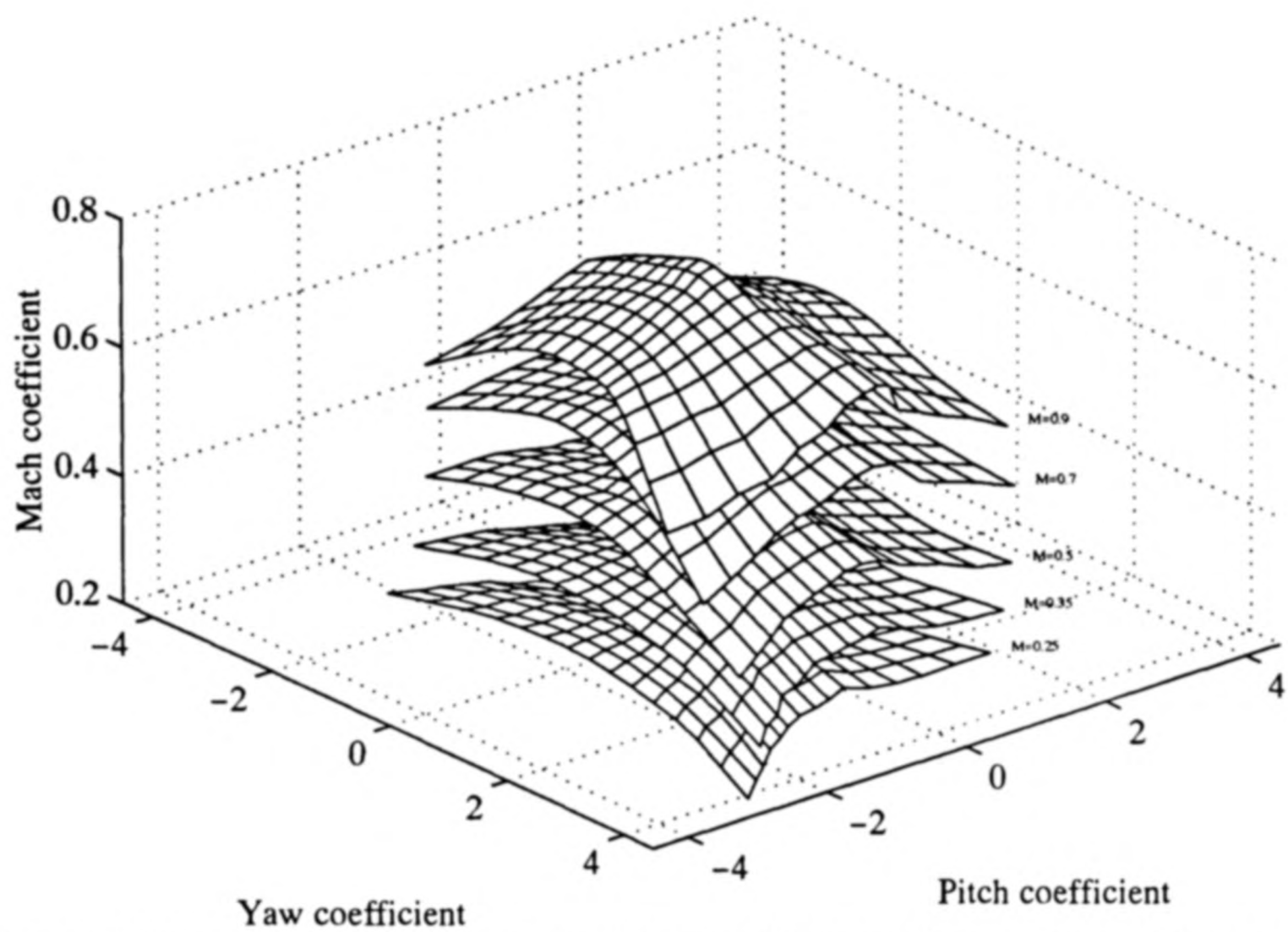


Figure 4.19 Complete calibration space showing planes of constant Mach number on which are plotted lines of constant yaw and pitch angles

5. Aerodynamic probes

5.0. Overview

This chapter details how an aerodynamic probe may be used to measure flow once it has been calibrated using the techniques outlined in the previous chapter. When the probe is used for measurement the aerodynamic calibration provides the mechanism to convert the pressures measured by the probe into Mach number, yaw angle, pitch angle, and total pressure. Before the calibration is applied it is inverted into a more manageable form using a technique called sequential interpolation. This inversion allows measurements to be derived directly from the pressure coefficients. The performance of a probe with respect to operating range and measurement error can be attributed to a variety of factors. A number of experiments on large scale models of aerodynamic probes are used to explain the important geometrical features.

5.1. Data inversion

5.1.1. Calibration data modelling

Once the calibration of an aerodynamic probe has been completed, it is ready for measurement of fluid flow. When the probe is placed in the flow, for instance behind a rotating turbine such as in chapters 6 and 7, the pressure sensors located on the probe will record the flow conditions in the vicinity of the probe. It is then possible to calculate coefficients in an identical way to those calculated during calibration. By comparison of these coefficients with those recorded during calibration the yaw angle, pitch angle, Mach number and total pressure can be found.

A probe will need to take measurements over a continuous range of combinations of yaw, pitch and Mach number, but it is only practical to calibrate an aerodynamic probe at a finite number of discrete flow combinations. Therefore, there needs to be a method by which the discrete calibration is extended to allow a continuous evaluation of real data. One approach to this transformation from a discrete calibration to continuous data is to use mathematical functions to fit the calibration data. This approach to data evaluation was used by Bohn and Simon [1975]. In their work the calibration

surfaces were approximated by a two variable polynomial function determined by a least squares method, equation 5.1.

$$z = f(x, y) = \sum_{i=0}^m \sum_{j=0}^n k_{ij} x^i y^j$$

Equation 5.1

For compressible flows a three variable polynomial function is required to allow for the variation of the calibration surfaces with Mach number. The additional variable greatly increases the difficulty of finding a good fit to the data. Koschel and Pretzsch [1988] found that using polynomial functions over a range of Mach numbers resulted in high residual errors and they resorted to applying separate approximations for a number of different Mach number ranges.

The alternative to a parametric model approach is where the experimental measurements are compared directly with the calibration. This may be achieved by storing the calibration data in the form of a 'look up' table. When experimental data is recorded all the results should fall between calibration points provided the probe is operating within the range of its calibration. The table allows interpolation between these calibration points to find the values at the positions recorded in the experiment. As it is only necessary to interpolate locally to the measurements good accuracy can be achieved. This contrasts with parametric models which can be compromised where, to find the best fit, the whole calibration space is considered. This last point is especially significant where the calibration data contains discontinuities.

5.1.2. Application of the calibration data

The calibration data may be used by directly comparing the values found during the experiment with those found during calibration. This approach may be demonstrated graphically if the calibration results are considered as a three-dimensional vector space with values of yaw angle, pitch angle and Mach number corresponding to yaw coefficient, pitch coefficient and Mach coefficient. This may be expressed as in equations 5.2-5.4.

$$C_\theta = C_\theta(\theta, \phi, M)$$

Equation 5.2

$$C_\phi = C_\phi(\theta, \phi, M)$$

Equation 5.3

$$C_M = C_M(\theta, \phi, M)$$

Equation 5.4

To show this graphically the calibration coefficients need to be plotted as a function of yaw, pitch and Mach number. This is how the data was plotted for the three-dimensional semiconductor probe in figure 4.19. To use the data presented in this way requires a process of guessing the yaw, pitch and Mach number, finding what the coefficients that this combination of conditions would give and then comparing these with the measured coefficients. To implement this process numerically requires iteration. The speed of convergence depends on the algorithm used and the behaviour of the calibration surface in the vicinity of the measurements. Epstein [1985] and Cooke [1989] used iterative methods to invert their probe measurements.

The second approach to applying the calibration data is to invert the calibration data before the experiment. The inverted data can then be used to directly find the experimental conditions measured by the probe. Returning to the graphical interpretation of this process, the data is transformed from the vector space defined in equations 5.2-5.4 to the vector space defined by equations 5.5-5.6 given below.

$$\theta = \theta(C_\theta, C_\phi, C_M)$$

Equation 5.5

$$\phi = \phi(C_\theta, C_\phi, C_M)$$

Equation 5.6

$$M = M(C_\theta, C_\phi, C_M)$$

Equation 5.7

The advantage of the direct method is that once the data has been inverted it becomes very easy to find the yaw, pitch and Mach number from any combination of coefficients measured by the probe. There is no need to use an iterative approach to find the final result. Whilst inversion of the data creates additional effort before the calibration data can be used, this effort need only be done once and final conversion of coefficients to measurements will be much faster.

Kuperferschmied and Gossweiler[1992] found that for their probe direct methods were up to seven times faster than indirect method using iteration. Although the comparison is specific to their calibration data and the algorithms used, this suggests that direct methods are more appropriate for the large data sets produced by fast response probes.

5.1.3. The sequential interpolation technique

At the same time as the author was developing the fast response aerodynamic probes for use in the Oxford rotor, Main [1994], was developing a four holed pneumatic pyramid probe for the measurement of loss in the Oxford Cold Heat Transfer Tunnel (CHTT), Jones [1978]. An effective method for applying the three-dimensional aerodynamic calibration data to experimental measurements was needed for both applications and resulted in the joint development of the sequential interpolation technique.

Parametric methods appear attractive because the calibration may be reduced to a small number of parameters, namely the parameters of the surface fits to the calibration data. However this advantage is lost because the calibration surfaces in practice are very complex and a surface fit is often compromised.

Under the method developed for these probes a 'look up' table is used. The calibration data is stored in a large matrix known as the probe calibration matrix in which each row in the matrix corresponds to one combination of yaw angle, pitch angle and Mach number. The values of the

coefficients are explicitly shown as a function of the yaw angle, pitch angle and Mach number. The probe calibration matrix may be expressed:

$$\begin{bmatrix} C_\theta & C_\phi & C_M & C_T \end{bmatrix} = f(\theta \quad \phi \quad M)$$

Equation 5.8

To facilitate obtaining experimental measurements this data is then inverted so that a direct method may be applied. The aim of the inversion is to get the calibration data in an array of uniform intervals of the values of the primary coefficients, C_θ C_ϕ C_M , and therefore the values θ , ϕ , M become an explicit function of the primary coefficients. The data in this form is called the probe application matrix:

$$\begin{bmatrix} \theta & \phi & M & C_T \end{bmatrix} = f(C_\theta \quad C_\phi \quad C_M)$$

Equation 5.9

With the data in this format it is very easy to look up any combination of the pressure coefficients C_θ C_ϕ C_M , to find the values of θ , ϕ , M . If a chosen combination of coefficients does not exist it is possible to interpolate between the points that lie closest to the chosen combination.

To convert from the probe calibration to the probe application matrix is non trivial and requires three-dimensional interpolation. M.G.L. Oldfield [1993] suggested that to generate the probe application matrix, sequential interpolation could be employed to get from the original matrix to the final matrix in three steps. This allowed the inversion of the entire calibration matrix with fast one-dimensional interpolation. The process, also described in Main [1994], can be summarised in the following steps:

Step 1: For each value of yaw and pitch interpolate between the Mach coefficient and the Mach number evenly spaced values of the Mach coefficient. Therefore the probe calibration matrix becomes a temporary matrix in which $\begin{bmatrix} C_\theta & C_\phi & M & C_T \end{bmatrix} = f(\theta \quad \phi \quad C_M)$

Step 2: For each value of Mach coefficient and yaw angle interpolate between the pitch angle and pitch coefficient evenly spaced values of the pitch coefficient. The first temporary matrix becomes the second temporary matrix $[C_\theta \ \phi \ M \ C_T] = f(\theta \ C_\phi \ C_M)$

Step 3: Finally for each value of Mach coefficient and pitch coefficient interpolate between the yaw angle and yaw coefficient evenly spaced values of the yaw coefficient. The second temporary matrix $[C_\theta \ \phi \ M \ C_T] = f(\theta \ C_\phi \ C_M)$ becomes the desired probe application matrix $[\theta \ \phi \ M \ C_T] = f(C_\theta \ C_\phi \ C_M)$.

It will be noticed that the total pressure coefficient, C_T , is included in the matrix during this interpolation. The total pressure coefficient is determined by which combination of yaw angle, pitch angle and Mach number the probe experiences and therefore takes no part in the inversion of the flow conditions. However the coefficient is included so that the total pressure may also be found once the yaw angle, pitch angle and Mach number are known.

For the probe calibrated in chapter 4, the calibrations were undertaken at 17 intervals of yaw angle, 15 intervals of pitch angle and at 10 Mach numbers. Generation of the probe application matrix takes minutes whilst the inversion of 8000 experimental results takes seconds on an IBM compatible 486DX2 with 16Mb of memory.

5.1.4. Inversion surfaces

After the aerodynamic calibration data from chapter 4 is inverted a convenient method of plotting the data is in the form of lines of constant yaw and pitch coefficient on axes of yaw and pitch angle. The data is plotted in this way for a constant Mach coefficient of 0.5 in figure 5.1. It is clear that the data plotted in this form is easier to use because for a given combination of yaw and pitch coefficient the corresponding value of pitch angle and yaw angle may be read directly from the axes.

To introduce the third coefficient for the Mach number requires a three-dimensional plot. In figure 5.2 three planes of constant Mach coefficient are shown, each with lines of constant pitch and

yaw coefficient plotted over their surface. In this plot the calibration space results have been shown from two different directions.

In chapter 4 it was explained that the form of the Mach coefficient given in equation 4.1 was chosen because it would be linear with Mach number. In figure 5.3 the Mach coefficient is plotted over a variety of Mach numbers and yaw angles, it can be seen to be approximately linear with Mach number. This contrasts with figure 5.4 where instead of the Mach coefficient, the probe pressure ratio has been plotted. It is more convenient to use a uniform grid when programming the inversion routine. If the coefficient is non-linear this means that a uniform grid is likely to be too dense in some areas and too sparse in others. This makes the Mach coefficient more suitable for linear interpolation.

5.2. Measurement errors

Measurement errors with the probe may be caused for a variety of reasons, the most significant factors are the errors in the calibration process, inversion errors and errors resulting from the pressure measured by the transducers of the probe.

5.2.1. Calibration errors

There are likely to be small errors arising from inaccuracies in the calibration. These will mainly be a result of the inaccuracy in the positioning the probe. The stepper motors have a repeatability to $\pm 0.2^\circ$ but there may be a systematic error in the positioning of the probe as a result in the difficulty of aligning a relatively small object with the axis of the calibration facility. An uncertainty of $\pm 1^\circ$ degree will comfortably accommodate the systematic error.

5.2.2. Inversion errors

The density of the grid chosen for inversion of the calibration data will affect the accuracy of the inversion process. The choice of grid size for the inverted data is a compromise between speed and accuracy. The denser the grid the more accurate the inversion will be, until the limitations of the accuracy of the original calibration data are reached. The denser the calibration grid, the larger the inverse data will be and the longer the inversion will take.

The quality of the inversion depends on where in the operating range of the probe the results are to be used. Table 5.1 gives the 95% confidence limits for data over the approximate range of expected flow conditions in the Oxford rotor as the grid size of the inversion is increased. A confidence interval is used because the calibration data errors are dependent on where within the probes operating envelope the measurements are being recorded. As the table shows the upper limit on the grid size was determined by the memory available of the computer used for the inversion. However as the results from a less fine grid were perfectly satisfactory it was not considered necessary to attempt to optimise the code for memory management or seek additional hardware. The inverted data generated in the earlier figures 5.1 and 5.2 was generated with data at intervals of 0.2 in both yaw and pitch coefficient, and 0.02 intervals of Mach coefficient.

Pitch coefficient grid size	Yaw coefficient grid size	Mach coefficient grid size	95% confidence pitch angle	95% confidence yaw angle	95% confidence Mach number	Inversion time (secs)
0.1	0.1	0.01	Insufficient memory			
0.15	0.15	0.015	0.35°	0.22°	0.004	126
0.2	0.2	0.02	0.48°	0.29°	0.005	92
0.4	0.4	0.04	0.7°	0.5°	0.01	29
0.8	0.8	0.8	1.1°	0.74°	0.013	13
1.6	1.6	0.16	1.8°	1.57°	0.014	9

Table 5.1 Influence of grid size on inversion

In figures 5.5-5.7 the original calibration points are compared with the values found after inversion of their corresponding coefficients through the probe application matrix. These plots show that the inversion process finds the correct value within an acceptable error.

The plots in figure 5.5-5.7 assume that the measurement errors are fully independent. However this is clearly not the case and therefore to demonstrate the cross dependence of errors a three-

dimensional plot is shown in figures 5.8-5.10. In the first figure the error in yaw has been plotted over the whole calibration space of yaw angle, pitch angle and Mach number. The largest errors occur at high angles of combined pitch and yaw. Similar effects can be seen in the second two figures for pitch and Mach number.

5.2.3. Pressure error

In chapter 3 it was shown that the pressure uncertainty of each transducer was 0.226% full scale. To generate each calibration coefficient requires four pressure measurements and therefore error in each coefficient can be estimated assuming that the errors are independent by taking the root mean square of the four components to give an error of the order 0.45% full scale. In absolute terms this level of error corresponds to approximately 1575 Pa. This figure has been used to predict the measurement error over the calibration space by considering a 1575 Pa error in the numerator of the coefficient. In figures 5.11-5.13 the errors are plotted in a similar fashion to the method used in figures 5.8-5.10 but this time after the error in the coefficient has been included. It can be seen that the Mach number becomes very important in determining the magnitude of error. At the lower Mach numbers the errors are the greatest. This is because the magnitude of the pressure differences between the faces at the lower Mach numbers are less than at higher Mach numbers. The result is that the pressure errors are proportionally larger at the lower Mach numbers.

It is important to put the inversion error in the context of the absolute pressure measurement error. Through all three figures the pressure errors are clearly more significant than those from the inversion of the data and therefore further optimisation of the inversion process is not necessary.

5.3. Large scale modelling

Experiments with large scale models of the probe allow the detailed flow field over the probe to be studied using visualisation and surface pressure measurement. A large scale model of the wedge geometry was studied by Ainsworth and Stickland [1992], and valuable conclusions for the design of

semiconductor versions of the wedge probe were made. It was shown that varying wedge angle had much less influence on the probe sensitivity than the position of the pressure tapping on the probe.

The three-dimensional probe has been made using the truncated pyramid to provide the four faces for pressure measurement. To understand the flow field around the truncated pyramid geometry a number of experiments on large scale models were undertaken. Four models of pyramid type probes were manufactured from perspex at a scale twenty-five times that of a semiconductor probe. These models were manufactured with semi-angles of 15° , 22.5° , 30° and 45° , a photograph of three of the models is shown in figure 5.14.

The models were tested in the 2m by 4m low speed wind tunnel at the Osney laboratory. The tunnel is capable of operating at continuous speeds up to 10 m/s. It has two constant speed, variable pitch fans with a feedback control system to set the flow speed. The upstream test section was designed to provide a smooth uniform flow and this has been measured to be uniform to within 1.2% and with a turbulence intensity of less than 0.8%. The low speed wind tunnel was clearly unable to match the Mach numbers encountered in turbomachinery, but the scale of the probes gives, for a tunnel speed of 10 m/s, a Reynolds number of 8.6×10^4 . On the Oxford rotor experiment, detailed in chapter 6, the probe Reynolds number was 1.4×10^5 , and therefore the aerodynamic behaviour of the probe in the large scale tests and in the rotor should be similar.

In the first series of experiments the probe heads were used for flow visualisation with paraffin smoke streamers. The most significant feature highlighted by flow visualisation was the laminar separation bubble formed over the leading edge of the probe. This separation bubble is characterised by separation over the sharp edge of the probe face and a re-attachment of the flow further down the face. Figure 5.15 is a photograph of a 30° semi-angle probe head positioned at 20° incidence to the flow in which the separation bubble has been highlighted with a smoke streamer. At very large angles of incidence the flow completely separates from the probe. As the semi-angle was reduced the

separation was found to occupy a larger proportion of the pressure face and the complete separation from the probe occurred at lower angles of incidence.

Close examination of the truncated pyramid geometry will reveal that there are nine prismatic edges over which it is possible for the flow to separate. The flow over the probe at combined pitch and yaw is a complex pattern of separations over the probe's many faces. These separations around the probe are therefore complicated and would be difficult to model analytically. Guzman et al [1994] undertook a computation solution to the flow by using a code that solved the incompressible steady state Navier-Stokes equations. Stream lines plotted from the computation results were very similar to the experimental results seen with flow visualisation.

Since separation is a viscous effect it was important to see if the Reynolds number had any affect on the separation behaviour. Over the limited range of Reynolds numbers possible in the low speed wind tunnel (from 1×10^4 to 1×10^5) it was noted that the separation was invariant. Whilst this test was not comprehensive, it suggested that for Reynolds numbers corresponding to rotor exit, there would be no Reynolds number effect on the separation bubble on the probe. This is important since if the separation behaviour were to change, any aerodynamic calibrations would be sensitive to Reynolds number.

Shepherd [1981] designed his pyramid probe with a 45° semi-angle head to give maximum angular sensitivity at low incidence angles. This same geometry was chosen for detailed experimentation on a large scale probe. The probe head was manufactured with pressure tapings located over the pyramid faces. These were used for pressure measurements over the probe faces as the incidence of the probe was altered. The probe was supported in a frame which allowed the pitch and yaw of the probe to be altered. The probe was positioned at angles over a range of $\pm 30^\circ$ in yaw and $\pm 20^\circ$ in pitch, at intervals of 5° whilst the pressure from the surface tapings were recorded.

The low speed wind tunnel is usually used for wind engineering research, rather than probe calibration, and is subject to a slow oscillation in flow speed with time. The cycle period for this

oscillation is approximately 5 seconds. The tunnel has a Scanivalve pressure measurement system in which multiple pressure tapings are sequentially measured by a single accurate pressure transducer. The Scanivalve pressure measurement system on the tunnel can cycle through 48 pressure tapings sequentially. The pressure at each tapping is sampled at 10ms intervals. The experimental scatter versus sample period was measured and the best compromise between scatter and sample period was found to be if the readings from each pressure tapping were sampled for one minute sequentially. This period allowed the data to be averaged over 12 cycles of the tunnel speed. The pressure sensor used in these experiments was a Setra 237 low range pressure transducer with a full scale of 34.4mB (0.5 psi).

In figure 5.16 the variation of pressure coefficient with pitch angle is plotted for face D of the pyramid probe at five locations along the centre chord of the face. The position along the length of the face have been non-dimensionalised by dividing its axial distance from the leading edge, x , by the face chord, c . The yaw angle for this data is zero and the pressure coefficient has been defined as the pressure relative to atmosphere divided by the tunnel dynamic pressure. The effect of the laminar separation bubble can be seen on the pressure measured by the forward pressure tapping, $x/c=0.1$. At an angle between 0° and -5° the laminar separation bubble forms and the pressure at the tapping drops dramatically beneath the bubble. As the pitch angle becomes more negative the separation bubble grows and between -20° and -25° it starts to enclose the second pressure tapping at $x/c=0.23$.

The separation bubble appears to affect the pressures of the pressure tapings behind the separation. As the pressure of the forward tapping drops between 0° and -5° there is also a change in the gradient of the pressure measured by the other tapings. This is because the separation bubble deflects the flow and changes the relative incidence of the flow further up the face. Once the separation bubble has formed the rearward pressure tapings become almost insensitive to any change in incidence.

The information from the four faces may be combined to generate a carpet plot of yaw and pitch coefficients for various combinations of pitch and yaw. Using the coefficients defined in

equations 2.19 and 2.20 it is possible to predict the calibration of the semiconductor probe and more specifically the influence of the transducer position on probe performance. This influence is quite dramatic. In figures 5.17-5.19 the variations of the calibration coefficients are plotted using the pressures measured at different axial locations for each of the probe faces.

The size of the aerodynamic coefficients increases markedly as the sensor site is moved closer to the prismatic edge, figure 5.17. This implies the probe is more sensitive to flow angle. Any experimental uncertainty in the pressure measurement will cause a corresponding uncertainty in the coefficients. For the same uncertainty in pressure measurement a more sensitive probe will have a lower overall uncertainty.

Close to the prismatic edge the separation bubble exaggerates the pressure differences measured, especially for small changes in incidence near to zero pitch and yaw. However, the cost of this increased sensitivity are coefficients that have cross dependency. Cross dependency is where a change in pitch affects the yaw coefficient and or a change in yaw affects the pitch coefficient. This is undesirable because an uncertainty in one measurement will cause an uncertainty in the other. The cross dependency of the coefficients can be tolerated if each combination of Mach number, yaw angle and pitch angle corresponds to a unique combination of Mach, yaw and pitch coefficient. However it is desirable to design the probe geometry to minimise this cross dependency because it also lowers the cross dependency of error.

Away from the prismatic edge, figure 5.19, the probe is less sensitive and, more seriously, collapses onto itself at higher angles of attack. The insensitivity of the rearward transducers to negative pitch angle contributes to make the overall sensitivity of the probe much lower. This suggests that positioning the transducer to the rear of the side faces would be undesirable.

A good compromise appears to occur with pressure tapings at an axial chord of 0.36, figure 5.18. The calibration plot shows good overall sensitivity and the cross dependency of the coefficients found with the forward pressure tapings has been reduced. This location is approximately where the

transducers on the semiconductor probes are located. It was interesting (and encouraging) to see many similar features occurring in both the large scale model data and that obtained from the semiconductor probe, (compare figure 5.18 with figure 4.15).

5.4. Discussion

The choice of the 45° truncated pyramid for an aerodynamic probe has worked well. This geometry allows three-dimensional measurement with four pressure measurements rather than five as normally associated with five-holed probes. This has allowed the probe head for the semiconductor probe to be smaller than it would have been if five transducers had been required in the probe head.

The calibration maps in which the yaw and pitch coefficient are plotted along lines of constant pitch and yaw are a convenient method of showing the probe's directional sensitivity. In general the calibration maps showed good uniformity and orthogonality over the calibration range.

An alternative choice of geometry might have been the three dimensional wedge probe, figure 2.11. The calibration surface for this geometry as measured by Cherrett et al [1992] is plotted in figure 5.20. The contrast between the uniformity of the grid with that of the pyramid geometry probe is quite clear. The problem with this probe will be its sensitivity to negative pitch angle. In the pitch plane only one pressure reading surface is sensitive to any changes in incidence. When the probe is at negative incidence the effect of the separation bubble on the top face gives it little pressure sensitivity to changes in pitch angle.

Closer examination of the pyramid probe's calibration surface reveals several distinct features. The calibration surface is symmetrical in yaw but not in pitch. In fact there was a distinct 120° division of the map surface, an effect which is a direct consequence of the pyramid probe geometry.

For the pyramid probe face D has a surface normal in the pitch direction. However for the side faces B and C the surface normal is neither in the pitch nor the yaw direction. For this to be the case, the co-ordinate system would have to be rotated clockwise by 120° for face B or 240° for face C. The yaw coefficient uses the pressure difference between the two side faces B and C to measure a yaw

pressure. Since movement of the probe in the yaw direction has an equal and opposite effect on the incidence of faces B and C, it is reasonable to expect symmetry in the yaw direction.

However, when moving the probe in the pitch direction the behaviour of face D will differ from that of faces B and C. Faces B and C are less sensitive to changes in pitch angle than face D and therefore the behaviour of the probe is not symmetrical in the pitch axis.

The separation bubble was seen to influence the pressure sensitivity of the probe face. This effect will be seen on all the three side faces. To consider when the flow separation bubble occurs on the side faces it is convenient to remember that if the co-ordinate system is rotated by either 120° or 240° the geometry of the probe faces are identical. Therefore it is when the probe is moved in the directions of the surface normal to sides B and C that we expect the separation bubble to influence the surface pressure in the same way in which the face D was influenced.

The difference in the experiments between the large scale and the semiconductor probes that has been ignored has been the small scale detail of the semiconductor probe. The large scale probes have edges that are sharp whereas the semiconductor probe, because of the enamel layer used in its fabrication, had much more rounded edges. The influence of this may be to change the nature of the separation bubble. The angle at which the separation bubble develops and its magnitude are very likely to be changed.

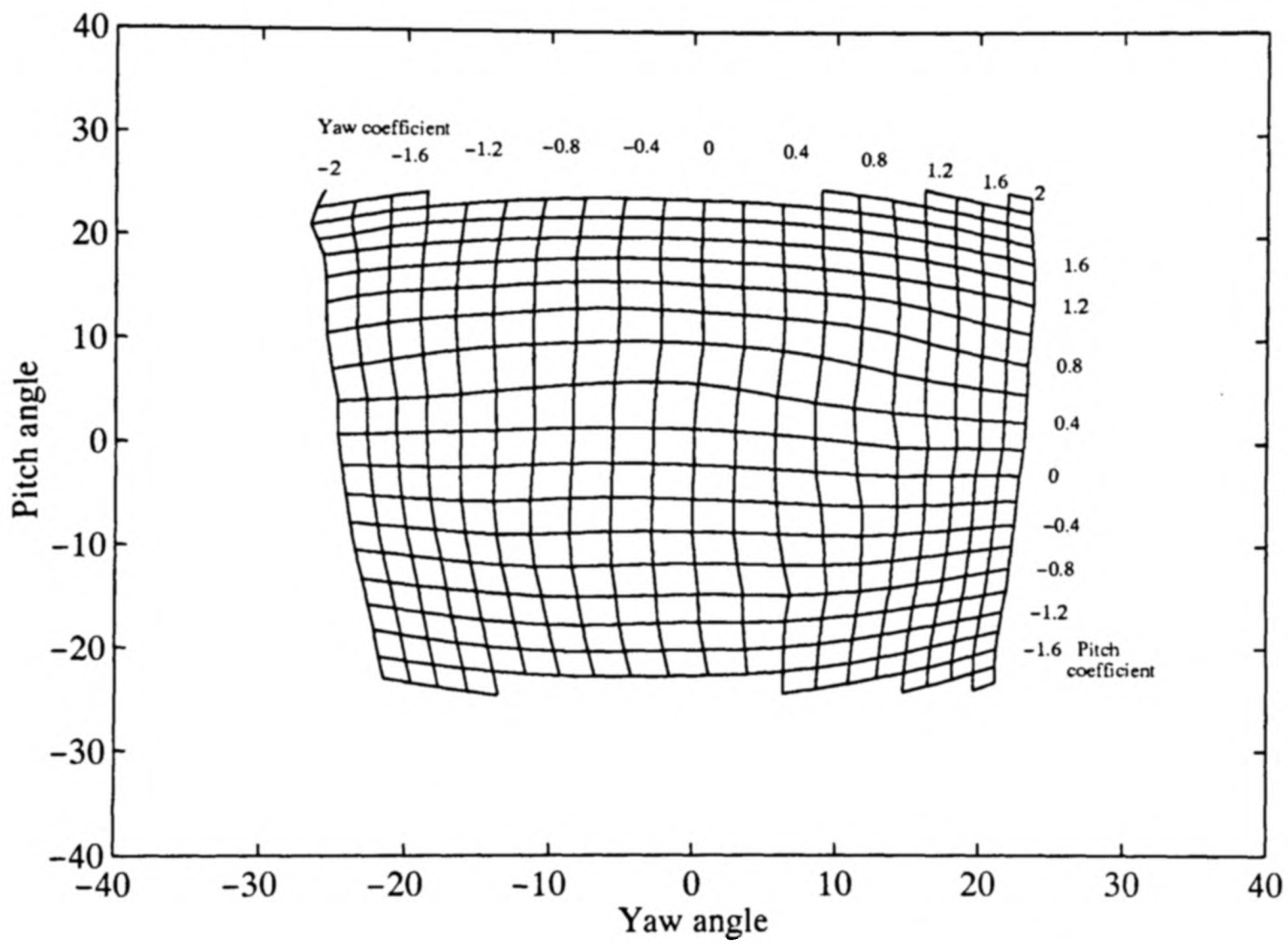


Figure 5.1 Lines of constant yaw and pitch coefficient plotted where Mach coefficient is 0.5, plotted at intervals of 0.2

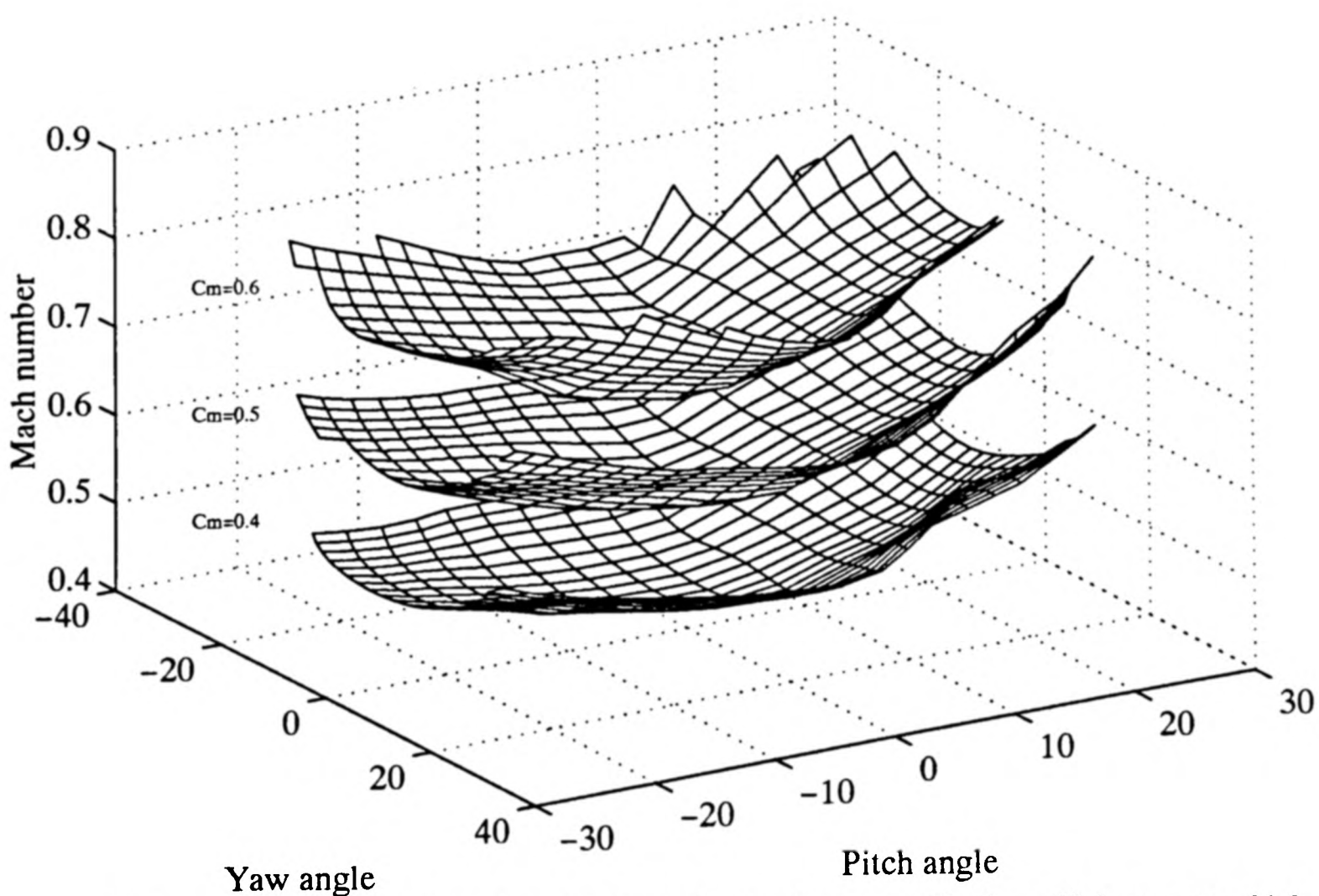
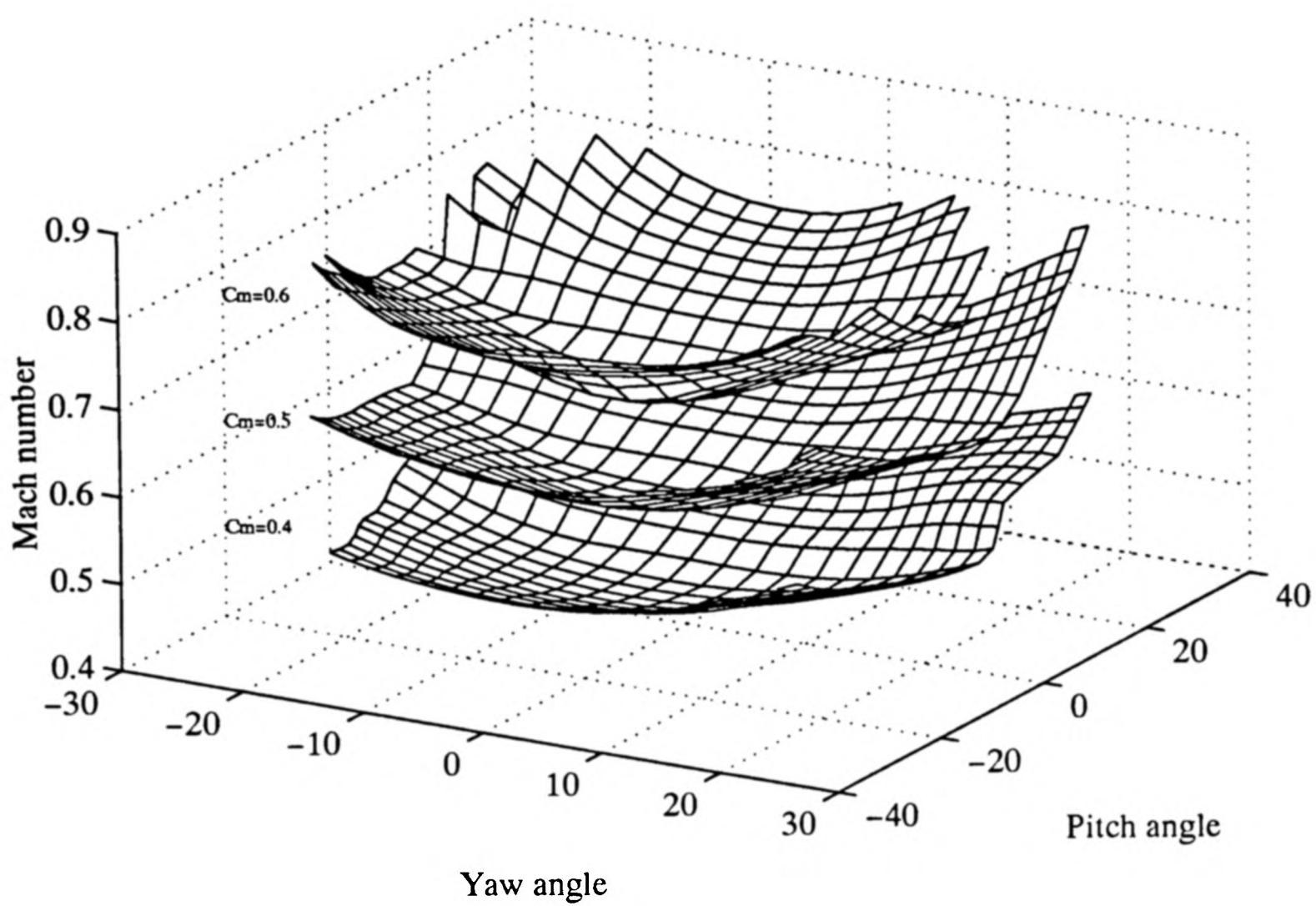


