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This report covers research performed under Grant AFOSR-85-0288. The work consisted of ten separate projects: 1) Turbomachinery Noise Reduction Through Distortion Amelioration, 2) Fluid Mechanics of Discrete Passage Diffusers, 3) Compressor Stabilization Through Structural Feedback, 4) Linearized Euler Solutions for Turbomachinery Flutter and Forced Response; 5) Vortical Flow Behavior Downstream of Convoluted Trailing Edges; 6) Turbine Blade Tip Forces; 7) Influence of Inlet Radial Temperature Distribution on Turbine Rotor Heat Transfer, 8) Effect of Circumferential Inlet Distortion on the Formation of Turbine Secondary Flow Vortices, 9) Inlet Distortion with Application to STOVL, 10) Instability and Distortion in Multistage Compressors. These were carried out as part of the Air Force Research in Aero-Propulsion Technology (AFRAPT) Program.

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PERIOD OF INVESTIGATION: September 1, 1989 - August 31, 1990

September 1990

Introduction

This report describes progress on the Air Force Research in Aero Propulsion Technology (AFRAPT) Program. The report covers the period September 1, 1989 to August 31, 1990 and consists of short descriptions of the research of each student during this period. The students, their advisors, and the research projects are as follows:

Trainee:	Scott Barton
Advisor:	Professor A.H. Epstein
Project:	Turbomachinery Noise Reduction Through Distortion Amelioration
Trainee:	Victor Filipenco
Advisor:	Professor E.M. Greitzer
Project:	Fluid Mechanics of Discrete Passage Diffusers
Trainee:	Daniel Gysling (6/1/90 - 8/31/90)
Advisor:	Professor J. Dugundji/Professor E.M. Greitzer
Project:	Compressor Stabilization Through Structural Feedback
Trainee:	Dana Lindquist
Advisor:	Professor M.B. Giles
Project:	Linearized Euler Solutions for Turbomachinery Flutter and Forced Response
Trainee:	Ted Manning
Advisor:	Dr. C.S. Tan/Professor E.M. Greitzer
Project:	Vortical Flow Behavior Downstream of Convoluted Trailing Edges
Trainee:	Knox Millsaps
Advisor:	Professor M. Martinez-Sanchez
Project:	Turbine Blade Tip Forces
Trainee:	George Pappas
Advisor:	Professor A.H. Epstein
Project:	Influence of Inlet Radial Temperature Distribution on Turbine Rotor Heat Transfer
Trainee:	Earl Renaud
Advisor:	Dr. C.S. Tan
Project:	Effect of Circumferential Inlet Distortion on the Formation of Turbine Secondary Flow Vortices
Trainee:	Eric Strang
Advisor:	Dr. C.S. Tan/Professor E.M. Greitzer
Project:	Inlet Distortion with Application to STOVL
Trainee:	Pete Silkowski
Advisor:	Dr. C.S. Tan/Professor E.M. Greitzer
Project:	Instability and Distortion in Multistage Compressors

Description of the Research

Brief descriptions of the different research projects are given below; detailed descriptions will be given in the M.S. and Ph.D. theses of the graduate student trainees.

Turbomachinery Noise Reduction Through Distortion Amelioration

This project is aimed at determining the effect of distorted inlet velocity profiles on the acoustics of turbomachines. As a first step in this, a pump loop has been designed and built to measure the acoustic output of a pump subject to different inlet profiles.

The loop consists of the pump, four lengths of 4" diameter rubber hose, one 600 gallon tank, one length of 1.5" diameter throttle hose, a magnetic flow meter, an 11 gallon expansion tank, and various test sections and inlet fittings (see Fig. 1). The pump is of the type used in nuclear submarines and was designed for minimal mechanical noise, so that the strongest components of its acoustic spectrum are flow related. The rubber hose is used to attenuate the pump signal far upstream and downstream, thereby preventing interference due to any resonance effects, where sound waves are able to travel around the entire loop. The 1.5" hose is used in place of a conventional throttle, in order to drop pressure without introducing extraneous noise into the loop.

Three 4" diameter stainless steel test sections have been designed and built on which to mount hydrophones and Pitot-static probes. One section, 8 ft in length, is currently mounted on the pump inlet, for measurement of inlet acoustic intensity as well as to detect the presence of a standing wave in the test section, using the two-microphone method. This measurement is important because the impedance mismatch at the steel pipe-rubber hose boundary causes acoustic reflection, and the resulting standing wave must be taken into account in measuring the acoustic spectrum. Other test sections are 28" and 8" long and are used for acoustic measurements at the pump outlet, as well as far upstream in order to estimate attenuation due to the hose.

At present, the loop is capable of producing two inlet profiles: 1) quasi-fully developed flow, obtained by attaching the 8' test section directly to the pump inlet, and 2) secondary flow from a 90° elbow connected directly to the inlet. Future plans call for a flow conditioner to create a "top hat" or uniform velocity inlet profile.

Acoustic spectra taken thus far match general expectations, with the most prominent peaks appearing at the operating speed and blade passing frequencies. The accuracy of these spectra is currently being confirmed by comparison to existing pump data.

Once the current shakedown phase is complete, the next step is to make a simple comparison between the acoustic spectra of the two available configurations, fully developed inlet flow and asymmetric inlet flow due to the 90° bend. In addition, other profiles will also be tested, including a uniform velocity, "top hat" profile to assess whether inlet distortion does indeed lead to significantly higher intensity at the expected frequencies.

