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Development of a Quiet Supersonic Wind Tunnel with a Cryogenic Adaptive Nozzle

Stephen W. D. Wolf

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ANNUAL PROGRESS REPORT

March 1991 - April 1992

NASA Ames Contract NCC 2-604

Principal Investigator - Dr. Stephen W. D. Wolf

Introduction:

The main objective of this work has now changed to the development of an interim *Quiet* (low-disturbance) supersonic wind tunnel for the NASA-Ames Fluid Mechanics Laboratory (FML). This change is a result of the need to bring the full-scale tunnel on-line as rapidly as possible to impact the NASA High Speed Research Program (HSRP). The development of a cryogenic adaptive nozzle and other sophisticated features of the tunnel will now happen later, after the full scale wind tunnel is in operation. The work under this contract for the period of this report can be summarized as follows:

1) Research to find design parameters for a unique Mach 2.5 drive system for the Fluid Mechanics Laboratory (FML) Laminar Flow Supersonic Wind Tunnel (LFSWT) using an 1/8th-scale Proof-of-Concept (PoC) tunnel.

2) Carry out research to aid the design of critical components of the LFSWT.

3) Appraise the *State of the Art* in *Quiet* supersonic wind tunnel design.

4) Help develop a supersonic research capability within the FML particularly in the area of instrumentation techniques.

The body of this annual report summarizes the work of the Principal Investigator and is presented under logical headings. The order is not significant.

Drive System Research:

The PoC supersonic wind tunnel has proved to be a valuable workhorse during the period of this report. Work with PoC has concentrated on Phase 1 and 2 (Investigation of drive system and instrumentation development) of the PoC experimental program outlined in the last progress report.[1] The development of the PoC Mach 2.5 drive system has been reported in the literature (See Appendix 1). Phase 3 (Investigation of adaptive wall and cryogenic technologies) is now postponed owing to time constraints, and this wind tunnel development will be undertaken using the actual LFSWT.

As part of the Phase 1 work, in August 1991, we demonstrated that the PoC secondary injectors can be moved closer together with no detriment to the drive system performance at Mach 2.5 (See Appendix 2). This finding shortens the LFSWT drive system by a significant 16.5 feet.

A new settling chamber was installed in the PoC in October 1991 as part of our quiet flow studies. This change also allowed the PoC test section length to be restored to the original length of 4 inches. Consequently, the PoC was better able to simulate the LFSWT test section flows, and the length change caused the minimum stagnation pressure (Po) for Mach 2.5 operation to rise from 5 psia to 5.4 psia. This happened despite some primary injector and supersonic diffuser tuning. We concluded that the additional run length of 3.335 inches had some effect on supersonic diffuser losses. In addition, the new use of dynamic instrumentation

probably made us more critical than before, particularly as minimum Po is determined subjectively by the tunnel operator.

In December 1991, we examined the reduction of the secondary injector mass flows. This work arose from a realization that the PoC drive system mass flow had grown too big to scale up 64-times for the LFSWT. This drive system tuning showed that, the original PoC drive system was operating close to the minimum mass flow. Any reductions of mass flow incurred an undesirable rise in the minimum Po. We concluded that the LFSWT Mach 2.5 drive system will require a minimum air mass flow of 184 lbs/sec with an exit pressure (PE) of 8 psia.

Since the expected mass flow capability of the FML compressor is only 163 lbs/sec with a PE of 8 psia, a series of compressor tests were carried out in February 1992 to find out if the compressor could drive the LFSWT. The findings of these tests are described in detail in Appendix 3. A series of misconceptions was uncovered leading to a conclusion that the FML compressor alone cannot drive the LFSWT at Mach 2.5. The decision was then made to concentrate the LFSWT design on Mach 1.6 operation with a single stage of ambient injectors, incorporating a high degree of adjustability. In the absence of any changes to the FML compressor operation, the PE is now raised to 8.3 psia for LFSWT operation.

Quiet Flow Research:

Settling Chamber

In November 1991, we installed a new PoC low-disturbance settling chamber (see Appendix 2). This settling chamber has been operated over a Po range from 5 to 15 psia. We found that the maximum pressure drop in the settling chamber was about 2.5 psia which occurred across the flat sheet of Rigimesh. The Rigimesh cone supports minimal pressure load, which simplifies the necessary support structure for the full-scale LFSWT cone.

Preliminary flow disturbance measurements were made in the plane of the settling chamber exit (at a single location on the tunnel centerline) using a Kulite total pressure probe and a 4 micron Tungsten hot-wire. The ratio of the Pressure rms with stagnation pressure shows a significant rise with the honeycomb and Rigimesh sheet removed. This pressure ratio drops with increasing Po. With all the settling chamber components in place, the pressure fluctuations are of the order 0.1%. This level of disturbances is considered to be comparable with other quiet wind tunnels (See Appendix 2).

Laminar Flow

The laminar flow studies with the PoC have involved the use of different types of instrumentation to confirm the state of the test section boundary layers. The detection of boundary layer transition tends to be qualitative and our goal was to find at least 2 measurement techniques which agreed about the location of transition.

We found during January 1992, that the hot-wire measurements made above the PoC test section floor, in the outer portions of the boundary layer, show a sharp rise in signal rms when Po is about 9 psia. The signal spectrums are broadband with no discrete frequencies. Furthermore, the associated rise in signal rms is independent of the signal bandwidth. The behaviour of the signal rms has been seen before in supersonic transition research, where the transition bursting reaches a maximum frequency in the transition process. Unfortunately, the uncalibrated hot-wire data can only be used qualitatively and alone these data could not prove the existence of laminar flow in the PoC.

To check the reliability of the hot-wire data from the PoC test section, the honeycomb and Rigimesh sheet were removed from the settling chamber. The increase of free stream turbulence had the effect of initiating the rise in signal rms at a lower Po of about 6 psia and hence a lower Re. This result shows the strong influence of free stream turbulence on supersonic boundary layers, as first reported by Laufer in 1956.

In another series of tests in March 1992, the test section hot-wire was replaced by a Preston tube. This tube was sized to fit in the lower half of the floor boundary layer. The Preston tube data shows that there is a significant rise in the probe Cp at a stagnation pressure of about 8.5 psia. This rise is associated with the known aerodynamic effects of transition onset, where the boundary layer profile starts changing from a laminar type to a turbulent type. This observation combined with the hot-wire data confirms that laminar flow does exist in the PoC over 84% of the test section length at Reynolds numbers up to about 2 million per foot.

Furthermore, the sidewall boundary layers were studied with a flush-surface-mounted hot-film gage. This gage had previously been calibrated for transition detection in a Mach 3 quiet tunnel. The hot-film data show that the boundary layer on the sidewall remained laminar over the entire Reynolds number range. The hot-film signal rms is seen to jump to expected levels for turbulent flow only when tunnel leaks caused the nozzle flow to unstart. This flow break down caused transition bypass to occur on the sidewall. In addition, the same leaks cause transition bypass to occur on the test section floor and ceiling, as measured by the hot-wire probe in the test section. These tunnel leaks have hampered this work but have now been solved with improved window potting.