Figure 5.2 Inverted calibration data showing three planes of constant Mach coefficient, onto which are plotted lines of constant pitch and yaw coefficient at intervals of 0.2

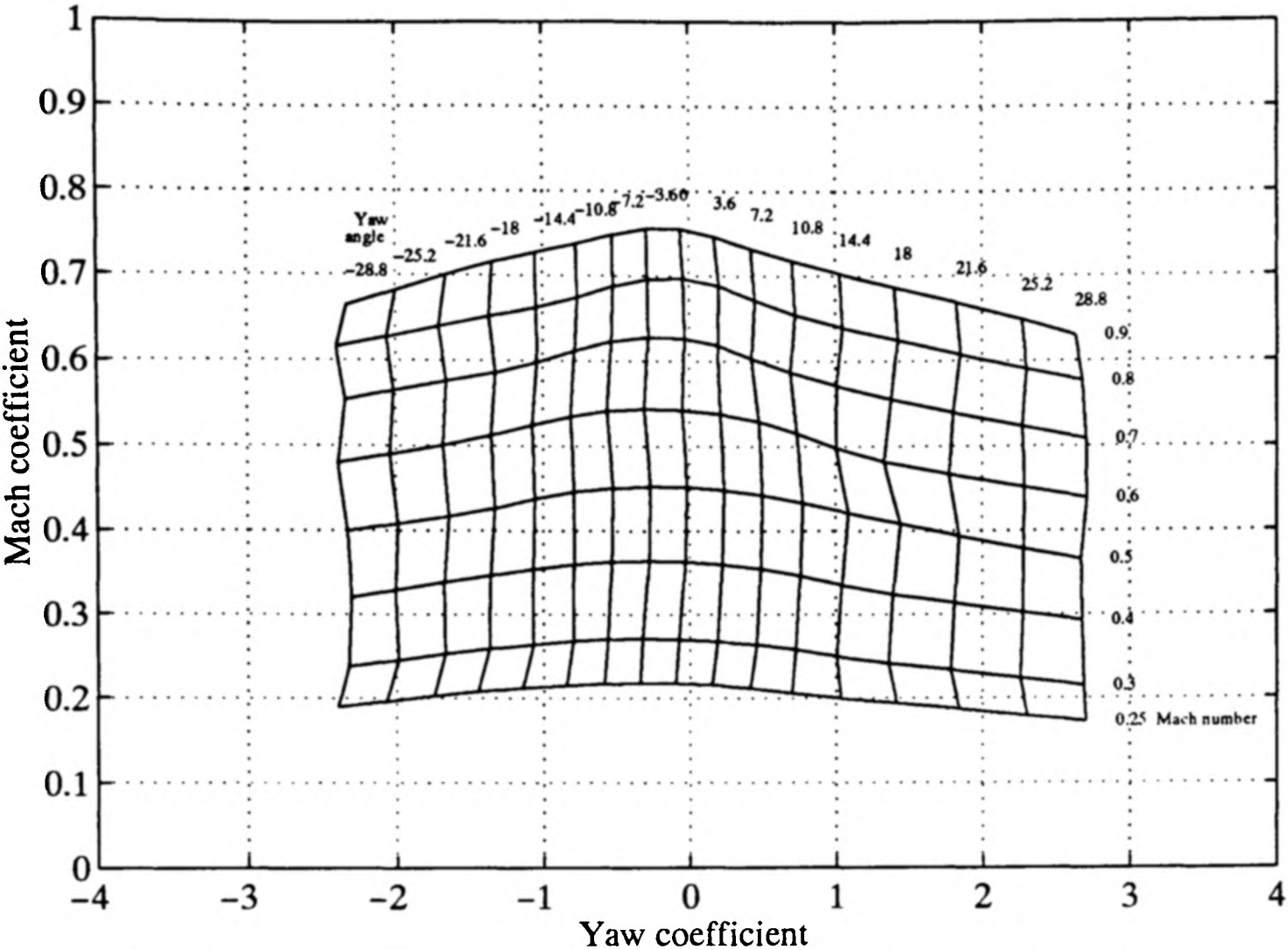


Figure 5.3 Probe Mach coefficient ratio plotted against yaw angle and Mach number

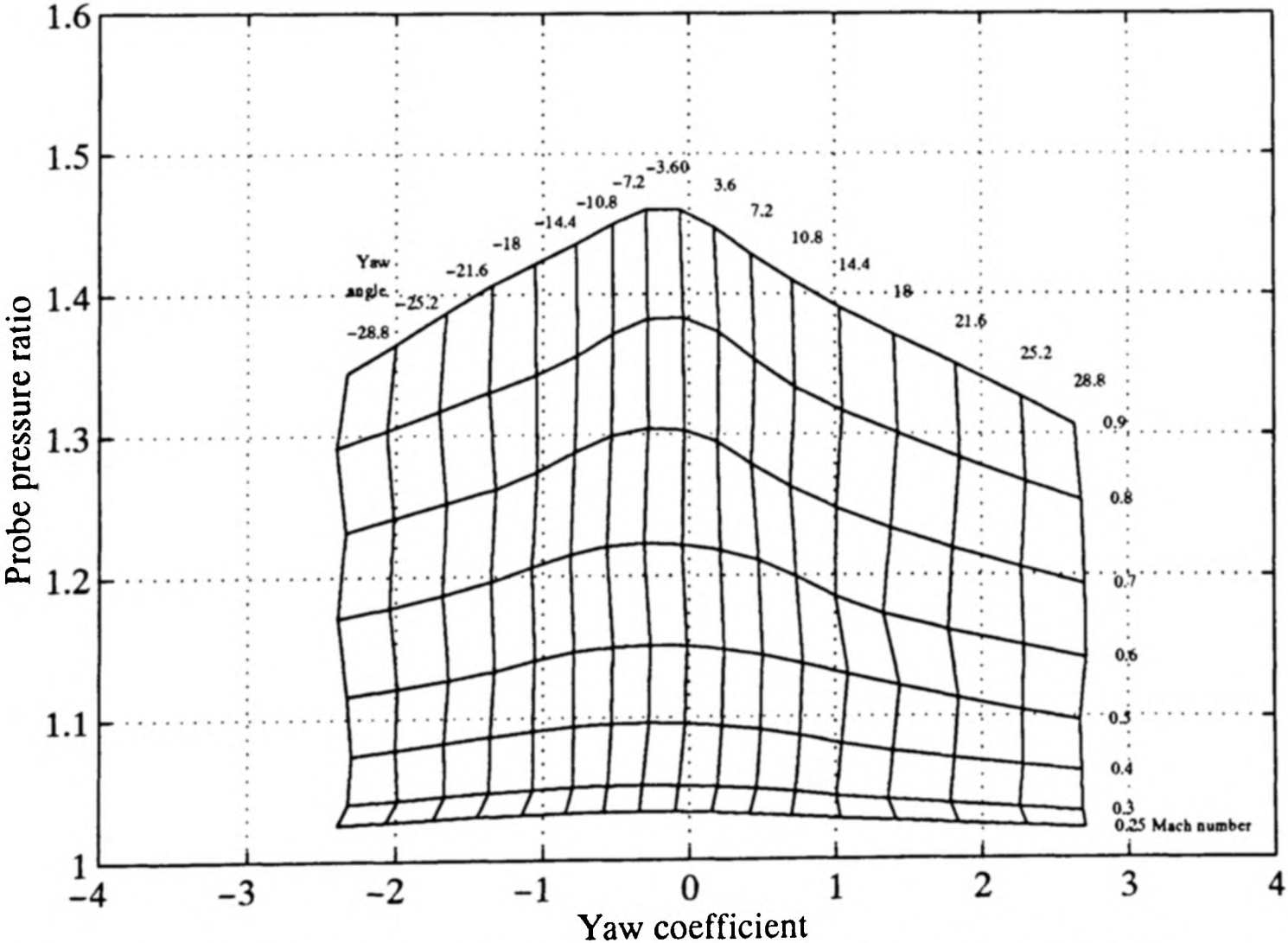


Figure 5.4 Probe pressure ratio coefficient plotted against yaw angle and Mach number

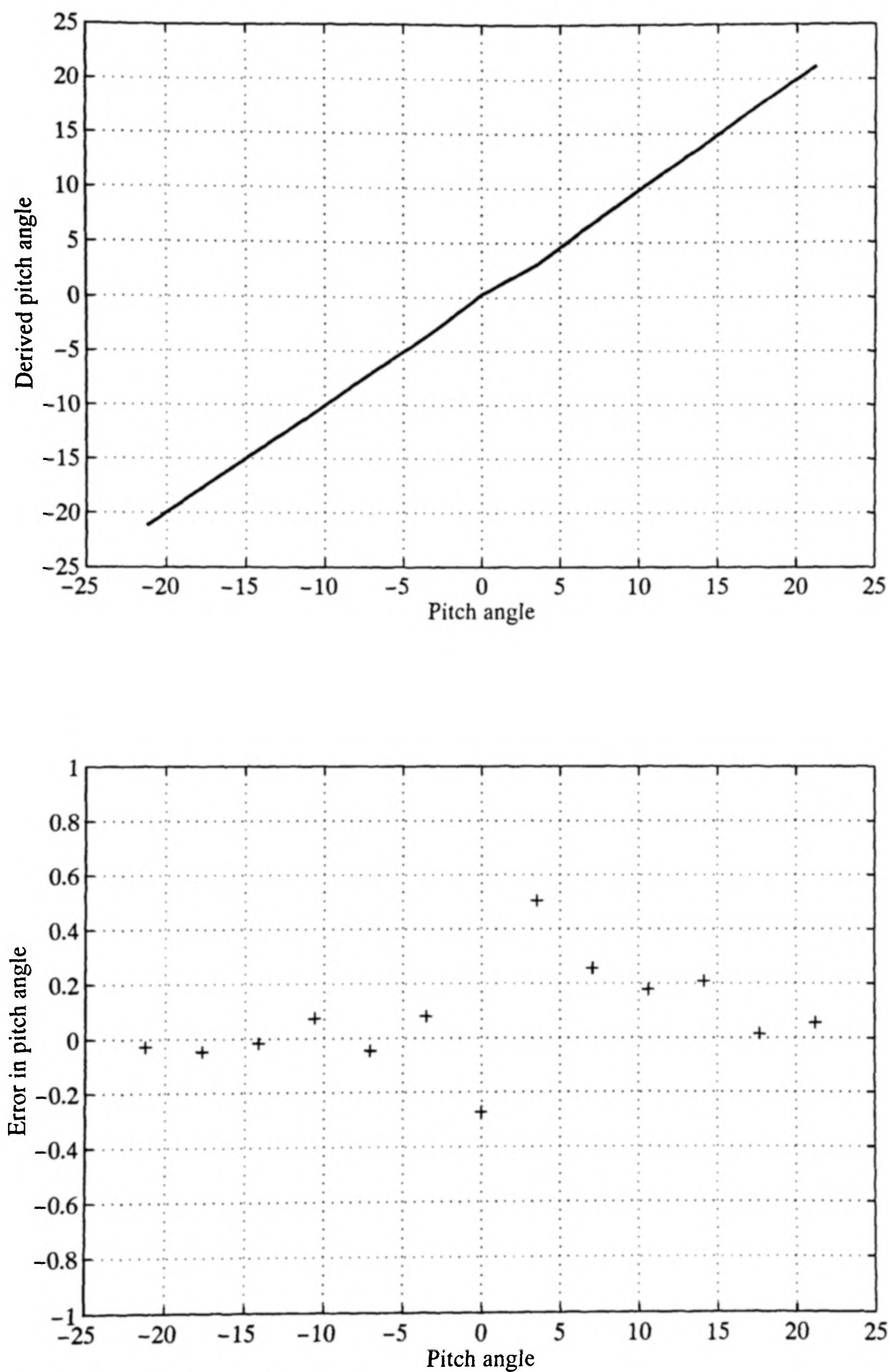


Figure 5.5 Error in inverting pitch angle

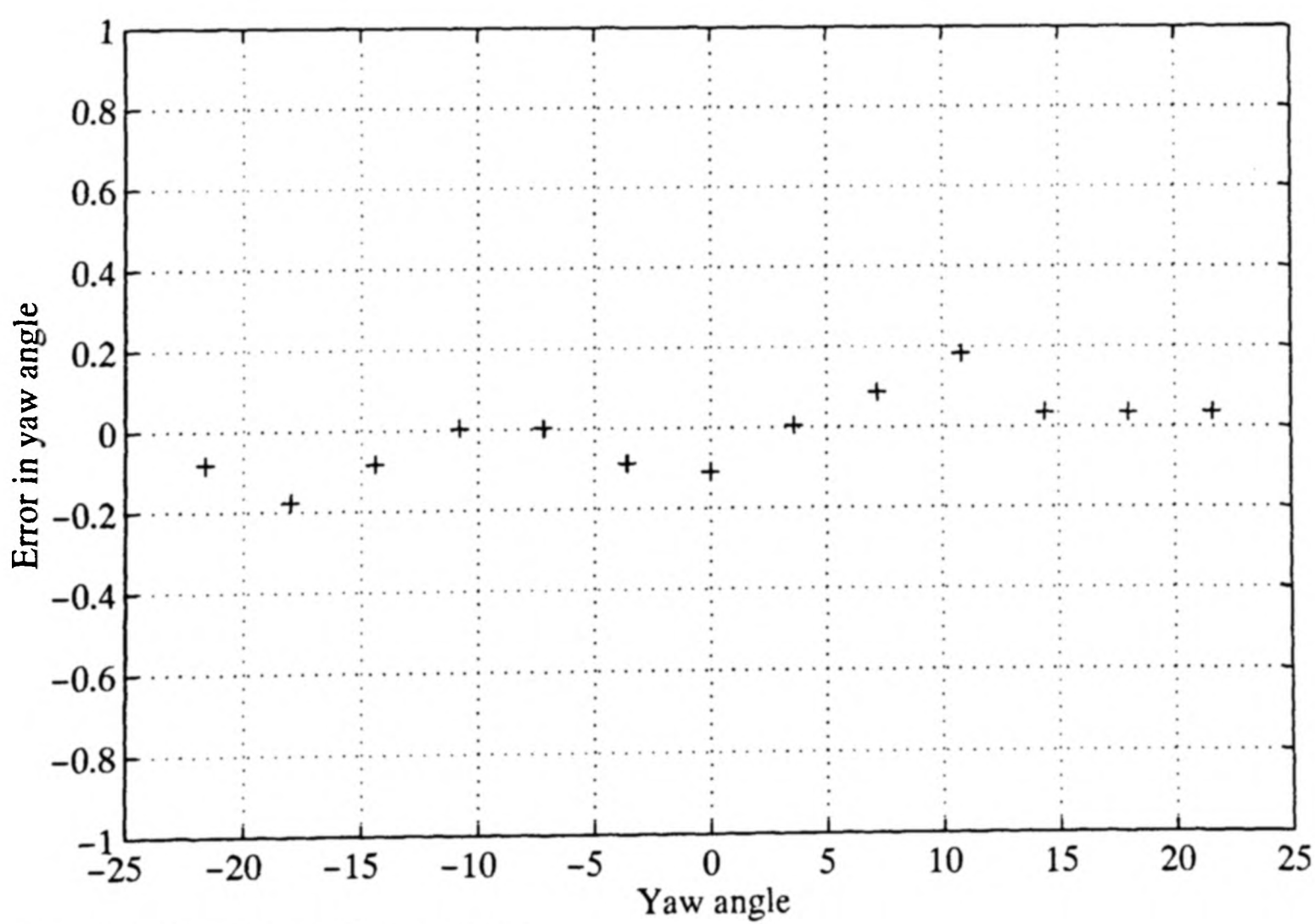
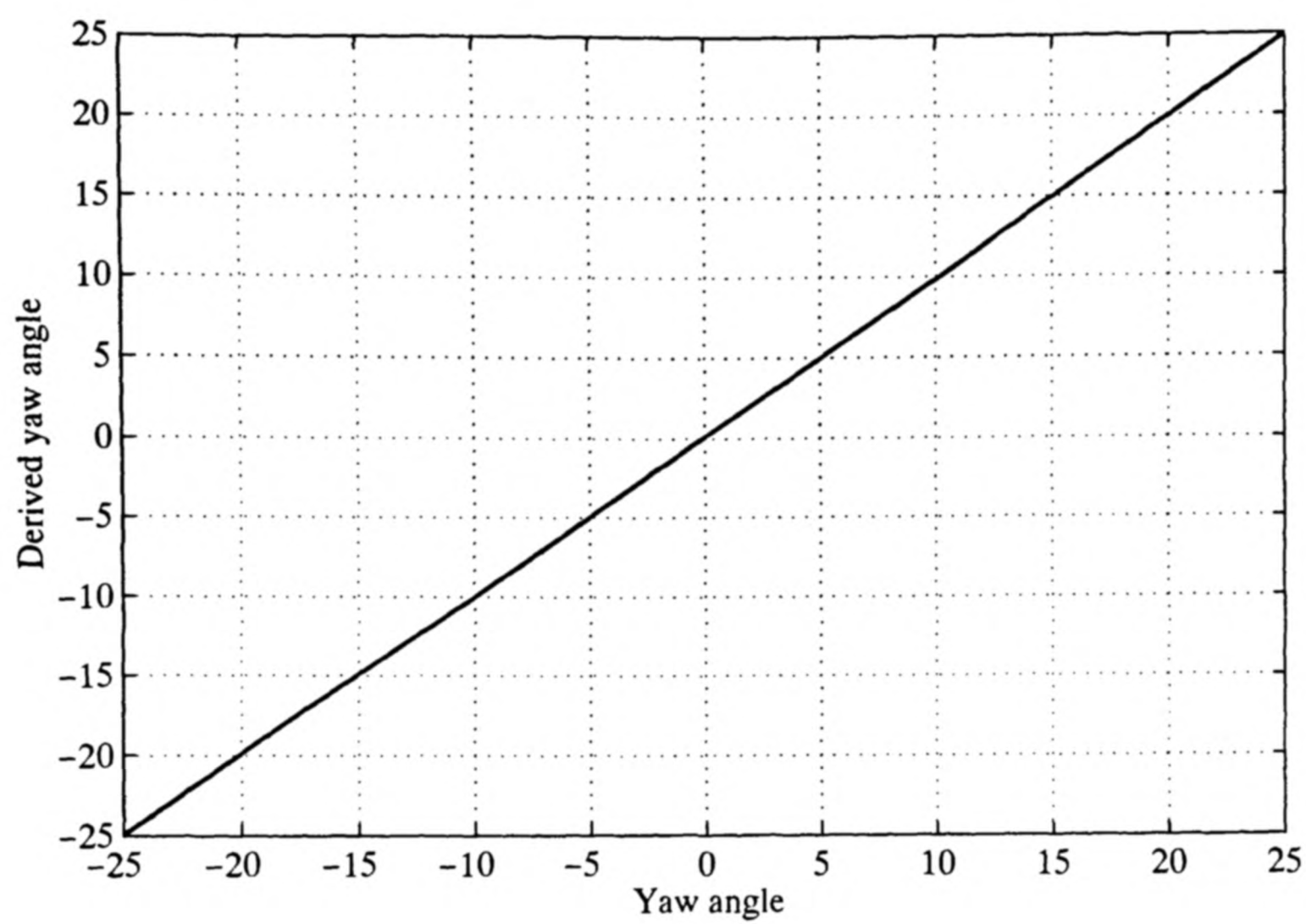