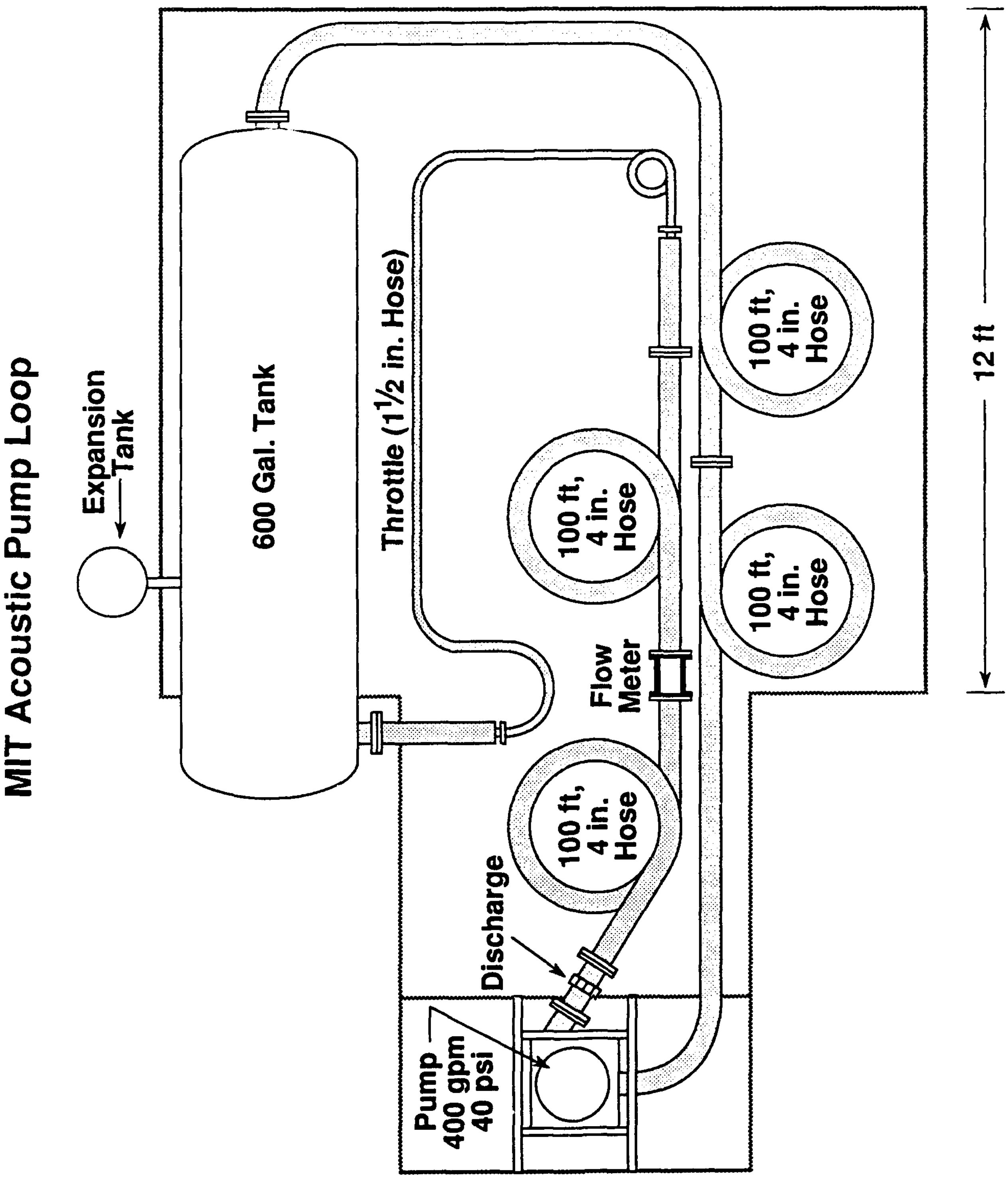

MIT Acoustic Pump Loop
Expansion Tank
600 Gal. Tank
Throttle (1½ in. Hose)
100 ft, 4 in. Hose
100 ft, 4 in. Hose
100 ft, 4 in. Hose
100 ft, 4 in. Hose
Flow Meter
Discharge
Pump 400 gpm 40 psi
12 ft

Figure 1

Fluid Mechanics of Discrete Passage Diffusers

The radial diffuser test rig became operational in Fall 1989, after almost five years of design, construction, and debugging. Over the last twelve months, the main accomplishments have been:

1) Verification of the performance of the diffuser test rig using a 1.2 radius ratio vaneless diffuser. This included:
   a) Pressure rise versus flow constant speed characteristics, at rotor speeds from 0 to 6200 rpm.
   b) Establishment of the influence of diffuser inlet flow conditions (suction and injection through the annular slots upstream and downstream of the rotor) on the diffuser inlet velocity profile.
   c) Determination of the rotor blade wake strength and turbulence level at a radius corresponding to the test diffuser inlet, over a wide range of operating conditions.
   d) Verification of circumferential uniformity of the flow at the diffuser inlet.
2) Installation of the General Electric 30 passage diffuser configuration. This included three high frequency response pressure transducers (Kulites) in the vaneless space between the rotor and diffuser inlet, with appropriate additions to the data acquisition system hardware/software.
3) Obtaining a comprehensive diffuser performance/stability data over a range of diffuser inlet conditions.

Diffuser pressure recovery was measured over a range of diffuser inlet Mach numbers and swirl angles, from choke to stall, with the inlet flow field as uniform as possible. These tests were then repeated with axial distortion of the inlet velocity profile. The distortion was created by means of a cross-flow (relative to the main flow) introduced through an annular slot in the vaneless space between the rotor exit and the diffuser inlet.

The onset of unstable operation was determined by monitoring the output of the Kulite transducers. Except for several special situations in which a system instability (surge) was encountered, it was clearly established that rotating stall was the form of instability.

4) Analysis of data, including definition of suitable parameters to quantify the diffuser inlet flow field distortion and of "average" parameters in terms of which the diffuser performance could be correlated. It was found that the diffuser pressure recovery is not sensitive to distortion of the inlet flow field and that the diffuser stalls at a critical "momentum averaged" flow angle and corresponding value of pressure recovery.

The latter point is demonstrated in Fig. 2, in which the overall diffuser pressure recovery coefficient (based on the above-mentioned considerations) is plotted, for the high and low diffuser inlet distortion cases, as a function of the diffuser inlet momentum averaged flow angle. The circled points correspond to the rotating stall threshold. As can be seen, stall occurs within an angle range of less than one degree, independent of profile distortion.

A simple analytical consideration of the effect of mixing on diffuser performance has also been carried out. This provides a qualitative explanation of the insensitivity of the diffuser performance to inlet distortion.

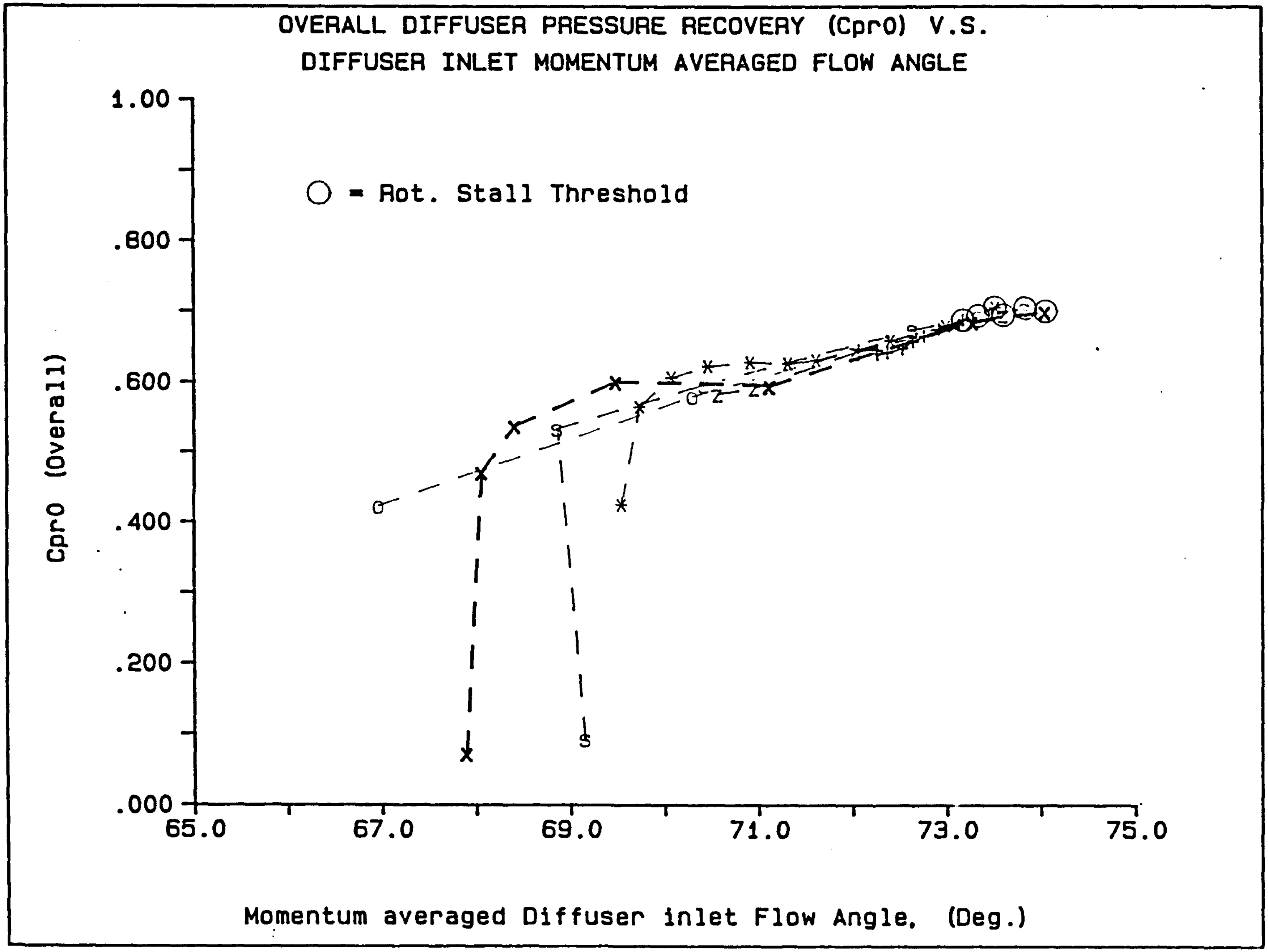

OVERALL DIFFUSER PRESSURE RECOVERY (Cpr0) V.S.
DIFFUSER INLET MOMENTUM AVERAGED FLOW ANGLE
○ = Rot. Stall Threshold
Cpr0 (Overall)
1.00
.800
.600
.400
.200
.000
65.0
67.0
69.0
71.0
73.0
75.0
Momentum averaged Diffuser inlet Flow Angle, (Deg.)