During these laminar flow studies there was some concern about the drift in temperature of the inlet air and the PoC nozzle/test section structure. The air supply to the PoC is not heated and the inlet air temperature is always lower than ambient due to the expansion across a single air regulator. We monitor the inlet air temperature on a regular basis to check for repeatability of test conditions. The thermal mass of the PoC is large compared to the heat transfer associated with the nozzle/test section flow. We have observed the PoC structure reaching near temperature equilibrium within about the first 5 minutes of running. This temperature equilibrium is affected only slightly by changes of inlet air mass flow, despite noticeable changes in the inlet air temperature. To assess the long term effects of temperature drift, we operated the PoC for 2 1/2 hours continuously and monitored our hot-wire and hot-film instrumentation. No significant changes in the test section flow were observed during this test.

LFSWT Support:

As project engineer for the LFSWT, I have been involved on a daily basis with the design and fabrication of the new 3000 psia air supply system and modifications to the PoC supersonic wind tunnel necessary to support the LFSWT effort. During May 1991, I assisted with the formulation of a detailed project plan for the interim LFSWT (LFSWT-I). This project plan reflects the support of the High Speed Research Program (HSRP) and should bring the LFSWT-I on-line earlier to impact HSRP. In addition, I have greatly assisted with the formulation of a detailed LFSWT-I design and work package outlines (See Appendix 4) during November/December 1990. These outlines are now the basis for ongoing LFSWT design being performed by NASA-Ames Code-E engineers. Regular meetings with Code-E designers and staff ensure that the design progresses in an orderly fashion.

I participated in several meetings with HSRP managers, in which we were able to emphasis the significance of ground testing in Supersonic Laminar Flow Control (SLFC) research. The reliance on flight test to validate CFD analyses is not valid. We now feel that supersonic ground testing at Ames and Langley has been accepted as an integral part of the HSRP SLFC Phase I effort.

Furthermore, I participated in five meetings with the NASA Ames Director of Aerophysics, Dr. Ron Bailey through the year. In these meetings, I was responsible for presenting an update of PoC/LFSWT technical issues. The continued financial support of Aerophysics Directorate relies on the success of these meetings.

The main features of the LFSWT will be research flexibility (ease of removing components and gaining access to the nozzle and test section), settling chamber/nozzle/test section vibration isolation and the ability to use 2 stages of ambient injectors. The fixed nozzle block will be

designed according to Riise. This design produces a long nozzle with minimized curvature which encourages natural laminar flow. We are designing the nozzle/contraction as one component so there are no steps or gaps on the floor and ceiling of the LFSWT. The step and gap between the nozzle and test section is being held to 0.001 inch or less. We consider this requirement essential to maintaining natural laminar flow through the LFSWT test section over a limited Reynolds number range.

We are designing the LFSWT in stages, the interim tunnel is the first stage and will operate at Mach 1.6 with the provision for Mach number increase. Subsequent stages will expand the Mach number envelope and the quiet test envelope.

Instrumentation Development:

Testing with the PoC provided an important means of developing instrumentation for transition detection. Our early drive system studies were concerned only with static pressure measurements. I have since helped assess the instrumentation requirements for future supersonic testing, with particular reference to hot-wire probes, hot-film gages and arrays, temperature and pressure probes and transducers, and schlieren optics.

In October 1991, I brought together a dynamic data acquisition system built around a Tektronix 2642A Fourier Analyzer coupled to a 486/33 MHz PC computer. This system has the capability of simultaneously sampling two channels of input at up to 512 KHz with 16 bit resolution. Furthermore, Fast Fourier Transforms can be averaged and displayed in a few seconds using a user-friendly graphics interface. All calculations are carried out by parallel processors in the Fourier Analyzer.

The dynamic data acquisition system is used for all hot-wire and hot-film data acquisition. We use a 5 micron Tungsten hot-wire made in house for supersonic measurements. In the subsonic settling chamber flow, we use a commercially available 4 micron Tungsten hot-wire. Both wires are driven by constant temperature bridges designed and built in the FML. The Platinum hot-film gage is commercially available is driven by a constant current bridge devised by Demetriades at Montana State University.

For thermocouple data, I purchased a Dianachart Therm-ACQ system which can handle up to 48 channels of input. The data from the Therm-ACQ system is transferred to a PC computer for display and storage. This system will be used to monitor nozzle/test section wall temperatures.

We now have a mark II version of our *Focusing Schlieren* system which has been designed by Weinstein of NASA Langley for increased sensitivity. We have improved the optical components and the mounting hardware, so that the system is more permanent. So far, we have been unsuccessful in using the system to observe the supersonic boundary layer at low stagnation pressures. We are currently exploring the use of a more intense spark light source and cylindrical lenses for boundary layer magnification.

For future testing, I have designed a hot-film array, which we intend to fix to the floor or ceiling of the PoC contraction/nozzle/test section. This non-intrusive array will allow the location of boundary layer transition to be traced against stagnation pressure (Reynolds number). This array should be available for testing in June 1992.

State-of-the-Art Appraisal:

I keep an ongoing library search in the following topics: supersonic wind tunnel and nozzle design; surface temperature effects on transition; effects of surface shape and roughness on transition; supersonic mixing layers; supersonic diffusers; transition detection instrumentation. This task is simplified by use of STAR and IAA combined with a PC computer database, I created. This database provides immediate access and sorting of all citations as these are found. Currently, the database contains 779 citations. An extract of the *Supersonic Nozzle*

Bibliography has now been published as a NASA contractor's report with 298 citations and abstracts.

An important aspect of appraising the *State-of-the-Art* is meeting other scientists at conferences. During the period of this report, I participated in the High Speed Research Workshop (Williamsburg, Virginia in May 1991), the International Conference on Adaptive Walls and Wall Interference (Xian, China in June 1991), the International Conference on Experimental Fluid Mechanics (Chengdu, China in June 1991) and the AIAA 9th Applied Aerodynamics Conference (Baltimore, Maryland in September 1991). In addition, I visited NASA Langley on two occasions to discuss quiet wind tunnel testing and instrumentation development with our East coast counterparts. I was fortunate to meet with many scientists from China and the Commonwealth of States (formally the USSR). From discussions with these scientists, I was able to learn that there are no quiet supersonic tunnels in their respective countries. It would appear that the only operating quiet supersonic tunnels are in the USA and France at Mach numbers of 3 and above. It is clear that the LFSWT will give NASA a unique capability in 1993.

Publication and Presentations

I presented Paper 1 entitled **ADAPTIVE WALL TECHNOLOGY FOR IMPROVED FREE AIR SIMULATIONS IN WIND TUNNELS** at the International Conference on Adaptive Walls and Wall Interference at Xian, China during June 1991. The abstract is as follows:

> This paper reviews adaptive wall technology for improving wind tunnel free air simulations. This technology uses a powerful marriage of experiment and theory to minimize wall interferences at the very source of the disturbances. The significant benefits of adaptive wall testing techniques are briefly discussed. An overview of Adaptive Wall Test Section (AWTS) design is presented to show the preference for 2 flexible walls for both 2-D and 3-D testing. The status of adaptive wall technology is discussed and future directions for research in 3-D testing proposed.