Figure 5.6 Error in inverting yaw angle

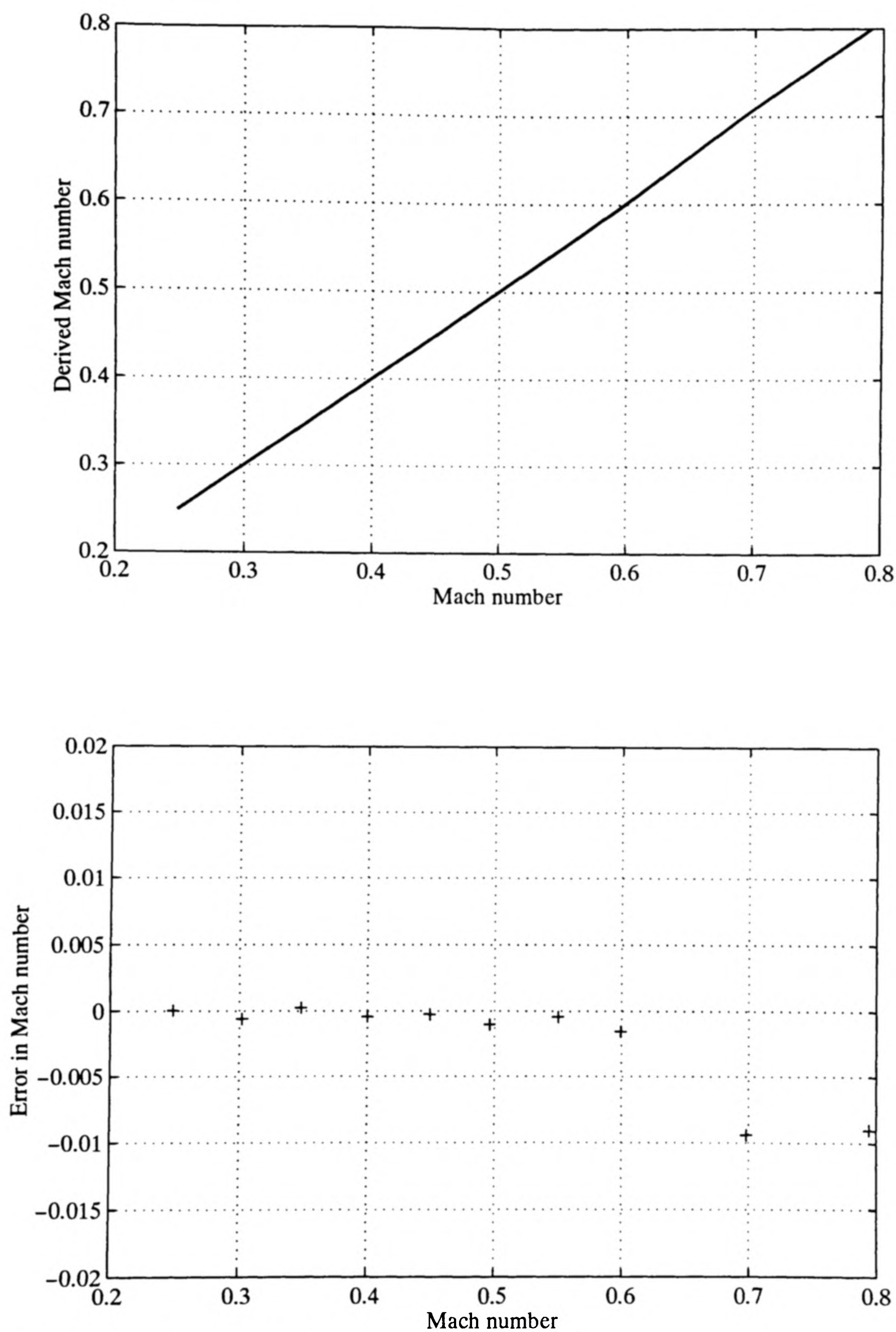


Figure 5.7 Error in inverting Mach number

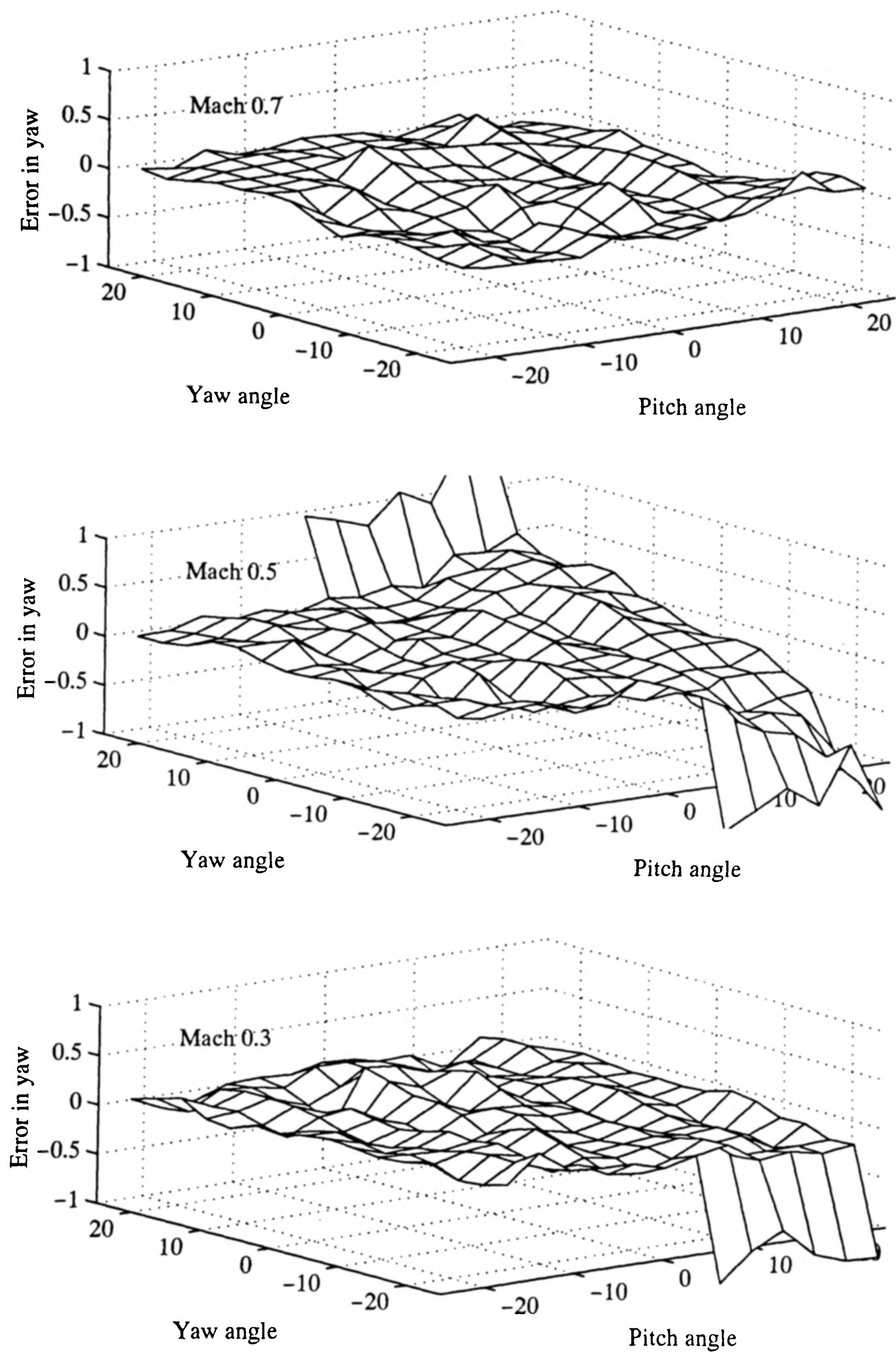


Figure 5.8 Error in inverting yaw angle over calibration space

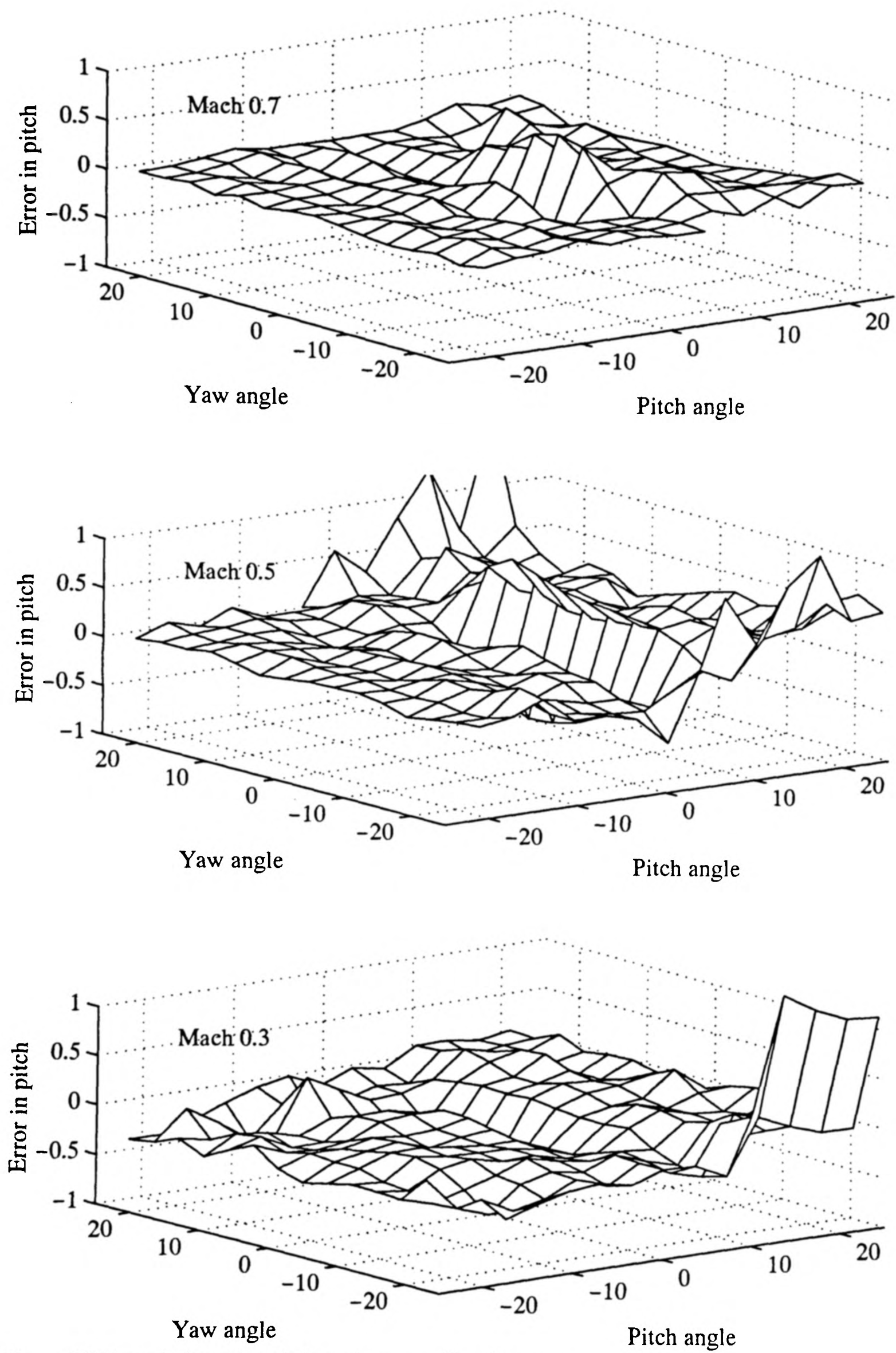


Figure 5.9 Error in inverting pitch angle over calibration space

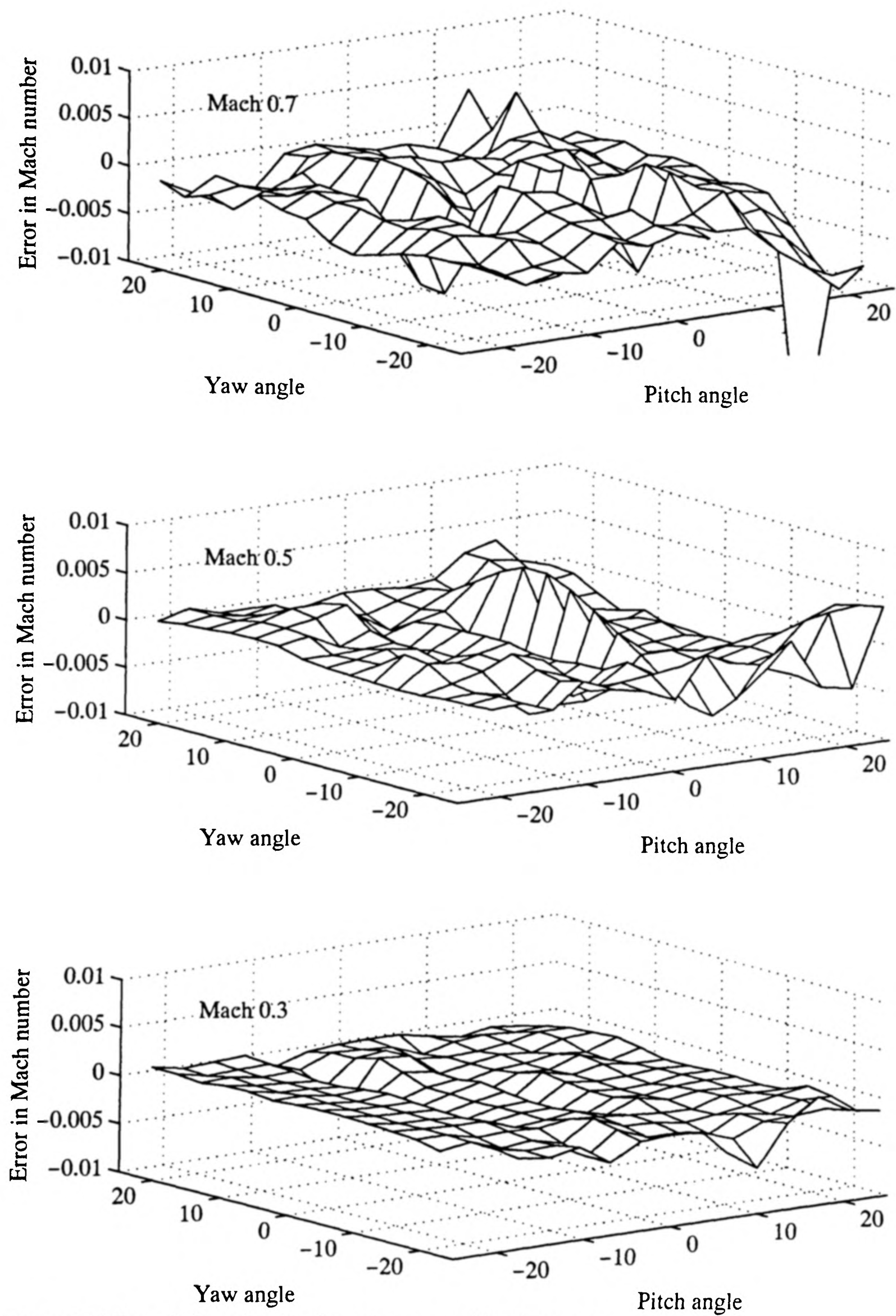


Figure 5.10 Error in inverting Mach number over calibration space

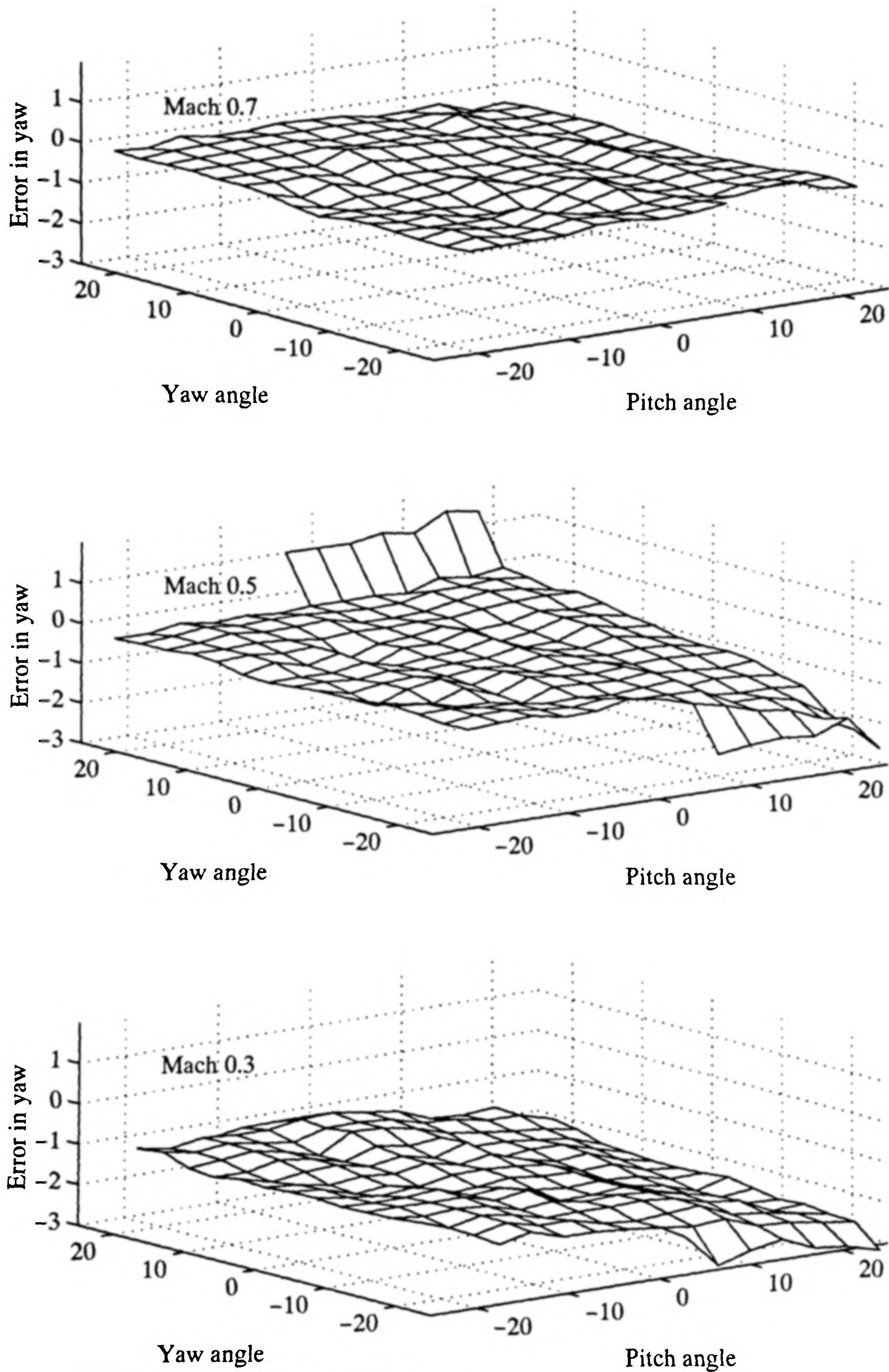


Figure 5.11 Error in yaw angle from 1575 Pa pressure error

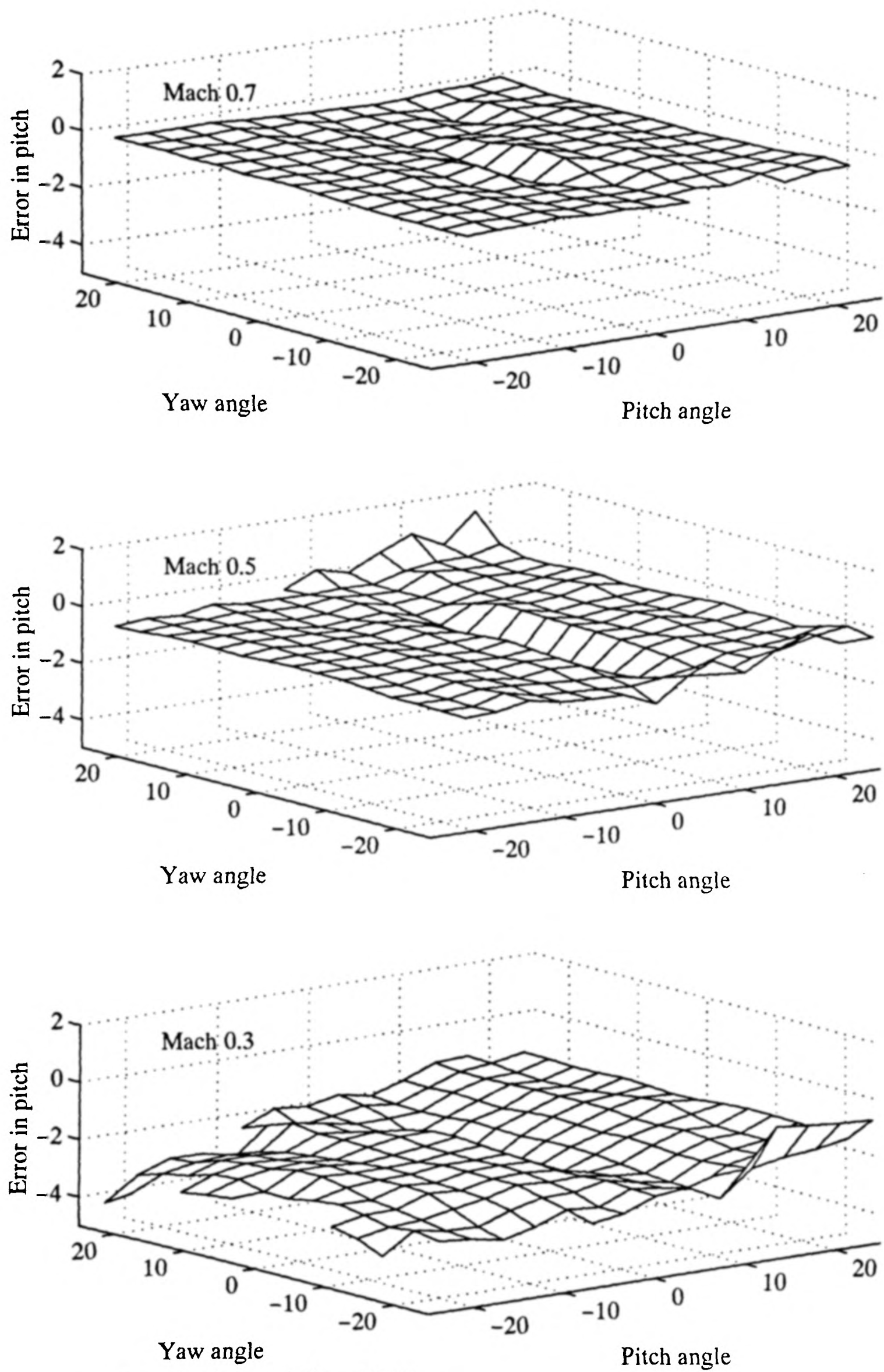


Figure 5.12 Error in pitch angle from 1575 Pa pressure error

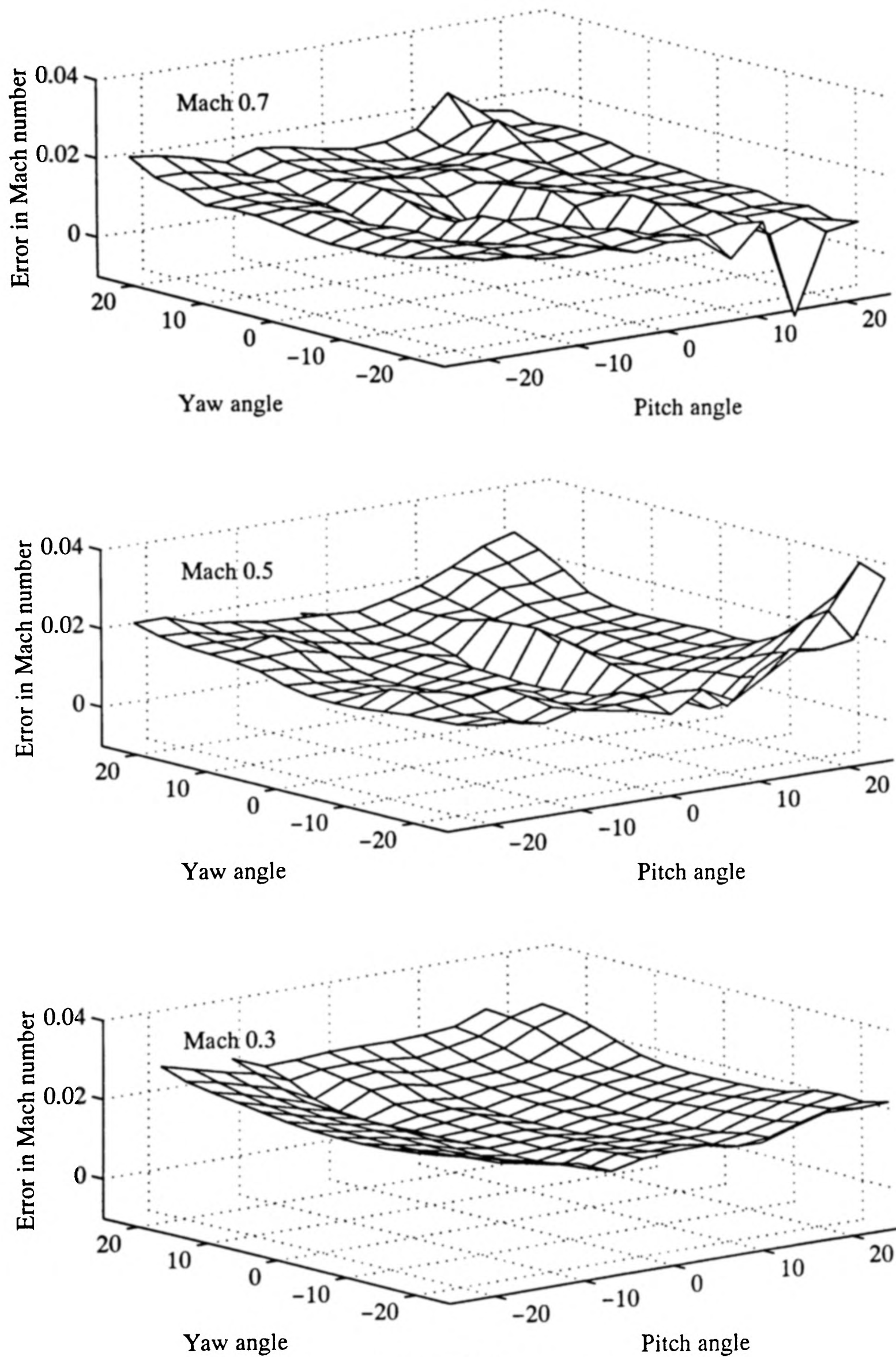


Figure 5.13 Error in Mach number angle from 1575 Pa pressure error

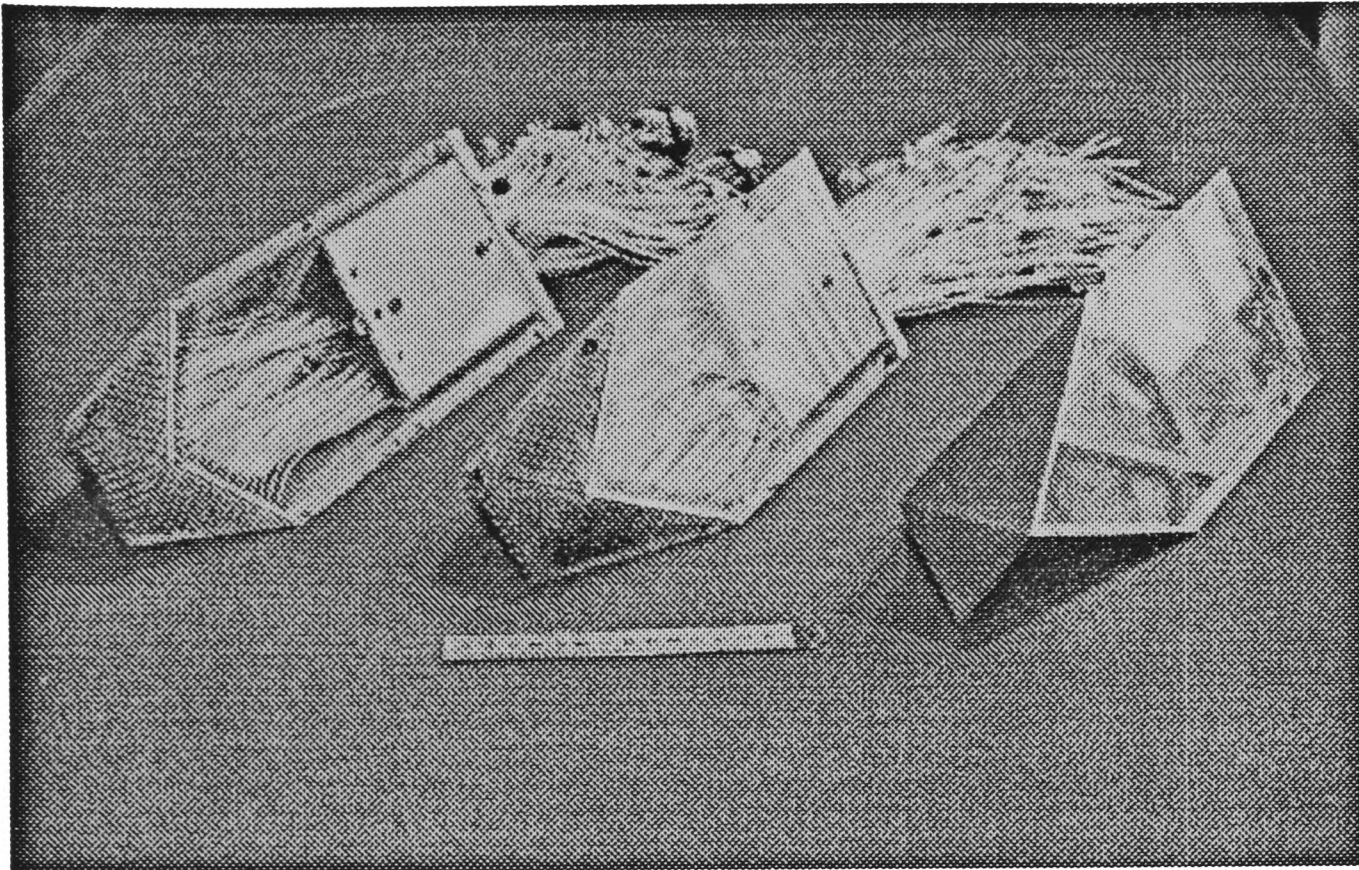


Figure 5.14 Large scale models of pyramid geometry probes

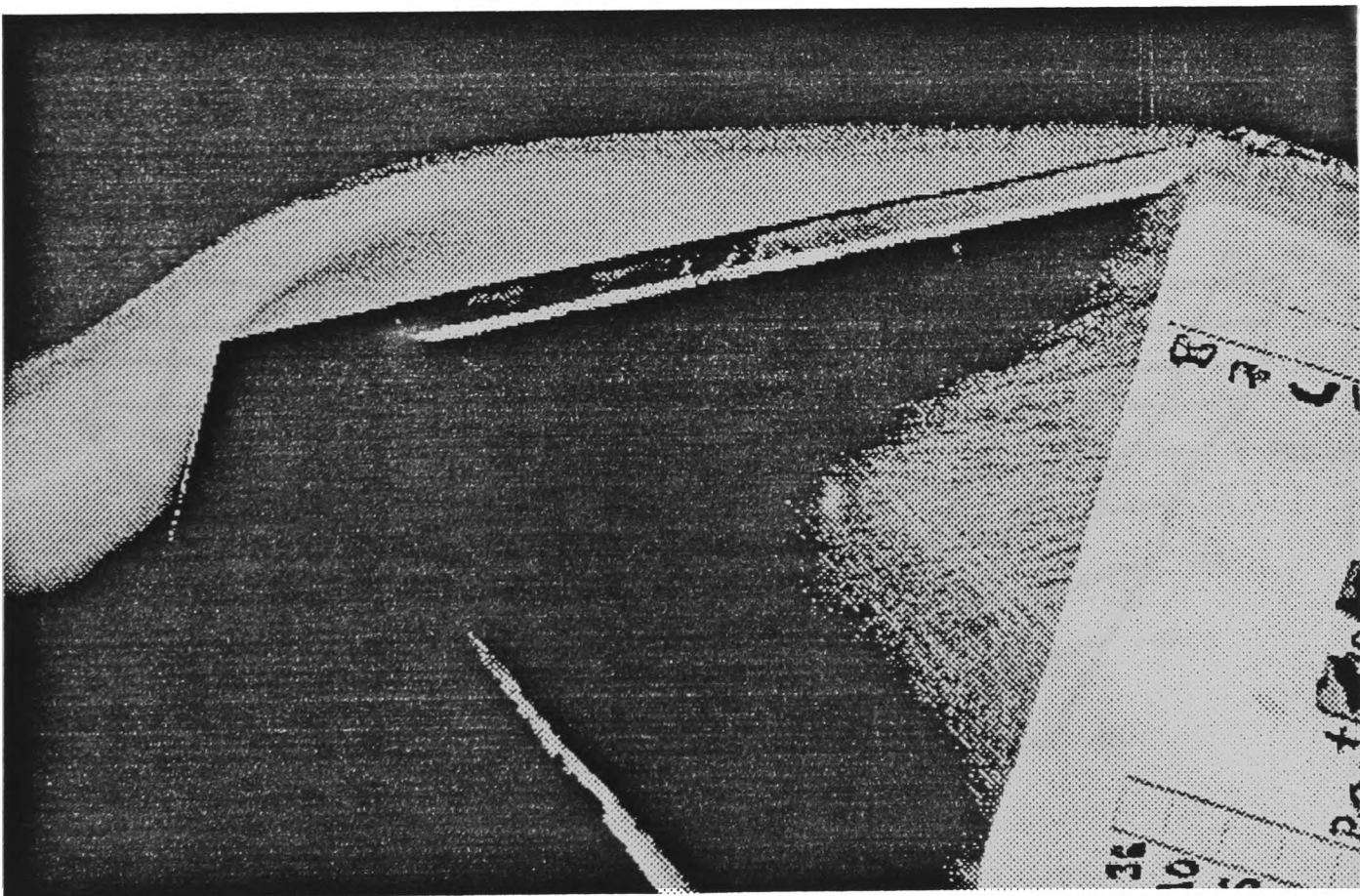


Figure 5.15 Flow separation over leading edge of 30° probe at -20° pitch

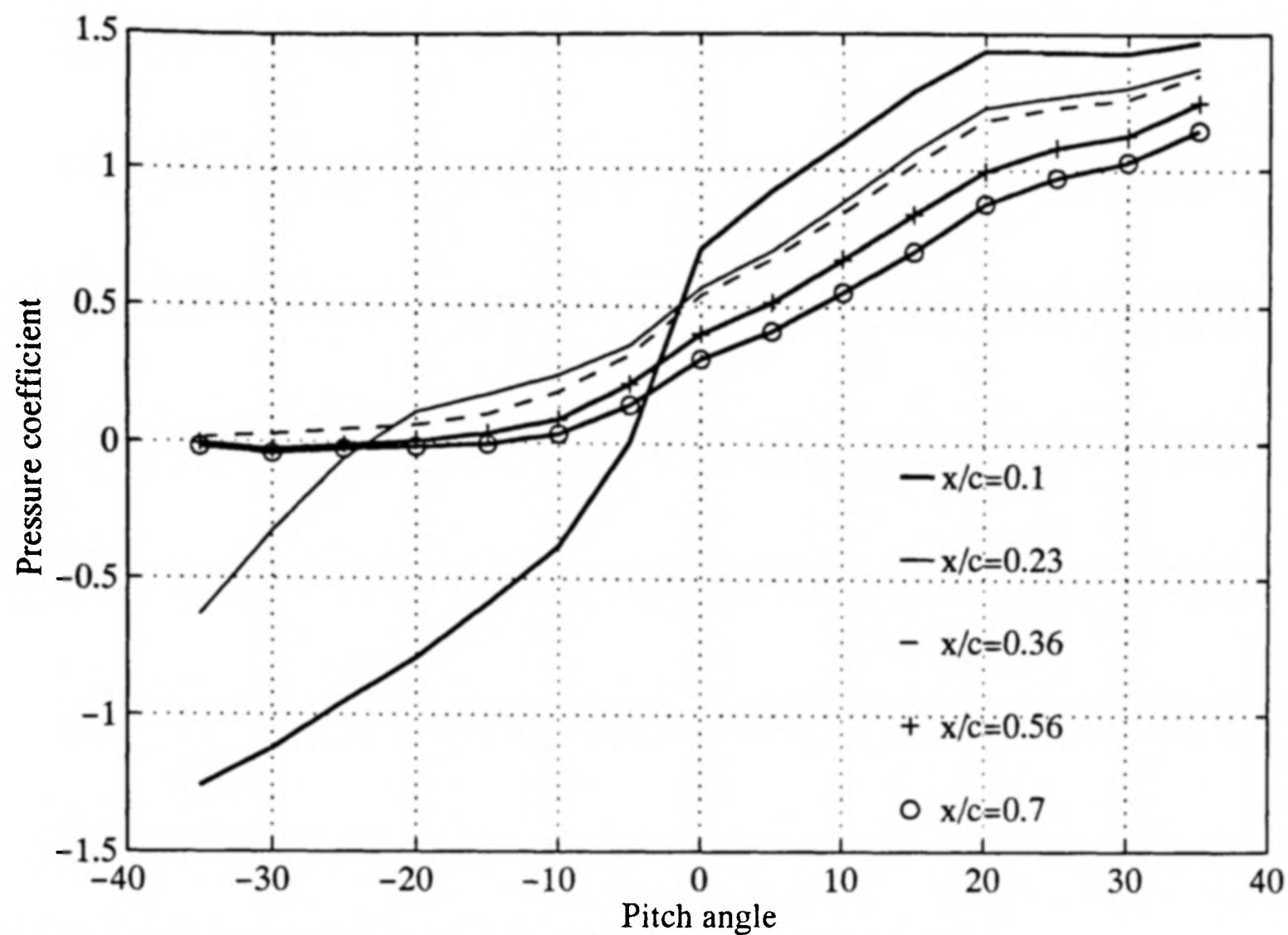


Figure 5.16 Pressure coefficient variation with incidence and axial position

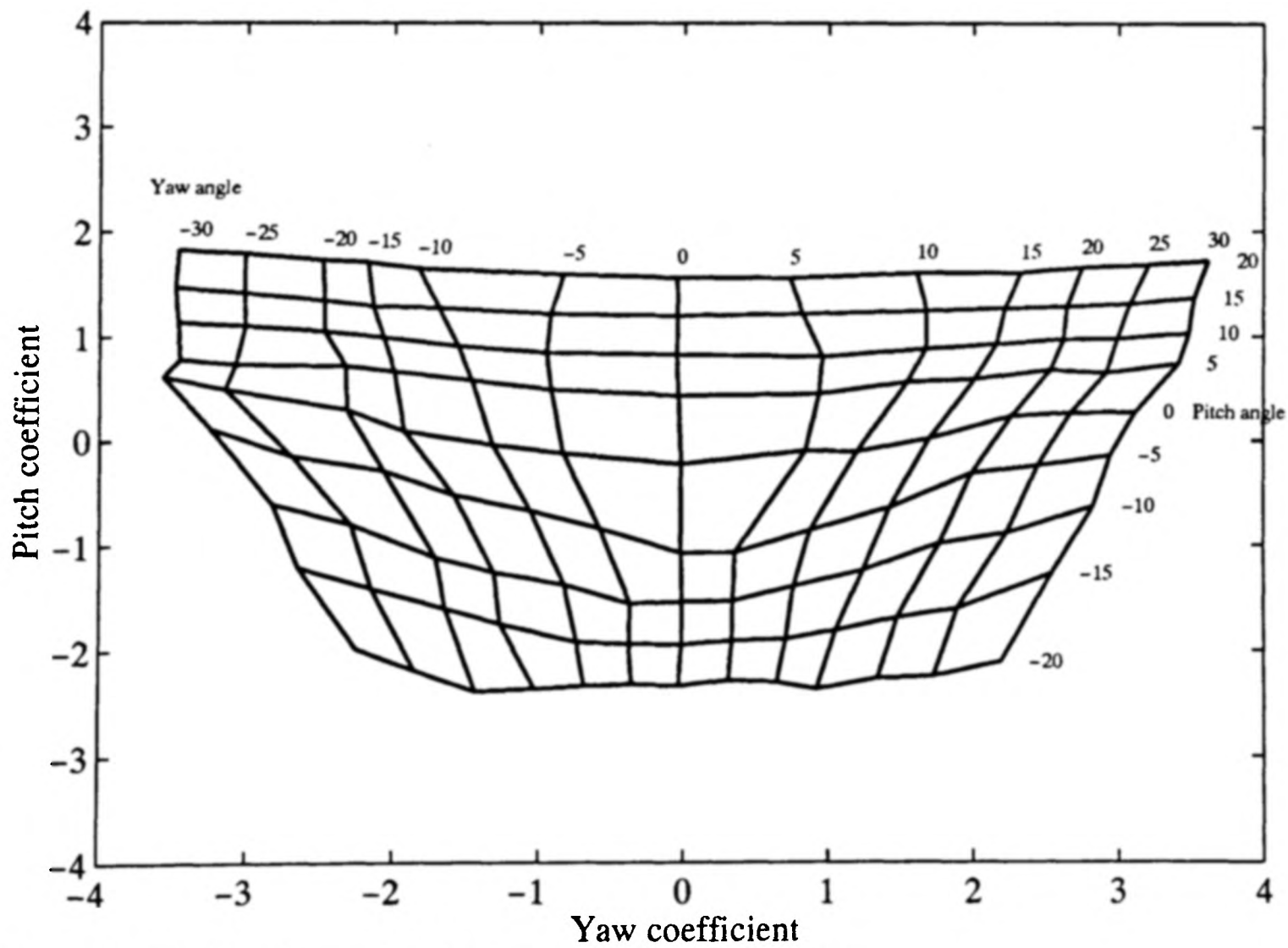


Figure 5.17 Calibration surface for forward mounted transducer

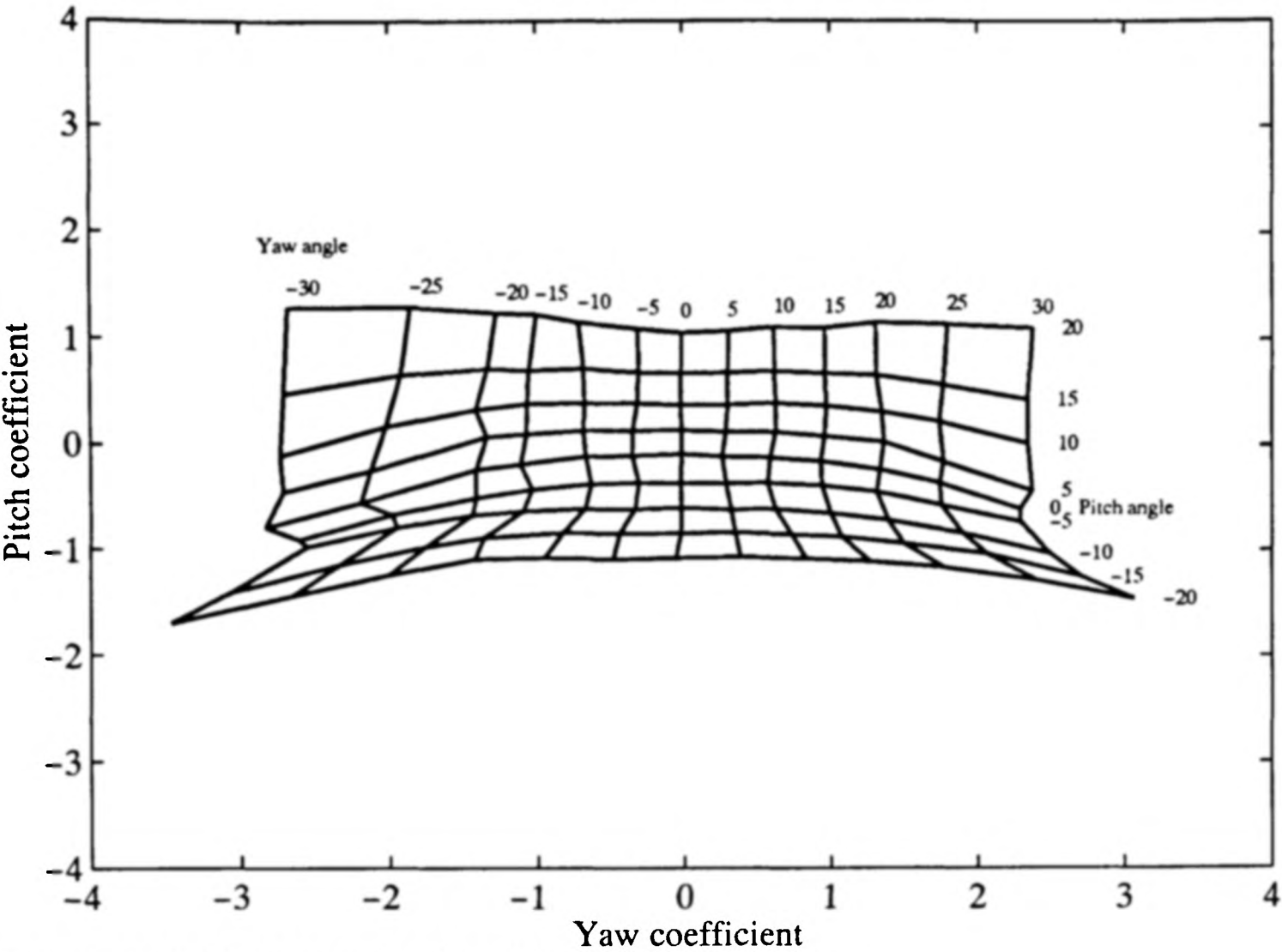


Figure 5.18 Calibration surface for mid mounted transducers

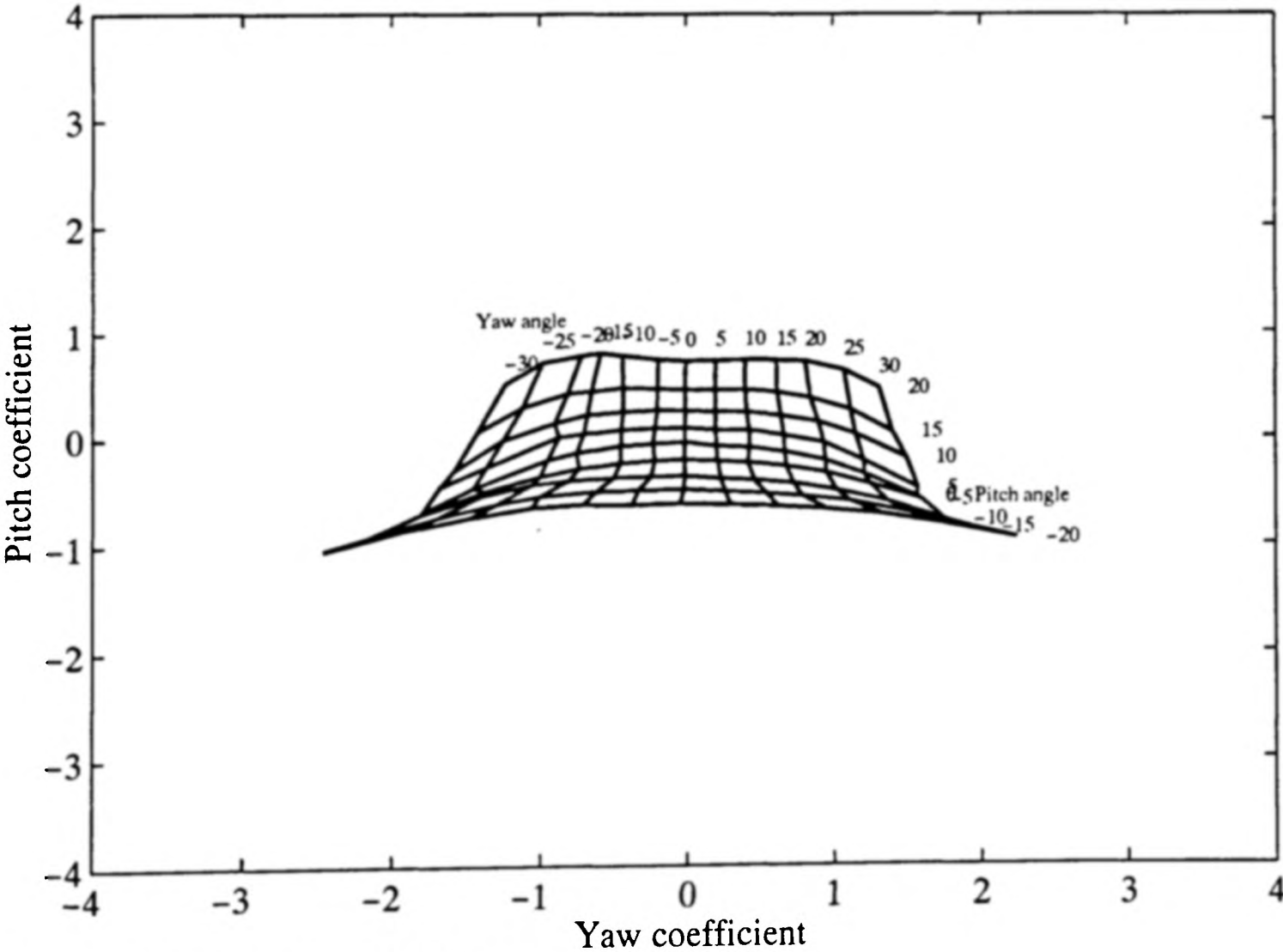


Figure 5.19 Calibration surface for rear mounted transducers

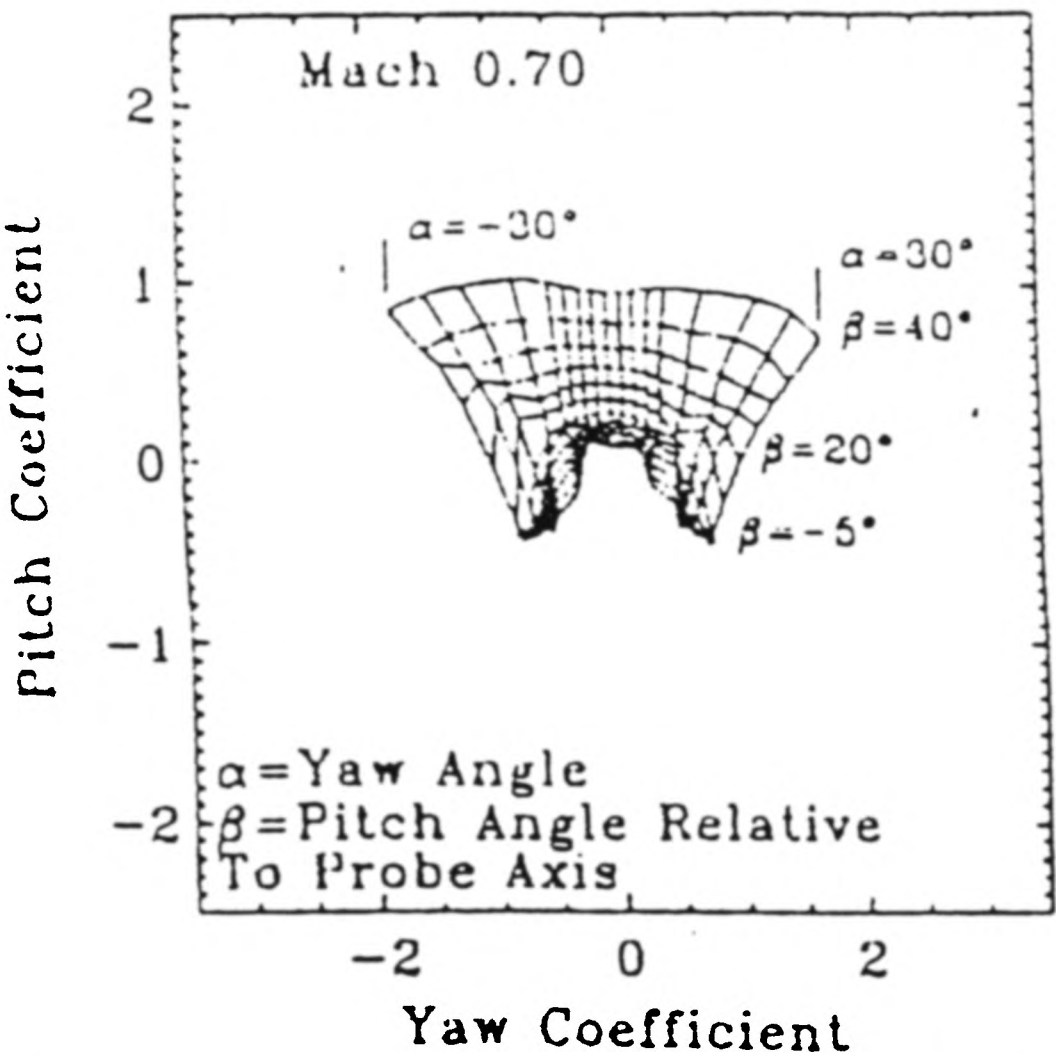


Figure 5.20 Three dimensional wedge probe calibration, Cherrett et al [1992]

6. The Oxford Rotor experiment

6.0. Overview

This chapter, along with chapter 7, details the flow measurements made with a three-dimensional semiconductor probe positioned at the exit of a model high pressure turbine. The Oxford Rotor is a fully rotating turbine which, for short periods of time, simulates the conditions in a real gas turbine engine by matching the relevant non-dimensional parameters. Transient facilities similar to the Oxford Rotor allow experimental research to be undertaken at a fraction of the cost and less complexity than running a continuous facility. A semiconductor probe mounted at the rotor exit served the dual function of measuring the run conditions over the whole rotor experiment whilst recording unsteady wake measurements as the turbine passes through design condition. The steady flow readings from the semiconductor probe are shown to be consistent with the measurements taken with other techniques. The unsteady data is explored in chapter 7.

6.1. The Oxford Rotor Facility

The Oxford Rotor is a fully rotating single stage model turbine simulating engine conditions. The facility was designed to meet the demand for measurements of turbomachinery flows in which the relevant non-dimensional parameters are matched. The Rotor is a 0.62 scale model high pressure turbine stage using Rolls-Royce's B22 rotor and B23 stator blade profiles. The B22 profile was designed by Rolls-Royce as an experimental profile representative of future trends in civil aircraft, high bypass, high pressure turbine stages.

The working fluid for the Oxford Rotor is test gas which is supplied to the working section from a pump tube containing a lightweight piston. The piston is driven forward by high pressure air from a separate high pressure reservoir and this compresses the air ahead of the piston head. At a pre-determined combination of temperature and pressure, a fast acting gate valve opens allowing flow through the working section. The flow generated is transient in nature, lasting approximately 200ms, but the flow-field inside the working section establishes rapidly, and measurements can to be taken in

the tunnel which are representative of those in the steady state environment. After the flow has passed through the working section it enters an evacuated dump tank. A schematic of the Oxford Rotor is shown in figure 6.1 which illustrates the major components.

A facility which takes its test gas from a pump tube in this way is referred to as an Isentropic Light Piston Tunnel (ILPT). This technique was developed for heat transfer studies where it was found the ILPT gave longer run times than a conventional shock tube, Jones et al [1973].

Transient tunnels have considerable advantages over continuous facilities because they have reduced energy requirements and since the stresses are also greatly decreased, materials may be chosen which are easier to work with. This allows the experimental research in a transient facility to be undertaken at a much lower cost.

6.2. *Previous work on the Oxford Rotor*

The Oxford Rotor facility has its origins in cascade research. An ILPT was commissioned for cascade testing where it was used to study the heat transfer on both the B22 and B23 blades, Nicholson [1981], Horton et al [1985]. After experiments to measure the steady flow heat transfer were completed, upstream rotating cylindrical bars were introduced to simulate upstream nozzle guide vane wakes. These experiments were to investigate the influence of the wakes on the downstream rotor. These wakes were found to have significant influence on the rotor blade boundary layer, Doorly and Oldfield [1985].