Figure 2

Compressor Stabilization Through Structural Feedback

Research is being conducted in developing methods to suppress rotating stall in axial flow compressors using structural feedback. During the Summer of 1990, the initial phases of developing and modelling various structural control concepts were conducted.

At this time, three possible passive control strategies have been identified and modelled. Each of the methods involves introduction of additional structural degrees of freedom to the basic fluid dynamic model which predicts the onset of rotating stall. The effectiveness of the structure in suppressing rotating stall is dependent on matching (i.e. tuning) the structural properties to the aerodynamics.

The first two methods involve the motion of the compressor blades in translation (bending) and torsion-oscillation. Analysis carried out to date employs a 2-D, incompressible, quasi-steady actuator disk model for the aerodynamics. Initial results show that the flow parameters at which rotating stall occurs appear to be significantly influenced by blade motion, but it is too early for any definite conclusions to be drawn. The other method examined is the use of dynamically controlled variable tip clearances to control rotating stall. The underlying mechanism in this method is based on modifying the local (circumferential) pressure rise characteristic of a blade row by varying the local tip clearance. A schematic illustrating this concept is shown in Fig. 3. As modelled, the rotor remains fixed and the wall responds to local pressure perturbations, thereby varying the tip clearance, either aerodynamically or mechanically.

As stated, this research is in the initial stages and no conclusions are yet available regarding the feasibility of these methods. Work is thus continuing on the methods described above, as well as on developing other possible methods to control rotating stall using structural feedback.

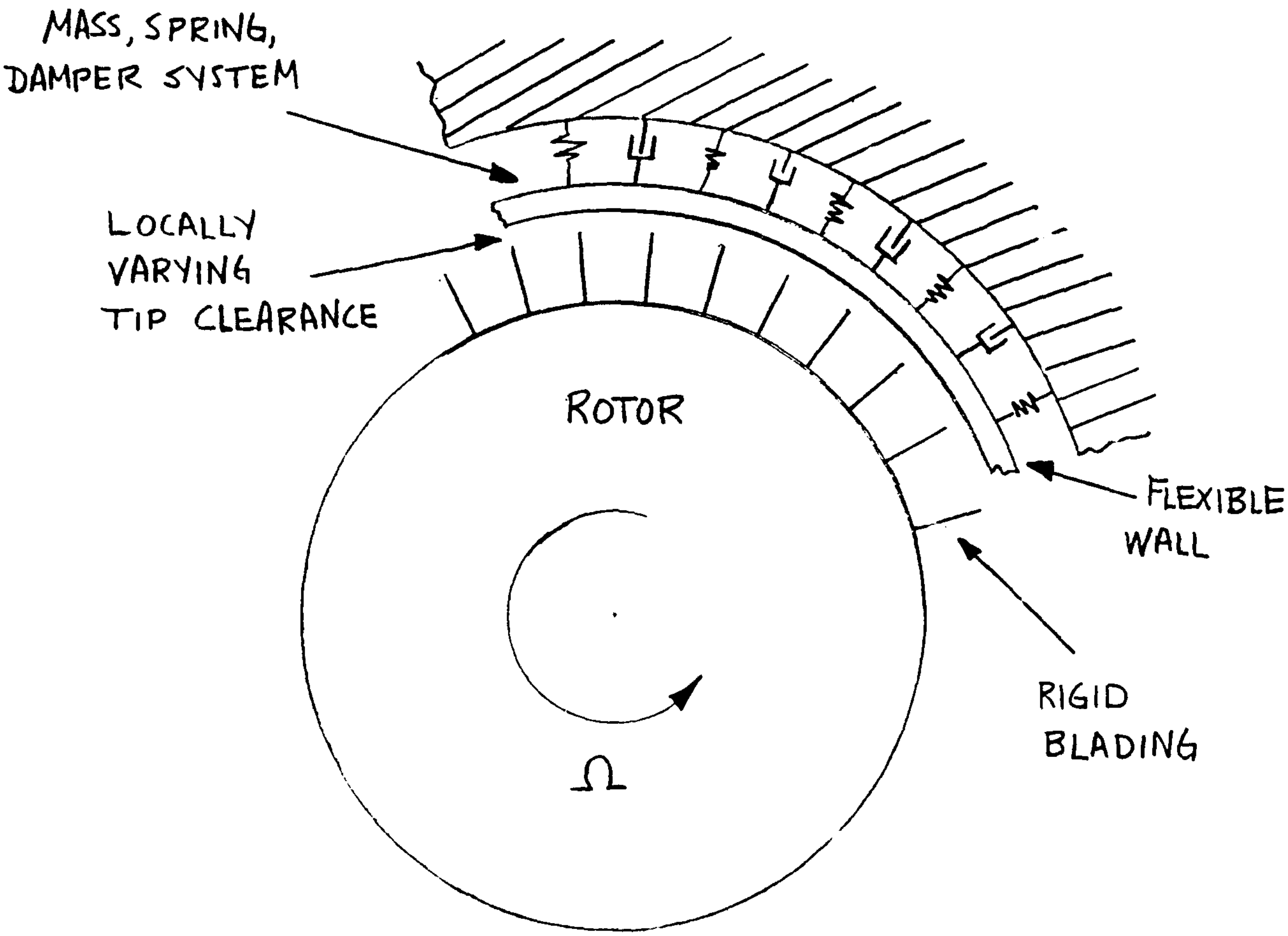


Fig. 3: Schematic of variable tip clearance using structural feedback

Linearized Euler Solutions for Turbomachinery Flutter and Forced Response

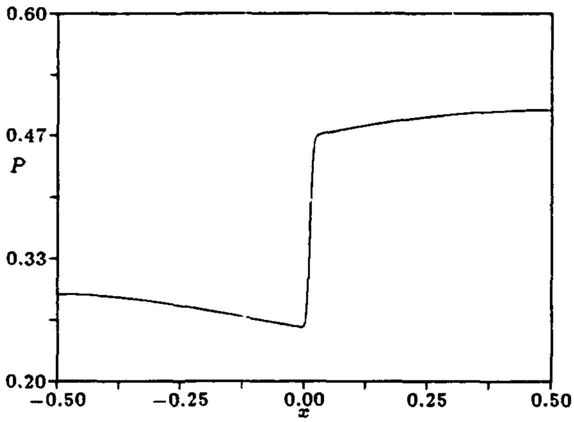
The focus of this research is on the unsteady fluid dynamics around a deforming blade, in particular those that affect aeroelastic behavior. Initial perturbations in these types of flows are small, and it is thus appropriate to perform the unsteady calculations using a linear perturbation code, which would be much cheaper to run than a full nonlinear unsteady code.