At the International Conference on Experimental Fluid Mechanics in Chengdu, China during June 1991, I presented a paper entitled **ADAPTIVE WALL TECHNOLOGY FOR MINIMIZATION OF WIND TUNNEL BOUNDARY INTERFERENCES - A REVIEW.** The abstract is as follows:

> This paper reviews adaptive wall technology for improving wind tunnel flow simulations. The technology relies on a tunnel/computer system to control the shapes of the test section boundaries. This powerful marriage of experiment and theory is used to minimize boundary interferences at the very source of the disturbances. The significant benefits of adaptive wall testing techniques are briefly discussed. A short historical overview describes the disjointed development and the status of these testing techniques from 1938 to present. Some of the currently operational Adaptive Wall Test Sections (AWTSs) for aerofoil and turbomachinery research are described. Some observations on the achievements and future directions of adaptive wall research are presented to stimulate round table discussion.

I presented AIAA paper 91-3260 entitled **AN EFFICIENT SUPERSONIC WIND TUNNEL DRIVE SYSTEM FOR MACH 2.5 FLOWS** at the AIAA 9th Applied Aerodynamic Conference, Baltimore, Maryland in September 1991. The abstract is as follows:

> The desire to drive a new, low-disturbance, Mach 2.5 wind tunnel

> with a non-specialist indraft compressor has spawned the development of an efficient drive system. This paper describes a combined experimental and numerical effort to find a novel drive system to operate continuously at double the efficiency of any previous drive system. A small pilot research tunnel was built to investigate an extreme derivative of the auxiliary injector concept, which utilizes the excessive mass flow capability of the designated compressor. We have successfully proven that a dual injector drive system can provide Mach 2.5 flow at an uniquely low compression ratio of 0.625:1 and above. This ratio is less than one, so the exit pressure is higher than the stagnation pressure. Furthermore, the drive system does not require the usual overpressure for start. The success of the drive system relies on the establishment of adequate local compression ratios to sustain supersonic flow, supersonic shear layer mixing, and supersonic diffuser pressure recovery. Observations on these aspects are discussed in the paper and the planned use of this efficient drive system in a larger wind tunnel is outlined.

In addition, I have first authored an invited paper to the AIAA 7th Aerospace Ground Testing Conference to be held in Nashville, Tennessee during July 1992. This AIAA paper no. 92-3909 is entitled **DEVELOPMENT OF THE NASA-AMES LOW DISTURBANCE SUPERSONIC WIND TUNNEL FOR TRANSITION RESEARCH UP TO MACH 2.5.** The abstract is as follows:

> A unique, low-disturbance supersonic wind tunnel is being developed at NASA-Ames to support supersonic laminar flow control research at cruise Mach numbers of the High Speed Civil Transport (HSCT). The distinctive aerodynamic features of this new quiet tunnel will be a low-disturbance settling chamber, laminar boundary layers on the nozzle walls and steady supersonic diffuser flow. Furthermore, this new wind tunnel will operate continuously at uniquely low compression ratios (less than unity). This feature allows an existing non-specialist compressor to be used as a major part of the drive system. In this paper, we highlight activities associated with drive system development, the establishment of natural laminar flow on the test section walls, and instrumentation development for transition detection. Experimental results from an 1/8th-scale model of the supersonic wind tunnel are presented and discussed in association with theoretical predictions. Plans are progressing to build the full-scale wind tunnel by the end of 1993.

<u>Summary of Progress</u>

1) The LFSWT project is now firmly established as an integral part of the HSRP SLFC studies, and the main emphasis is on Mach 1.6 operation to support F-16XL SLFC flight tests.

2) We have documented natural laminar flow in the PoC up to a unit Reynolds number of about 2 million per foot at Mach 2.5.

3) An efficient tunnel drive system has been developed for Mach 2.5 operation, but the mass flow requirement is too large for the FML compressor alone.

4) We now have available an array of instrumentation for transition detection which we can use with the LFSWT when it comes on-line.

5) The LFSWT-I design is progressing towards an August 1992 completion.

6) A detailed project plan is in place which will bring the LFSWT-I on-line by October 1993.

References:

1. Wolf, S.W.D.: **Development of a Quiet Supersonic Wind Tunnel with a Cryogenic Adaptive Nozzle - Annual Progress Report, May 1990 - April 1991.** NASA CR-186769, February 1991, 106 pp. N91-23195.

APPENDIX A

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AIAA

AIAA Paper 91-3260
AN EFFICIENT SUPERSONIC WIND TUNNEL DRIVE SYSTEM FOR MACH 2.5 FLOWS

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AIAA 9th Applied Aerodynamics Conference
September 23-25, 1991/ Baltimore, Maryland

AN EFFICIENT SUPERSONIC WIND TUNNEL DRIVE SYSTEM FOR MACH 2.5 FLOWS

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Abstract

The desire to drive a new, low-disturbance, Mach 2.5 wind tunnel with a non-specialist indraft compressor has spawned the development of an efficient drive system. This paper describes a combined experimental and numerical effort to find a novel drive system to operate continuously at double the efficiency of any previous drive system. A small pilot research tunnel was built to investigate an extreme derivative of the auxiliary injector concept, which utilizes the excessive mass flow capability of the designated compressor. We have successfully proven that a dual injector drive system can provide Mach 2.5 flow at an uniquely low compression ratio of 0.625:1 and above. This ratio is less than one, so the exit pressure is higher than the stagnation pressure. Furthermore, the drive system does not require the usual overpressure for start. The success of the drive system relies on the establishment of adequate local compression ratios to sustain supersonic flow, supersonic shear layer mixing, and supersonic diffuser pressure recovery. Observations on these aspects are discussed in the paper and the planned use of this efficient drive system in a larger wind tunnel is outlined.

Symbols

A*	Sonic throat area
Ats	Test section area
Adif	Supersonic diffuser throat area
$\dot{m}_{inj}$	Primary injector mass flow
Minj	Primary injector exit Mach number
Me	Supersonic diffuser exit Mach number
P	Local static pressure
Po	Stagnation pressure
PE	Exit (manifold) total pressure
Pe	Supersonic diffuser exit pressure
Pinj	Primary injector exit pressure
P_∞	Ambient pressure
Pts	Test section static pressure
Re	Unit Reynolds number
Si	Arc movement of injector block relative to datum position
X	Streamwise position relative to Mach 2.5 nozzle throat station
Xend	Streamwise position of diffuser exit relative to nozzle throat station
Xn	Movement of nozzle block relative to datum position
y	Vertical position relative to tunnel centerline

1. Introduction

The drive system of a supersonic wind tunnel must provide sufficiently large compression (pressure) ratios across the test section to produce and sustain the desired test velocity. In a typical Mach 2.5 wind tunnel, the start compression ratio, Po/PE, is of the order 3-4:1 and the run compression ratio is less, of the order 2:1 as predicted by classical theory. The variation from wind tunnel to wind tunnel is mainly attributed to the use of different diffusers and model supports. In fact, prior to this work, the minimum run compression ratio at Mach 2.5 is reported as 1.41:1.[1]

It is interesting to note that the majority of supersonic wind tunnels operational today were built back in the 1940s and 1950s, when the main goal was simply to provide a supersonic testing capability. Consequently, these original supersonic tunnels are inefficient to operate and require special expensive compressors to drive them. Today, this attitude cannot be tolerated in the design of new supersonic wind tunnels, because of limited resources.