The rig was subsequently modified to incorporate a fully rotating turbine so that unsteady flow-fields at engine conditions could be measured. The ILPT and control system of the old rig was maintained but the working section required the conversion from a linear to an annular cascade, Ainsworth et al [1988b].

Since the commissioning of the Oxford Rotor, comprehensive heat transfer studies have been done on the turbine blades, Hilditch and Ainsworth[1990]. Measurements of the unsteady pressure on the rotor blades have also been measured, using the surface mounted transducer technique, Dietz and

Ainsworth [1992]. These unsteady measurements have subsequently shown to have good agreement with computational fluid dynamic predictions, Moss et al [1995].

Slater [1993] used two-dimensional hot wires for measurements of the rotor exit in one radial plane and compared the results with the Moore Elliptical Flow Program (MEFP). The rotor tip flow was shown to be dominated by tip leakage effects, whereas near the hub there were considerable secondary flows. The solution predicted considerable radial flow in the exit, but it was not possible to measure this using a two-dimensional technique. As the results were recorded at one radial plane the influence of the upstream nozzle guide vane were not measured.

The direction of rotation of the rotor was subsequently reversed and the nozzle guide vane row removed to allow measurements to be made of the blade heat transfer in the absence of nozzle guide vane wakes, Garside et al[1994]. These measurements were used to compare the computational predictions of the heat transfer which were found to overestimate the experimental values by approximately 30% on the pressure surface.

Following these experiments the rotor was then returned back to the forwards rotation configuration. Hot wires, with a novel in-shaft hot wire signal conditioning board, were used to take unsteady two-dimensional velocity measurements at mid-height of the leading edge of the rotating turbine blade, Sheldrake and Ainsworth [1995]. The signal conditioning board rotated with the rotor to avoid the difficulties of using hot wires through slip rings. These results were compared with computation fluid dynamics predictions based on the unsteady two-dimensional code Unsflo, Giles[1988]. The experiment and prediction were found to show good agreement on the structure of the flow, although Unsflo was found to under-predict the Mach number.

6.3. *Current Rotor programme*

It is known that the flow-fields between stages of an engine are coupled. The presence of downstream stator row will modify the aerodynamic behaviour of the upstream turbine but the extent of this coupling is not fully understood or determined. A programme of work is underway in which the

Oxford Rotor will be modified into a one and a half stage turbine with the installation of an Intermediate Pressure (IP) nozzle guide vane stage. This is to allow the interaction between the IP nozzle guide vane and the High Pressure (HP) rotor to be measured. These experiments require a detailed programme of time averaged and fast response aerodynamic measurements to be undertaken before, and after, the IP nozzle guide vane stage is installed.

The results in this and the following chapter were recorded on the Oxford Rotor to ensure that the fast response probe techniques were capable of taking the measurements required for this programme of work. Complete area traverses are required at the rotor exit and these have been subsequently taken using the probe techniques described herein, combined with a fast traverse mechanism, Miller and Ainsworth [1996]. These area traverses have shown the influence of the upstream NGV, the hub and tip losses. However the work presented in this thesis is concerned with the development of the technique which allowed these measurements to be made.

6.4. *Operating point*

To make detailed measurements at engine conditions would be impractical not least because the temperature of hot gas would destroy instrumentation. However if the relevant non-dimensional parameters are matched between experiment and engine operating conditions the aerodynamics of the turbomachine may be considered the same.

A standard derivation of the relevant non-dimensional parameters for a turbomachine is outlined in Horlock [1966] from which the following analysis is taken. The dependent variables for the performance of a turbomachine between the inlet and exit of a stage may be considered to be the mass flow rate, $\dot{m}$, the turbine efficiency, η , and the temperature drop across the turbine, ΔT . These three dependent variables can be defined as a function of the independent variables given in equation 6.1. The thermodynamic properties use the station definitions shown in figure 6.2.

$$\dot{m}, \eta, \Delta T, = f(P_{01}, P_{05}, T_{01}, N, l, R, \gamma, \mu)$$

Equation 6.1

There are four dimensions present in equation 6.1; Mass, Length, Time and Temperature. From the Buckingham-Pi theorem, eleven independent variables with four dimensions requires the definition of seven non-dimensional groups.

$$\frac{\dot{m}\sqrt{RT_{01}}}{P_{01}l^2}, \eta, \frac{\Delta T_0}{T_{01}} = f\left(\frac{Nl}{\sqrt{RT_{01}}}, \frac{P_{01}}{P_{05}}, \frac{\dot{m}}{\mu l}, \gamma\right)$$

Equation 6.2

The more practical form of equation 6.2 can be found with a number of simplifications and modifications. Firstly, the efficiency of the turbine may be eliminated from equation 6.2 using the isentropic relation between efficiency and the temperature change across the turbine.

$$\eta = \frac{\Delta T_0}{T_{01} \left(1 - \left(\frac{P_{05}}{P_{01}} \right)^{\frac{\gamma-1}{\gamma}} \right)}$$

Equation 6.3

When concerned with the performance of a machine of fixed size expanding a specific gas, the values of the gas constant, R , and characteristic dimension, l , may be considered constant. It is convenient to drop these from the non-dimensional groups to create parameters that are not truly non-dimensional but specific to an individual machine.

To express these parameters in values that may be measured easily in an experiment the pressure ratio is re-defined in terms of the exit static pressure, P_5 , rather than the exit total pressure, P_{05} . This is because it is easier to obtain reliable readings of the exit static pressure. Finally the Reynolds number, Re , as defined based on the nozzle guide vane absolute exit velocity and mid-height

chord, is used instead of the term $\frac{\dot{m}}{\mu l}$. The resultant set of parameters needed to be matched for dimensional similarity are given in equation 6.4.

$$\frac{\dot{m}\sqrt{T_{01}}}{P_{01}}, \quad \frac{C_p\Delta T_0}{T_{01}} = f_2\left(\frac{N}{\sqrt{T_{01}}}, \quad \frac{P_{01}}{P_5}, \quad Re, \quad \gamma\right)$$

Equation 6.4

Therefore we expect that the absolute values of the quantities used in equation 6.4 would not affect the experimental readings provided the pseudo non-dimensional parameters between experiment and engine were the same.

Specific speed	$\frac{N}{\sqrt{T_{01}}}$	460.5 rpmK ^{-0.5}
Stage Pressure ratio	$\frac{P_{01}}{P_5}$	3.08
Reynolds number	$\frac{\rho_2 u_2 c}{\mu_2}$	2.7×10 ⁶
Ratio of specific heats	γ	1.4
Mass flow number	$\frac{\dot{m}\sqrt{T_{01}}}{P_{01}}$	7.023×10 ⁻⁴ ms K ^{0.5}
Specific work	$\frac{c_p \Delta T}{T_{01}}$	226 J Kg ⁻¹ K ⁻¹

Table 6.1 Operating point

The Oxford Rotor is a 0.62 model high pressure turbine stage designed by Rolls-Royce. The operating point is defined by Rolls-Royce using bulk properties from an in-house design code, known as Q263. This code is an inviscid, axi-symmetric streamline curvature prediction code used for

through-flow analysis of turbomachinery. The values provided by Rolls-Royce for the operating point are given in table 6.1.¹

The ratio of the gas temperature to the wall temperature as can be seen from equation 6.4, does not affect the aerodynamic similarity. However the temperature gradient is expected to affect the development of the boundary layer and therefore the secondary flows. To allow heat transfer measurements to be made requires a temperature gradient between the gas and the engine wall. Earlier heat transfer studies on the rotor have used a temperature ratio of 1.3 and, although no heat transfer measurements were being recorded, this ratio was again used to be consistent with other experiments.

The value of the ratio of specific heats in a real engine will be lower than the value for air because of the products of combustion. However as air is used as the working fluid for the experiment a value of 1.4 is chosen rather than the true value in an engine, which might be in the range of 1.25-1.3. It is possible to simulate the engine values using a mixture of Noble gases, Epstein and Guenette [1984], but the advantages of using air in terms of cost and handling out weight the benefits of matching the value of γ exactly.

NGV axial chord	0.0312m
Mean radius	0.254 m
NGV exit metal angle	70.15°
Rotor inlet metal angle	42.56°
Rotor exit metal angle	-62.47°

Table 6.2 Tunnel dimensions

The most important tunnel dimensions for the Oxford Rotor facility are listed in table 6.2 and the tunnel parameters at various stages through the tunnel are given in table 6.3.

¹ Q263 version 155 16/11/94

<i>Piston tube</i>		
Temperature ratio		1.3
Temperature	T_t	288
Pressure	P_t	3.2×10^5
<i>Inlet conditions</i>		
Total temperature	T_{01}	374 K
Total pressure	P_{01}	8.04×10^5 Pa
<i>NGV exit conditions</i>		
NGV exit Gas angle	α_2	70.9°
NGV exit Mach number	M_2	0.910
NGV exit static temp	T_2	321.4
<i>Rotor relative exit conditions</i>		
Relative Mach number	M_4	0.977
Relative angle	β_4	-65.46°
Total temperature	T_{04}	329.9
Total pressure	P_{04}	4.77×10^5
<i>Absolute exit conditions</i>		
Mach number	M_5	0.4455
Angle	α_5	-23.7°
Total temperature	T_{05}	287
Static temperature	T_5	276
Total pressure	P_{05}	2.96×10^5
Static pressure	P_5	2.58×10^5

Table 6.3 Tunnel parameters

6.5. Operation of the tunnel

The tunnel is controlled from a Hewlett Packard HP7475 computer communicating with the tunnel via a CIL² interface. Plotted in figure 6.3 is the variation of the rotor speed during an experiment. At the beginning, indicated in the figure by the letter A, an air motor is used to spin the rotor. At this point there is a vacuum in the working section. The air supply to this motor is controlled from the computer and this allows the tunnel operator to set the rotor speed.

Before the experiment an annular gate valve is used to close off the working section from the air in the piston tube. The main working section is then evacuated to approximately 5000 Pa. This vacuum serves the purpose of ensuring the downstream throat is choked all the way through the experiment and therefore prevents propagation of any information from downstream of the second throat to the working section. The air motor is then used to spin the rotor up to a speed of approximately 8000rpm at which point the air motor is switched off, B. The turbine continues to rotate but decelerates until at a pre-determined speed the piston is released, C.

The piston is then driven by high pressure air to compress the air enclosed in the piston tube. The piston travels at approximately 30ms⁻¹ which is slow relative to the sonic speed and therefore the compression process is close to isentropic. The total pressure and temperature of the experiment may be varied by choice of the initial tube pressure and the compression ratio, using the standard isentropic relation, equation 6.5.

$$\frac{P_{01}}{P_t} = \left(\frac{T_{01}}{T_t} \right)^{\frac{\gamma}{\gamma-1}}$$

Equation 6.5

The exit to the piston tube is blocked by an annular gate valve which is held closed by a pneumatic actuator. A small perspex rod holds the actuator in place until the pressure in the tube

² Computer Instrumentation Limited

reaches the total pressure chosen for the experiment. At this point a small charge of explosive is detonated which destroys the perspex rod. With the perspex rod destroyed, the actuator is released and the shutter opens. The gate valve arrangement was designed in this way to achieve fast opening times, typically 30ms.

As the working fluid passes into the working section it expands through the turbine. The turbine has no brake to absorb the work and therefore the rotor accelerates through and beyond the design speed, D . A fast data acquisition system is triggered as the rotor passes design speed to allow unsteady measurements to be recorded, as indicated in the expanded view in figure 6.4. When all the gas from the piston tube has passed through the working section the rotor slows under the action of friction, E .

The speed of the rotor is measured from three optical encoders mounted on the end of the rotor shaft. A one line encoder is used by the HP7475 to monitor the shaft speed through the experiment. The one line encoder is further supplemented by a thirty-six and a sixty line encoder which allow either the position of the rotor blades relative to the nozzle guide vanes or the rotor position relative to a fixed point to be indexed.

The working section is fitted with an adjustable downstream throat which is used to set the exit Mach number independently of the inlet conditions. Once the annular gate valve has been opened the mass flow into the tunnel may be adjusted from the tube driver pressure. If the mass flow into and out of the tunnel is correctly matched the mean pressure level during the experiment will remain constant. If the pressure rises or falls during the experiment the inlet and exit mass flow rates have not been correctly matched.

6.6. Instrumentation

The tunnel data acquisition system consists on a sixty-four channel 12 bit 434Hz analogue to digital board mounted in a PDP 64 computer. The data recorded from this system may be processed using a program written in Fortran called TOPSY (Tunnel Operating System), Oldfield et al [1978].

This program is used to interface with the ILPT and stores information such as the sensor calibration constants, channel connections and comments on the details of the current experiment. TOPSY is used to record information during the experiment over approximately 0.5 seconds. This is sufficient to capture the period from when the piston is released to the end of the flow through the working section.

Data recorded on the PDP-11 is pre-amplified by a CAS 8A differential amplifier which inverts the signal and has variable gain. The bandwidth of these amplifiers is set to 90 Hz and thereby acts as the anti aliasing filter for the PDP-11.

This data acquisition system is supplemented by higher specification Datalab DL1200 and DL2400 transient recorders. These are capable of recording data up to 500KHz, but because the memory on the recorders is limited to only 32,786 data points, they capture data only as the rotor passes through design speed. This is achieved by careful triggering of the fast data recorders so that the rotor reaches design speed half way through the data capture envelope. The data from the DL1200 is saved directly onto an IBM compatible PC using the data acquisition software DUCKS, Moss [1991]. Once downloaded, the data can be processed using the commercial data analysis package MATLAB, MathWorks [1991].

The high frequency data is amplified using the AMP 05 amplifier manufactured by Precision Monolithic Inc. which has variable gain and a nominal bandwidth of 100 kHz. They may be either AC or DC coupled. This amplifier has the dual role of amplifying the signal and acting as an anti-aliasing filter for the 500 kHz sampled data on the Datalab DL1200 recorders.

Side wall static pressure tapings are located at the nozzle guide vane exit (referred to as the C-plane) and at rotor exit (F-plane). These pressure tapings are located around both the inner and outer annulus and positioned so that the influence of the nozzle guide vanes is averaged out. The side wall static tapings are connected to National Semiconductor differential transducers which require a reference pressure. All the pressure transducers in the facility are regularly calibrated by pressurisation of the whole facility and comparison of on-line values with a precision Bourdon pressure gauge.

The downstream conditions are measured from a total pressure probe, which uses a Kistler transducer. A second Kistler probe is used to measure the total pressure upstream of the rotor. These probes are absolute transducers and therefore do not require a reference pressure.

6.7. *Three-dimensional steady flow measurement*

6.7.1. Experimental set up

The flow-fields at rotor exit are three-dimensional and unsteady, therefore the three-dimensional pyramid probe was expected to be ideal for flow measurement in these conditions. The pyramid probe was mounted on a removable cassette which contained part of the nozzle guide vane blade row. This cassette was designed to allow the probe to move in both the radial and circumferential directions, Miller and Ainsworth [1995]. During an experiment the probe can be traversed rapidly across the rotor exit in the circumferential direction or used to hold the probe in a fixed position, as in these experiments.

In this configuration the front face on the pyramid was located 3.5 mm, or at 14 % of axial chord, downstream from the rotor trailing edge. A view of the working section is shown in figure 6.4. Connection to the probe was facilitated by a small loom which gives the wiring flexibility to move with the probe. The output voltage from each of the transducers was recorded on the PDP-11 together with the temperature sense voltage. The output signal was also recorded on the higher specification Datalab DL1200 recorders to measure the unsteady signals as individual rotor wakes pass the probe. It was not necessary to record the temperature sense voltage on the Datalab recorders because the thermal inertia of the probe prevents high frequency fluctuations in the probe temperature.

The velocity triangle at rotor exit is given in figure 6.5 based on the values given by the flow prediction program Q263, table 6.2. The mean flow angle for the mid-height is predicted to be -23.7° , and therefore to operate over the full range without exceeding the limits of its calibration, the probe needs to be inserted with an offset angle. The probe mounting mechanism was designed to align the probe using a slot machined in the rear of the probe stem. The slot gave an offset angle of -19° , an

angle which corresponded with the mid-height exit flow angle predicted for the operating point of the rotor during earlier work on the rotor, Slater [1993], but perfectly satisfactory for the current work.

6.7.2. Measured operating point

Before results from the three-dimensional probe experiments can be considered, it is important to ensure that the facility is recording measurements at the operating point. The upstream total pressure for an experiment is plotted in figure 6.6. The pressure is subject to a fluctuation because the finite mass of the piston causes it to oscillate in the piston tube after it has compressed the working fluid from the point in time where the gate valve opens. These oscillations decay with time but affect the experiment by causing variations in the flow. The oscillations are overcome by timing the data acquisition so that it occurs as close as possible to mid-oscillation. In figures 6.7-6.9 the stage pressure ratio, specific speed and Reynolds number variation during the experiment are plotted. Again it can be seen that the piston oscillation has an influence on the readings, but at the time of the data acquisition, indicated by the trigger, the turbine is close to the design values.

The nozzle guide vane and rotor exit Mach number are used to ensure that the rotor is operating with the correct mass flow and specific work number. The values predicted by Q263 for the operating point are compared with the measured values in figure 6.10.

6.7.3. Flow measurements by the three-dimensional pyramid probe

In figure 6.11 the pressure traces from the four pressure transducers mounted on the probe are plotted for the period while there is flow through the working section. As the turbine approaches the design speed, a trigger, indicated on the plot by the vertical line, starts data acquisition on the Datalab recorders for the unsteady measurements. The results from the unsteady measurements will be returned to in the following chapter.

The pressure rises rapidly as the upstream shutters open and flow enters the working section. The temperature compensation circuit allows measurement of the changes in the probes temperature over the course of the experiment. In contrast to the temperature changes experienced during the

calibration of the probe, the temperature change is small, figure 6.12, and has little influence on the measured pressures. The length of the experiment is considerably shorter than during an aerodynamic calibration and by the time the gases have been expanded, the temperature difference between the probe and expanded gases is small.

As the rotor speed accelerates the flow angle at exit varies. This is reflected in the way that the pressures on faces B and C in figure 6.11 differ early in the experiment and approach each other close to when the trigger is fired. This indicates the probe has been set to approximately the correct angle of incidence.

This can be seen clearer in the first subplot of figure 6.13 in which the yaw and pitch pressure during the experiment have been plotted. The first 0.02 seconds of this plot represent when the flow first starts through the working section and the total pressure rises. The yaw pressure then starts to fall as the rotor speed increases and the flow incidence to the probe changes. The pitch pressure is also plotted in the same subplot and appears to be approximately constant during the experiment.

In the second subplot the non-dimensional form of this data is plotted and in the third subplot the Mach coefficient is included. Early in the experiment the low total pressure means that the coefficients are subject to large experimental error and are therefore meaningless. From 0.02 seconds onwards the yaw and pitch coefficient reflect the measured flow angle. Finally after 0.15 seconds the flow through the turbine reduces and the total pressure falls. Once again the experimental error associated with low pressures cause the coefficients to be misleading but since the experiment is over this is of no concern.

By using the inversion techniques detailed in chapter 5 these coefficients may be converted into the measured primary flow parameters yaw, pitch and Mach number. Once these values have been found the static and total pressure may also be derived using the static and total pressure coefficients for each given combination of yaw, pitch and Mach number. Figure 6.14 shows these variations over the duration of the experiment.

The measured yaw angle and Mach number change as the rotor accelerates through the design speed. As the rotor accelerates the relative rotor exit Mach number will remain constant because the mass flow through the turbine is approximately constant. With reference to the rotor exit velocity triangle, figure 6.5, the absolute exit Mach number and magnitude of the yaw angle will therefore fall as the rotor accelerates. This can be seen in the experimental results. The flow in the turbine at mid-height is predominately axial and therefore the flow pitch angle, which corresponds to flow in the radial direction, is expected to be close to zero as found by the probe.

6.7.4. Comparisons between probe and rig instrumentation

The total pressure measured by the probe may be compared with the total pressure measured by the pneumatic Kistler total probe. Figure 6.15 shows the comparison between the two probes. This plot shows the total pressure from inverting the three-dimensional probe gives a marginally higher value than from the Kistler probe measurement. This reflects the fact that the Kistler probe has not been calibrated for incidence or compressible effects, whereas the loss coefficient measured during calibration is applied to the three-dimensional pyramid probe. It is interesting to note that the Kistler transducer output lags that of the faster response semiconductor probe, as one might expect.

In a similar fashion to the comparison of the measured total pressure, the measured static pressure from the three-dimensional probe may be compared with rig instrumentation. In this case static pressure tapings located in the inner and outer annulus at the rotor exit may be compared with the probe measurement. In figure 6.16 the comparison between the static pressures is shown to be very favourable.

To compare with the exit Mach number measured by the probe, the rotor exit Mach number is calculated using the average value of the static pressures from the inner and outer annulus pressure tapings combined with the total pressure by the Kistler probe. The comparison plotted in figure 6.17 shows that the three-dimensional probe measures a marginally higher Mach number than the Kistler probe. The Kistler probe, like the three-dimensional probe, is mounted at -19° to the flow but earlier in the experiment the magnitude of flow incidence is much greater, as shown in figure 6.14.

When the flow is at incidence to the Kistler probe, the face does not recover the true total pressure and corresponding Mach number is measured too low. In contrast, the three-dimensional probe has been calibrated for these incidence effects and the measured values are corrected. As the incidence reduces, the error in the measurement made by the Kistler diminishes, and the agreement between the Mach number derived in these two ways improves.

6.8. Conclusions for steady state measurements

The steady state measurements demonstrate that the semiconductor probe can record reliable steady state measurements. The three-dimensional pyramid probe has advantages over methods in that one probe can be used to measure Mach number, yaw angle, pitch angle, total and static pressure. Unlike simple total or static pressure probes the measurements with the three-dimensional probe are corrected for both Mach number and incidence. The results from the probe have been shown to be in good agreement with those found by independent measurements. It is worth highlighting that these measurements have been recorded with a semiconductor probe which have hitherto been considered unsuitable for steady state measurements.

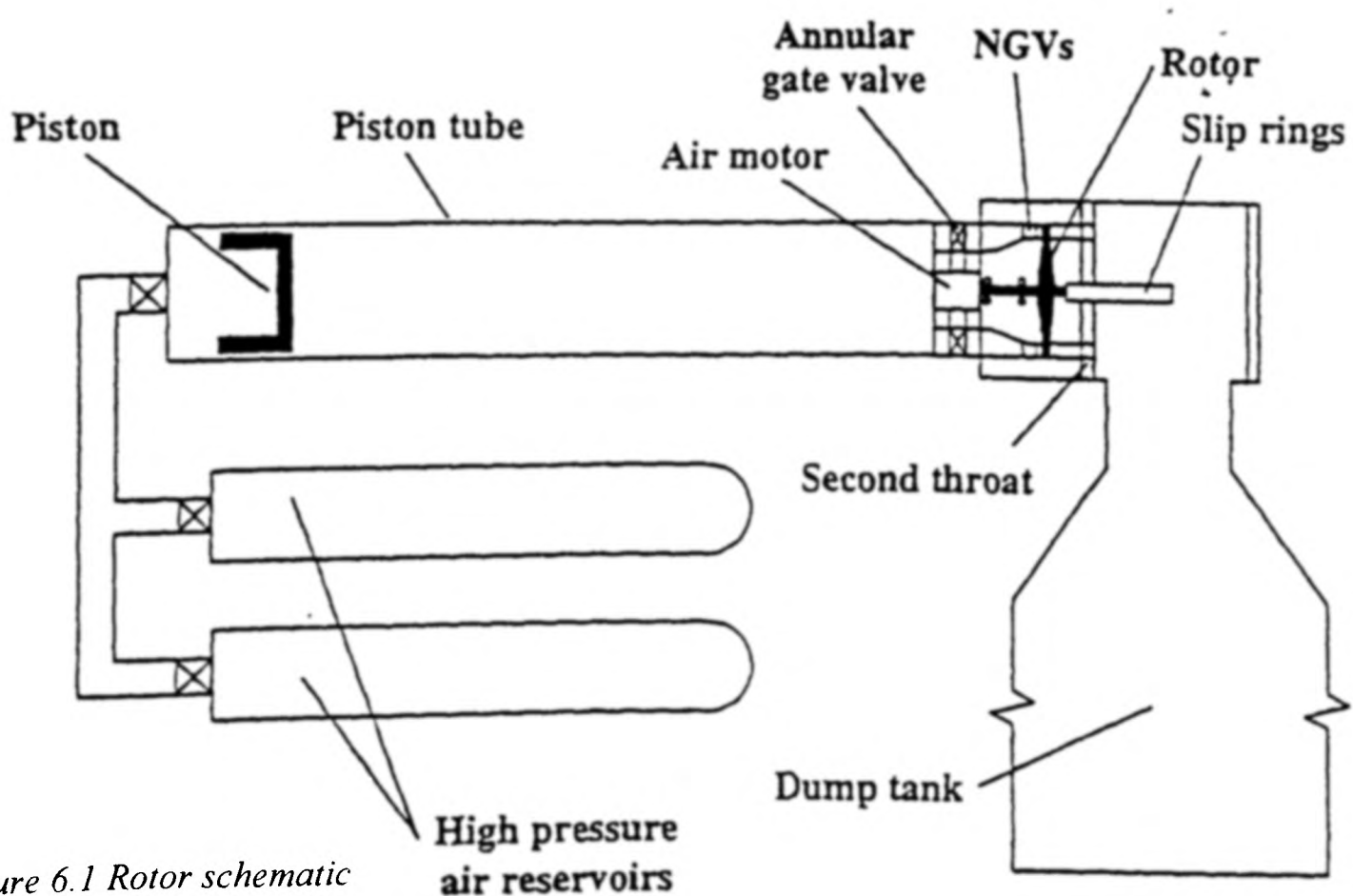


Figure 6.1 Rotor schematic

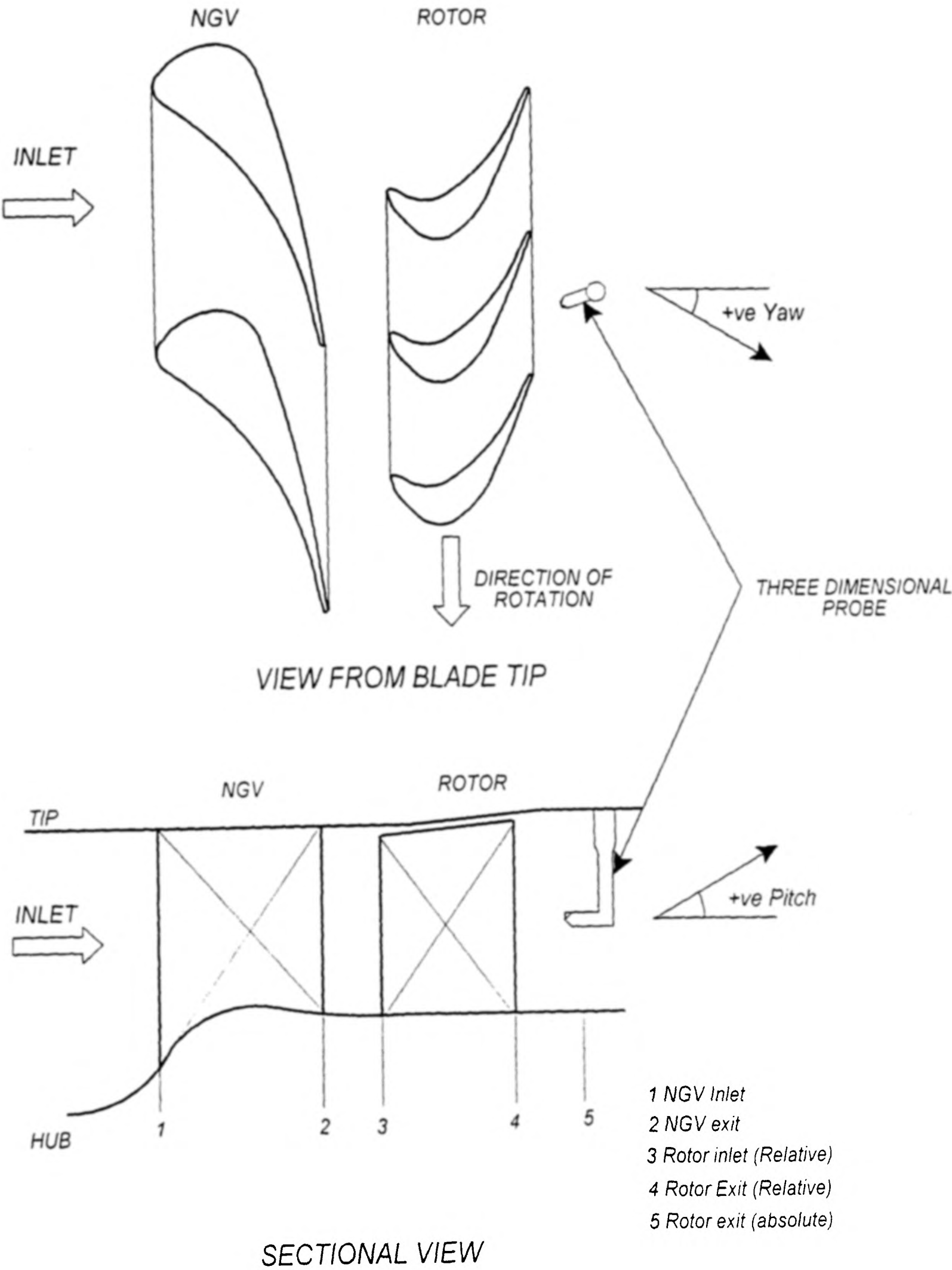


Figure 6.2 Turbine stage definition and sign convention for flow measurements

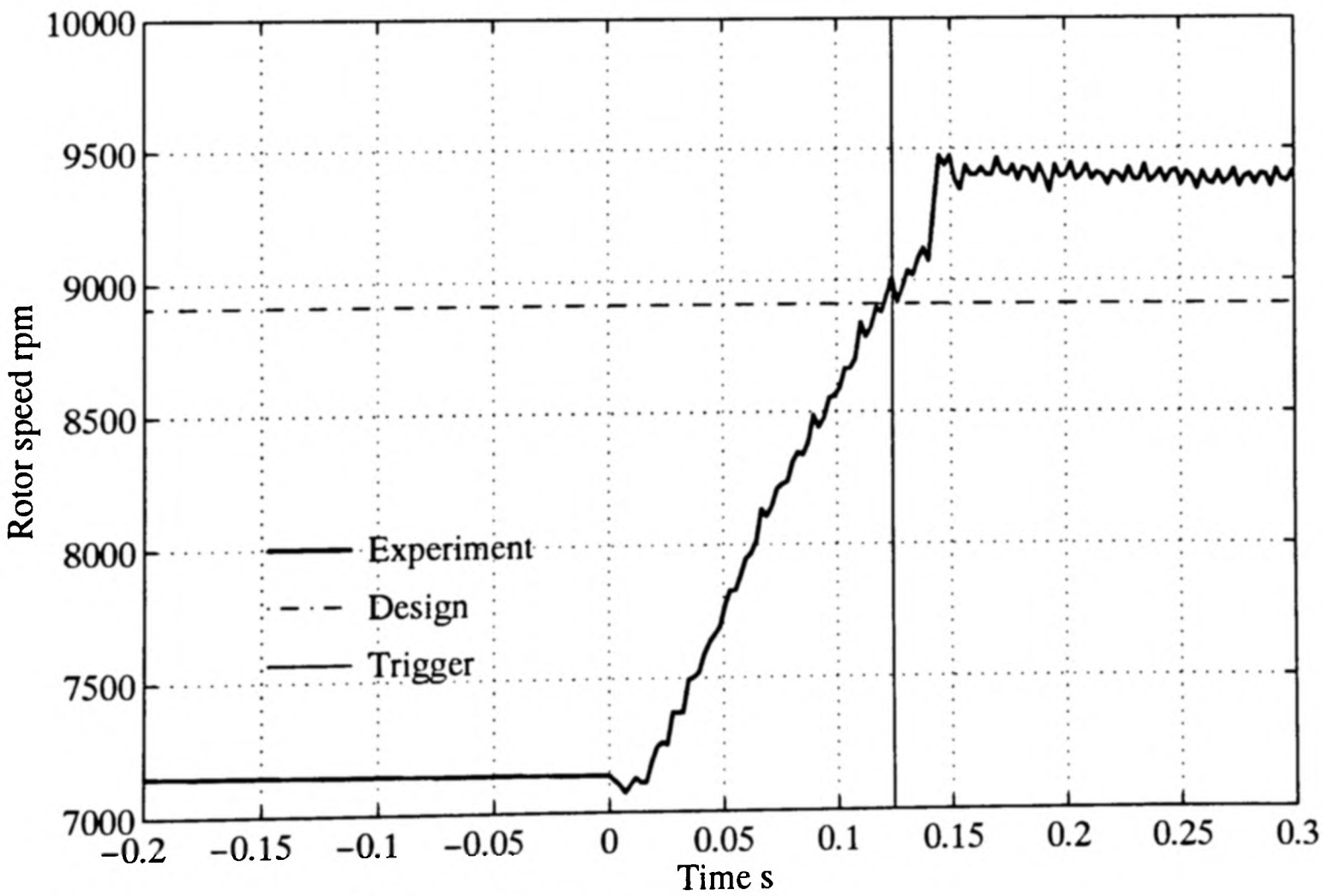
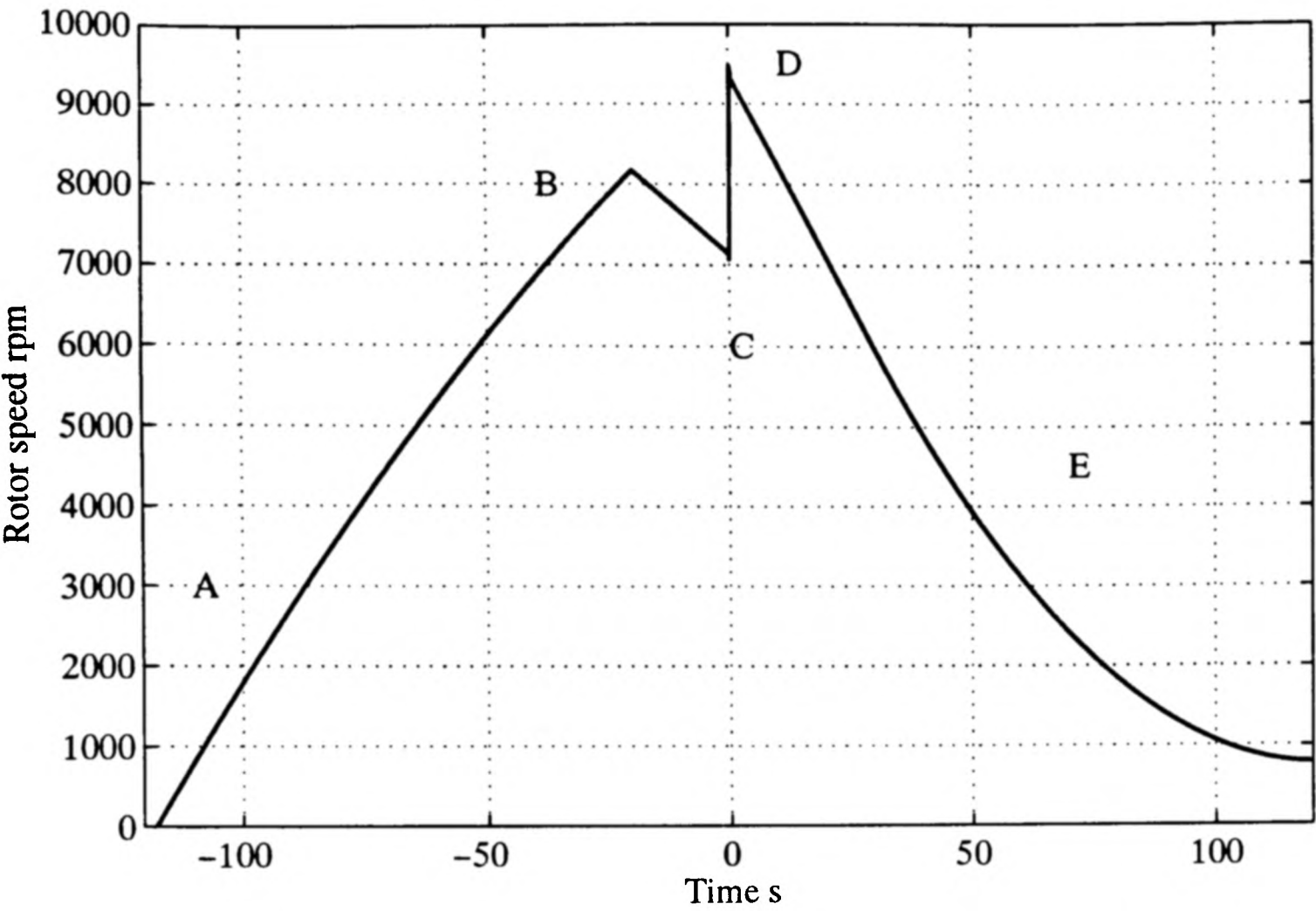