In a linear code, the steady state nonlinear flowfield is first calculated about the mean location of the blade which produces a solution $\overline{U}(x)$. Next, for each mode of the excitation, an unsteady perturbation is found. This perturbation has form $\widetilde{U}(x)e^{i\omega t}$, so the full unsteady solution looks like

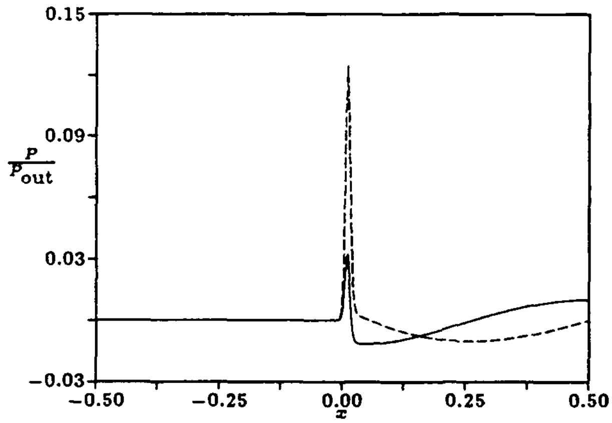
$$U(x,t) = \overline{U}(x) + \widetilde{U}(x)e^{i\omega t}$$

This solution procedure has been used for some time with flowfields which do not contain discontinuities, and recent work has been done for flowfields which contain shocks where the shock is fit, i.e. a grid line fits along the discontinuity. The purpose of the present work is to *develop methods to capture these discontinuities.*

To date, the research has involved investigations of quasi one-dimensional duct flow. With this configuration, a great deal can be learned about the mechanisms involved without undue computational complexity. The results of these studies have been very encouraging. In Fig. 4a, the steady state pressure for a transonic duct is shown as a function of x. In Fig. 4b, the real and imaginary part of the unsteady pressure perturbation is shown. This code, which captures the shock as discussed above, gives an unsteady lift (the integral of the pressure perturbation) of $\widetilde{L} = 0.0000260 - 0.0006692i$ and a code which fits the shock gives $\widetilde{L} = 0.0000256 - 0.0006723i$, so the accuracy is quite good. The questions being examined now are what kind of resolution is needed and what kind of numerical smoothing formulation gives the best results. The next phase of the work will be to examine two-dimensional flows where the unsteady perturbation will be caused by the movement of a blade.



a) Steady pressure



b) Unsteady pressure perturbation

Fig. 4: Pressure along duct

Vortical Flow Behavior Downstream of Convoluted Trailing Edges

The project involves research in the role of streamwise vorticity in enhancing fluid dynamic mixing. Lobed trailing edges have been shown to enhance the rate of mixing of adjacent parallel flows. Two phenomena are responsible for the increased mixing: increased interfacial contact area and the secondary circulation due to streamwise vorticity. The current objective is to obtain quantitative and qualitative information from flow visualization studies using the Gas Turbine Laboratory Water Tunnel Mixing Facility.

The tunnel is a gravity driven facility with interchangeable splitter (mixing) plates. Two 800 liter supply tanks provide independent speed control (up to 80 cm/sec) and chemical seeding (for flow visualization) of two streams. Maximum Reynolds number based on test section half width is 3×10^4.

The techniques for flow visualization consist of the following: a) a chemical reaction based on pH, which produces a red product at the mixing interface (phenolphthalein/sodium hydroxide); b) laser induced fluorescence (LIF), a method in which a dye, disodium fluorescein, is illuminated in an isolated plane with green laser light to reveal a cross-sectional view of the flow; and c) chemical LIF, a combination of methods a) and b), in which the fluorescence property of the dye is suppressed with acid, then re-enabled with a base from the second stream, so that the fluorescent region is limited to the region where a degree of molecular scale mixing has occurred.

Mixedness profiles downstream from the trailing edge of the splitter plates can be obtained with a technique that measures the light intensity distribution of the backlit test section during chemical (technique a) above) flow visualization. Methods for obtaining mixedness data directly from LIF cross-sectional images are being investigated.

The activities of the past year were devoted to maintenance and update of the water tunnel apparatus, calibration of the velocity measurement system, and proof of concept studies for using LIF. Tunnel maintenance consisted of realignment of modular segments, resealing of joints, repair of fractured nozzle block, reinforcement of center plates, and installation of automatic shut-off valves. Instrumentation was updated for real-time computer-based velocity readout. LIF concept studies with video and 35 mm photography indicate reasonable potential for acquiring quantitative data from geometry of mixing interface.

An experiment to compare the mixedness profiles of three splitter plate configurations will be conducted shortly. Its purpose is to isolate those effects due to increased surface area from those due to streamwise vorticity. The plates consist of a forced mixer (baseline mixer), a convoluted plate (produces little or no streamwise vorticity, but maintains same trailing edge geometry), and a flat plate. LIF based imaging experiments are being developed and will be carried out in the next year.

Fig. 5: Laser Induced Flourescence (LIF). Cross-sectional view of mixing interface, originating from one lobe on forced mixer, 3.5 lobe wave lengths downstream. Velocity ratio 2:1; mean velocity 10 cm/sec.

Turbine Blade Tip Forces

This research program is an investigation of the unsteady aerodynamics of swirling flow in labyrinth seals and its impact on rotordynamic stability. Air (or other gas) flowing through the clearances of labyrinth seals can induce an asymmetric pressure distribution which leads to non-synchronous vibration. The problem has direct application to the rotor whirl problem of high power density turbomachinery.

The work over the previous year has focused mostly on collecting and processing data from the Labyrinth Seal Test Facility (LSTF) and interpreting those results. The instrumentation in the LSTF has been upgraded to reduce some of the early uncertainties with respect to (knife-to-knife) carryover flow coefficients and axial distributions of pressure fluctuations. Slight adjustments have been made to the lumped parameter computer model when experimental results suggested modifications were necessary.

Two of four planned builds have been completed. In these first two builds, a narrow sharp seal was tested against a smooth land. A relatively full test matrix has been done on this configuration to examine the effects of pressure ratio across the seal, inlet swirl, spin rate, and whirl rate. The data from the second build was also thoroughly compared with our model [1]. Among the important results is that the cross forces generated by different inlet pressures can be approximately collapsed to a single curve by dividing by a function (given by theory) of inlet to exit pressure ratio and the ratio of effective flow areas. Figure 6 shows the measured cross force versus whirl rpm for three inlet pressures at zero inlet swirl. Figure 7 shows the "corrected cross force" versus whirl rpm. For non-zero inlet swirl, it is necessary to plot this versus a non-dimensional whirl speed to effectively collapse the data.

A third build is currently being tested using a different rotor seal geometry (wider knife spacing and flat sealing areas) but still a smooth land. Preliminary results demonstrate excellent linearity of forces with swirl orbit radius.

Reference

1. Millsaps, K.T., Martinez-Sanchez, M., "Static and Dynamic Pressure Distributions in a Short Labyrinth Seal," Sixth Conference on Rotordynamic Instability, College Station, TX, May 1990.