This work, to provide a more efficient drive system, is part of the development of a new continuously-operating Mach 2.5 Laminar Flow Supersonic Wind Tunnel (LFSWT) for boundary layer transition research in the Fluid Mechanics Laboratory (FML). The desired test envelope for the LFSWT is a Re range of 1-3

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million per foot, which matches the cruise conditions of the High Speed Civil Transport (HSCT). To minimize project costs, it was decided that an existing non-specialist indraft compressor should power the LFSWT. This compressor has a measured capacity of 228,000 icfm (about 143 lbs/sec - 65 kg/sec with a minimum P_E of 8 psia - 0.55 bar) and a pressure ratio of 1.8:1. Consequently, to achieve the low end of the Re range, the LFSWT has to operate with a Po which is less than the minimum P_E. This means that the LFSWT must run with very low, unusual, compression ratios (down to 0.625:1 with Re = 1 million per foot at Po = 5 psia - 0.34 bar). So , the utilization of the FML compressor precludes the use of a conventional drive system to achieve the desired Re range. This situation brought about the need for a novel drive system to reduce the best previous run compression ratio by more than 50%.

The concept for a novel drive system was based on the effective use of all of the mass flow capability of the FML compressor. Since the LFSWT test section flow is limited to a mass flow of 21 lbs/sec (9.52 kg/sec) by the dry air supply system, this theoretically leaves a total of 122 lbs/sec (55 kg/sec) for the drive system. In 1953, Spiegel et al[2] had demonstrated reductions in compression ratios for supersonic wind tunnels by using auxiliary injectors. Based on this work, it was decided to use the excess mass flow of the FML compressor to drive ambient injectors. However, to achieve very low compression ratios, the use of the Spiegel design analysis indicates the need for a huge injector mass flow ratio of the order 15:1. Previous successful use of auxiliary injectors (which assist the drive system rather than perform as the primary drive) have involved only small mass flow ratios, up to about 2:1. Furthermore, the Spiegel design analysis contains assumptions about perfect flow mixing which are overly simplistic for the high injector mass flow ratios required. In view of these concerns, an 1/8th-scale model of the LFSWT, called the Proof-of-Concept (PoC) supersonic wind tunnel, was built. The primary objective of this small research tunnel was to study our proposed extreme derivative of the Spiegel drive system. Our doubts about the original design were confirmed and we embarked on a research program to develop design principles for a new LFSWT drive system.

The LFSWT project actually started in January 1989 at the NASA Ames FML. The decision to build PoC was made in April 1989. The design of PoC was completed by July 1989 and PoC ran in October 1989, which is a remarkable feat, due in part to the small scale of the tunnel. By November 1989, we had PoC operating at Mach 2.5. For 15 months until February 1991, we carried out a joint experimental and theoretical research program to develop an efficient LFSWT drive system.

This paper includes a description of PoC and its modifications. Theoretical and experimental research efforts are described, showing how we focus on developing the optimum drive system as rapidly as possible. Here we define optimum as the configuration which best meets our requirements to drive LFSWT. We only summarize the results of our studies here, but intend to publish all the results in a future NASA report to provide a database. The discussion covers observations of the PoC drive system. The conclusions are specific to the LFSWT drive system and, in the interest of conciseness, do not cover the complex details of supersonic shear layers or supersonic diffusers.

2. PoC Tunnel Hardware and Instrumentation

A schematic of the PoC layout is shown in Figure 1, with a single-stage injector system. The PoC is an 1/8th-scale model of the full scale facility, yet to be designed. The test section in PoC is 1 inch (2.54 cm) high, 2 inches (5.08 cm) wide and nominally 4 inches (10.16 cm) long. The PoC is made up of five major components: cover plates, injector blocks, a nozzle block, a mixing region and a subsonic diffuser. We incorporated in the PoC design the ability to independently vary the mass flow and the supersonic exit Mach number of the 2-D ambient injectors.

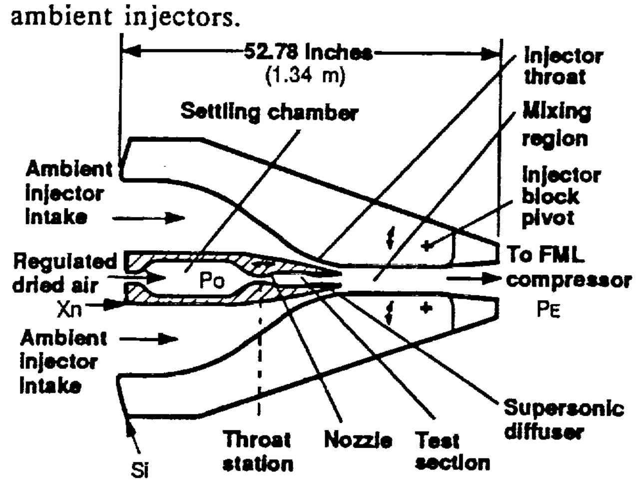

Fig. 1 - Schematic of the PoC supersonic wind tunnel with a single-stage injector system.

The two injector blocks are able to rotate about pivots normal to the flow, and the nozzle block translates upstream and downstream as indicated on Figure 1. This movement adjusts the relative position of the two circular arcs which form the walls of each 2-D injector nozzle. We defined a datum configuration for the injectors as that configuration indicated by the Spiegel design analysis for operation at Po = 5 psia (0.34 bar). This datum is that the injector throat area is 1.16 square inches (7.48 cm^2) and the injector exit area is 1.32 square inches (8.52 cm^2). The translation of the nozzle block relative to this datum is defined as Xn inches, positive upstream and negative downstream. The rotation of the injector blocks relative to the datum is defined as Si inches, an arc measurement made at a radius of 44.9 inches (1.14 m) for convenience, positive is away from the tunnel centerline. The ranges of Si and Xn were extended by PoC modifications as testing proceeded. The final injector test envelope is shown in Figure 2 as a carpet plot of exit Mach number versus mass flow for the range of Xn and Si available. The shape of the plot indicates the testing trend towards higher exit Mach numbers and lower mass flows. It should be noted that the extent of the achievable injector test envelope depends on the compression ratio across the injectors, as discussed later.