Figure 6.3 Rotor speed variation during experiment

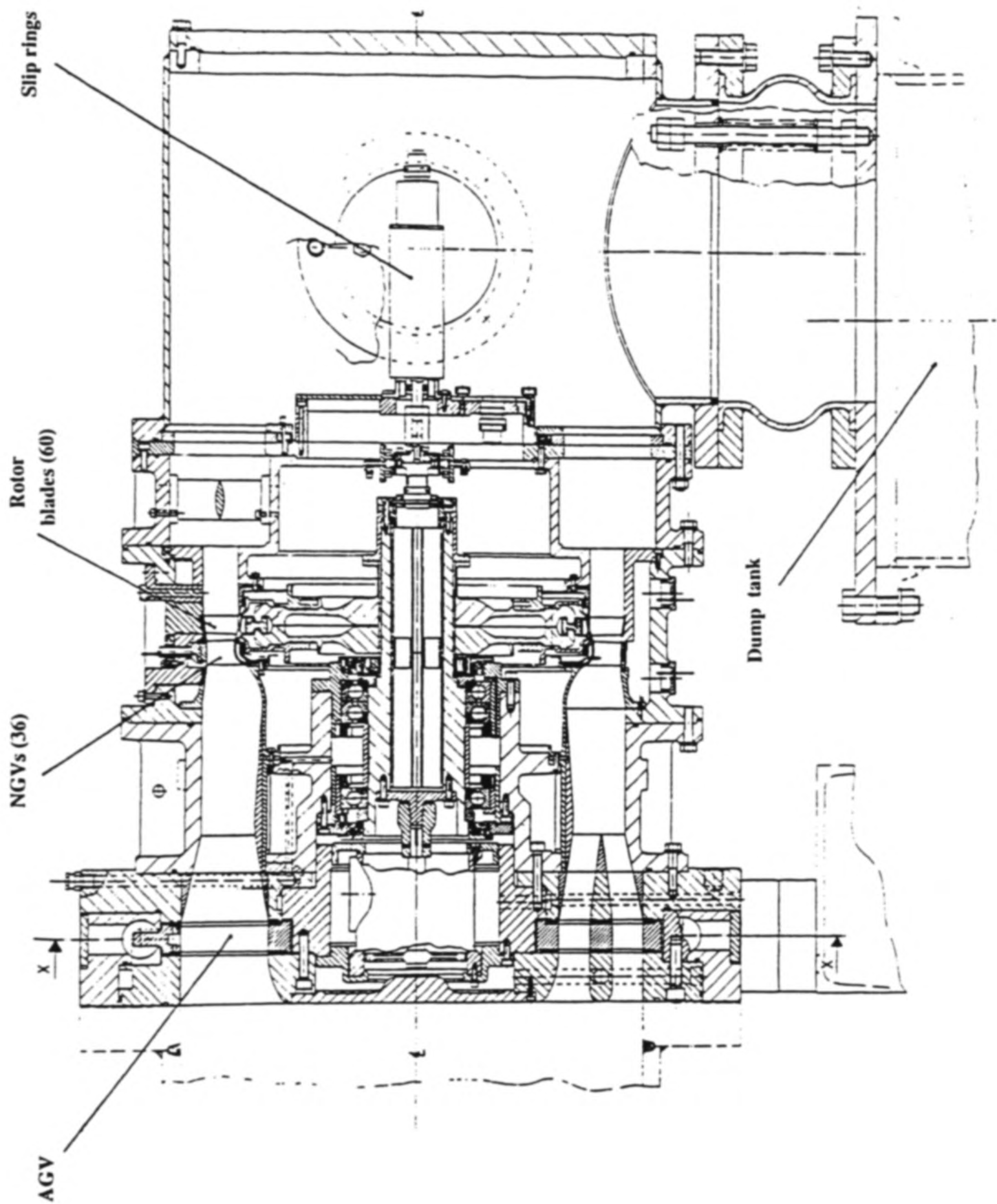


Figure 6.4 Cross section of the turbine stage

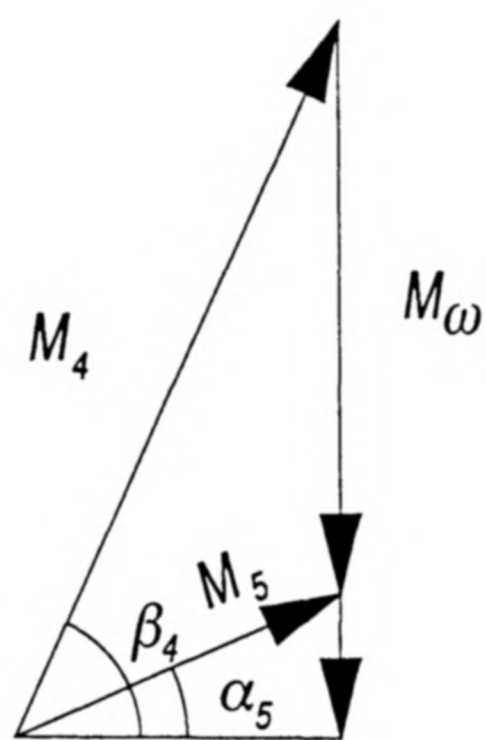


Figure 6.5 Rotor exit velocity triangle

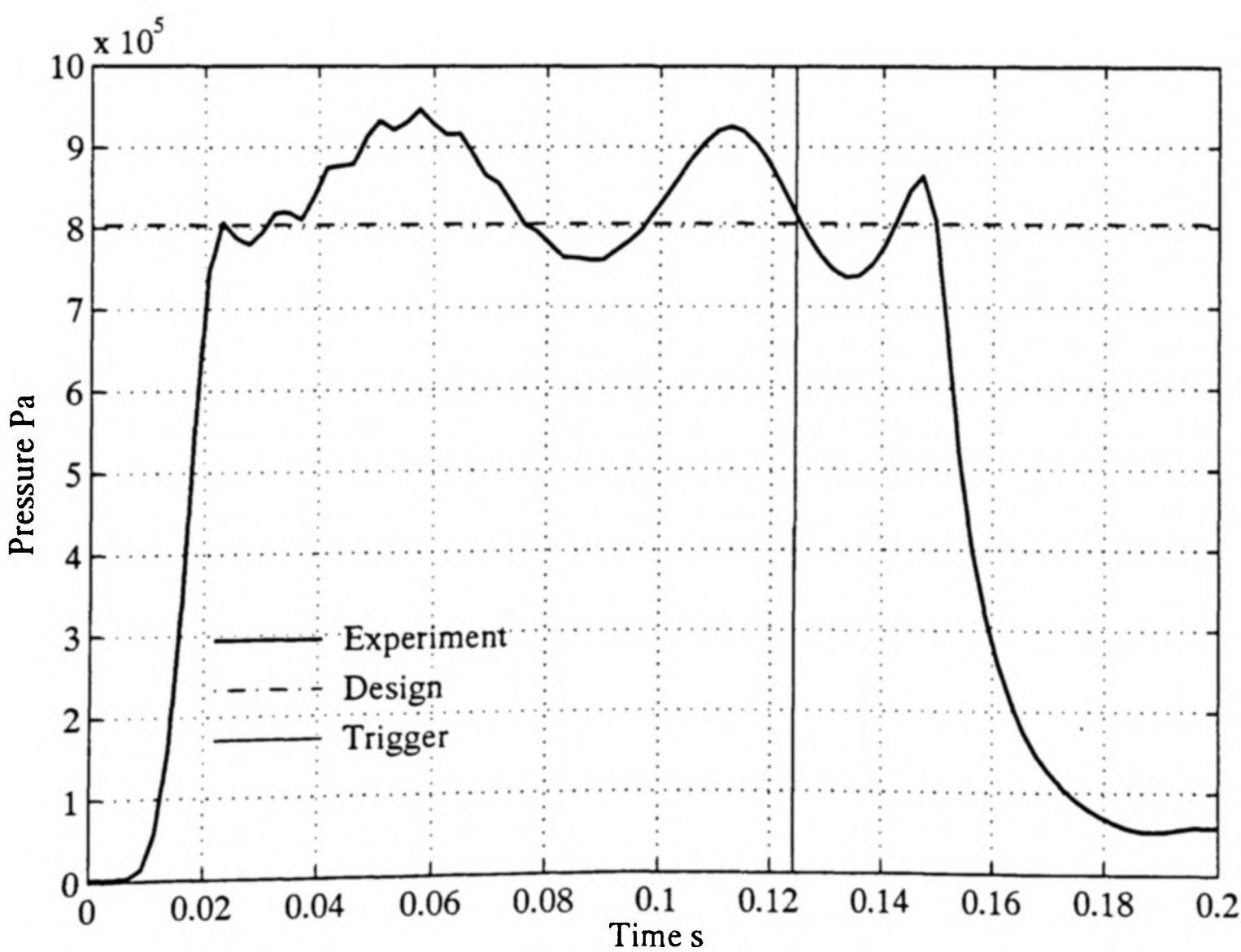


Figure 6.6 Inlet total pressure variation during the experiment

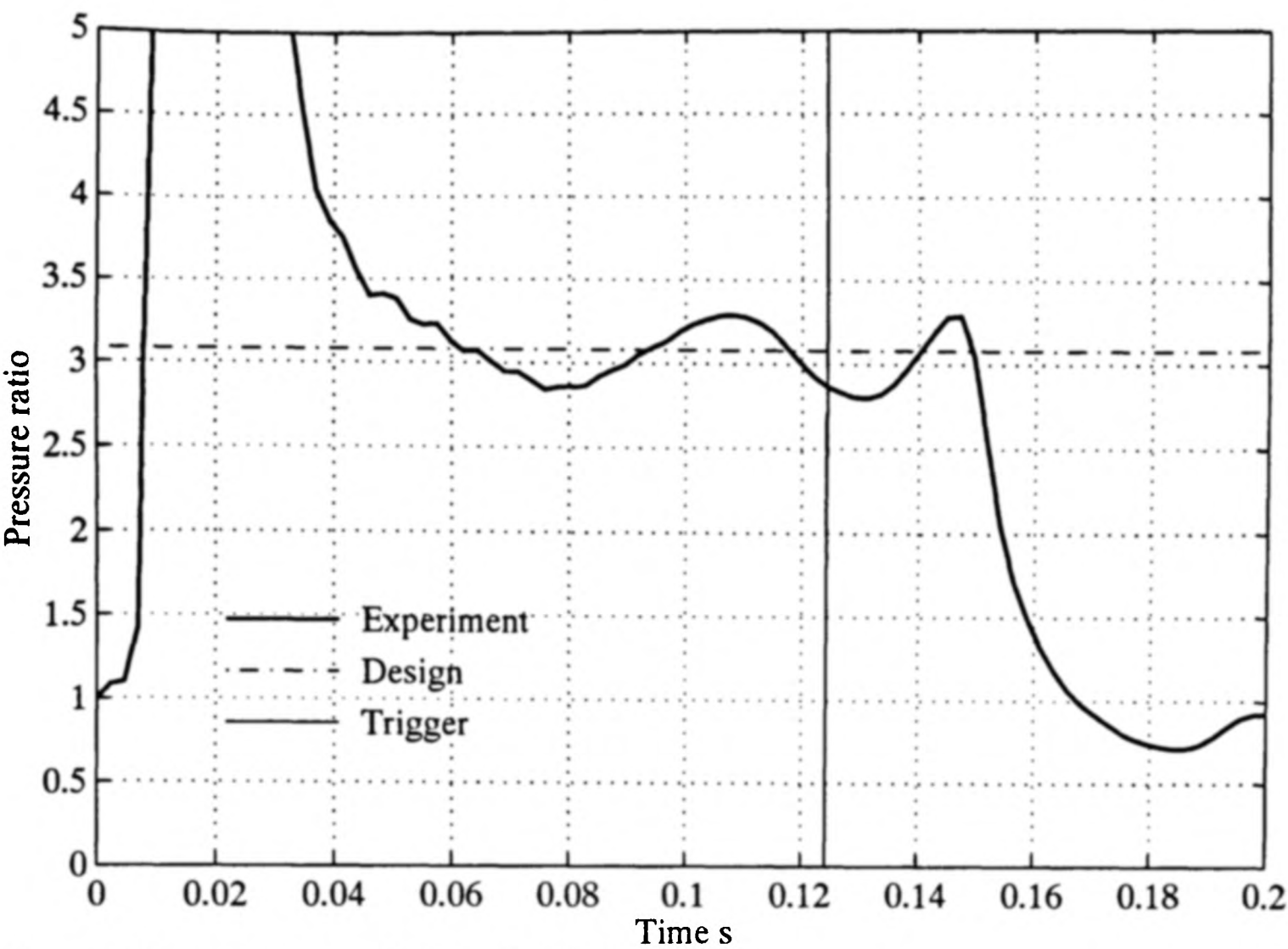


Figure 6.7 Stage pressure ratio variation during experiment

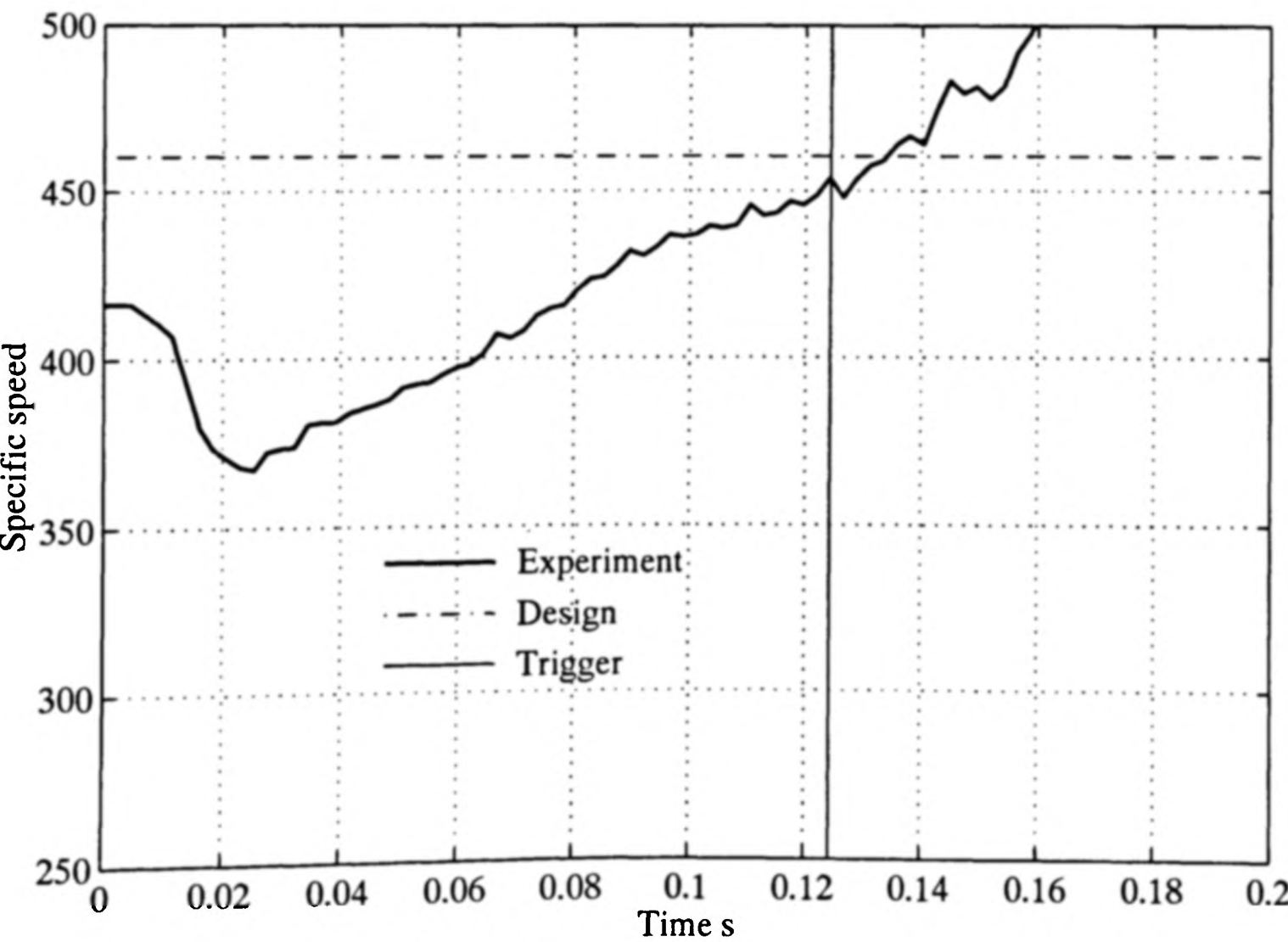


Figure 6.8 Specific speed variation during experiment

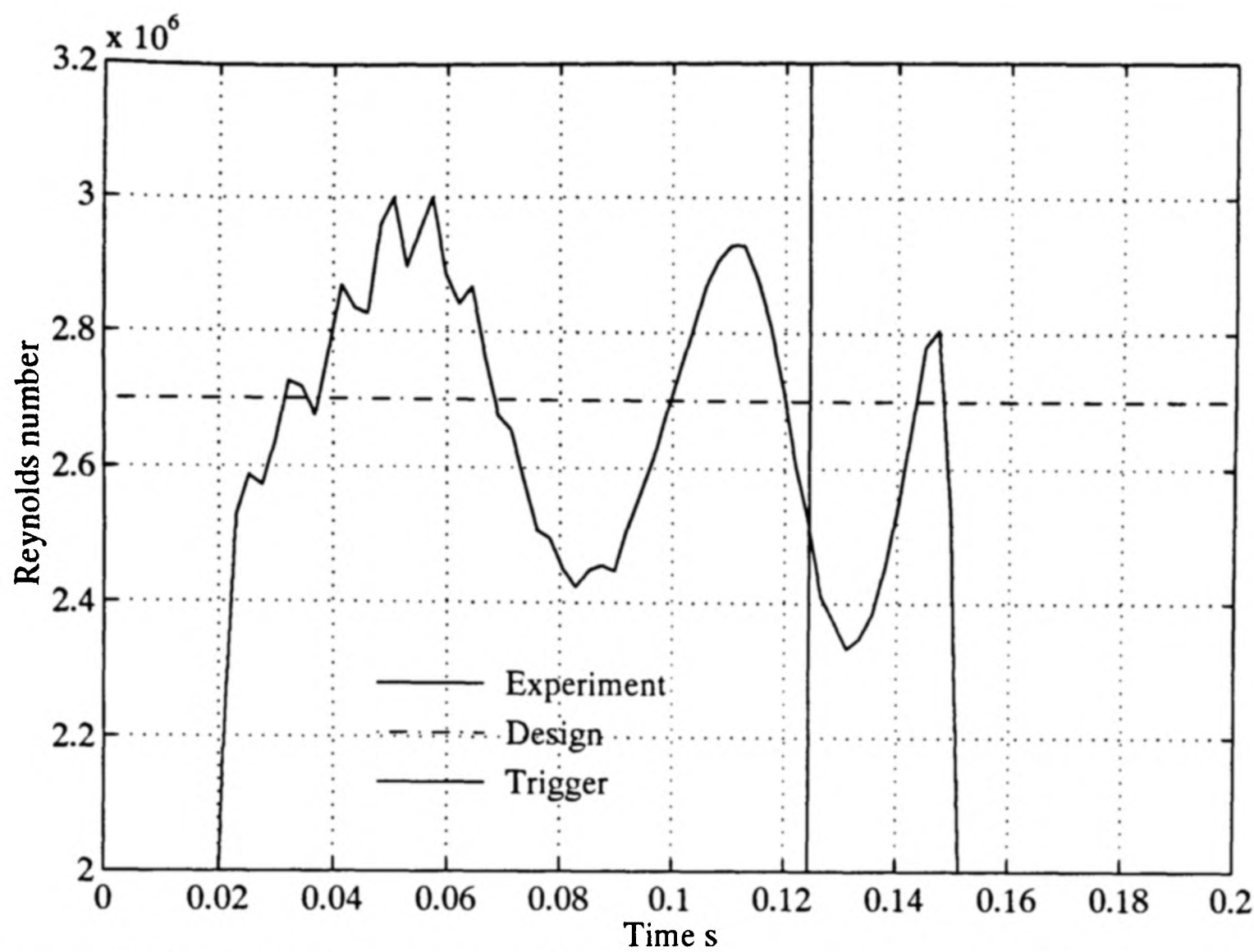


Figure 6.9 Reynolds number variation during experiment

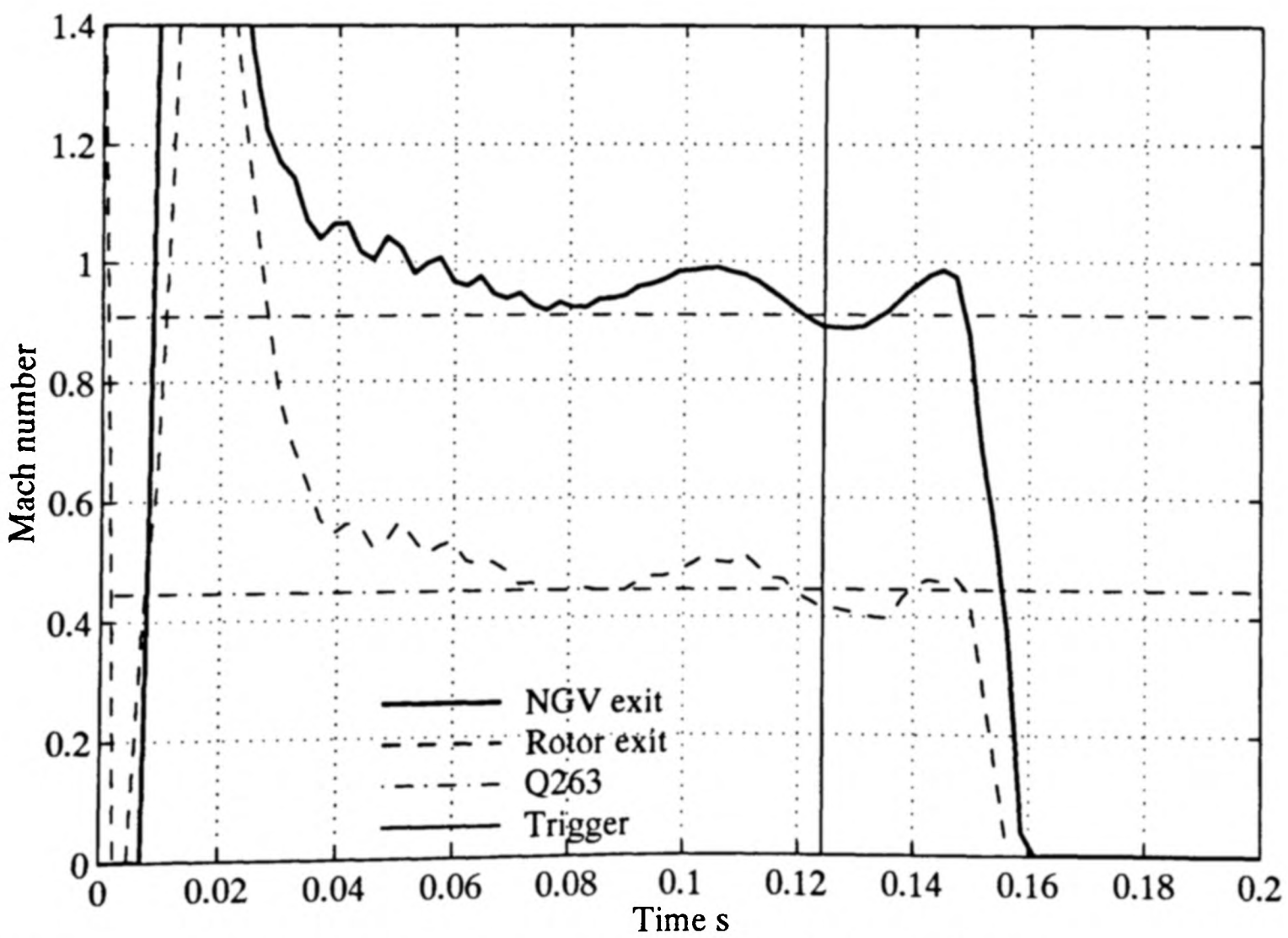


Figure 6.10 Nozzle guide vane and rotor exit Mach number during experiment

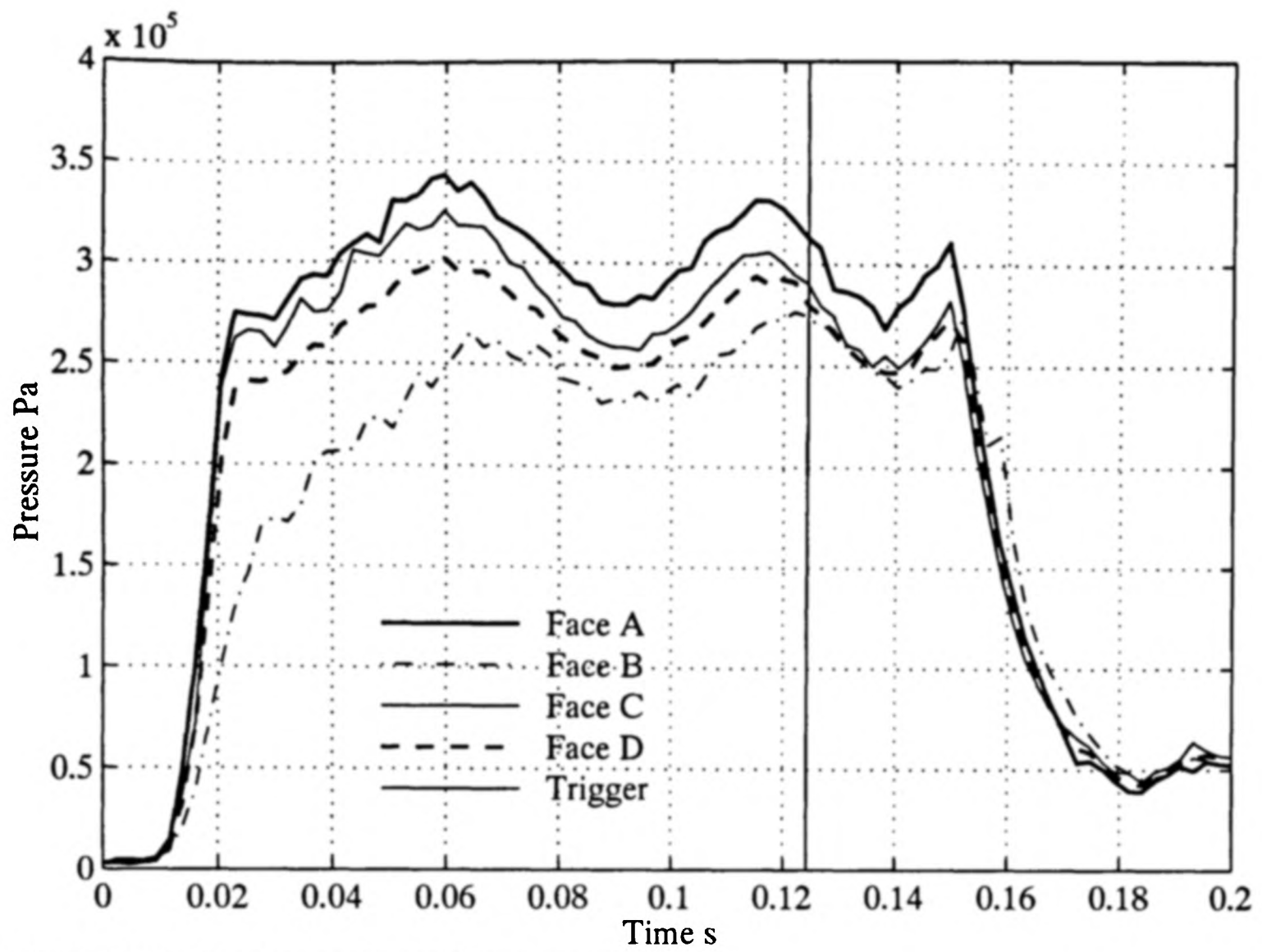


Figure 6.11 Probe face pressures during experiment

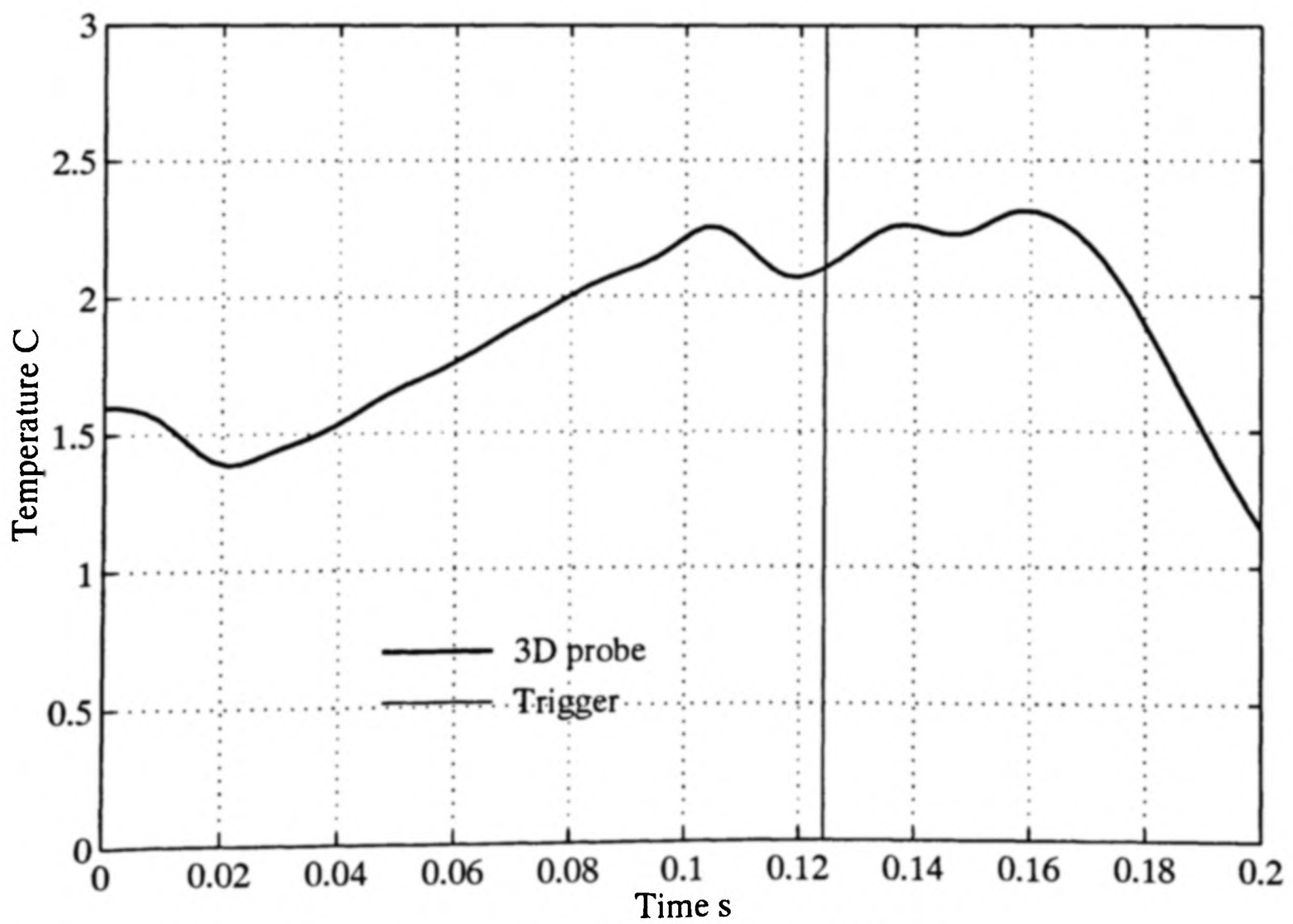


Figure 6.12 Average transducer temperature change during experiment

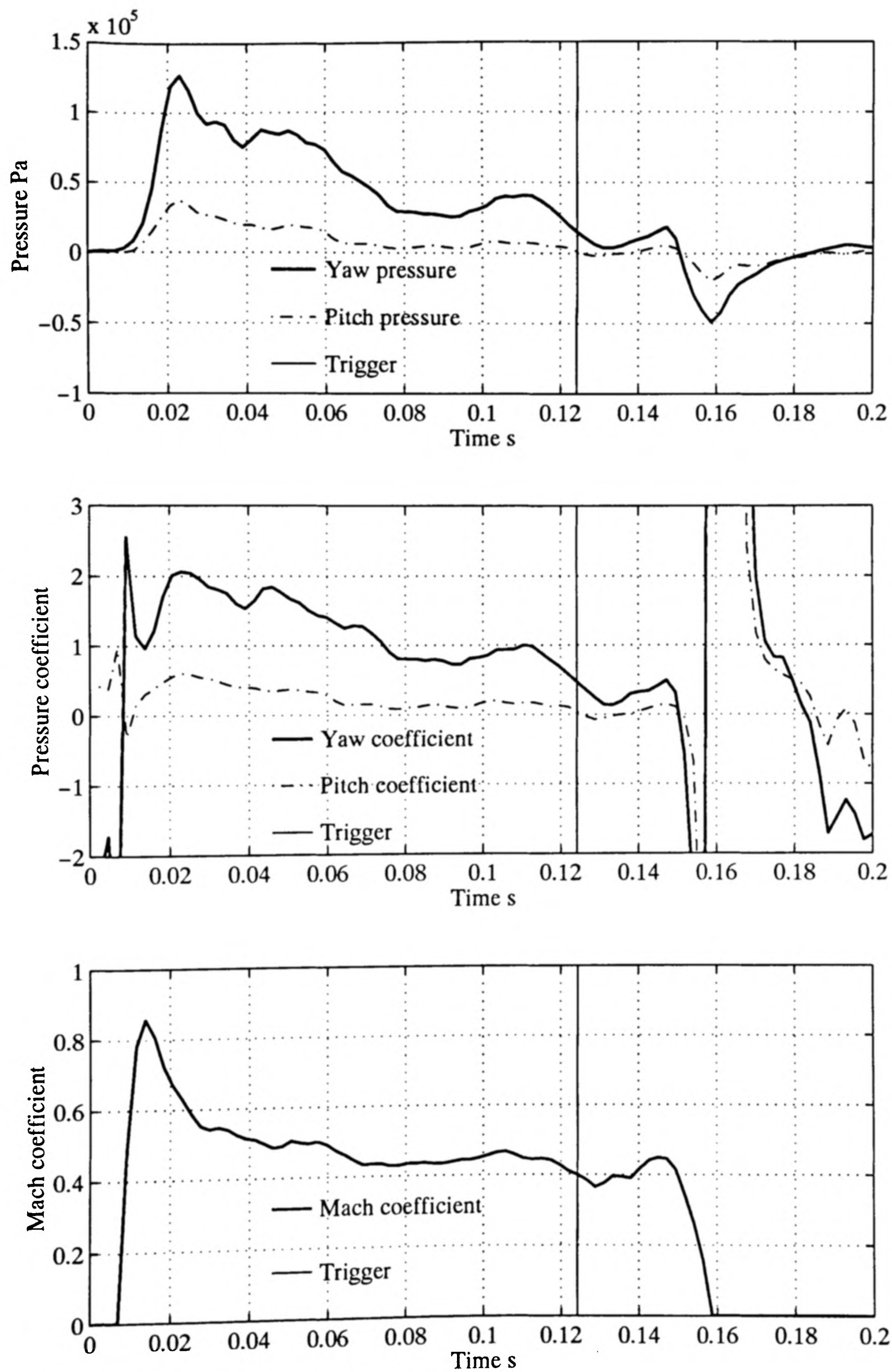


Figure 6.13 Coefficients derived from probe face pressures

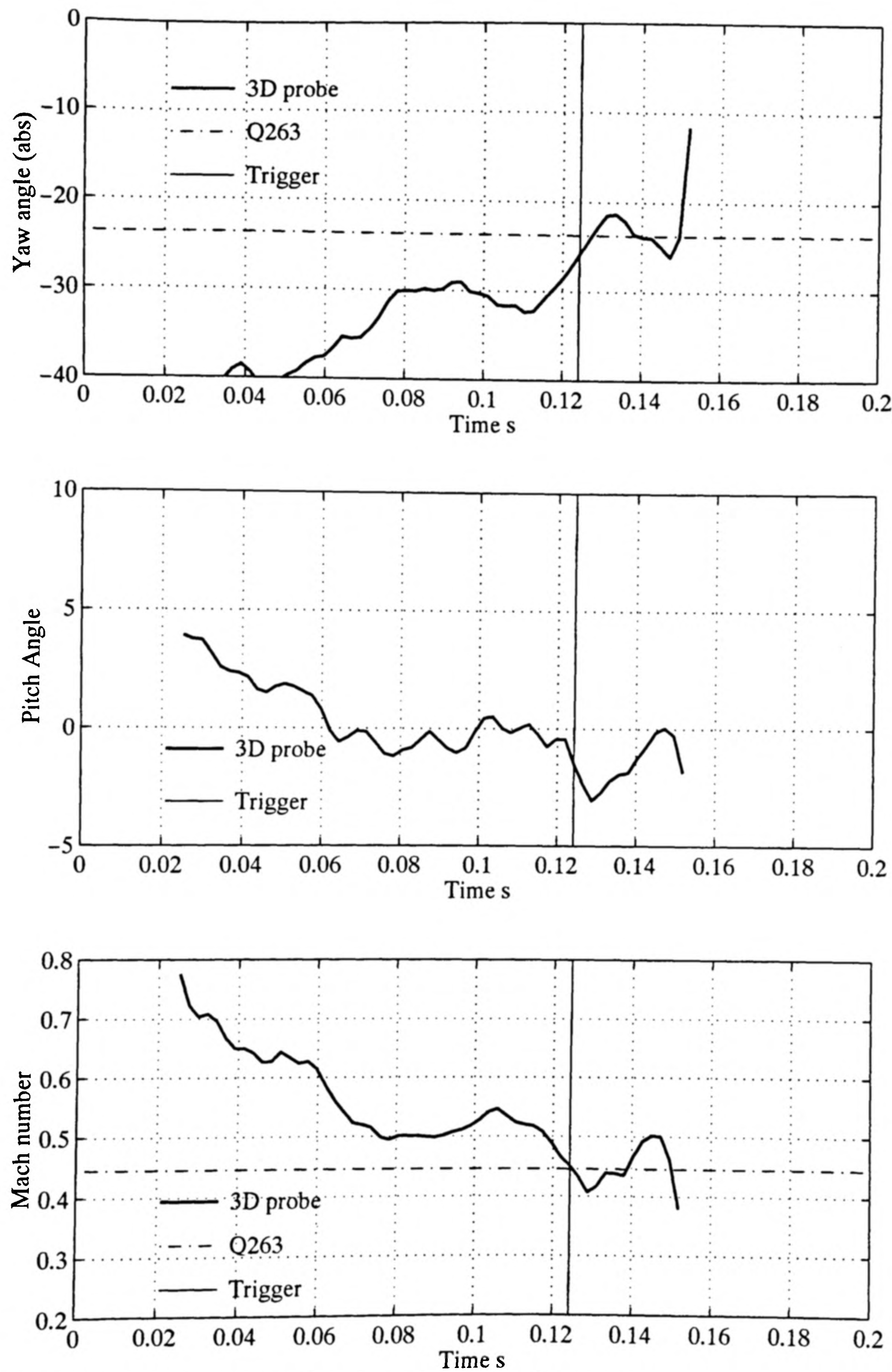


Figure 6.14 Flow measurements derived from three-dimensional probe

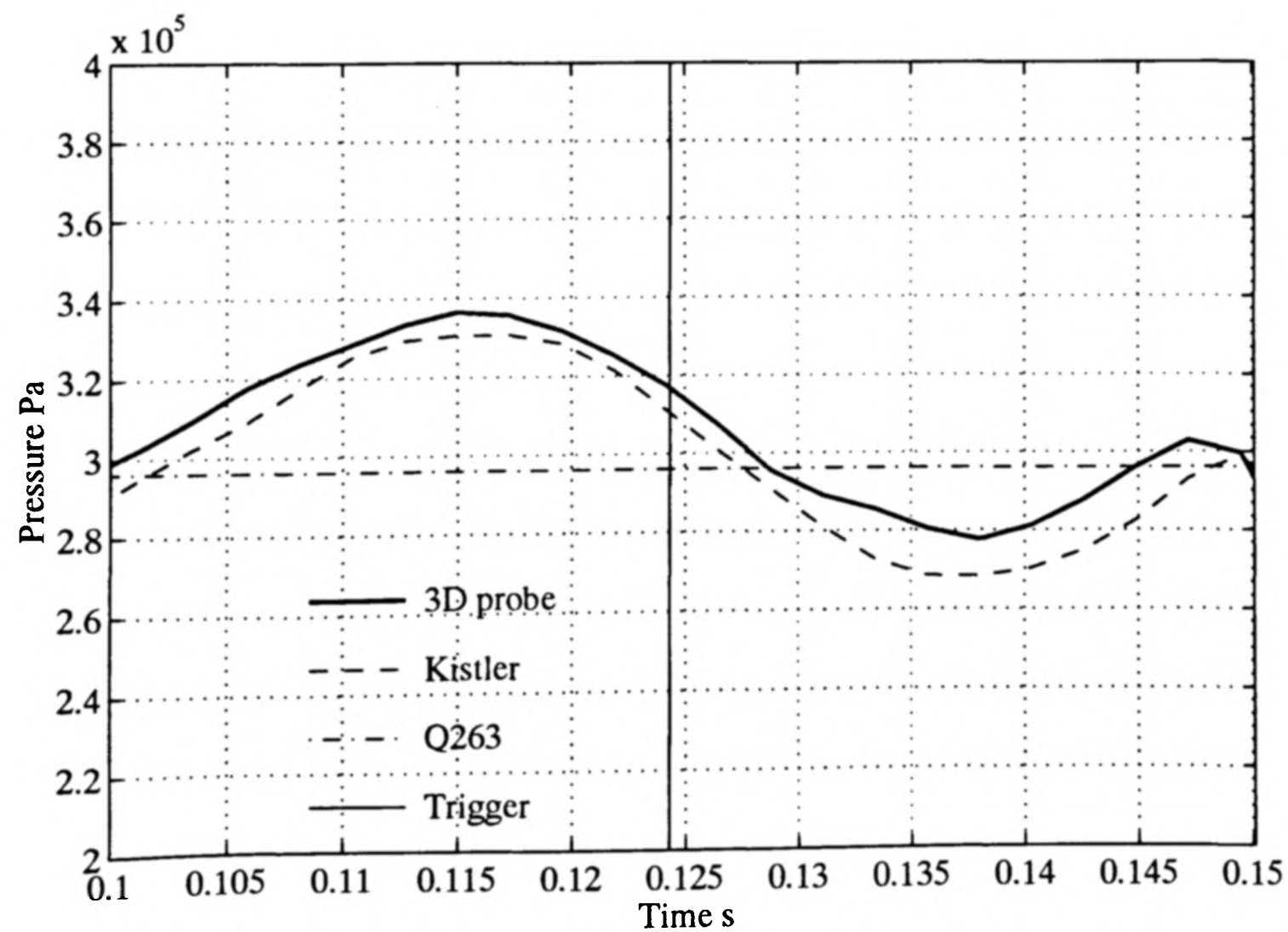
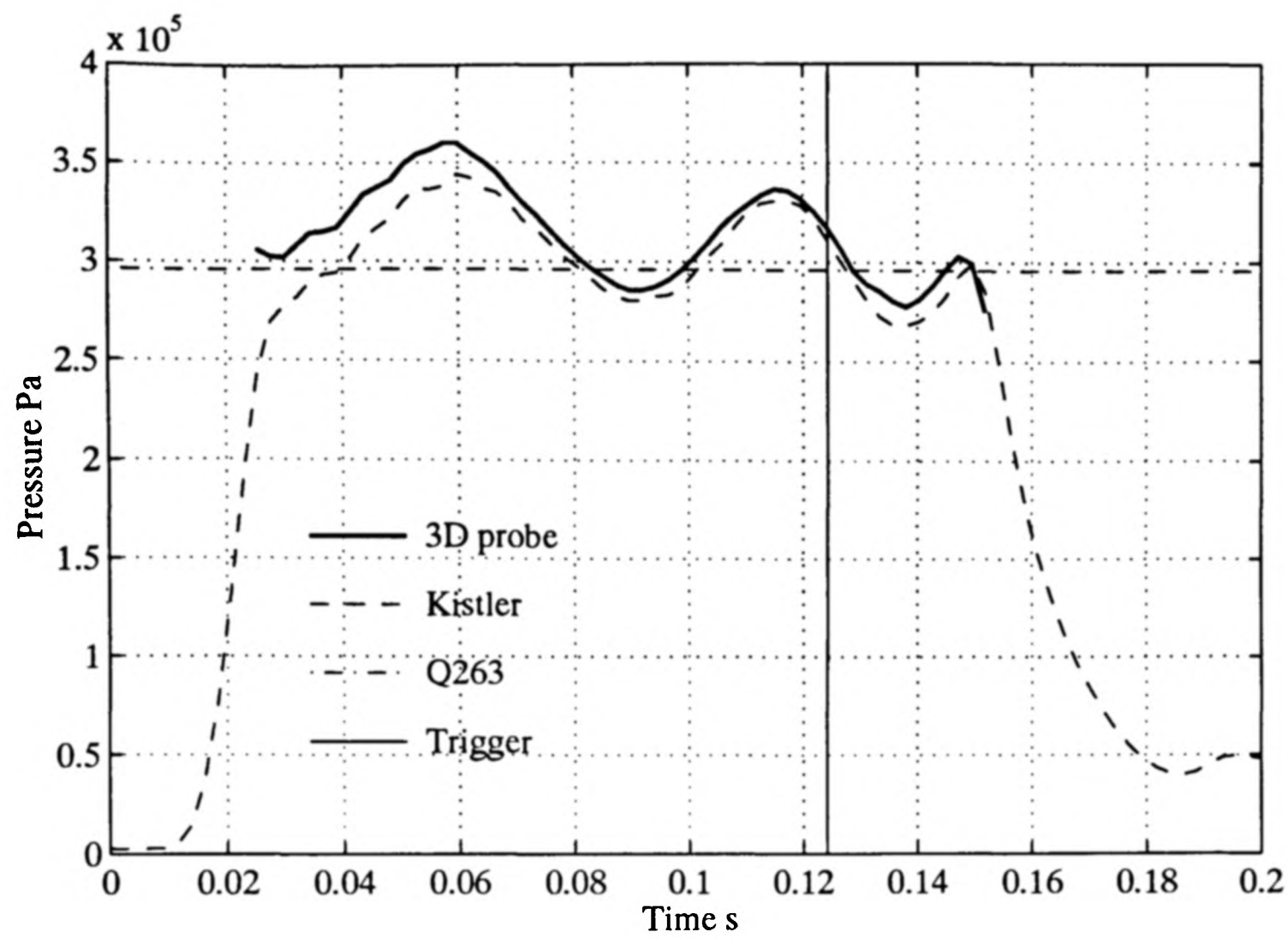


Figure 6.15 Comparison of measured total pressure

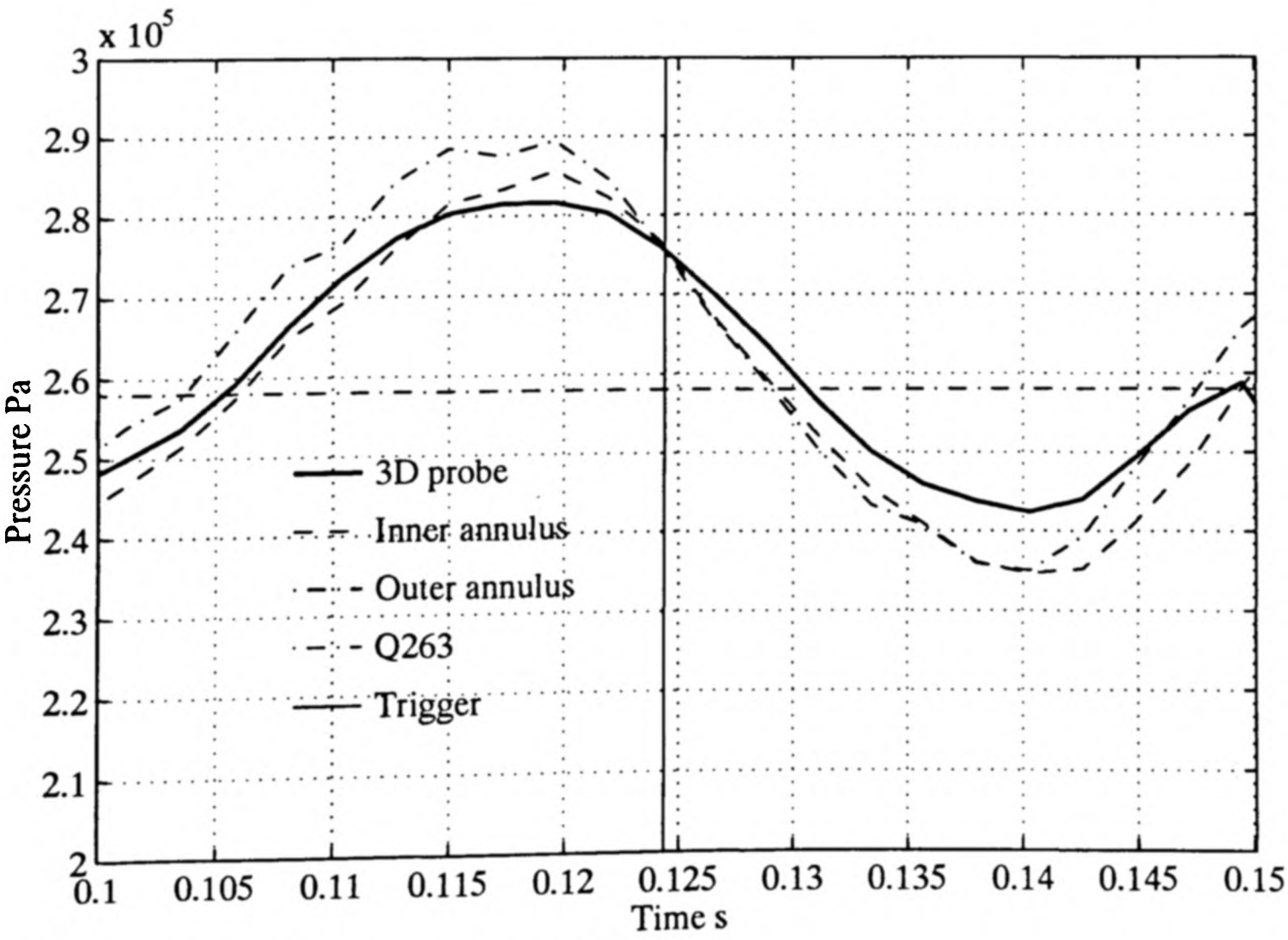
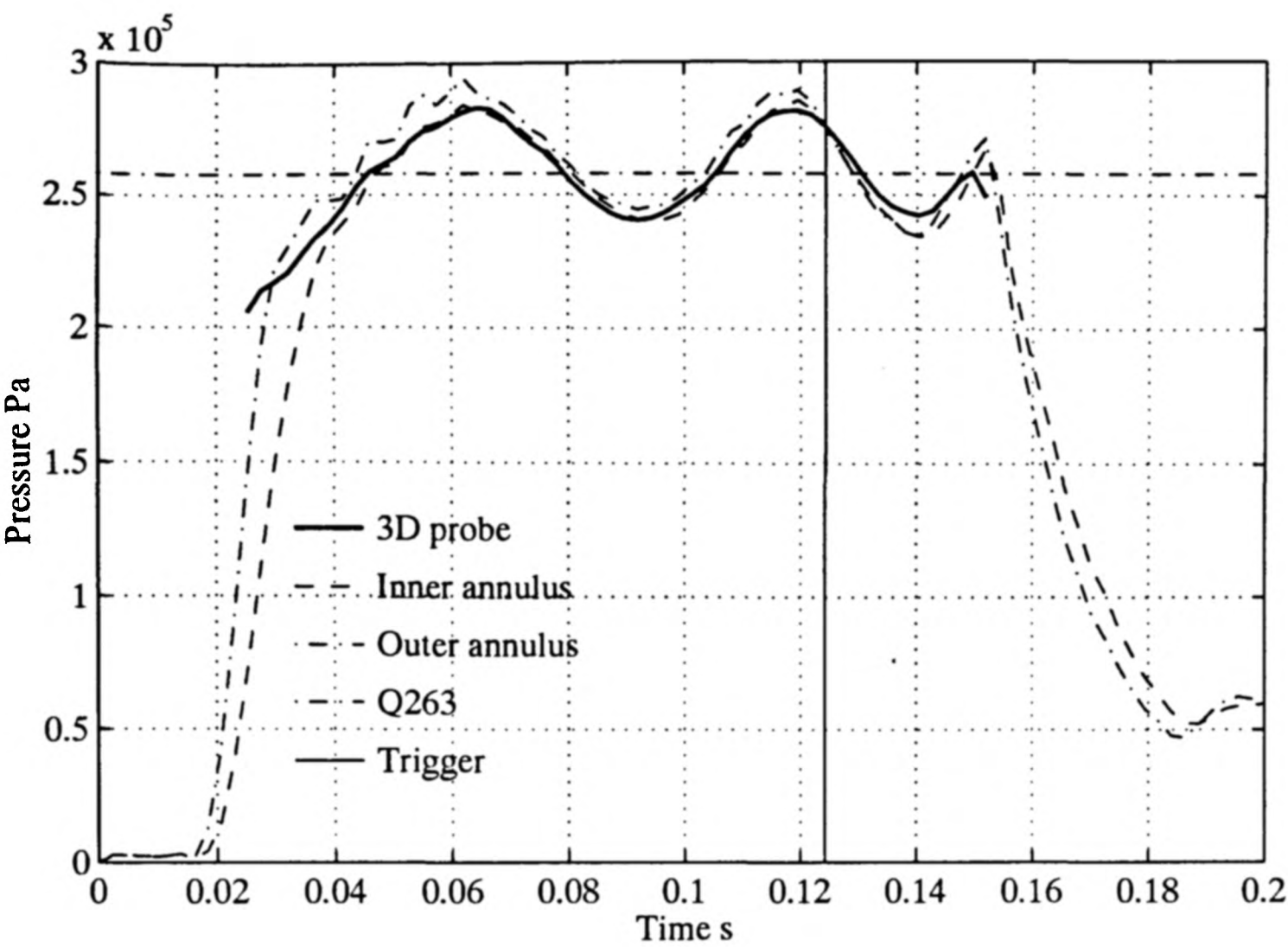


Figure 6.16 Comparison of measured static pressure

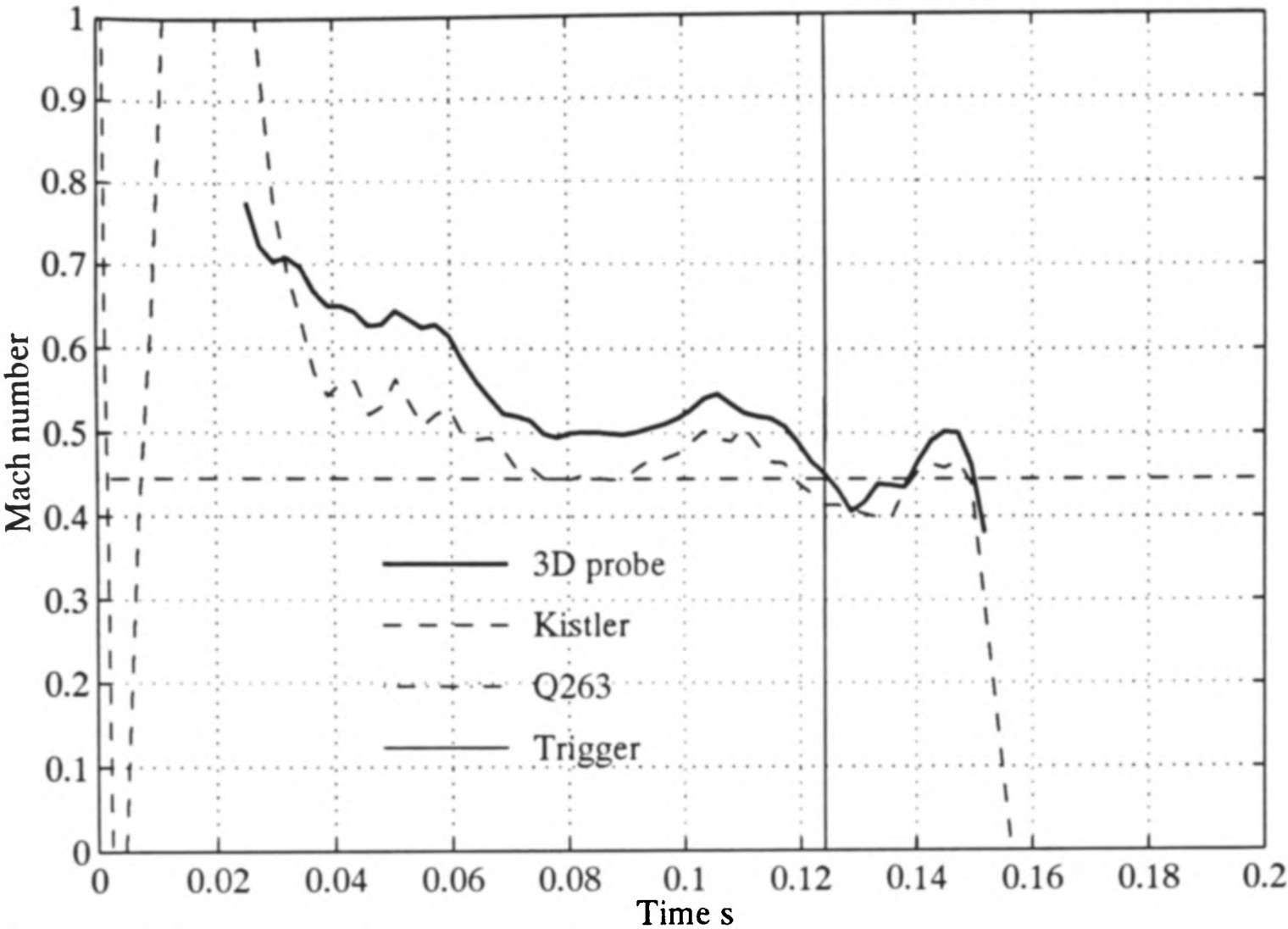


Figure 6.17 Comparison of measured Mach number

7. Unsteady flow measurements

7.0. Overview

This chapter details the measurements of unsteady flow in the Oxford rotor facility using a semiconductor probe. The Oxford rotor facility is a high pressure turbine which, at design speed, rotates at 8900 rpm. The rotor exit flow-field is three-dimensional and highly unsteady. The flow at mid-height is dominated by a periodic wake which is generated by each of the 60 rotor turbine blades. The bandwidth of the probe allowed these features to be measured.

Measurements with the probe show that the rotor exit flow is influenced by the upstream nozzle guide vane (NGV) wake. The NGV wake propagates through the rotor and can act to enhance the rotor exit wake deficit. These experimental measurements were compared with a prediction of the flow-field obtained using the unsteady viscous flow solver, Unsflo. The flow solver also predicts the NGV wake propagation through the rotor and gives good agreement with measured results.

7.1. Unsteady Flow

The flow-field in a turbomachine is highly unsteady as a result of the relative motion of the rotor blades combined with the spatial variation of pressure, Mach number and temperature. Linear cascades have been used to measure losses in turbomachines but because they do not include unsteady effects their measurements of loss will generally underestimate the true value. The main unsteady effects of relevance to turbomachinery are as follows:

7.1.1. Wake passing

Behind the trailing edge of an aerofoil a wake region will form. The trailing edge of the blade causes a flow discontinuity and the boundary layers from the suction and pressure side separate. This results in the formation of counter rotating vortices shed from opposite sides of the aerofoil. Close to the trailing edge the wake region will be narrow but defined by a sharp velocity deficit whereas far from the trailing edge the wake entrains the surrounding fluid to create a wider wake but with a shallower velocity deficit.

Wakes are generated from both the NGV and the rotor blades. The motion of the rotor causes the NGV wake to be unsteady in the rotor relative frame and therefore causes a periodic disturbance at rotor inlet. At the exit of the rotor the rotor wakes are steady in the relative frame but for a stationary object, such as an aerodynamic probe, the wakes become periodic.

7.1.2. Inviscid interaction

Inviscid interaction is where the potential flow-field distribution from the upstream blade row interacts with the potential flow-field of the downstream blade row. The relative motion of the blade rows cause the interaction to be an unsteady effect. The magnitude of the effect is a function of the distance between rotor and NGV. Parker and Watson [1972] found that for blades with spacing larger than 30% of the blade cord the potential interaction was insignificant compared with wake passing. For the Oxford rotor the spacing between NGV and rotor is approximately 40% and therefore wake passing can be expected to be the dominant feature.

7.1.3. Shock wave passing

Shocks generated in the upstream passage move relative to the downstream blades. The formation of the shock in itself causes loss because it represents an irreversible process. However most of the shocks in a turbine stage should be relatively weak. A more significant effect is the passing of shock waves over the boundary layer of the following blades. Doorly and Oldfield [1985] have shown that shock passing has an effect on the behaviour of the boundary layer. The shock impacting on the boundary layer creates a separation bubble which propagates downstream before collapsing and leaving a turbulent boundary layer.

7.2. Unsteady flow measurement

There has been relatively little published work detailing measurement of the unsteady flow-field at the exit of a rotating turbine. Of the published work on unsteady turbine flows, Joslyn et al [1982] used three-dimensional hot wires to measure the flow behind a low speed turbine. The flow was only at Mach 0.2 and therefore almost incompressible. They found that the flow was three-dimensional and had difficulty with their measurements because the pitch angle of the flow exceeded

the calibration range of their probe. This was despite the fact that the steady flow had no pitch component. They concluded that this pitch angle unsteadiness was caused by the centrifuge action in the rotor blade trailing edge wake region.

Zaccaria and Lakshminarayana [1995] used the Laser Doppler Velocimetry (LDV) to take two-dimensional unsteady measurements through a rotor stage. These measurements showed the NGV wake propagation through the rotor and interaction with the rotor wake. The way in which the wakes interacted depended on the phasing of the NGV and rotor wakes.

Virtually all unsteady measurement with semiconductor probes have been used for measurements in compressor flow-fields, Cook [1989], Ruck and Stetter [1990], Cherrett et al [1994]. Of these, Ruck and Stetter [1990] used a pneumatic probe to obtain the steady component of their pressure signals, while Cook [1989] and Cherrett et al [1994] only measured with two-dimensional techniques. The unsteady turbine flow will differ from a compressor because the flow is accelerating rather than decelerating. Furthermore, the turning angles in a turbine are much greater than in a compressor and this causes much greater secondary flows.

7.3. Experimental measurements

In the previous chapter it was shown that the semiconductor probe correctly measured the steady flow conditions. Of interest here are the unsteady measurements made with the probe obtained by the fast data acquisition system as the rotor passes through design speed. The Oxford rotor has sixty rotor blades spinning at 8900 rpm at design condition. There will be a wake generated by each of these blades and therefore rotor wakes pass the stationary probe positioned at rotor exit at a frequency of 8.9kHz .

The three-dimensional semiconductor probe was designed with the pressure transducers mounted on the surface so that the transducer bandwidth can be optimised. Each transducer on the probe had approximately 400kHz bandwidth and therefore gave the probe sufficient response to allow measurements of the unsteady wake.

When the rotor reaches design speed the data acquisition boards are triggered to record 15 ms of data at 500kHz. This time envelope is a compromise between recording data at constant conditions and allowing sufficient quantity of data to be recorded. It is also the current limit of the memory available on the data acquisition boards.¹

Plotted in figure 7.1 are the unsteady pressures measured by the four transducers on the faces of the probe during this period. This unsteady data results from the passing of wakes from the rotor. Over the 15ms of data capture the rotor rotates approximately 2.5 times. Therefore, the plot in figure 7.1 is the result of approximately 150 individual wake events. A one-line and a sixty-line disk are attached to the rotor and are recorded by an optical encoder. These enable the indexing of the rotor relative to a fixed point, in this case the stationary probe. In figure 7.2 the current rotor blade versus time is plotted for a typical experiment. The position of the rotor when the data acquisition system is triggered depends on the experiment but by using the one-line disk it can be seen in this figure that blade 42 was at the 12 o'clock position at the start of the experiment and one revolution takes approximately 7ms.

The results from the probe may be plotted against rotor blade number to synchronise periodic events in the flow with the passing rotor blades. In figure 7.3 the measurements from the probe have been plotted with front face, pA , mean of the side faces, pM , yaw pressure, and pitch pressure as five rotor blades pass the probe. The mean pressure, pM , represents the pseudo static pressure whilst the front face pressure, pA , represents the pseudo total pressure. The similarities between the measured pressures as the individual blades pass is clear.

The results plotted in figure 7.3 are sensitive to absolute pressure and can be seen falling slowly over the course of the experiment in figure 7.1. The yaw, pitch and Mach coefficients are non-dimensional and therefore the influence of the absolute pressure is removed. In figure 7.4 the experimental data is plotted in this way.

¹ A new data acquisition system has been subsequently installed which has improved the memory capacity.

The inversion techniques described earlier in chapter 5 may be applied to these coefficients to find the measured yaw angle, pitch angle and Mach number. These results are plotted in figure 7.5. The flow Mach number clearly drops in a periodic fashion with the passing of the rotor blades. This drop in Mach number accompanied by a change in yaw and pitch angle can be attributed to the rotor wake. Whilst the previous plots were continuous there are breaks in these plots. These breaks occur when the measured coefficients are out of the range of the probe calibration which causes the inversion algorithm to return a null value. As the wake deficit passes over the probe it has difficulty resolving the detail because of the low Mach number and extreme angles.

To see how much of the trace is random unsteadiness rather than cyclic unsteadiness the measured flow from the same wake on its second revolution can be compared with the first revolution, as in figure 7.6. Clearly there is a considerable degree of similarity between the trace on the first and second revolution. This demonstrates how repeatable the flow is. The difference between the individual blade wakes is because each blade has a characteristic signature onto which a random unsteadiness is superimposed. The characteristic signature is a result of the slight differences in the manufacturing and mounting of each of the blades.

7.3.1. Ensemble averaged values

Whilst it would be desirable to average out the random unsteadiness for a single blade over a multiple of revolutions this is not possible in a transient facility. The only way to get an equivalent result is to ensemble average the signal from all the rotor blades to find an average blade wake profile. R.W. Moss developed an ensemble averaging technique for use in reducing the experimental data obtained from blade mounted instrumentation. The ensemble averaging uses macro files written in MATLAB to manipulate the data. Whilst the blade data was recorded in the rotating frame of reference, the three-dimensional probe is recording data in the stationary frame. However, the ensemble averaging technique is similar and therefore only a few modifications were required to allow the ensemble averaging of the probe data.

The technique works by firstly using the sixty-line disk to cut the measured wake into sixty segments per revolution, corresponding to each rotor blade. Speed variation during the experiment can cause the phase of the sampling to become distorted. This speed variation is removed so that the data from the first and second passing of the same blade may be averaged without distortion to generate a one revolution average. The time envelope of 15ms over which data was captured meant that the average will be over three revolutions for some blades and two revolutions for others. Finally the data from the sixty rotor blades are averaged to give the mean blade flow profile.