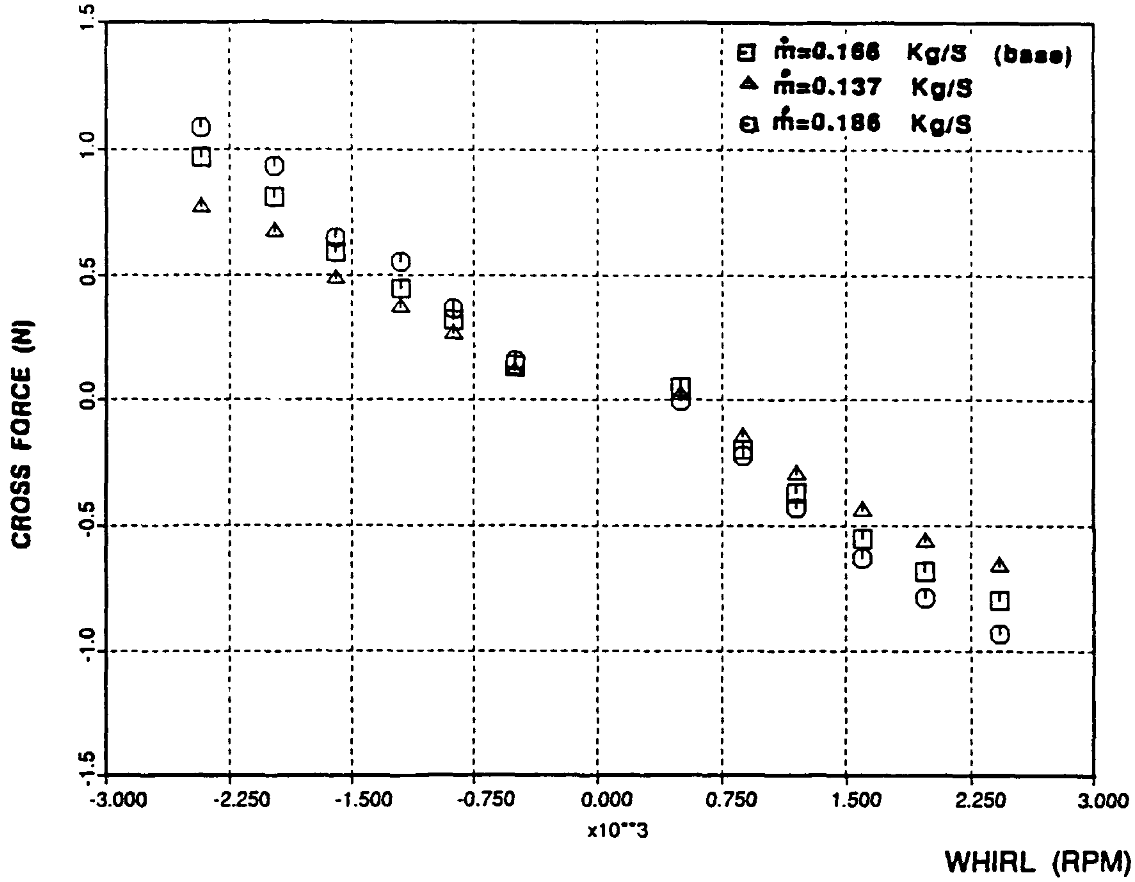


Figure 6. Cross force vs. whirl rpm for three different mass flows.

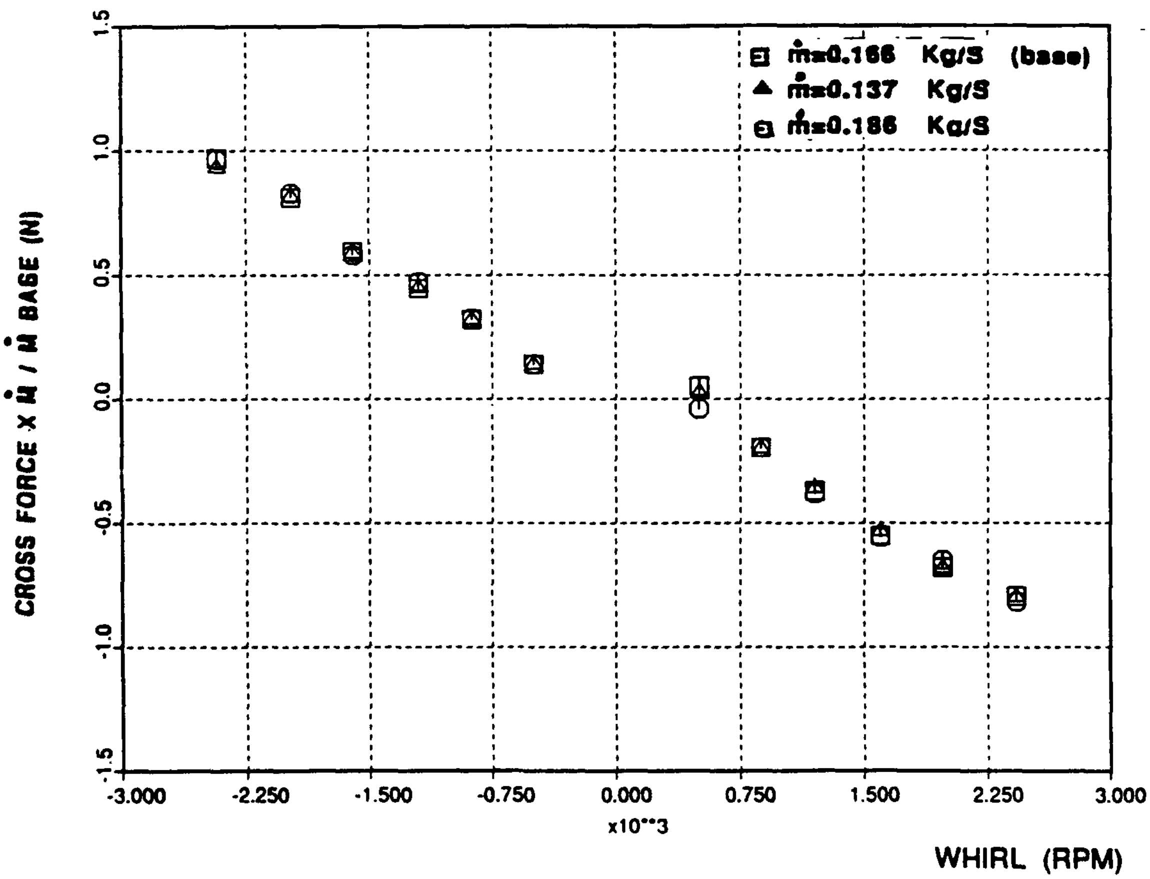


Figure 7. Cross force normalized by mass flow rate divided by design mass flow rate for the same three mass flows as in previous figure.

Influence of Inlet Radial Temperature Distribution on Turbine Rotor Heat Transfer

The effect of radial inlet temperature distortion on turbine rotor blade heat transfer has been studied experimentally and computationally. Heat flux distributions at the rotor tip and midspan were measured for several levels of inlet temperature distortion. Computational fluid dynamics (CFD) results were used in the analysis of the turbine flow field and experimental heat transfer data in an effort to ascertain the importance of inviscid effects. A comparison of Nusselt and Stanton numbers was made.

Measurements of rotor heat flux were taken in the MIT Blowdown Turbine facility which employed a storage type heat exchanger for the generation of an inlet radial temperature distribution factor (RTDF). The RTDF is a measure of the combustor exit temperature distortion normalized by the combustor temperature rise. The study focused on the comparison between a uniform inlet temperature and a 15% RTDF inlet profile (this is believed typical for a high performance combustor). Thin film heat flux gauges applied to the rotor blade along the midspan showed increases of 50% to 95% in the DC heat flux resulting from the introduction of the 15% RTDF. Measured heat flux levels on the pressure surface of the rotor tip were nearly doubled as compared to the uniform inlet temperature case (Figure 8).

Three-dimensional inviscid calculations show that secondary flows are introduced by the inlet temperature variation and cause a redistribution of the flow. A migration of the hot fluid at the center of the rotor inlet passage toward the blade pressure surface is seen, and movement of hot gas to the tip of the pressure surface is also observed. The trends in the heat flux data seem to be corroborated by the inviscid temperature redistribution.

Analysis of the non-dimensional heat transfer coefficients, supplemented by the use of CFD, indicates that both viscous and inviscid effects play a role in turbine rotor heat transfer. Nusselt numbers increase by 30% on the midspan suction surface and by 70% on the midspan pressure surface upon the introduction of the 15% RTDF. Nusselt number increases are also seen at the rotor tip, but are not as significant. The 15% RTDF also causes 10% to 30% increases in Stanton number, with values higher at the tip pressure surface than at midspan. In addition, there are increases of several hundred percent in Stanton number from the trailing to leading edges of the blade.