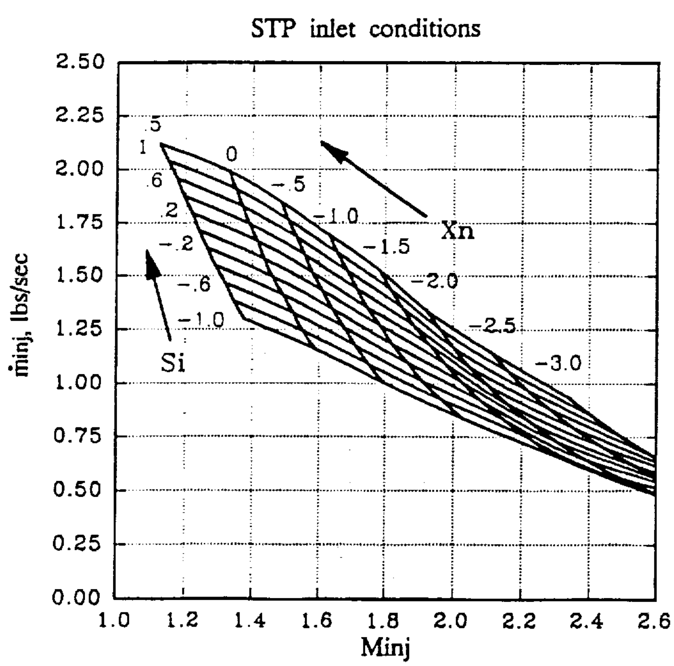

Fig. 2 - PoC primary injector test envelope.

The nozzle block and injector blocks are all 2 inches (5.08 cm) wide. The sidewalls of the 2-D injector nozzles, settling chamber, tunnel Mach 2.5 nozzle, test section, and mixing region are formed by the two cover plates. Precision fits and vacuum grease seal the flow channels.

The nozzle block is fed with regulated, dried air with a dew point of about -14°F (247 K) to avoid condensation effects in the test section. The nozzle block contains an open two-dimensional settling chamber, a 2-D Mach 2.5 nozzle, and a test section with a supersonic diffuser at its exit. The injectors draw air from the surrounding room through large intakes shown in Figure 1. The test section flow and the injector flows meet in the mixing region (a slightly convergent flow channel with a nominal inlet cross-section of 6.94 square inches - 44.77 cm^2) at the coincident supersonic diffuser and injector exits. Based on previous injector experience, the injector flows were deflected 10° towards the tunnel centerline. A subsonic diffuser, with a 7° half angle, is fitted downstream of the mixing region and channels the test section and injector flows into the FML compressor manifold.

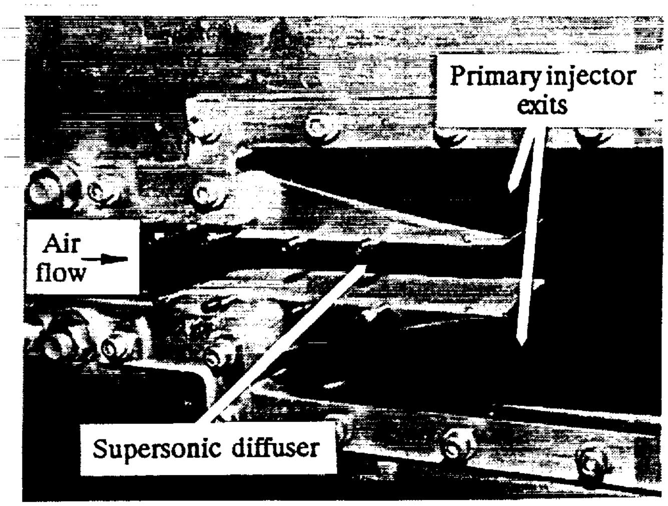

Fig. 3 - Optimum supersonic diffuser in situ.

During testing it became necessary to modify the supersonic diffuser. The original diffuser, which consisted of a 2-D compression ramp, was removed by extending the slightly divergent straight walls of the test section to the diffuser exit. (The slight divergence of the walls is to make an allowance for boundary layer growth.) This modification provided two 5.79 inch (14.7 cm) lengths of wall to which different diffusers could be attached. All but one of the 26 diffuser geometries tested were fixed and formed by gluing molded plates and spacers to the machined diffuser surface, as shown on Figure 3. The one variable diffuser was manually adjustable, so the throat height could be varied during a test. In addition, during July 1990, new cover plates incorporating Lexan-type windows were fitted to provide optical access to the Mach 2.5 nozzle, test section, supersonic diffuser and mixing region (see Figure 4a).

In January 1991, two fixed-throat ambient injectors were added to PoC (see Figure 4a) to form what we call a dual injector drive system. These secondary injectors are of similar layout to the primary injectors, as shown in Figure 4b. The injectors were designed with an exit Mach number of 2 and a mass flow of 1.648 lbs/sec (0.75 kg/sec). The injectors are attached to the downstream end of the mixing region, as shown in Figure 4a. The separation between the two injector stages is 31.24 inches (0.79 m). A new diffuser/mixing chamber connected the secondary injectors to the FML compressor manifold. For mechanical simplicity, the secondary injectors were installed at 90° to the primary (original single-stage) injectors.

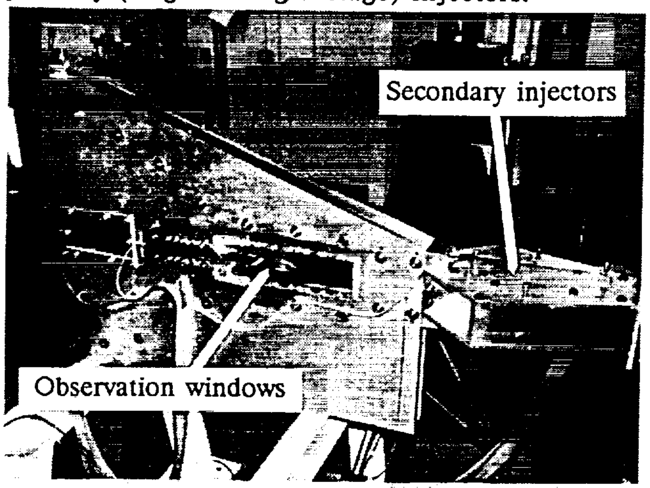

Fig. 4a - A general view of PoC with windows and the secondary injectors fitted.

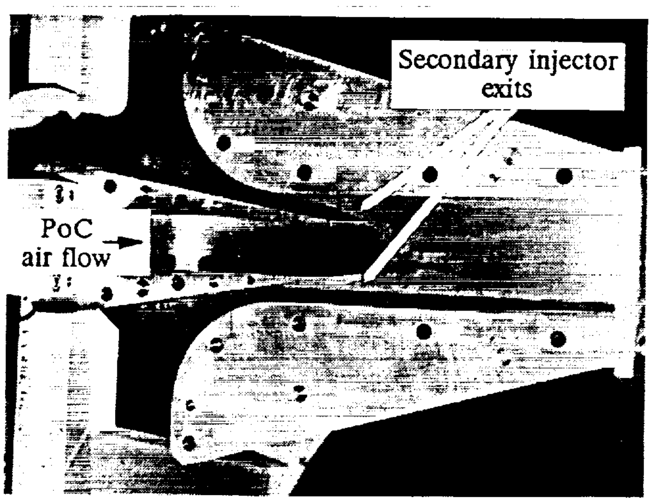

Fig. 4b - The secondary injectors with the top cover plate removed.