Before the ensemble averaging process is undertaken it must be considered which quantity should be ensemble averaged. It would be possible to ensemble one of three sets of data:

- the raw pressures
- the non-dimensional coefficients
- the measured velocity (Mach, yaw and pitch angle)

However, ensemble averaging the raw pressures has the disadvantage that any trend in the total pressure during the experiment is likely to affect the resulting ensemble average. This trend is avoided by choosing either the non-dimensional coefficient or the measured flow velocity for ensemble averaging. There is some loss of data when the non-dimensional coefficients are inverted in regions where measurements fall outside the calibration grid. Where these regions are a result of random unsteadiness averaging the non-dimensional coefficients will remove the random component and reduce the amount of lost data. Therefore the non-dimensional coefficients were chosen for the ensemble average process rather than the other two choices. Once the coefficients had been ensemble averaged they were inverted to find the ensemble measured Mach number, yaw angle and pitch angle.

In figure 7.7 the results of the ensemble average are plotted superimposed over the instantaneous values which were plotted earlier. For the sake of comparison the ensemble averages have been repeated five times. The ensemble average of the sixty wake profiles is clearly a good

representation of the raw wake profiles. When measuring unsteady flow behind a transonic compressor using an aspirating probe, Ng and Epstein [1983] found most of their features destroyed by ensemble averaging. They concluded that variations were unsteady in the rotor relative frame and caused by a shed vortex in the blade wake. This is clearly not the case for these measurements because the periodic features are quite distinct. They are steady in the rotor relative frame and clearly a function of the rotor wake.

7.3.2. Variation with circumferential position

In their measurements of rotor wakes using laser techniques, Zaccaria and Lakshminarayana [1995] found that the NGV wake can influence the rotor wake. To investigate whether the position of the probe relative to the upstream NGV had any influence on the measured results a number of experiments were undertaken at different positions relative to the NGV. The probe mounting mechanism consisted of a circular rail which allowed the probe be moved and positioned in the circumferential direction. An arbitrary co-ordinate system can be defined by taking the trailing edges of the NGVs and projecting them in the axial direction so that the position of the probe can be defined in a range from 0-100% of the NGV blade pitch.

The experiment was undertaken at four circumferential positions and the results from these experiments are plotted in figures 7.8-7.10. At each location the experiment was repeated at least twice to ensure that the results were repeatable. The run to run agreement is very satisfactory in terms of both the magnitude and structure of the measured values.

The unsteady Mach number shows a small variation as the probe is moved relative to the upstream NGV. The Mach number falls twice during each passing of the blade with a slight recovery of the Mach number between the deficits. The magnitude of the first drop in Mach number varies depending on the position relative to the NGV whereas the second velocity deficit appears invariant with position. This suggests that the first deficit is generated from the NGV wake profile whereas the second deficit is caused by the rotor wake. The yaw angle variation with NGV position can be seen in figure 7.9 although attributing the characteristic signature to the rotor and NGV component is less

clear. Finally, the unsteady pitch variation with NGV position in figure 7.10 is quite dramatic, with a particularly sharp change to positive pitch during the rotor wake indicated when the probe is at 50% NGV.

7.4. Computational predictions

R.W. Ainsworth has used the unsteady flow prediction program Unsflo to model the flow in the Oxford rotor. This program is a two-dimensional viscous, time resolved code that has been developed by Giles[1988] to predict the interactions of both the stator pressure field and the stator wakes with the blade row.

Predictions from models like this are very useful for illustrating the main features in the flow-field. In figure 7.11 plots of entropy contours are used to indicate the propagation of the NGV wake downstream. It can be seen from this plot that the upstream wake interacts with the flow through the rotor passage depending on the relative timing of the NGV wake and the rotor position. Entropy contours are plotted because these highlight the secondary flows very clearly. The loss region associated with the viscous wake appears as a region of entropy increase.

The wake cycle has been arbitrarily defined to start when the wake from the upper NGV blade is incident on the leading edge of the rotor blade. In frame 1, at the beginning of the cycle the wake produced from the upper NGV can also be seen to give a feature mid-passage of the middle and lower turbine blade in the plot. As the cycle continues this NGV wake feature is propagated through the turbine row to rotor exit. The result is that at rotor exit there are two distinct loss regions. The first is generated from the rotor wake, whilst the second is a remnant of the upstream NGV wake. The relative positions of the two features depend on the position in the wake cycle.

There is considerable confidence in the predictions generated by Unsflo. Abhari et al [1991] have successfully compared Unsflo predictions with time resolved heat transfer data taken in the MIT rotor facility and similarly Moss et al [1995] compared unsteady blade pressure measurements to Oxford rotor facility results.

7.4.1. Comparison between experiment and CFD

In an experiment the manufacturing tolerances will influence the flow through a turbine, especially in regions where the flow approaches sonic since small area changes will be critical. In computational solutions the influence of small manufacturing differences is unlikely to be seen. It is therefore only realistic to compare the computational predictions with the ensemble average data.

Using the Unsflo prediction run by R.W. Ainsworth of the Oxford rotor exit flow-field it is possible to probe the flow conditions at the same position as the probe was positioned in the experiment. In figures 7.12-7.15 the predictions from Unsflo are compared with the ensemble averaged values found from the experiments at two different positions relative to the upstream NGV.

The experimental measurements give a higher average Mach number than the Unsflo but the characteristic shapes of the Mach number profiles show very similar features. It was suggested earlier that the first dip in the Mach number on the ensemble average trace is the result of the deficit caused by the upstream NGV wake, whilst the second is a result of the rotor wake deficit. The two dips can also be seen in the CFD predictions, with the timing showing very good agreement although the magnitudes are slightly different. In comparison with the experiment the CFD predicts a stronger and broader influence of the upstream NGV wake but a similar strength rotor wake.

In figure 7.13 and 7.15 the measured ensemble average yaw angles are compared with the predicted yaw angles. As the probe was mounted at an angle of -19° to the flow the offset has been incorporated into the experimental results. There is a slight difference in the absolute levels of the measured angles but this could be a function of the timing of the data acquisition given that the mean flow angles are changing during the experiment. However the comparison of the characteristic shapes of the yaw angle variation between experiment and Unsflo is very good.

Whilst it would be desirable to compare measured pitch angle with a predicted value, three-dimensional unsteady flow solvers are only in their infancy and the predictions currently are unavailable. It will be interesting to compare the measured pitch variation with three-dimensional

unsteady predictions when they become available, in particular to see if they predict the sharp pitch angle variation seen at 50% NGV.

7.5. Discussion on performance

7.5.1. Experimental results

The aerodynamic probe has been shown to be able to measure both the steady flow and unsteady flow components of the rotor exit flow-field. This is a significant achievement; it has been achieved without resorting to the use of a pneumatic probe for the steady component.

Measurements from the probe show that the rotor exit flow is influenced by the upstream NGV wake flow. This can be attributed to the interaction between rotor blade wake and the NGV wake passing through the rotor stage. The unsteady measurements have provided valuable support to the computational predictions.

7.5.2. Dynamic effects

The probe is constructed with surface mounted semiconductor pressure transducers which give a bandwidth which is sufficient to measure the unsteady surface pressure to a frequency in excess of 100kHz. The probe static calibrations are based on the Bernoulli equation, 2.10, but when the probe is used to measure unsteady flow this equation may no longer be valid. In chapter two the steady flow energy equation was assumed in the derivation of the Bernoulli equation and this is clearly not appropriate where the flow is unsteady.

An alternative method of deriving the Bernoulli equation which does not use the steady flow energy equation is to start from the Euler equation 7.1.

$$-\frac{1}{\rho} \frac{dP}{dx} = \frac{\partial u}{\partial t} + u \frac{du}{dx} + g \frac{dz}{dx}$$

Equation 7.1

Where the flow is steady, the $\frac{\partial u}{\partial t}$ term is zero and it is relatively trivial to integrate equation

7.1 to derive the Bernoulli equation. In the case when the flow is unsteady, the ratio of the two

velocity terms will indicate the validity of applying the Bernoulli equation because where the ratio is small the unsteady terms may be ignored. The condition for when the unsteady effects can be ignored is given in figure 7.2.

$$\frac{\frac{\partial u}{\partial t}}{u \frac{du}{dx}} \ll 1$$

Equation 7.2

To provide an estimate of the magnitude of this ratio, the numerator may be estimated by assuming the flow is sinusoidal in nature which implies that $\left| \frac{\partial u}{\partial t} \right|$ is proportional to the product of the magnitude of the flow velocity and frequency, Uf .

In the case of the denominator, consider first that the presence of the probe in the flow creates a stagnation point at which the fluid is at rest. If it is assumed that the fluid is brought to rest in a distance which is of the same order as the probe size then the $u \frac{du}{dx}$ term may be estimated to be of the order of $U \times \frac{U}{L}$. Combining these two assumptions gives:

$$\left| \frac{\frac{\partial u}{\partial t}}{u \frac{du}{dx}} \right| \approx o \left(\frac{Uf}{\left(\frac{U^2}{L} \right)} \right) = o \left(\frac{fL}{U} \right)$$

Equation 7.3

The form of the expression in equation 7.3 will be recognised as the Strouhal number, the non-dimensional parameter widely use to quantify the magnitude of the unsteady effects.

$$S_t = \frac{fL}{U}$$

Equation 7.4

Where the Strouhal number is very much less than unity the unsteady terms will be negligible compared with other terms and the flow can be expected to behave like steady flow (quasi steady).

For the geometry of the experiment the rotor wake passing frequency, 8.9kHz, gives a probe Strouhal number of approximately 0.08. Steady flow past a circular cylinder gives rise to unsteadiness in the wake and periodic vortex shedding at Strouhal numbers above 0.21. This Strouhal number is achieved for the probe at a frequency of 23.1 kHz and therefore at frequencies above this caution with measurements should be taken.

Given that the velocity wake passing frequency are often fixed by the experiment the greatest scope for improving probe bandwidth is by reducing its size. Future generations of probe with higher bandwidths may be achievable through further minimisation. This is provided manufacturers can develop even smaller transducers and the constructional challenges can be overcome.