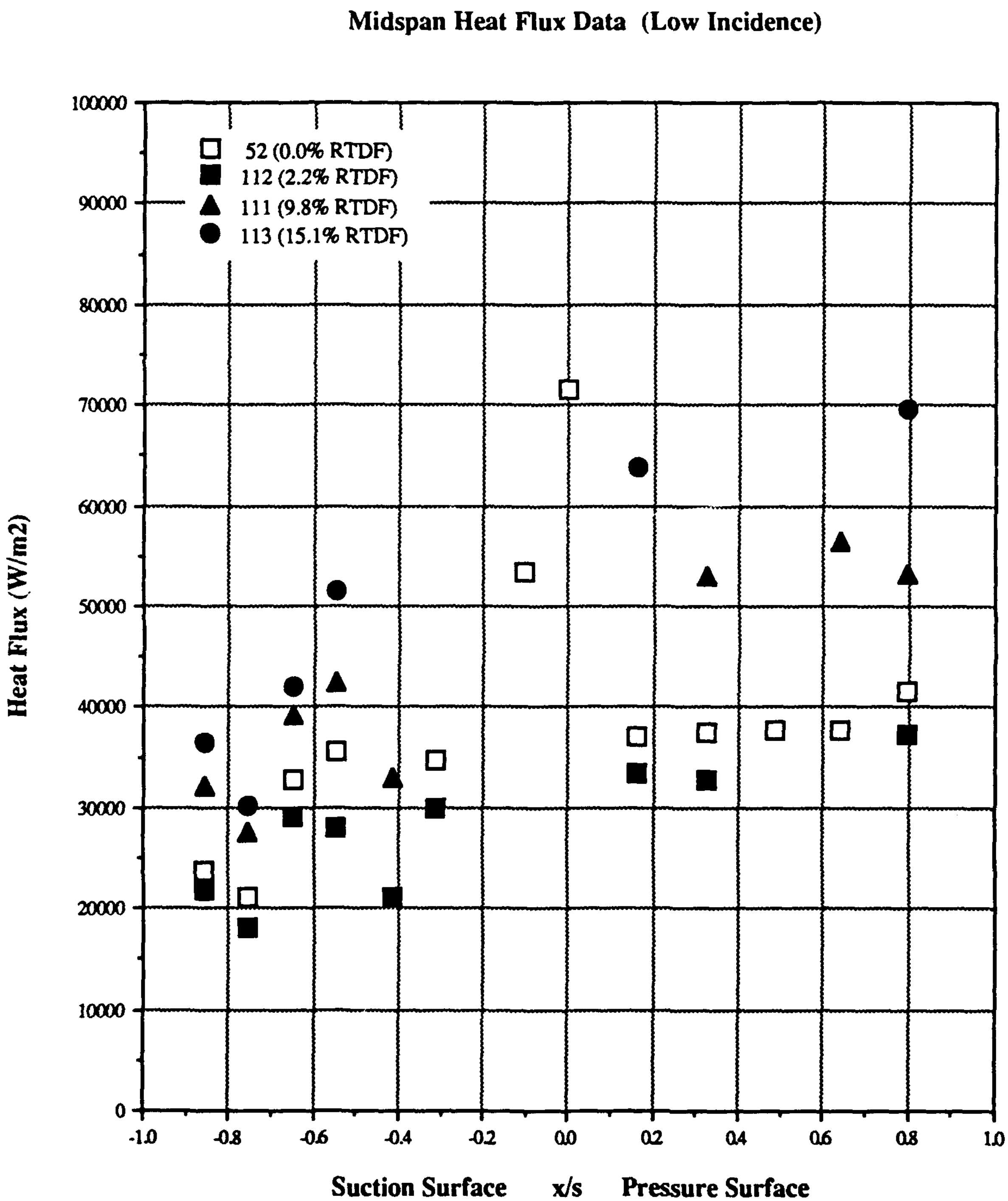

Midspan Heat Flux Data (Low Incidence)
52 (0.0% RTDF)
112 (2.2% RTDF)
111 (9.8% RTDF)
113 (15.1% RTDF)
Heat Flux (W/m2)
100000
90000
80000
70000
60000
50000
40000
30000
20000
10000
0
-1.0
-0.8
-0.6
-0.4
-0.2
0.0
0.2
0.4
0.6
0.8
1.0
Suction Surface x/s Pressure Surface

Figure 8

Effect of Circumferential Inlet Distortion on the Formation of Turbine Secondary Flow Vortices

The flow through the rotors of axial flow turbines is inherently unsteady due to the periodic ingestion of upstream vane wakes. The circumferential distortions in the rotor inlet flowfield lead to periodic temporal variations in the basic rotor flow parameters, and hence to temporal variations in the rotor passage performance. The research in this project examines how these circumferential distortions in turbine rotor inlet flowfields affect the secondary flow structure and associated total pressure loss within rotor passages.

Recent experimental investigations at the United Technologies Research Center suggest that large variations in the strength and structure of the secondary flow vortices occur as a turbine rotor passage ingests wakes from upstream airfoils. Analysis has revealed that the unsteadiness in vortex strength is tied to upstream vane passing frequency, indicating that the ingestion of the vane wakes is affecting the formation of the normal secondary flow vortex system. Since these vortices account for a significant percentage of the total pressure loss within a blade row, changes in their strength and structure may have large effects on overall turbine stage performance.

A numeric simulation of the flow through a turbine rotor passage is being used to gain understanding of how vane wake ingestion can affect secondary flow vortex formation. The particular algorithm is a three-dimensional spectral element technique, well suited to this investigation due to its high order accuracy and freedom from numerically induced artificial dissipation. The geometry chosen is that studied by Langston in his benchmark cascade investigation, with a blade aspect ratio of 1.0 and a solidity of 1.0.

Results have been obtained for the undistorted flow through the cascade. These show strong secondary flows in the endwall region, with three-dimensionality extending nearly to midspan. A dominant feature of the flow is the formation of a strong horseshoe vortex at the blade leading edges, which extends across the passage to become the passage vortex. In addition, there is a counter-vortex underneath and upstream of the horseshoe vortex, produced due to the endwall shear under the horseshoe system. The region of strong gradient in streamwise vorticity corresponds to the region of large streamwise gradient in total pressure. In addition to high shear on the endwall, the vortex system results in a significant thickening of the midspan boundary layers, yielding a midspan flowfield that is not the same as a simple two-dimensional cascade flow at the same Reynolds number.

Calculations using the same geometry but with distorted inflow profiles are currently proceeding, in order to assess the effect of the circumferential distortions on the secondary flow formation and the resulting total pressure loss.