The instrumentation in PoC was originally 40 pressure taps connected to a mercury manometer. Twenty-six of these pressure taps were positioned along the centerline of the tunnel sidewall, through the settling chamber, test section and mixing region. Two taps were positioned at $y = \pm 1$ inch (± 2.54 cm) in the mixing region to help monitor the shear layers. The remaining 12 pressure taps were placed along the sidewall of one of the primary injectors. When the new cover plates were fitted, the number of pressure taps along the sidewall of the tunnel centerline was reduced to 9. This change left 2 taps in the settling chamber, 4 in the test section and 3 in the mixing region. When the secondary injectors were added one pressure tap was fitted upstream of the new injectors and one downstream. During testing a scanivalve/PC computer system was introduced to automate PoC data acquisition. For flow visualization, we used a novel focusing schlieren system.[3] We selected this schlieren system because its simplicity allowed rapid construction without the need for either glass windows or a point light source. Schlieren pictures were recorded on film and video at up to 1000 frames per second.

3. Computational Method

The flow through the PoC 2-D Mach 2.5 nozzle, test section, and supersonic diffuser is computed with a Navier-Stokes code, which is a modified version of ARC2D. In the ARC2D code, the Reynolds-averaged Navier-Stokes equations are solved on a generalized body-fitted coordinate system, with an approximately factorized implicit algorithm. A time-accurate version of the code was first reported by Steger.[4] Since then many improvements in efficiency and convergence, particularly for steady flows, have been made by Pulliam and co-workers. A more detailed description of the code may be found in reference 5.

Modifications were made to the upstream and downstream boundary conditions. Upstream in the settling chamber, the incoming flow is an (almost) radial flow obtained from an analytical solution (the farfield of flow through a slit) and matched to the mass flow through the nozzle throat. The method of characteristics is employed to allow outgoing disturbances to pass through the upstream computational boundary. Downstream, the method of characteristics is again employed. For situations in which the exit flow is subsonic (e.g. startup), a constant exit pressure is imposed in lieu of an upstreampropagating characteristic. When the flow at the diffuser exit is supersonic, pressure may vary in the exit plane but is assumed constant across the subsonic part of the boundary layer.

In this analysis, the primary injectors are not considered, except that we assume the injectors provide a ratio of test section total pressure to primary injector exit pressure, Po/Pinj, which is sufficiently high to obtain supersonic flow at the exit of the supersonic diffuser. For this case, the flow from the diffuser is an overexpanded jet with an exit pressure, *P*e, less than Pinj, and pressure adjustments are made external to the diffuser in a system of oblique shocks. In practice, supersonic flow through the nozzle and test section can be maintained at lower values of Po/Pinj, allowing operation at lower values of Po. For these cases, however, pressure adjustments are no longer made external to the diffuser, and subsonic flow, and usually, separation exists in the rearward portion of the diffuser. For such a situation the present method does not apply.

Most of the calculations have been performed assuming laminar boundary layers along the nozzle and test section, but with turbulent boundary layers in the diffuser. The turbulent boundary layers are modeled with the Johnson-King[6,7,8] nonequilibrium model incorporated into the modified ARC2D code.

4. Experimental Program

The experimental program evolved into three parts, after the inadequacies of the Spiegel drive system were known. The first part naturally consisted of efforts to improve the Spiegel design we had built, by independently varying the Mach number and mass flow of the single-stage injectors. We tested 25 different nozzle and injector block settings (Xn and Si combinations) covering part of the injector test envelope shown in Figure 2. For these tests, there was no optical access and the only data available was static pressure distributions. The minimum Po was found intuitively by observing the test section pressure distribution and noticing when supersonic flow was lost, i.e. unstart occurred. The accuracy of the minimum Po is therefore only about ±0.2 psia (±14 mbar), due to pressure lag in the manometer system.

Optimization of the Mach number and mass flow of the single-stage injectors was influenced by reported experiences with supersonic mixing layers,[9] since these shear layers exist in the PoC mixing region. It has been shown that a reduction of the Mach number imbalance across the shear layer improves mixing of the primary and secondary flows. In the case of PoC, the primary flow is the injector flows and the secondary flow is the test section flow, and a convective Mach number for the shear layers can be defined as (Minj - Me)/2. So in PoC, the convective Mach number could be varied by adjusting either the injector exit Mach number or the exit Mach number of the supersonic diffuser. In general, these two Mach numbers are dependent variables, since the shear layers in the mixing region cannot support a pressure imbalance. To further enhance mixing, we put boundary layer trips on the walls of the injectors and the test section, to ensure that all the boundary layers entering the mixing region were turbulent.

The purpose of a supersonic diffuser is to efficiently decelerate the flow to maximize static pressure recovery. Maximizing the diffuser exit pressure, *P*e, should allow the tunnel to operate at a lower Po, and thus a lower Re. CFD analysis (previously described) indicated that we could improve the static pressure recovery of the PoC supersonic diffuser. To study this possibility, we tested a total of 26 different supersonic diffusers in a modified PoC fitted with windows. We varied the diffuser length, shape and height guided by the observed wave patterns and knowledge from the literature. Pressure distributions were acquired together with schlieren photographs. The minimum Po for each test condition was found by observing the unstart, via the schlieren system, and by monitoring the test section pressures. Interestingly, it was also possible to hear the unstart, because the shear layer in the mixing region began to buzz with flow breakdown in the test section. Therefore, the determination of minimum Po was better than before, but was still subjective.

When the diffuser and single-stage injectors were optimized, the minimum Po was still too high. We noticed that the PoC drive system was not using as much mass flow as it could. So, we decided to investigate the use of a second-stage of ambient injectors to raise the PoC mass flow. Based on our single-stage injector experience, we hoped that the secondary injectors would reduce the static pressure in the mixing region and allow the exit Mach number of the primary injectors to be increased. The secondary injectors were sized to try and make the total PoC mass flow less than 1/64th of the mass flow capacity of the FML compressor, to allow for scale-up to LFSWT requirements. The exit Mach number of the secondary injectors was designed to be Mach 2, based on primary injector experience. Hence, we were only free to optimize the exit Mach number and mass flow of the primary injectors in the dual injector system.

5. Results Summary

The exit Mach number of the single-stage injectors was varied from about Mach 1.4 up to a maximum of about Mach 2. (The exit Mach numbers are necessarily approximate because this parameter is not measured directly.) We summarize the influence of the injector exit Mach number by comparing the PoC pressure distributions with the initial (Spiegel) injector conditions (Xn = 0; Si = 0) and the optimum single-stage injector conditions (Xn = -1.25; Si = -0.4) in Figure 5. For these two cases, the increase in exit Mach number from about 1.4 to 2 reduced *Pe* from 3.64 psia (0.25 bar) to 2 psia (0.14 bar). The injector mass flow reduced from 1.58 lbs/sec (0.717 kg/sec) to 1.033 lbs/sec (0.468 kg/sec). While, the minimum Po dropped dramatically from 22.65 psia (1.56 bar) to 12.57 psia (0.87 bar). These changes resulted in the injector mass flow ratio increasing from 3.98:1 to 4.68:1 during these tests.