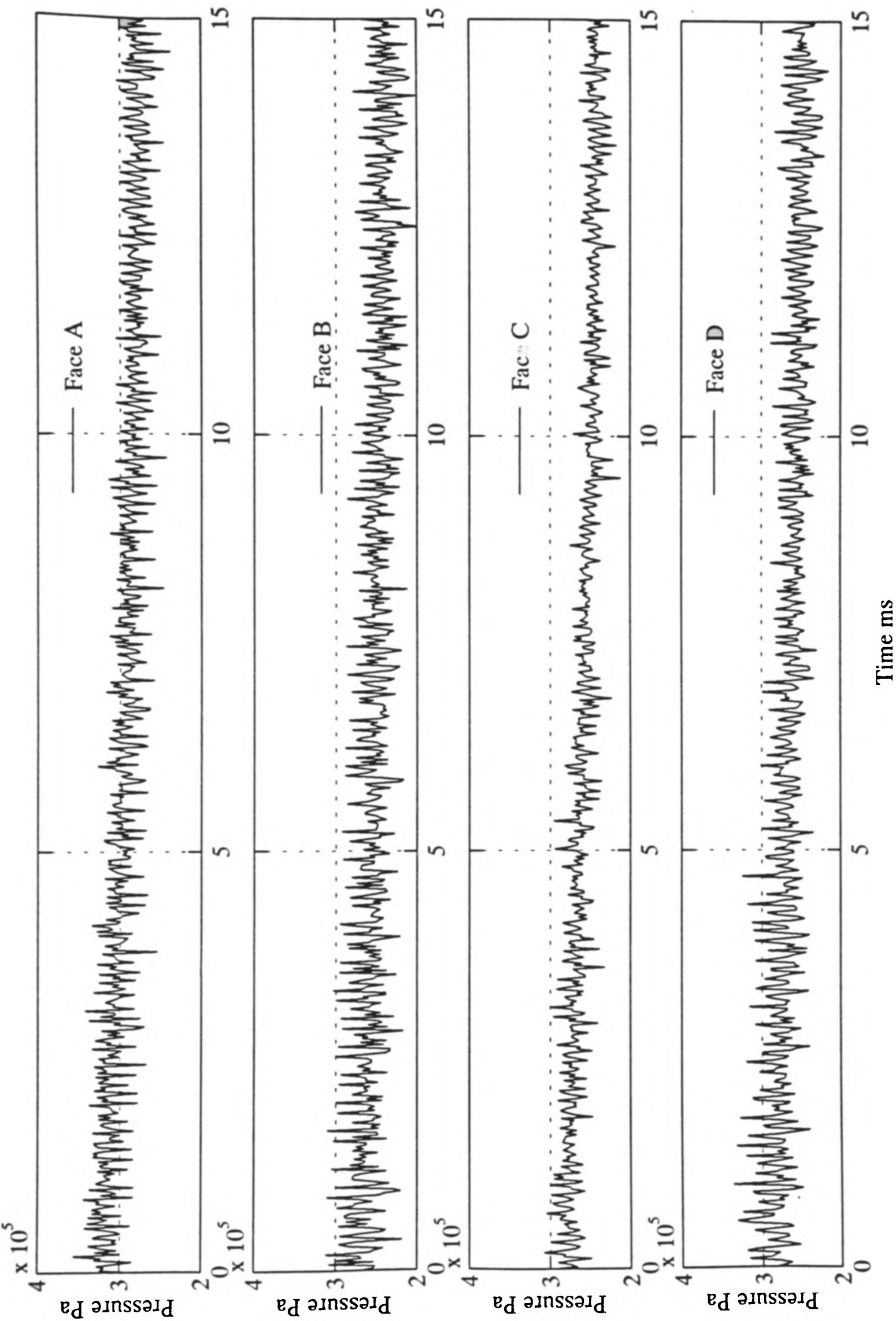


Figure 7.1 Unsteady pressures measured on probe faces

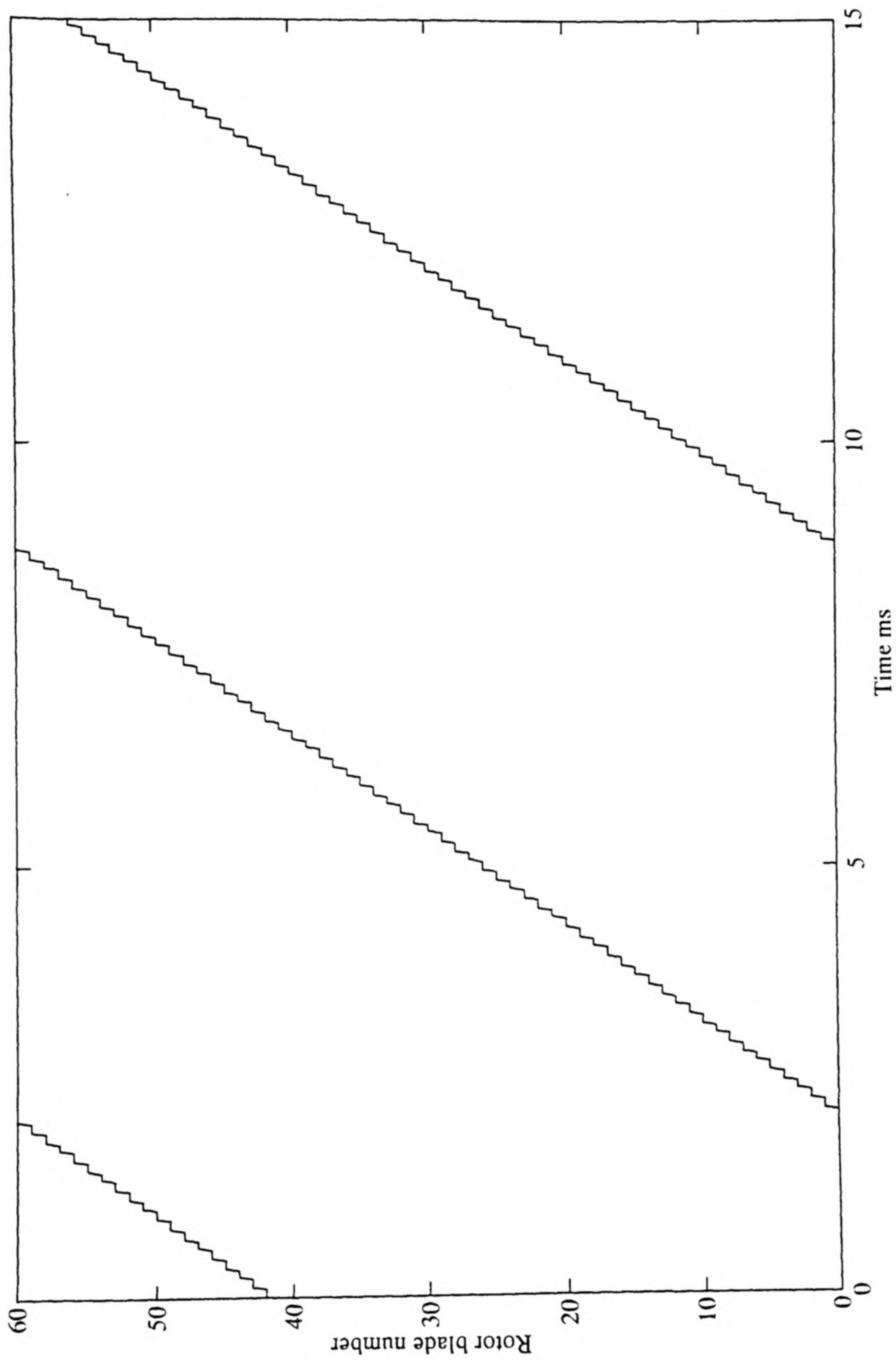


Figure 7.2 Rotor blade number versus time

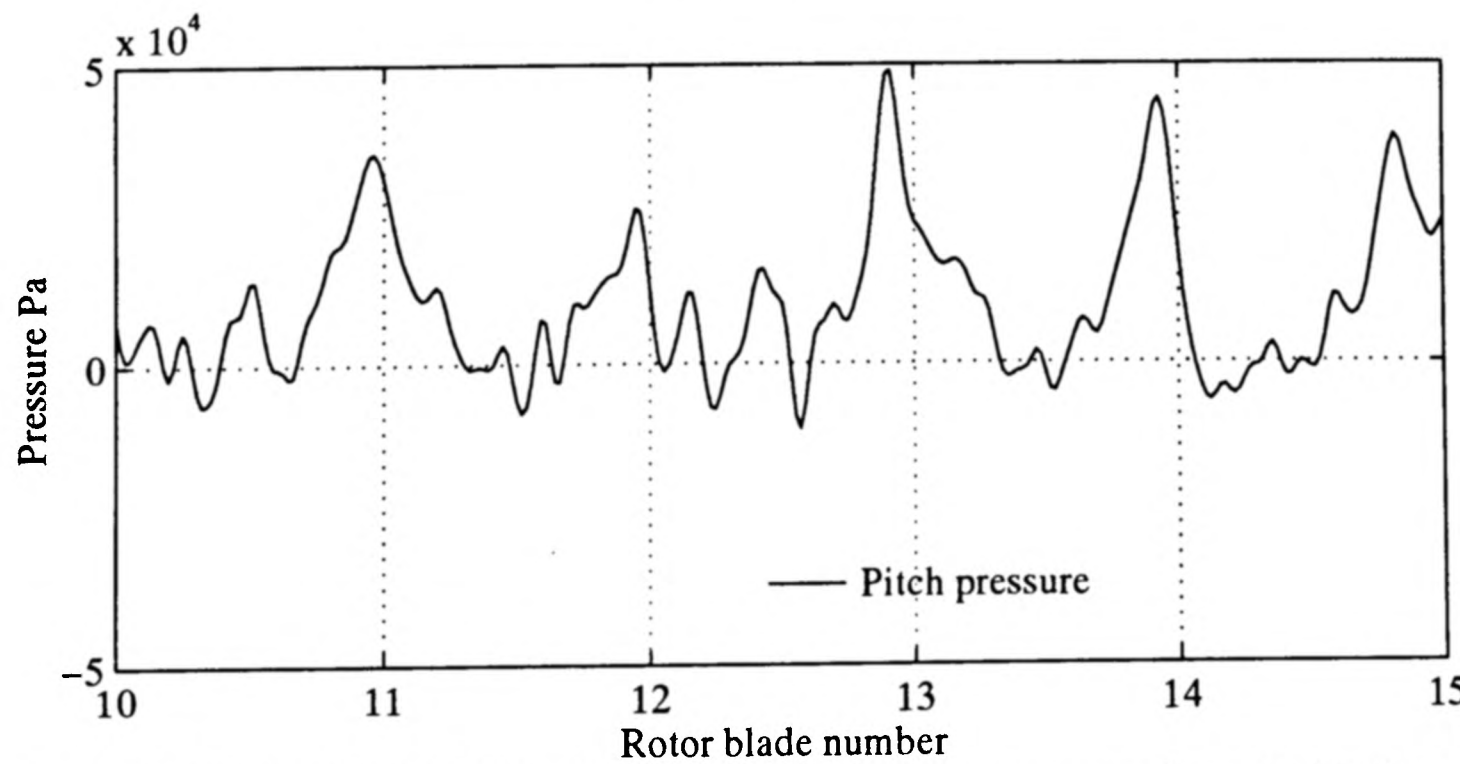
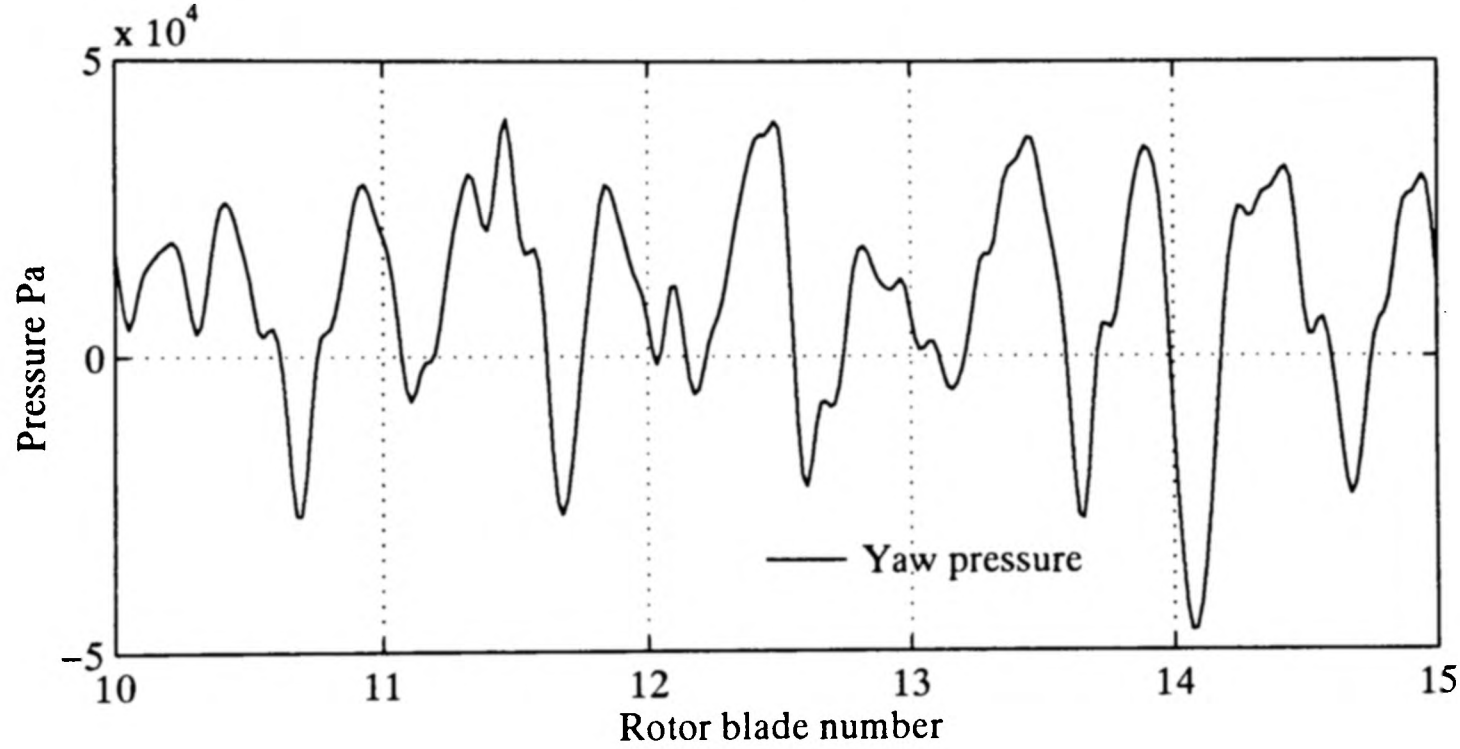
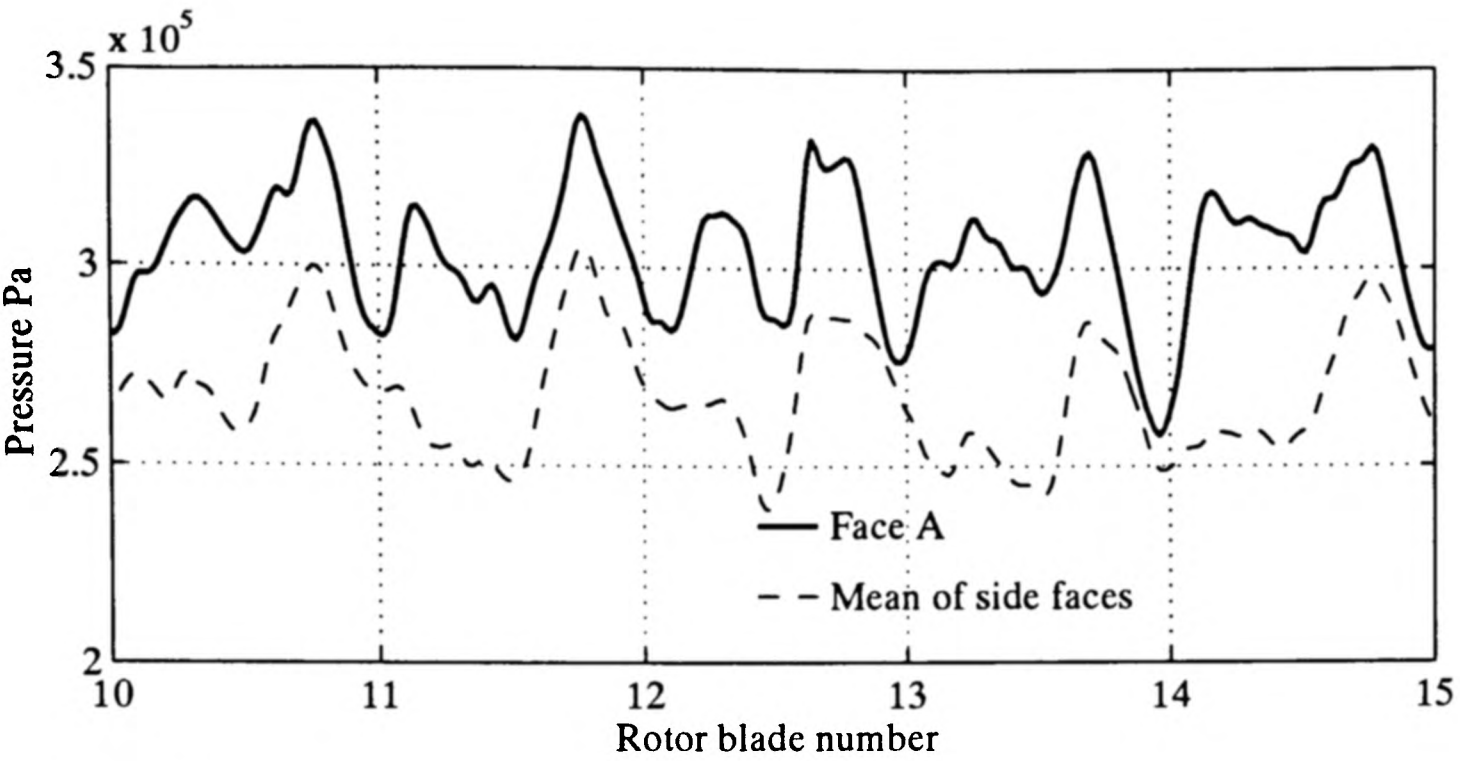


Figure 7.3 Front face, mean of the side faces, yaw and pitch pressures as five rotor blades pass the probe

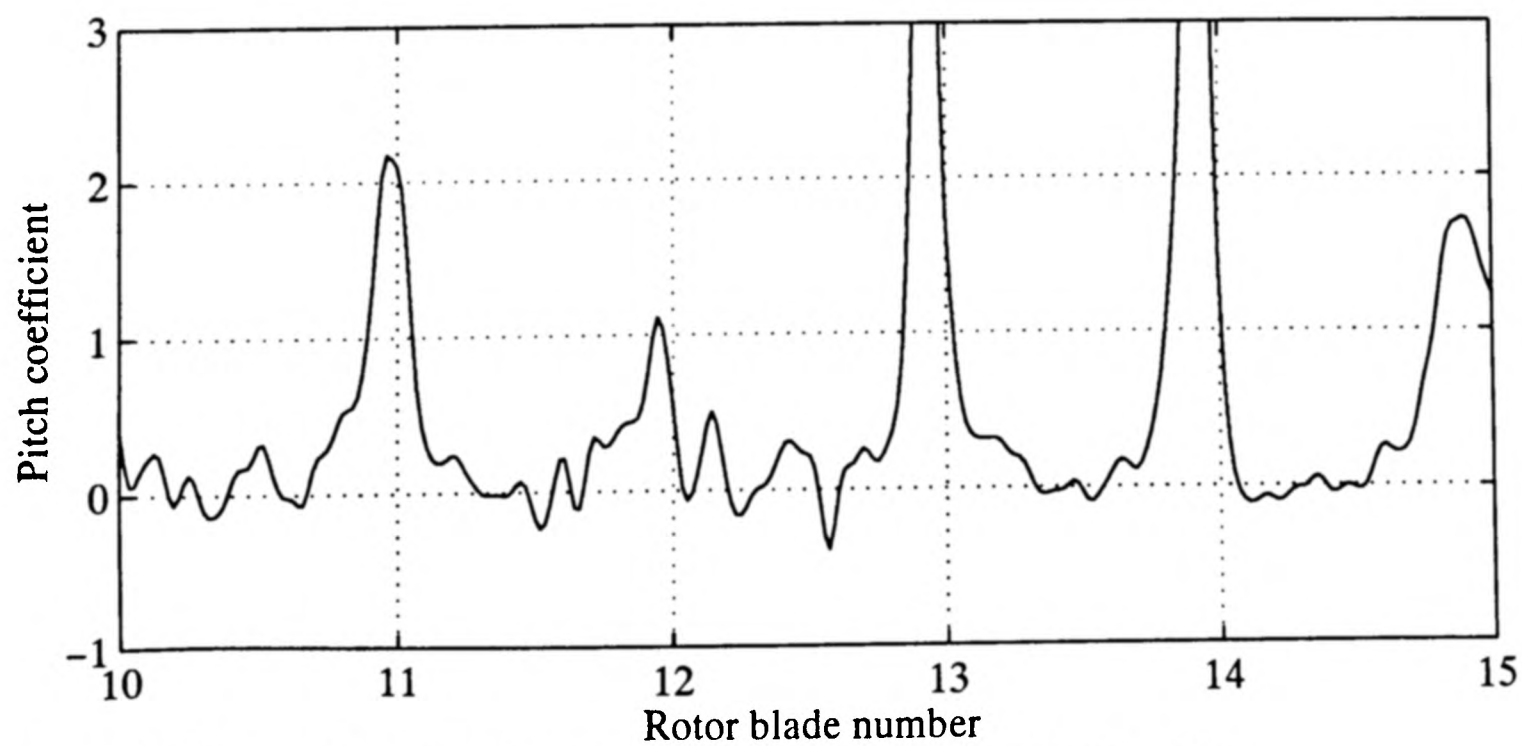
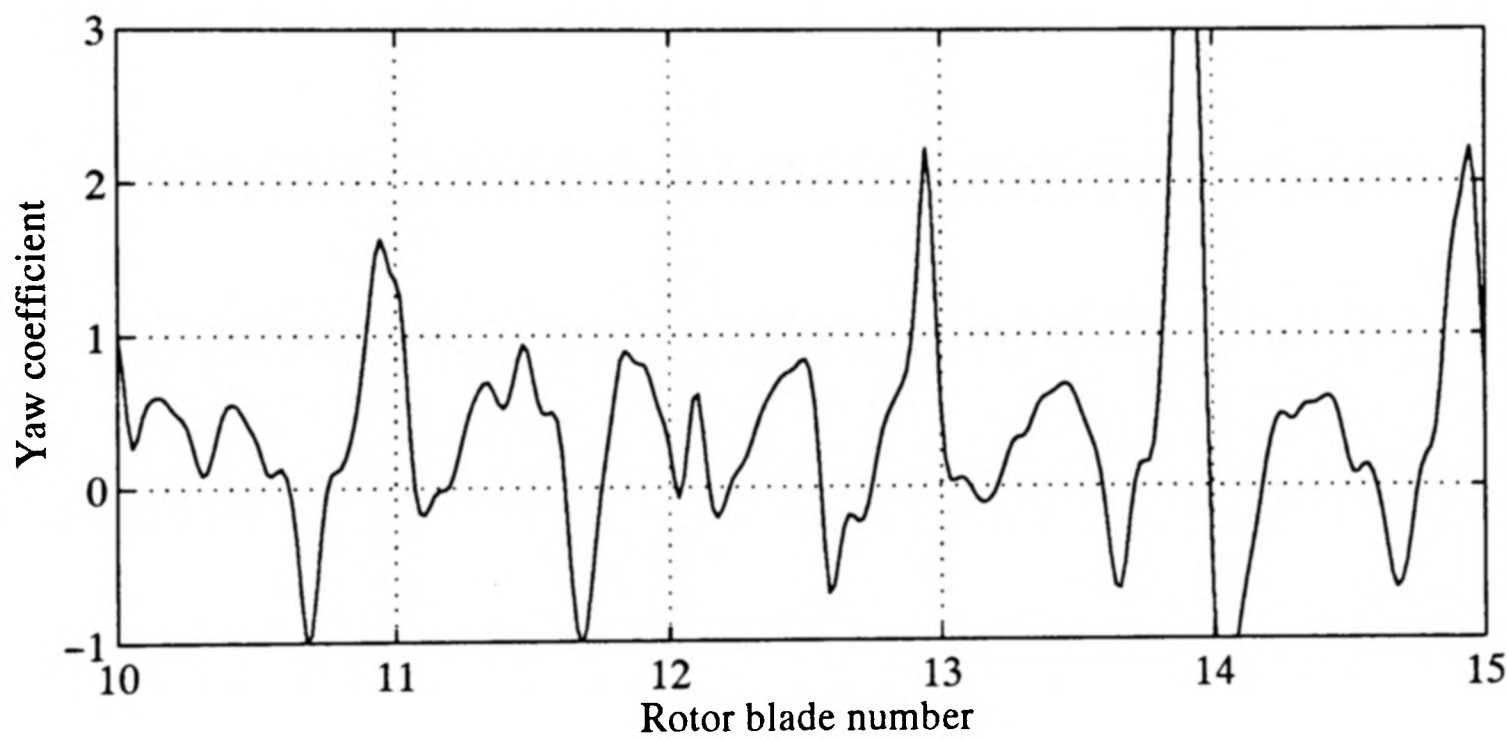
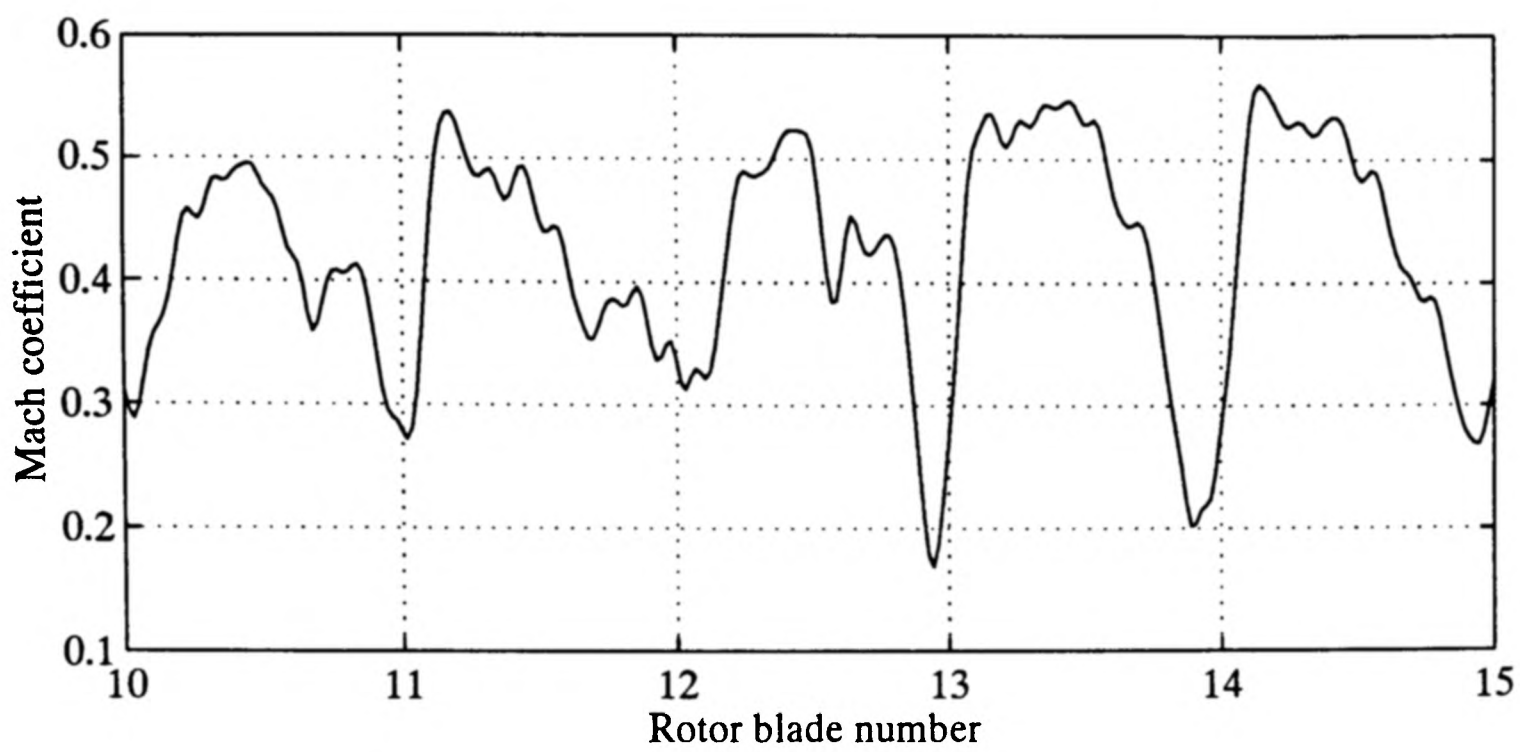


Figure 7.4 The non-dimensional coefficient as five rotor blades pass the probe

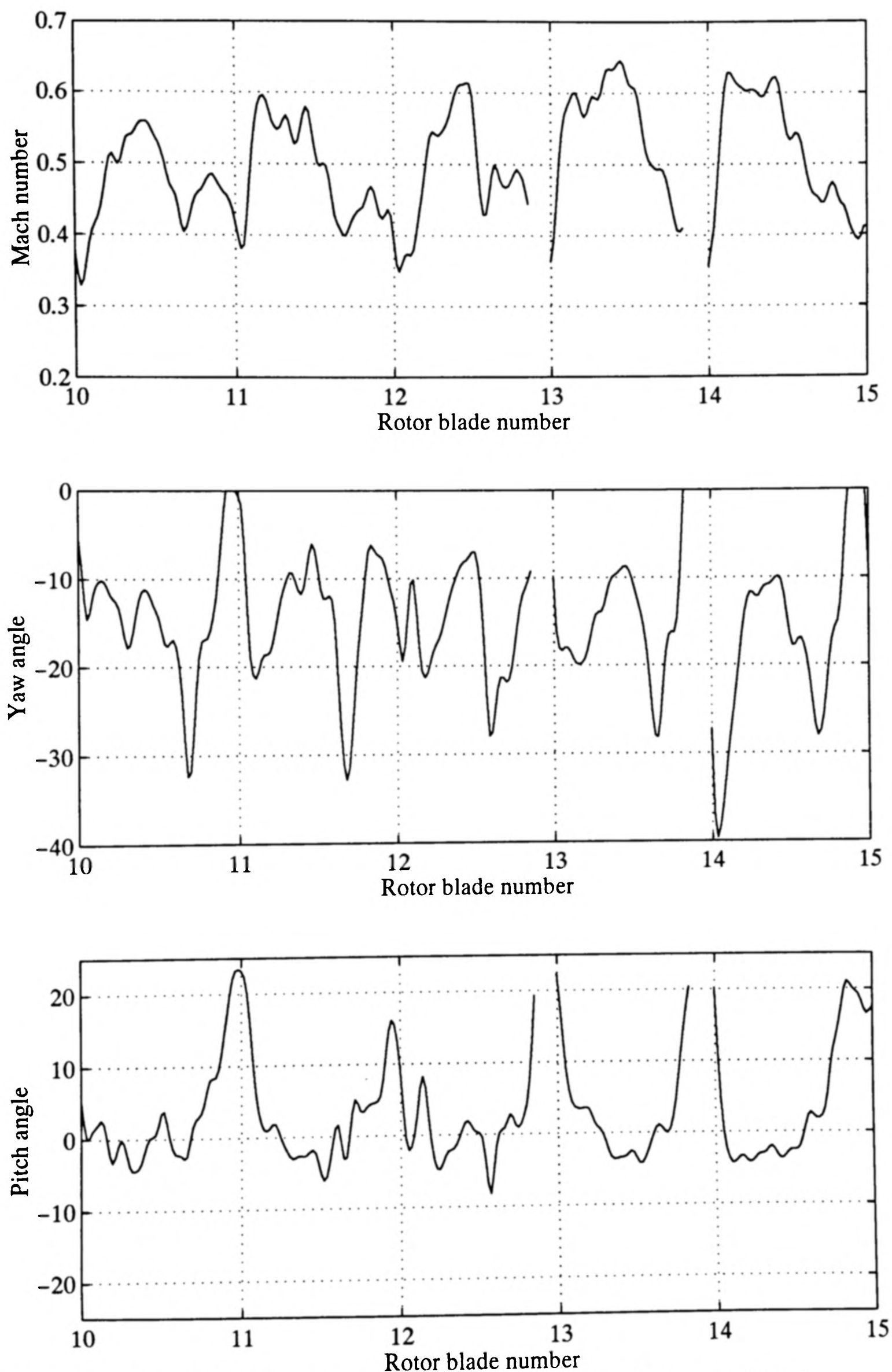


Figure 7.5 Derived Mach number, yaw and pitch angle as five rotor blades pass the probe

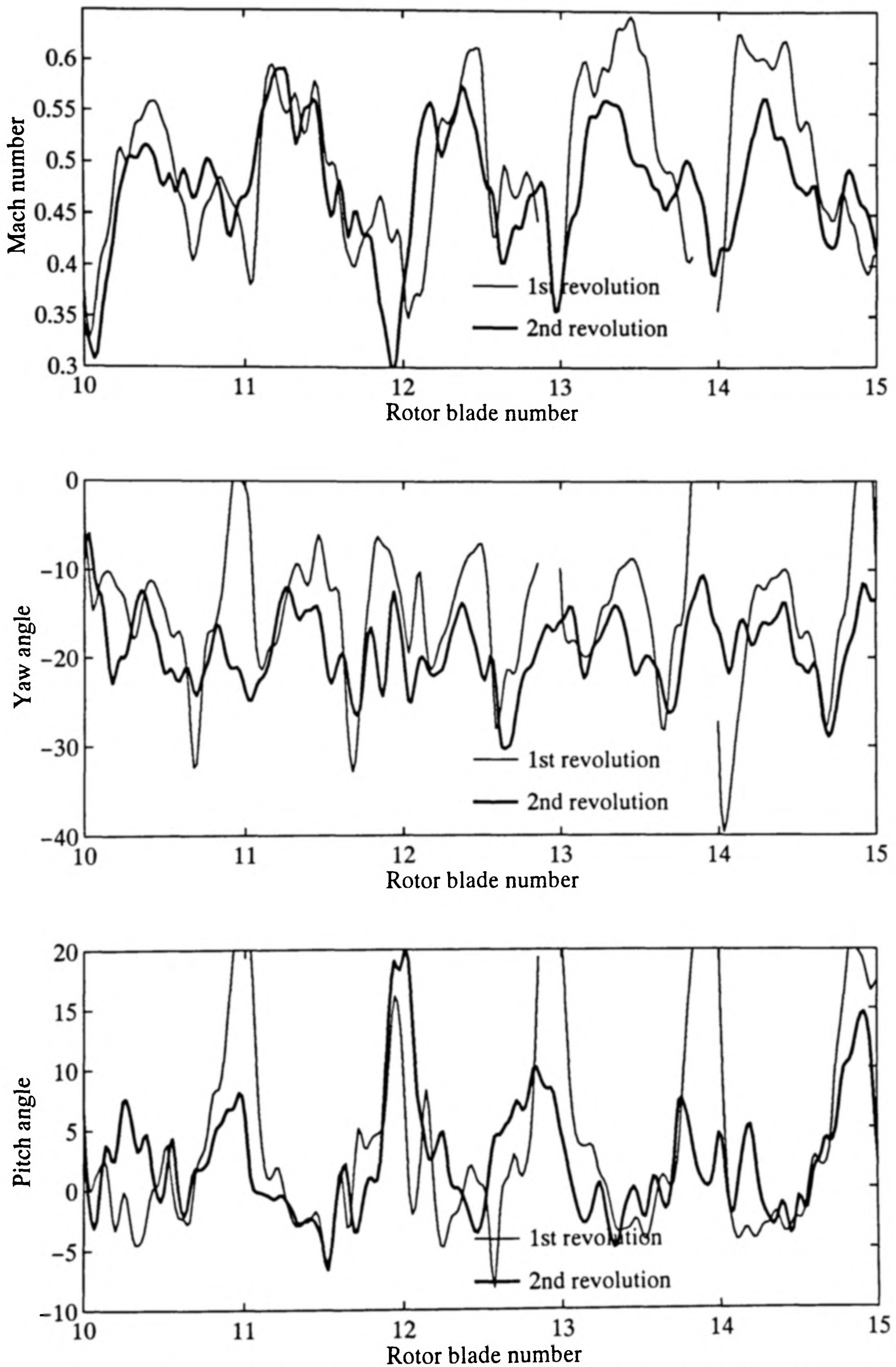


Figure 7.6 Comparison between the measurements on the first and second revolution of the rotor