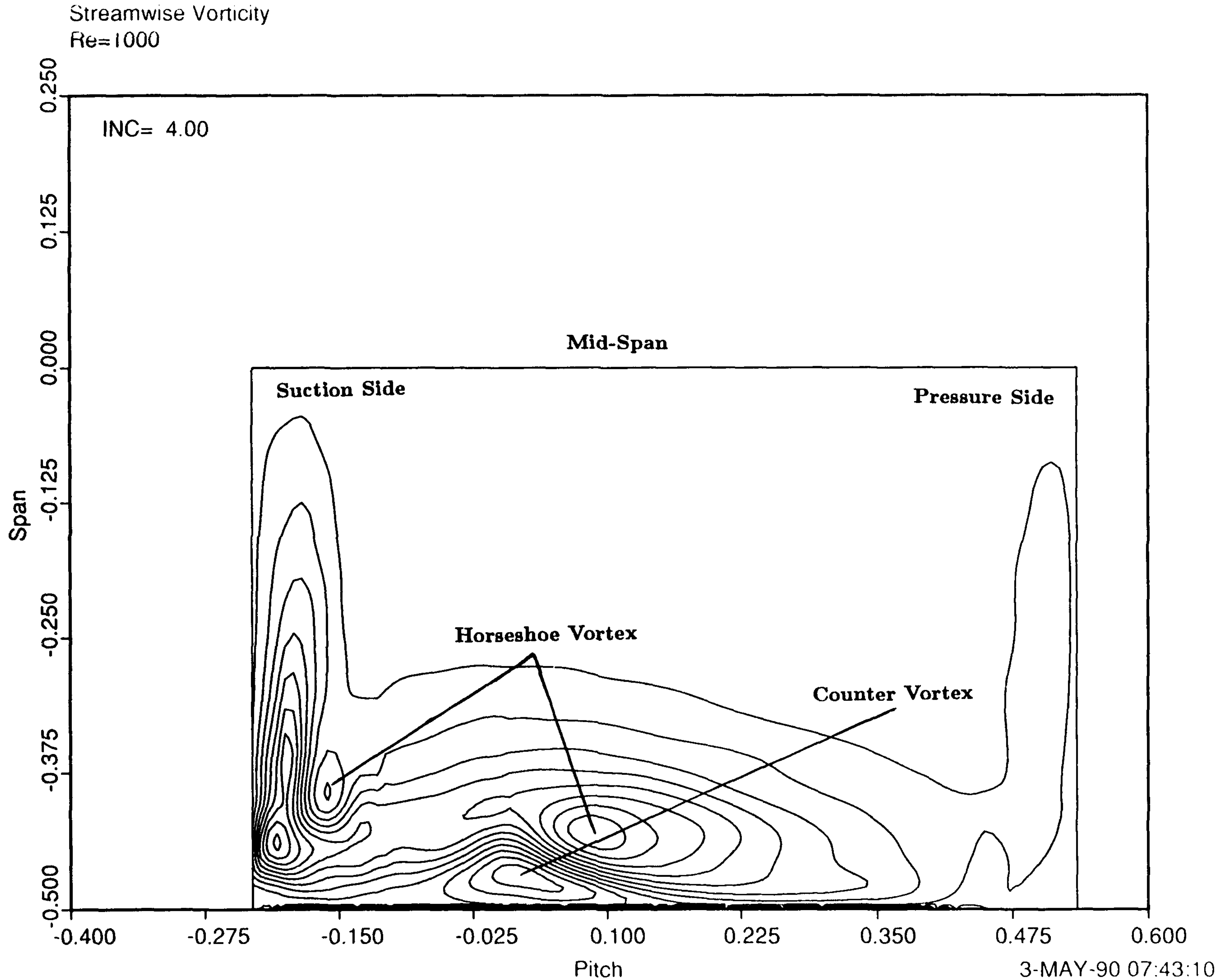

Streamwise Vorticity
Re=1000
INC= 4.00
Mid-Span
Suction Side
Pressure Side
Horseshoe Vortex
Counter Vortex
Span
0.250
0.125
0.000
-0.125
-0.250
-0.375
-0.500
-0.400
-0.275
-0.150
-0.025
0.100
0.225
0.350
0.475
0.600
Pitch
3-MAY-90 07:43:10

Figure 9

Inlet Distortion with Application to STOVL

Existing theories for predicting compressor stability with inlet distortions are based on models of the compressor performance that include only one aspect of the unsteady flow in the compressor. In particular, these models use an inertia correction to the local axisymmetric performance, in order to account for the deceleration and acceleration of fluid in the blade passage. The purpose here is to fully assess the extent to which this theory predicts compression system stability in circumferentially distorted flowfields by including nonsteady effects previously not accounted for.

The current effort involves inclusion of the effects of non quasi-steady losses and exit angle deviations on distorted compressor performance. When the compressor is subjected to an inlet distortion, the instantaneous flow quantity (loss, deviation) can lag behind the steady state due to delays in boundary layer response to changes in incidence. A fluid dynamic model has been developed to describe these small length scale effects. (Small length scale effects here denote phenomena whose characteristic scale is on the order of a blade pitch, whereas the distortion length scale is on the order of the compressor circumference.)

The solution procedure consists of (a) determining the steady state flowfields in the presence of a distortion, and (b) analyzing the stability of the flow by subjecting it to a general unsteady flow disturbance. Calculations have now been carried out to quantify the effects of non-steady losses. The calculations were made for a range of compressor pressure rise characteristics and a range of parameters, specifically inertia parameters and distortion magnitudes of practical interest. The overall effect of unsteadiness on stability for compressors of various design is shown in Fig. 10. The figure shows the loss of performance for eight different compressors with inlet distortion. Including nonsteady losses results in a smaller predicted loss of performance, i.e. the results show that, in most cases, predicted compressor instability occurs at a lower flow coefficient and corresponding higher distorted pressure rise. In addition, nonsteady losses tend to stabilize disturbance modes having strongest harmonic content greater than the first harmonic. Further analysis of these results indicates that, under certain circumstances, a simpler nonsteady loss model can be used, resulting in computational efficiency.

A flow model for nonsteady angle deviations has also been developed. This model has been computationally implemented and at this time is in the final stages of diagnostics. A similar study will be performed to determine the effects on distorted compressor performance. The model will also enable assessment to be made of compressor inlet swirl sensitivity with distortion. (Man flow calculations have shown that this effect causes a significant increase in rotor turning and an increase in total pressure loss through the inlet guide vanes in the region of negative swirl.) Future work will involve quantification of these effects and will essentially complete the 2-D modelling aspects of the problem.

Further research will include extending the theory to three-dimensional flows. In this, a new compressor model will have to be developed which includes those fluid dynamic effects intrinsic to three-dimensional turbomachinery flowfields.