In addition, the mass flow of the single-stage injectors was varied from 1.1 lbs/sec (0.499 kg/sec) to 0.83 lb/sec (0.376 kg/sec) with the injector exit Mach number held at about 2. The effect on minimum Po was very slight. Figure 6 shows a comparison of PoC pressure distributions at the extremes of the mass flow range investigated, with injector conditions (Xn = -1.5; Si = 0) and (Xn = -1.0; Si = -1.0) for high and low mass flows respectively. However, it was observed that at lower mass flows the flow in the original supersonic diffuser tended to be less steady. Further limited mass flow studies were carried out with an intermediate supersonic diffuser fitted. These tests revealed that reducing the injector mass flow by 9% from 0.926 lb/sec (0.42 kg/sec) to 0.844 lb/sec (0.38 kg/sec) increased the minimum Po from 10.1 psia (0.67 bar) to 11.5 psia (0.79 bar). Conversely, increasing the mass flow above 0.926 lb/sec (0.42 kg/sec) had little or no effect on the observed minimum Po.

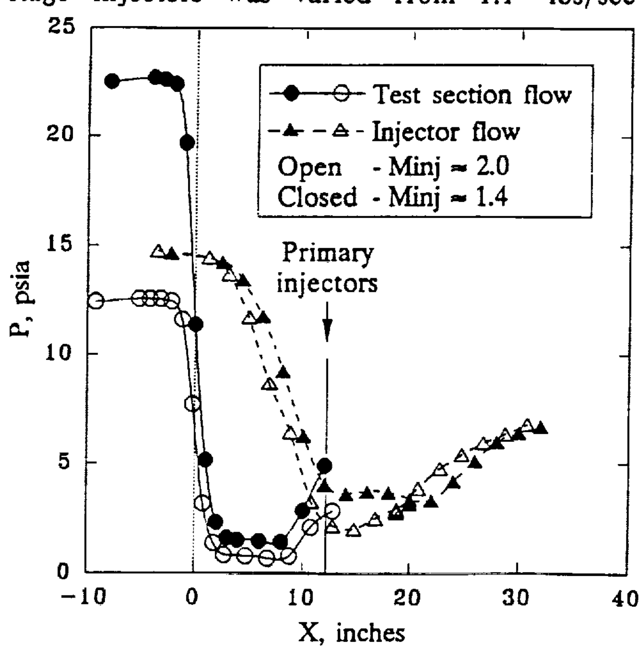

Fig. 5 - PoC pressure distributions show the effects of increasing the exit Mach number of the single-stage injectors.

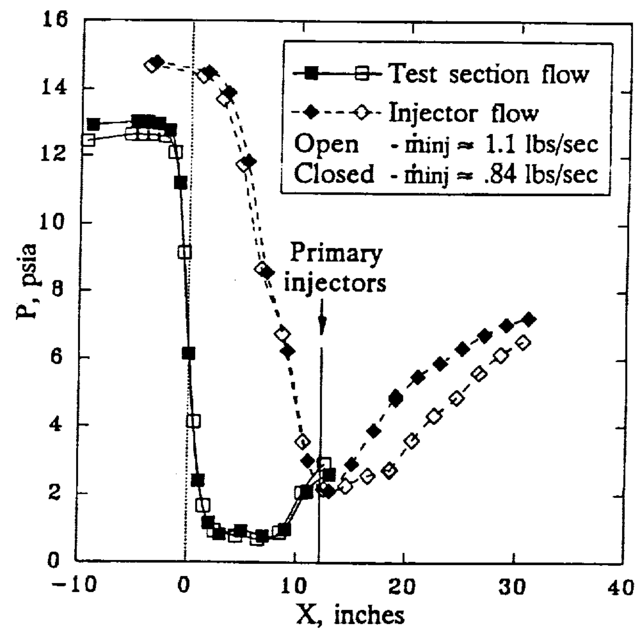

Fig. 6 - PoC pressure distributions show the effects of varying the mass flow of the single-stage injectors.

A comparison of CFD predictions and measured pressures in PoC fitted with single-stage injectors, at a minimum Po of 12.57 psia (0.87 bar), is shown in Figure 7. There is excellent agreement up to the diffuser exit and then the PoC data shows a more rapid pressure recovery than predicted. Nevertheless, CFD predictions did show that the static pressure recovery in the original supersonic diffuser could be improved. These predictions are summarized in Figure 8 by comparing the predicted centerline pressure distributions in the original and optimum diffusers. This comparison shows that the predicted static pressure recovery in the diffuser, Pe/Pts, increases from 1.42 to 1.51. Static pressures measured in PoC are also shown for comparison. The optimum diffuser flow was computed using actual Mach 2.5 nozzle coordinates while the original diffuser flow was computed using the Mach 2.5 nozzle design coordinates. This difference explains the small discrepancy in static pressure, P, in the test section.

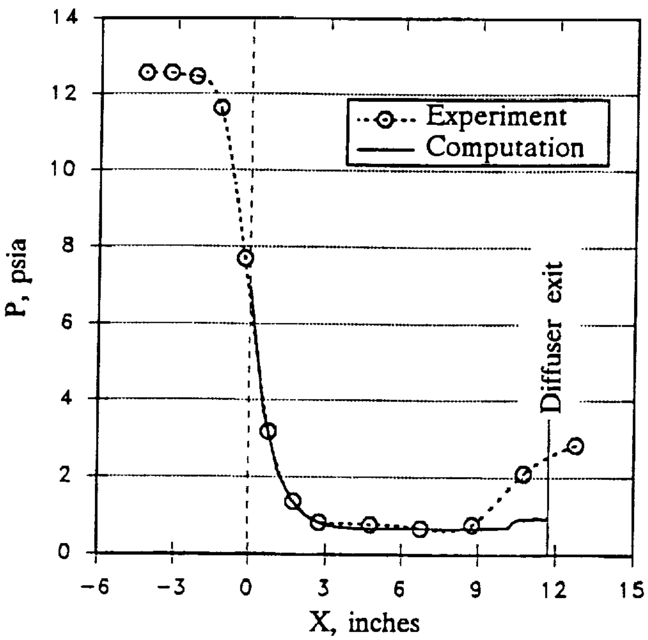

Fig. 7 - Comparison of predicted and measured centerline pressures through the PoC test section using the optimum single-stage injectors.

The shapes of the original and optimum 2-D diffusers are also compared in Figure 8. The optimum diffuser is an effectively parallel-walled type which is 5.125 inches (13 cm) long, and has a throat height of 0.76 inch (1.93 cm). The top and bottom walls of the optimum diffuser have a 1 inch (2.54 cm) long, leading-edge, compression ramp, with a thick trailing edge. Using this optimum diffuser, the minimum Po was reduced to 8.1 psia (0.56 bar) from 12.57 psia (0.87 bar) still using the single-stage injector system. An adjustable version of this diffuser was tested but any further reduction of the diffuser throat area caused unstart. We could see only faint, oscillating, wave patterns in the optimum diffuser at the minimum Po. However at higher pressures, the wave patterns became very distinct and stable as shown in Figure 9, and compare well with CFD predictions. In this case, Po was about 15 psia (1.03 bar) which is well above the minimum, so flow separations were greatly reduced, and the present computational method has proven to be an useful design tool in this situation.