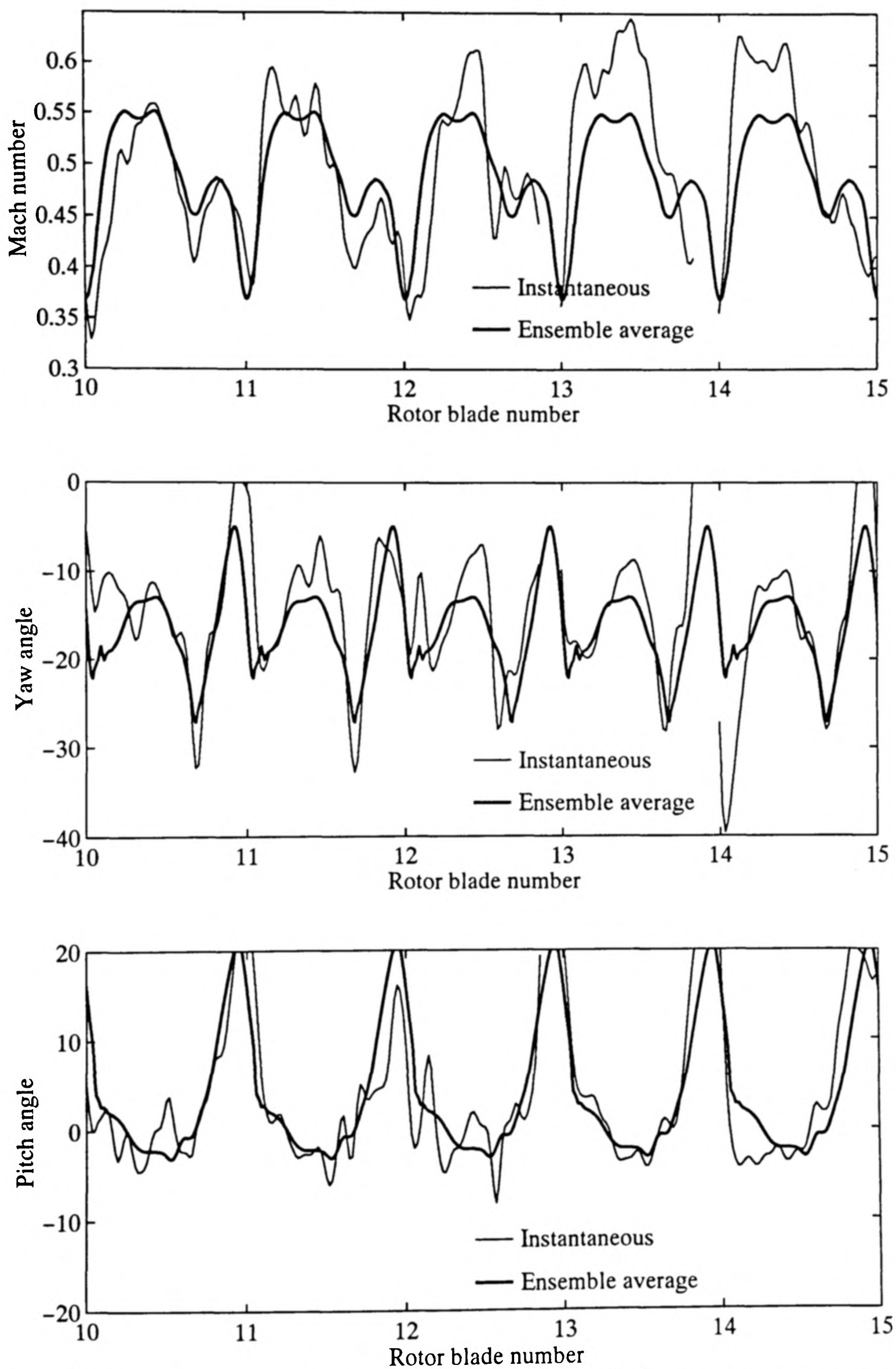


Figure 7.7 Comparison between the ensemble average values and the instantaneous measurement

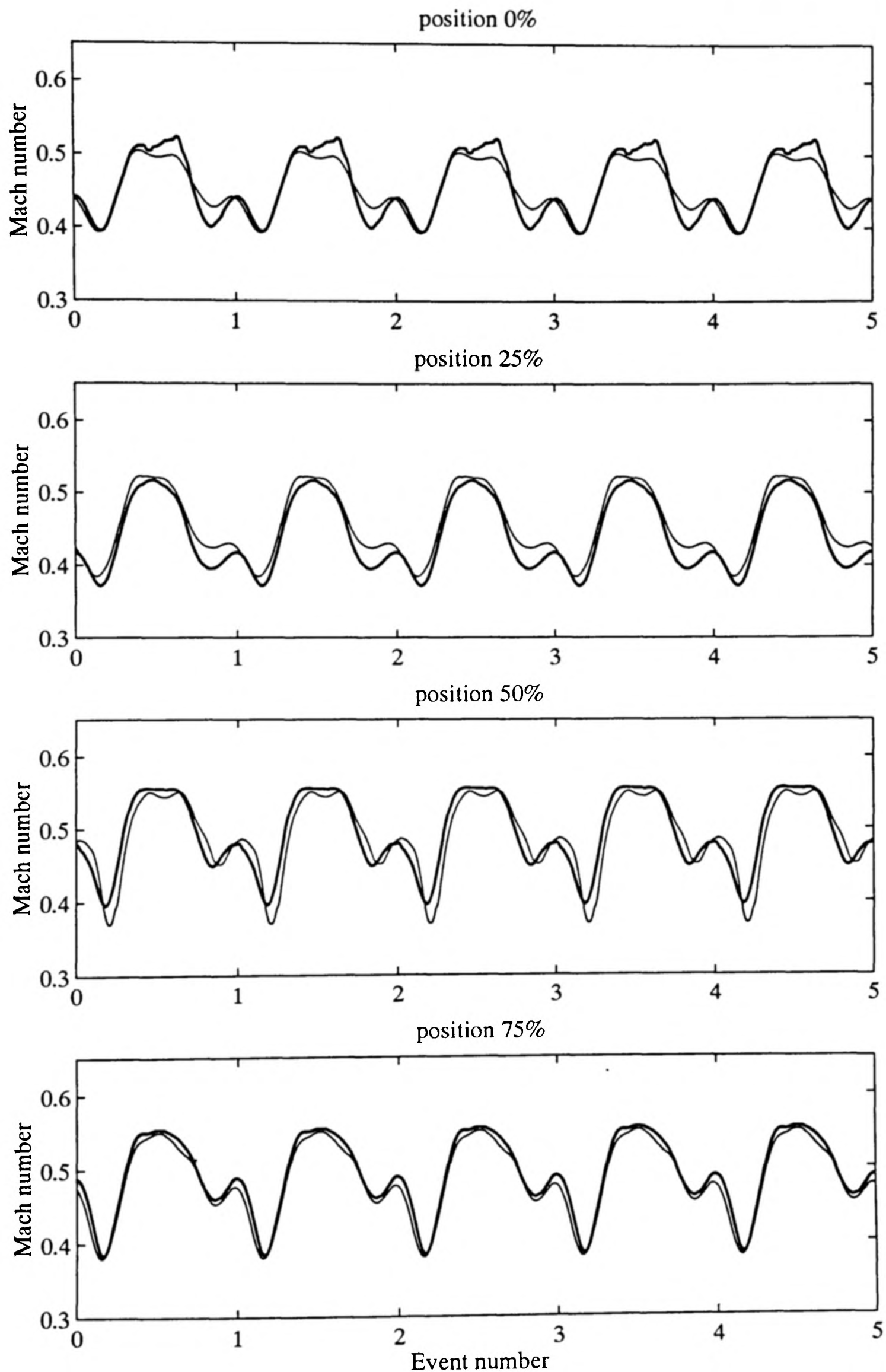


Figure 7.8 Variation in Mach number as circumferential position relative to upstream NGV is changed. Results plotted for two experiments to demonstrate repeatability.

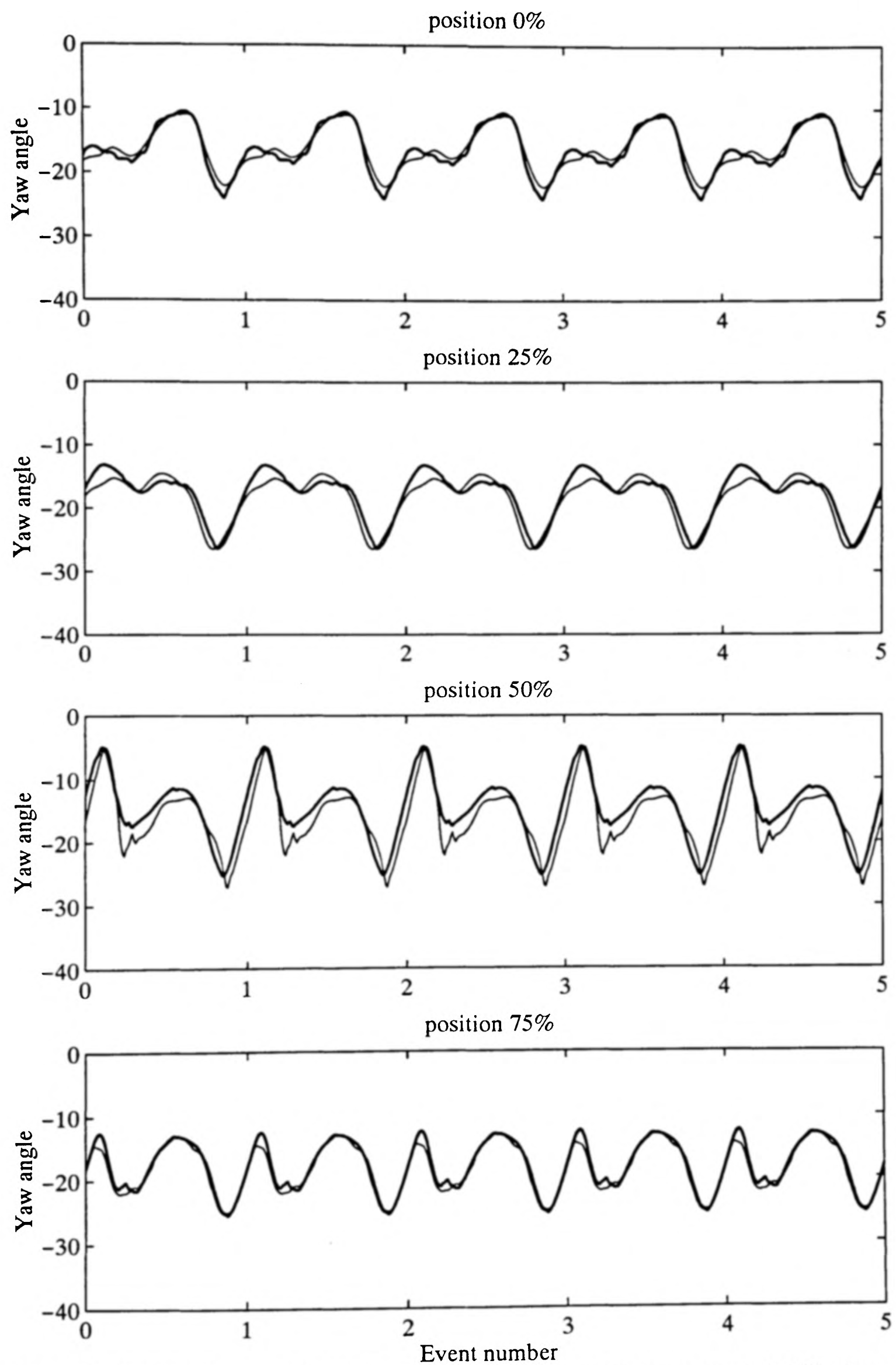


Figure 7.9 Variation in yaw angle as circumferential position relative to upstream NGV is changed. Results plotted for two experiments to demonstrate repeatability.

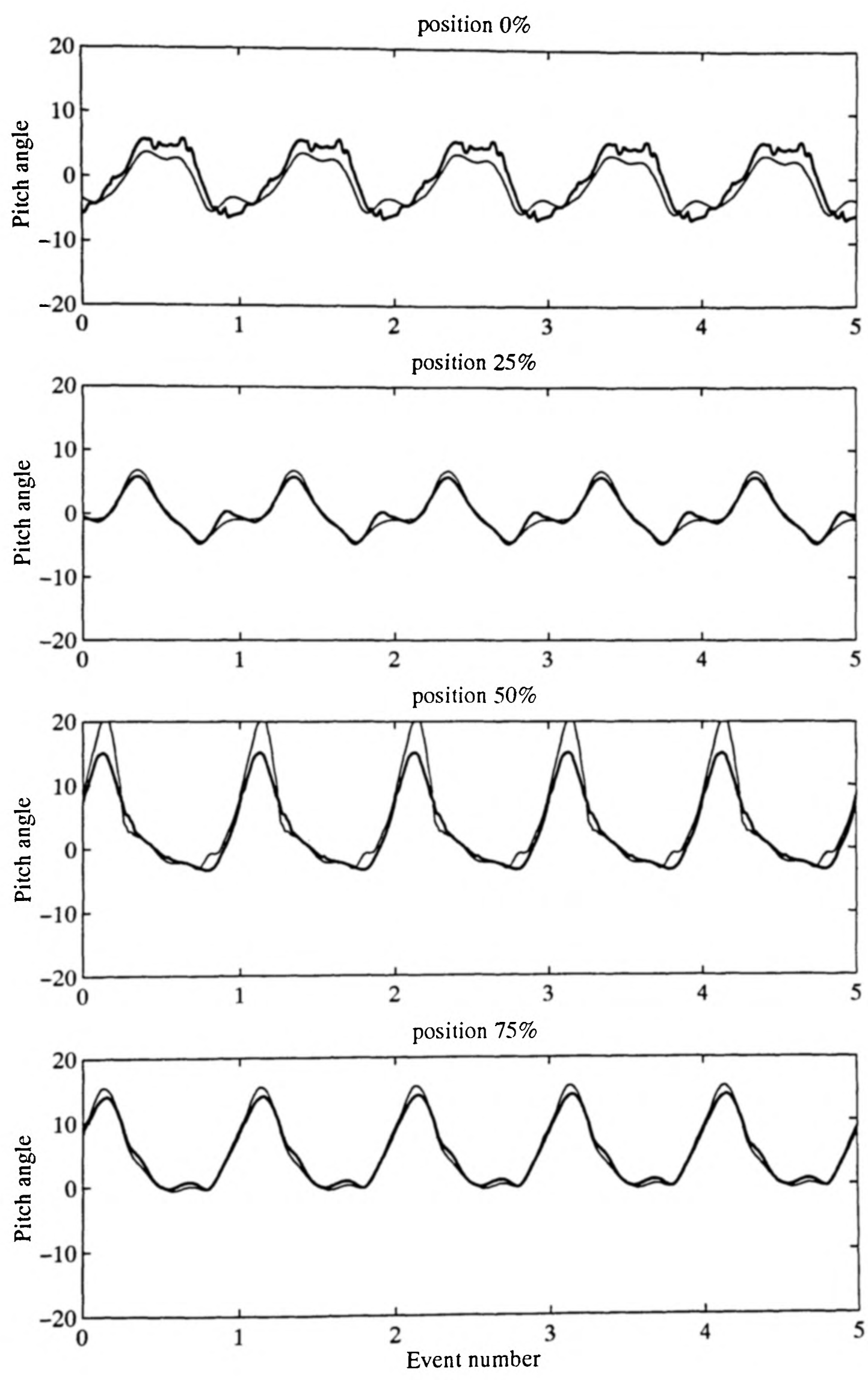


Figure 7.10 Variation in pitch angle as circumferential position relative to upstream NGV is changed. Results plotted for two experiments to demonstrate repeatability.

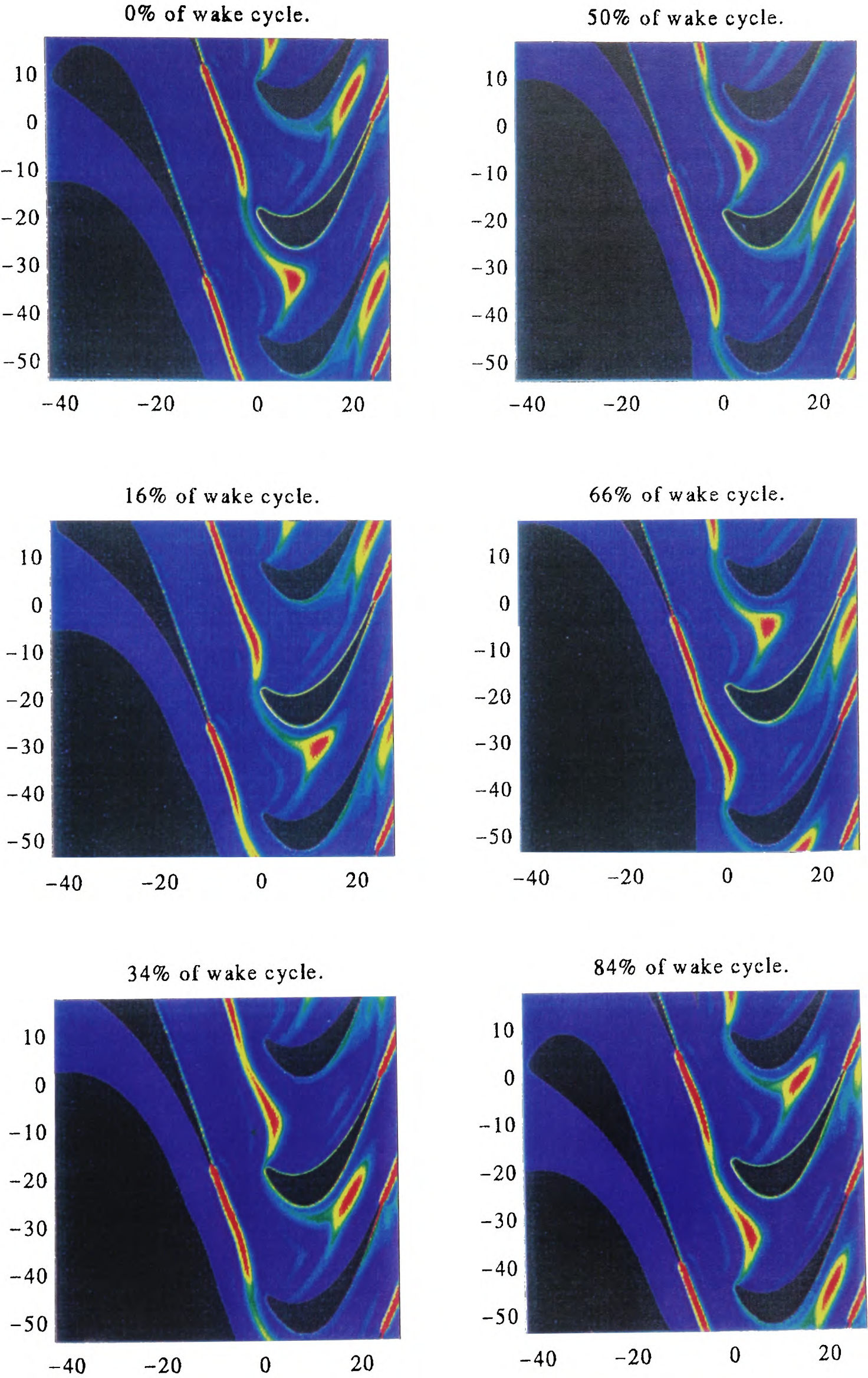


Figure 7.11 Entropy contours in the turbine stage

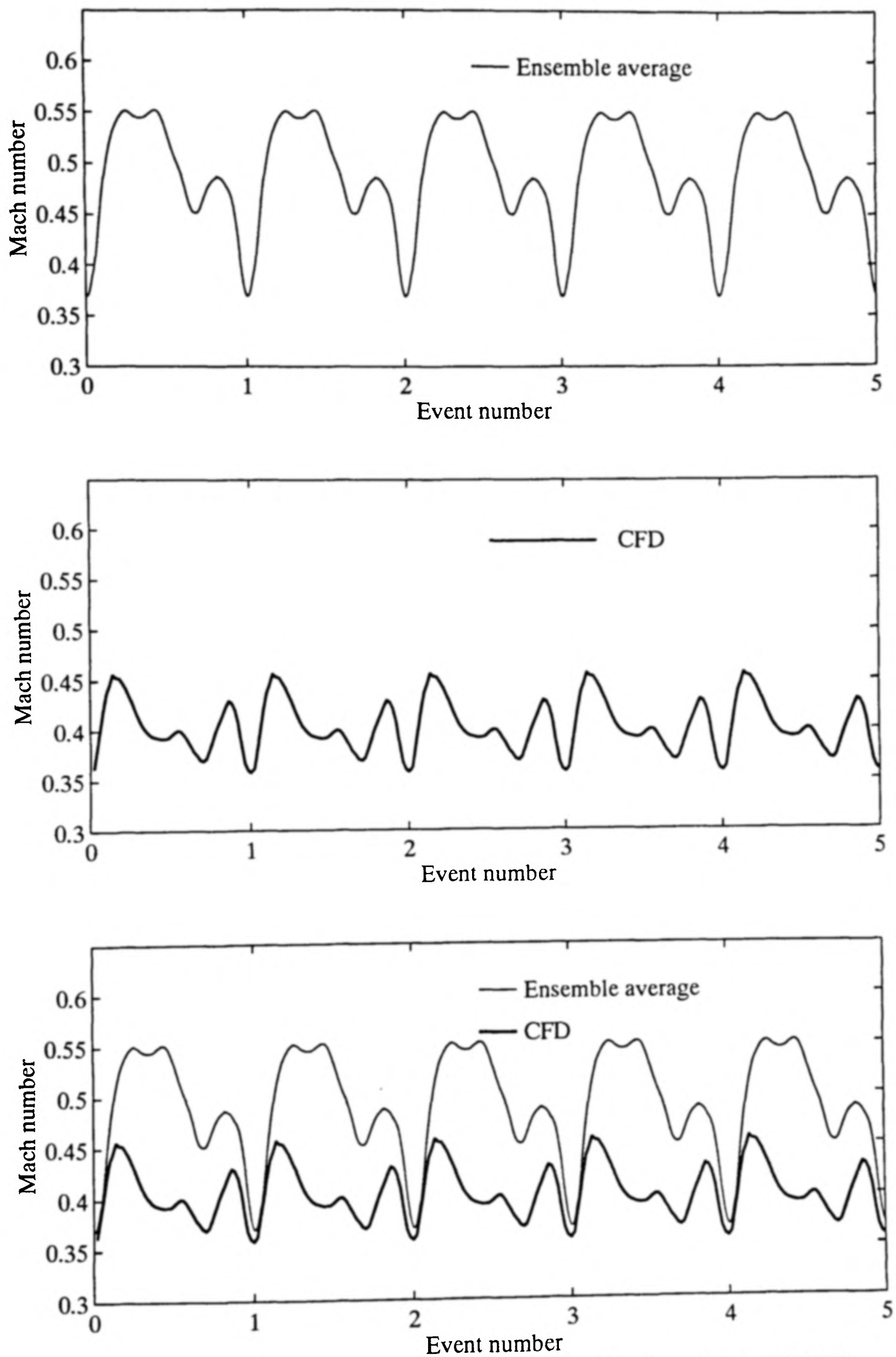


Figure 7.12 Comparison of the ensemble average and predicted Mach number at 50% NGV circumferential position

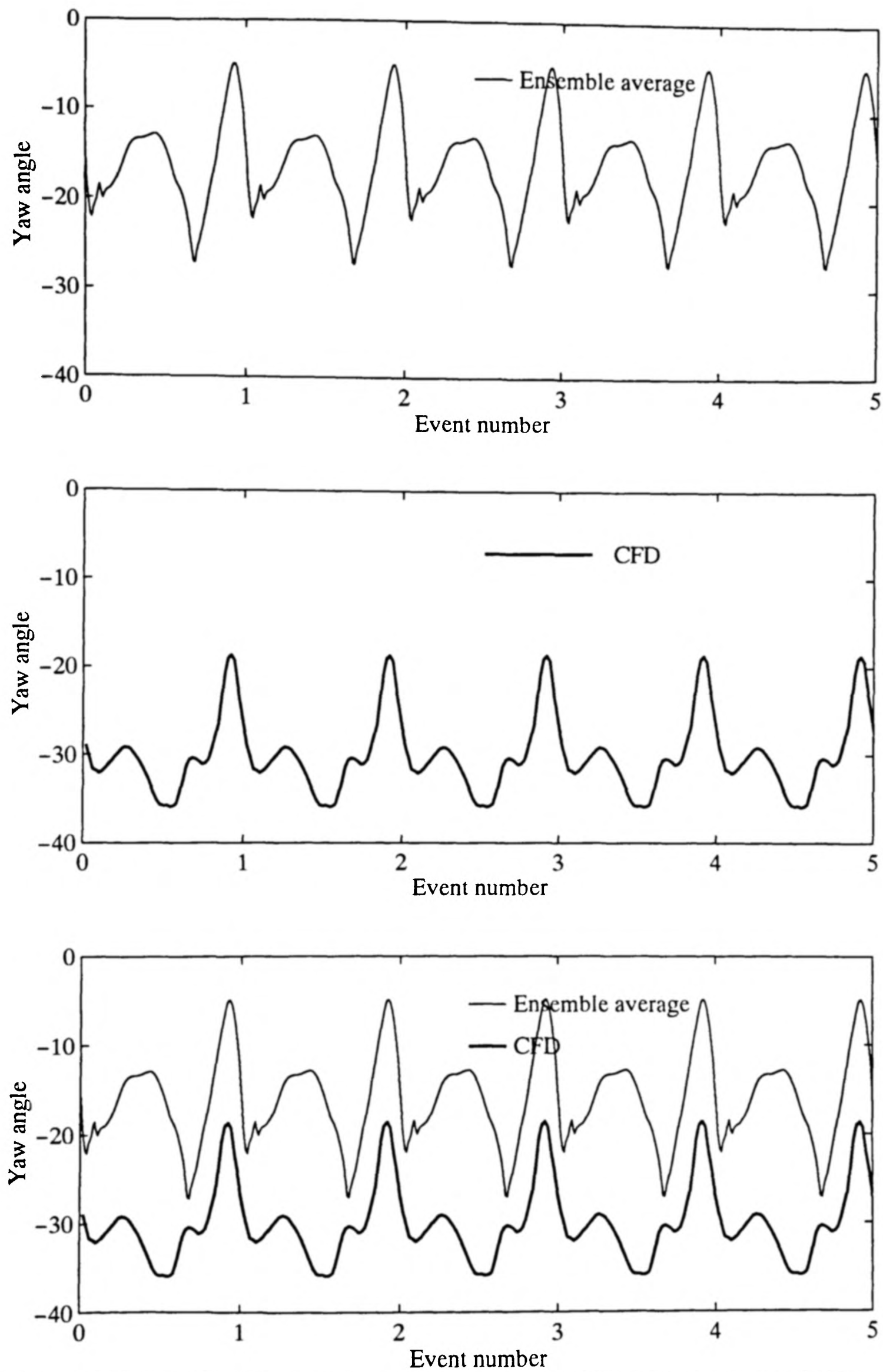


Figure 7.13 Comparison between the ensemble average and predicted yaw angle at 50% NGV circumferential position

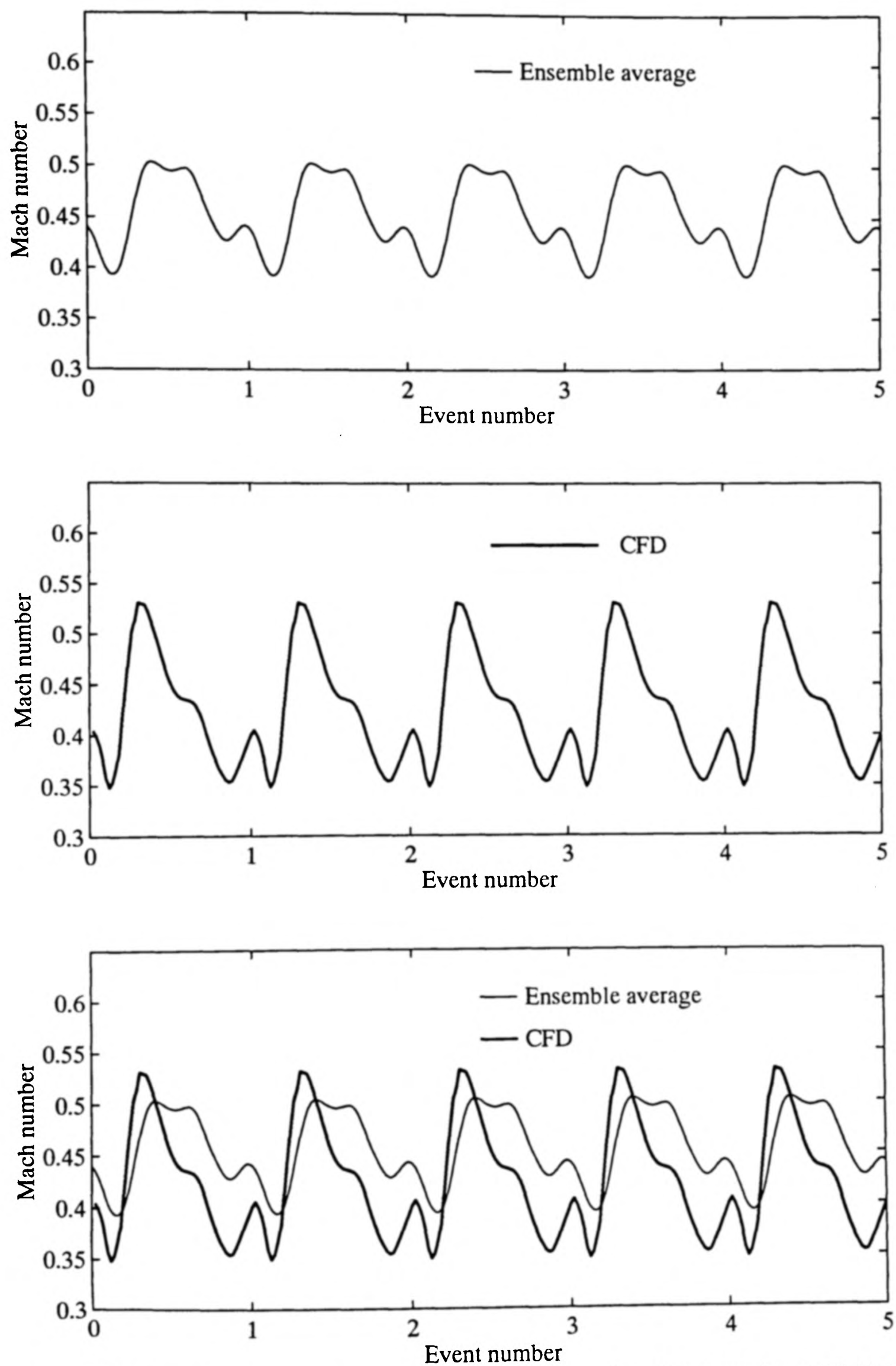


Figure 7.14 Comparison between the ensemble average and predicted Mach number at 0% NGV circumferential position

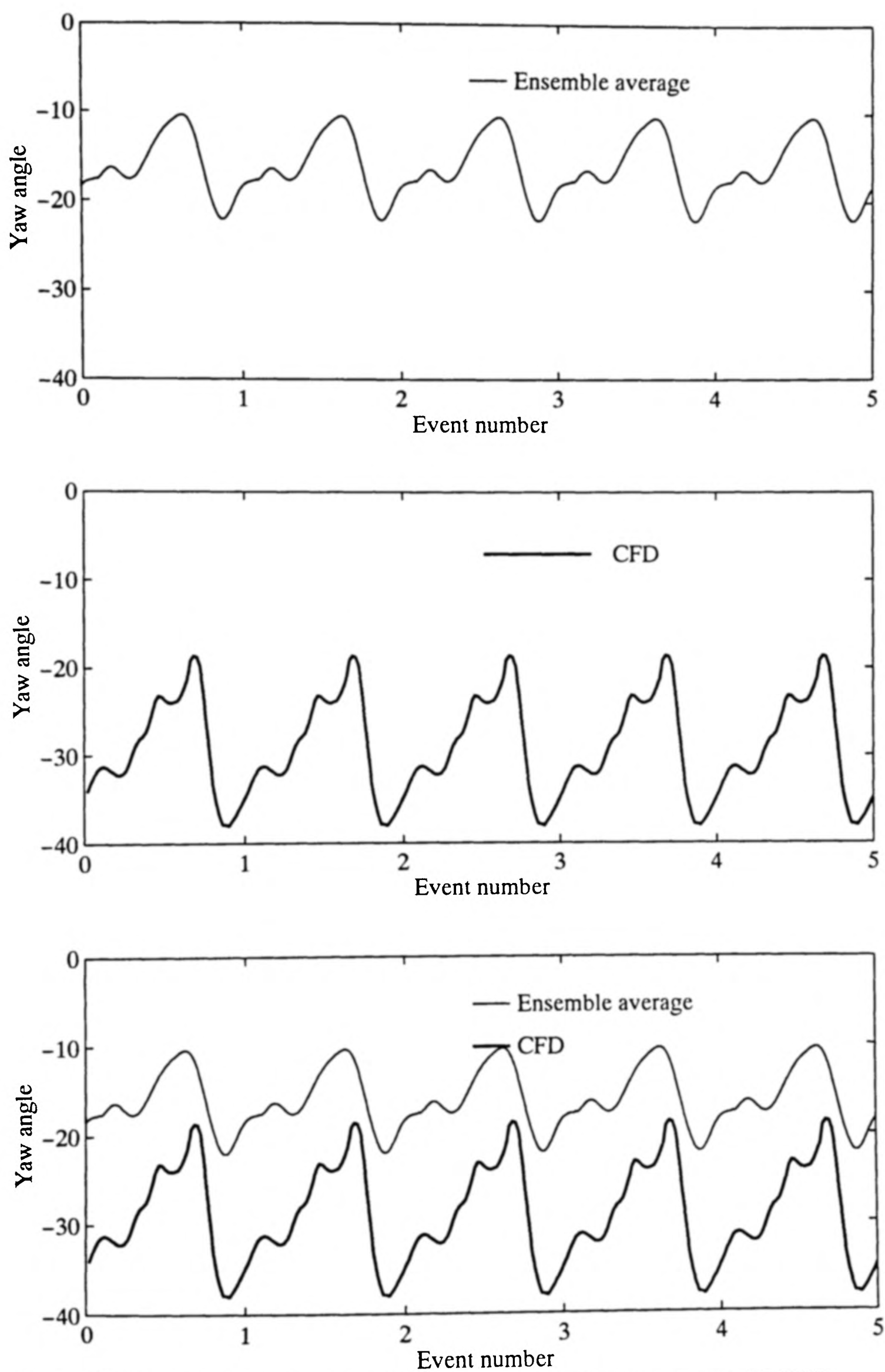


Figure 7.15 Comparison between ensemble average and predicted yaw angle at 0% NGV circumferential position

8. Summary and conclusions

8.0. Author's contribution

This work fits within a wider programme in which the objective is to increase the understanding of unsteady aerodynamics by providing a comprehensive data from a model gas turbine operating at engine conditions. The aim of the author's effort has been to develop a technique for the measurement of three-dimensional unsteady flows for use in turbomachines. The technique developed is based on semiconductor aerodynamic probes and these probes have been progressed to a sufficient level for them to be routinely used for research.

Research using this technique has continued in Oxford. It is currently being used on the Oxford Rotor experiment to measure the changes in performance of a turbine when an intermediate stage of stator blades is added downstream.

Demonstration of the technique's application has been shown through measurements of three-dimensional unsteady flow at the exit of a model high pressure gas turbine. It is believed to be the first time that a full three-dimensional calibration of a semiconductor probe followed by use in a high pressure turbine operating a engine conditions has been achieved.

It is very satisfying that as a direct result of this research, measurements based on similar aerodynamic probes to those presented here have been used for research outside the university. These probes were manufactured in Oxford and calibrated by the author.

8.1. The three-dimensional semiconductor probe

An aerodynamic probe was designed which incorporated four semiconductor pressure sensors, to allow wide bandwidth measurements. The probe was constructed with a head shaped in the form of truncated pyramid with four prismatic sides, to which the transducers were mounted using a novel surface mounting technique. This geometry allowed three-dimensional measurements to be recorded

with four pressure measurements. The result has been a semiconductor probe smaller than it could have been achieved if a conventional pneumatic five-holed geometry had been selected.

Semiconductor probes have been used for measurements in turbomachines before, but poor transducer steady state performance has restricted their application to unsteady measurements. Careful attention was placed on temperature compensation to the measured pressures both during calibration and use in the Oxford rotor. It was found that the combination of the technique used to surface mount the transducers and a robust temperature compensation system have allowed the transducer steady state performance issues to be avoided.

A method by which the pressures measured by the probe are converted into Mach number, yaw angle, pitch angle, and total pressure was developed. The probe was calibrated over a range of Mach numbers from 0.25-0.9 and over angles $\pm 28.8^\circ$ in yaw and $\pm 24.7^\circ$ in pitch in a dedicated calibration facility. The results of an aerodynamic calibration were built up piecemeal over a series of experiments. These results are then inverted into a 'look up' table using a technique called sequential interpolation. This inversion allows measurements to be derived directly from the pressures from the probe. The sequential interpolation technique has provided a fast and efficient method for conversion of the raw pressures to the measurements of interest.

An investigation into the errors associated with this process has shown that the operating range of the probe has a significant influence on the order of the error. Certain regions in the probe's operating envelope have been found to be more susceptible to error than others. In general the more extreme the incidence to the probe the higher the measurement error can be expected to be. It is clear, however that a more significant influence on the measurement accuracy of the probe than the magnitude of the inversion error is in fact a result of the absolute pressure accuracy of the transducers. This is particularly true at the lower Mach numbers where the pressure differences between each face are much lower in absolute terms.

A number of experiments on large scale models of aerodynamic probes were used to explain the aerodynamic behaviour of the probes. An apparent 120° division of the probe's sensitivity to direction was noted and attributed to the 120° symmetry of the probe head. The large scale tests highlighted separation bubbles over the prismatic faces which dominate the aerodynamic behaviour of the probe. These separations acted to increase the sensitivity to flow direction of pressure tapings located beneath the separation bubble. Pressure tapings located rearward of the separation bubble were found to have lower direction sensitivity because of the deflection of the main flow around the separation bubble.

8.2. *Experimental measurements*

The Oxford rotor experiment provided the ideal opportunity to test the measurement technique's integrity and robustness. The semiconductor probe mounted at the rotor exit served the dual function of measuring the run conditions over the whole rotor experiment and recording unsteady wake measurements when the turbine is at design condition.

Comparison of the Mach number, static and total pressure measured from the probe were made against independent rig instrumentation and were found to be in good general agreement. These measurements, together with the yaw and pitch angle for the flow, were also found to be in agreement with Rolls-Royce's design code Q263. The steady state measurements demonstrate that the semiconductor probe can record reliable steady state measurements. Hitherto it has been considered that semiconductor probes are unsuitable for steady state measurements.

Unsteady measurements from the probe indicated that the rotor exit flow is dominated by wakes from the rotating turbine trailing edge. The wakes result in a sharp drop in the Mach number and a change in the yaw and pitch angle. The position of the probe was moved relative to the upstream NGV row and the relative position was found to influence the structure of the rotor exit flow.

Visualisation of entropy generation by the unsteady viscous flow solver Unsflo suggested that the upstream NGV wake propagates through the turbine and interacts with the rotor wake. The relative

timing of the NGV and rotor wake determines how the two wakes will behave. This mechanism of the two wake sources interacting is consistent with differing flow structures at different positions relative to the upstream NGV.

Comparison of the prediction from Unsflo provides one of the few sources of data, albeit two-dimensional, with which the three-dimensional probe measurements can be compared. The similarity of the structure of the flow was found to be most encouraging. The absolute levels appeared to differ, but this may be a result of subtle differences in the design speeds of the prediction and the experiment.

It is reasonable to conclude from the success of these experiments that three-dimensional semiconductor probes of this type will be a valuable tool for future turbomachinery research.

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