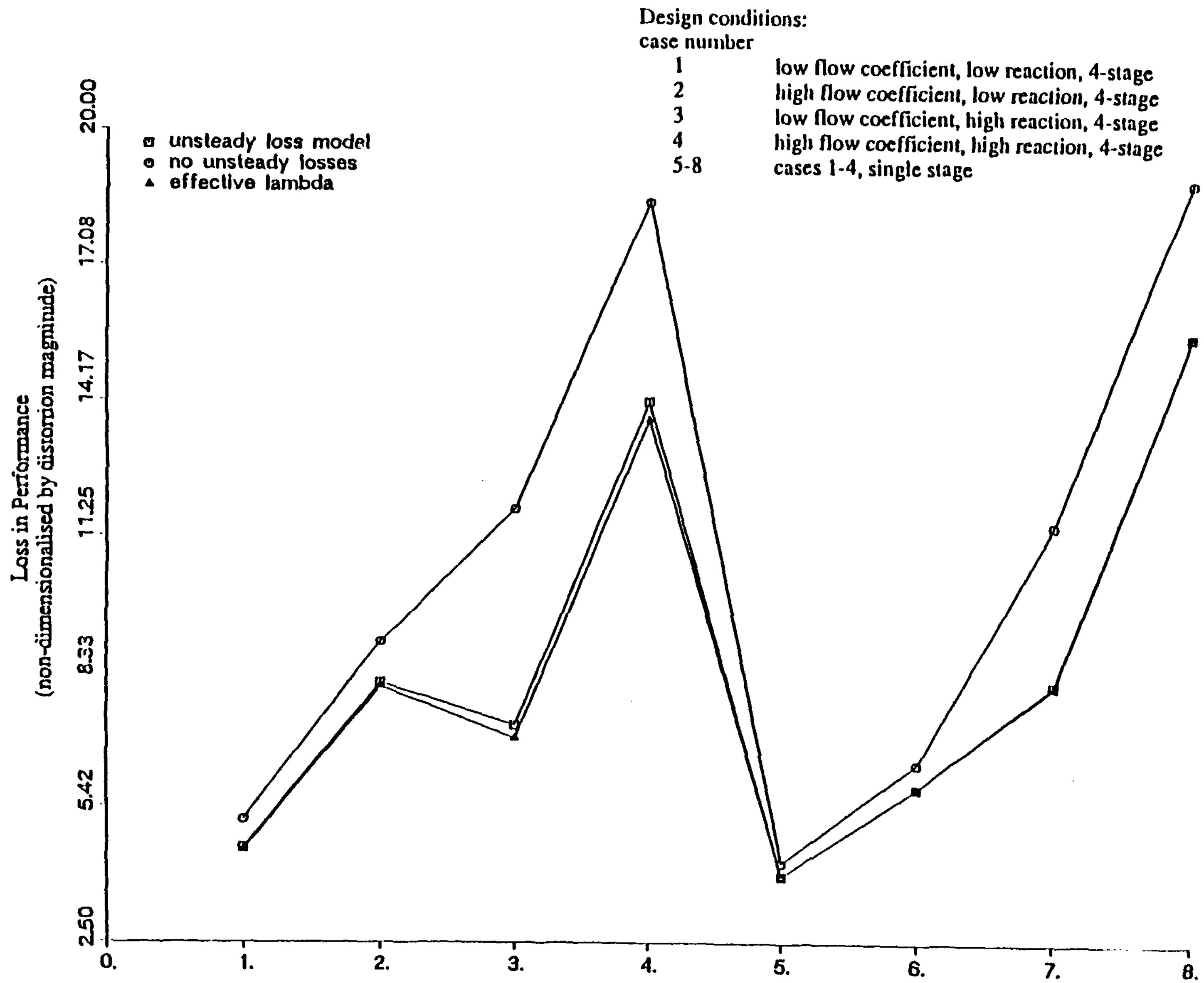


Figure 10 : Effec: of Nonsteady Losses on Stability Prediction for Compressors of Various Design Conditions (τ=0.2).

<u>Instability and Distortion in Multistage Compressors</u>

This program is an investigation of blade passage and rotating stall fluid mechanics. It represents a unique industry-university collaboration on an international level between the MIT Gas Turbine Laboratory, the Aerodynamic Research Laboratory at the General Electric Aircraft Engine Group, and the Whittle Laboratory at Cambridge University. The GE Low Speed Research Compressor will be the primary facility for the investigation, and this, with the assembled team, affords this project a potential not often possible in a university environment.

The program is laid out along two paths. First is an investigation of the existence of "precursors" to rotating stall; these precursors are of key interest for compressor control. There are, at present, two different views. One view sees the precursor as a long wavelength (scale of the circumference) disturbance that grows smoothly into rotating stall. The other view is that the stall process starts with short wavelength (1-2 pitch long) disturbances. As an aid in flow diagnosis, therefore, the experiments will be carried out with the baseline machine and with the rear three stages re-staggered to suppress the resulting long wave disturbances.

The second task is to look at the passage fluid mechanics in detail to determine the key fluid dynamic aspects of rotating stall inception. Measurements will be taken on the rotor and at the rotor exit and inlet (in combination with flow visualization) to examine specific flow structures such as boundary layers and tip vortices. This information will be used to examine current hypotheses about the growth of blockage in the passage, in particular the role of the tip vortex, as stall is approached.

The specific questions to be addressed in the overall program are:

1) Is there a precursor to rotating stall?
2) Is the precursor a short wave phenomena, a long wave phenomena, or some combination of the two?
3) Is the precursor rig dependent?
4) What is the fluid mechanic behavior of the blade passage, in particular the endwall boundary layers and the tip vortex, as stall is approached?

5) What are the mechanisms of rotating stall inception?

During the past several months, a test plan has been written, in an iterative fashion, with all members of the team. Data reduction programs have been written and tested for analyzing the data. The student is currently at GE making preparations for the upcoming experiment. A variety of instruments will be used, in particular an array of hot wires and pressure transducers will be used. Several radial and axial locations in the machine will be examined, first with rows of ten hot wires and ten pressure transducers around the circumference and then with rows of eight hot wires and eight pressure transducers over a two-pitch circumferential region to focus on any short wavelength phenomena. As part of the preparation, two of the pressure transducers have been installed into the rig during other GE testing; a fully developed stall cell travelling at roughly one half of rotor speed (800 rpm) is clearly visible in the voltage time trace of one of the transducers (Fig. 11).

The full series of experiments are scheduled to be run in two parts: the first part this October and the remaining part during the summer of 1991.

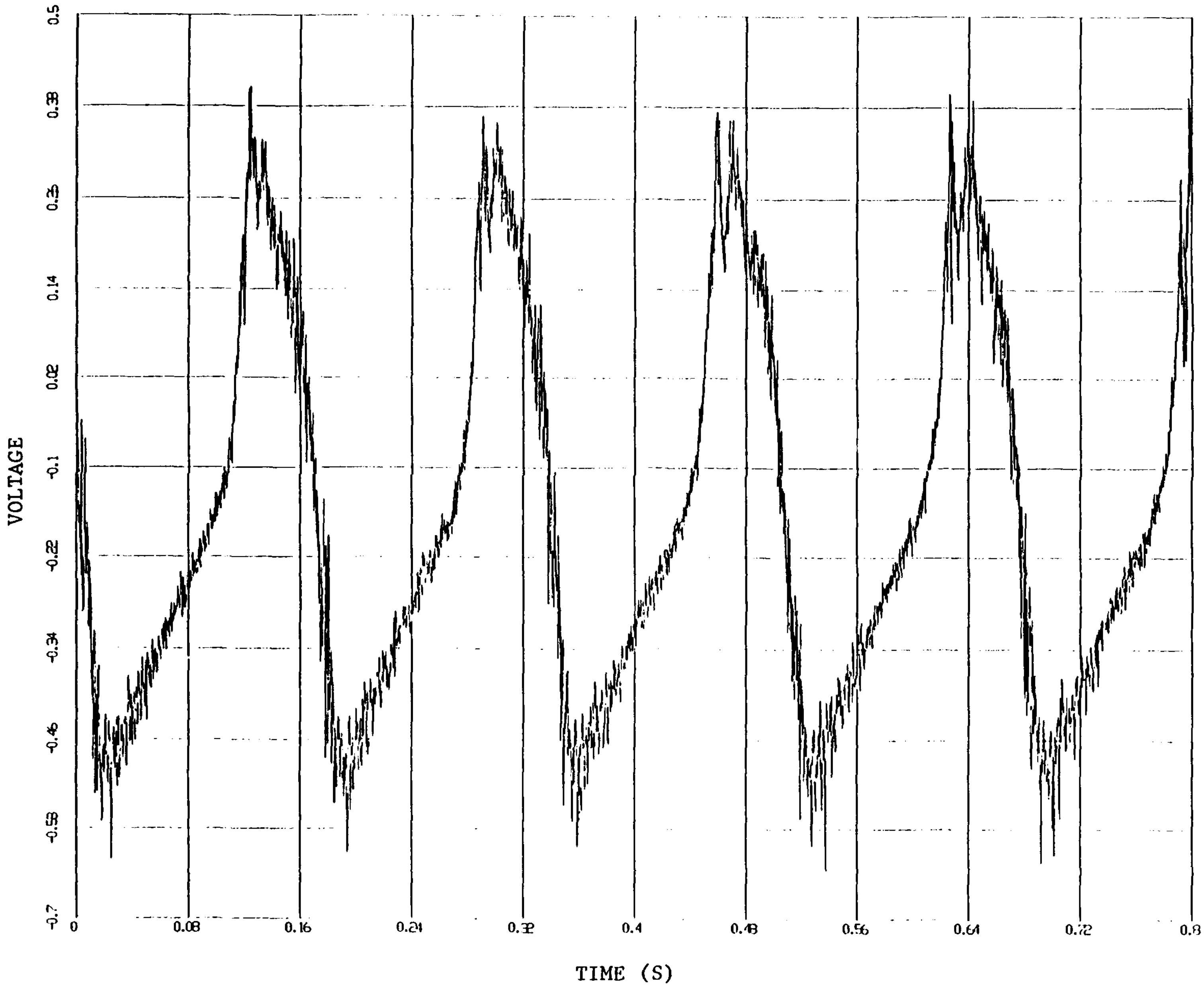


Figure 11: Hot wire output; multistage compressor in rotating stall