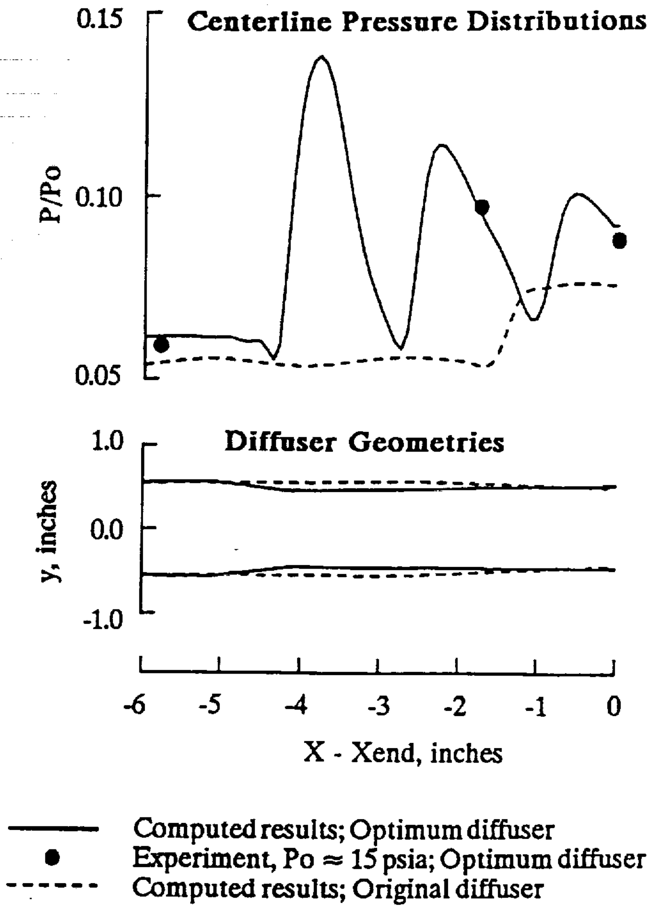

Fig. 8 - Comparison of the predicted and measured centerline pressures through the original and optimum supersonic diffusers.

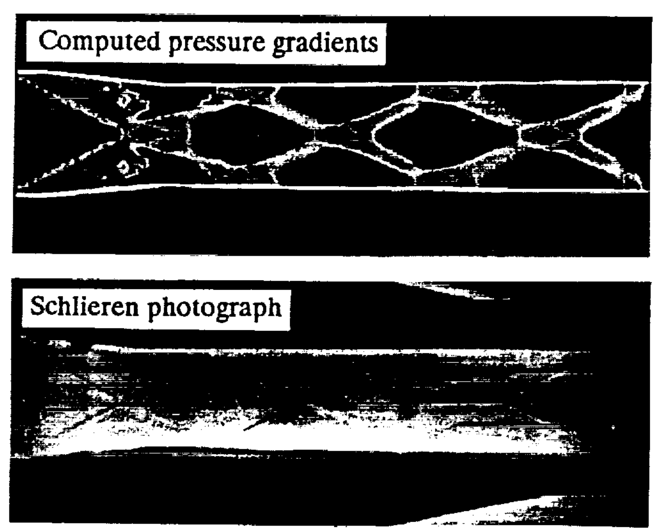

Fig. 9 - Comparison of predicted and observed wave patterns in the optimum supersonic diffuser with Po ≃ 15 psia (1.03 bar).

The introduction of a second-stage of ambient injectors significantly lowered the pressure in the downstream end of the mixing region from about 7.3 psia (0.5 bar) down to 1.57 psia (0.11 bar) as shown in Figure 10. The flow in the mixing region with the dual injectors appears to be supersonic at the downstream end, at least near the tunnel centerline. The comparison of pressure distributions along the single and dual injector systems, with Po held near constant, highlights the strong influence of the second-stage injectors. The change from single injectors to dual injectors also reduced *Pe* by 1.18 psia (81 mbar) with the primary injector exit Mach number held constant at about 2.0.

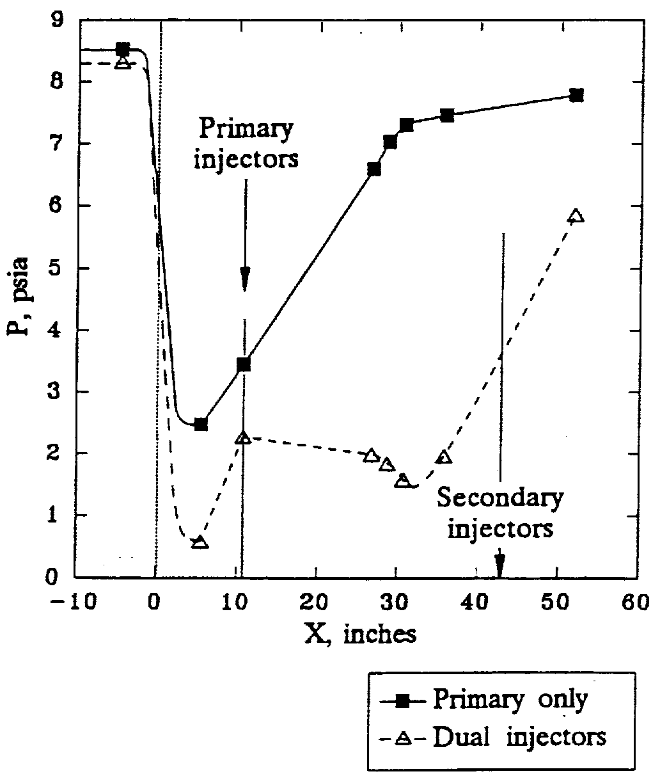

Fig. 10 - Comparison of measured PoC pressure distributions with the single and dual injector systems.

The lower pressure in the mixing region, with the secondary injectors operating, caused the local compression ratio across the primary injectors to rise, which allowed a further increase in the exit Mach number. By experiment, we found that the maximum exit Mach number was about Mach 2.4 with the dual injector system. With PoC in this configuration, the minimum Po was reduced to 5 psia (0.34 bar) with near Mach 2.5 flow in the test section, as shown by the pressure distribution in Figure 11. This unique event was confirmed by observing oblique waves in the supersonic diffuser at Po = 5 psia (0.34 bar). The wave patterns were jittery and could only be recorded as faint images, which could not be reproduced here. Also in Figure 11, the PoC pressure distribution at Po = 5 psia (0.34 bar) is compared with that of a typical wind tunnel operating with the same PE of 8 psia (0.55 bar). This comparison highlights the fact that typical tunnels can only operate with a minimum Po of about 16 psia (1.1 bar) in this situation.

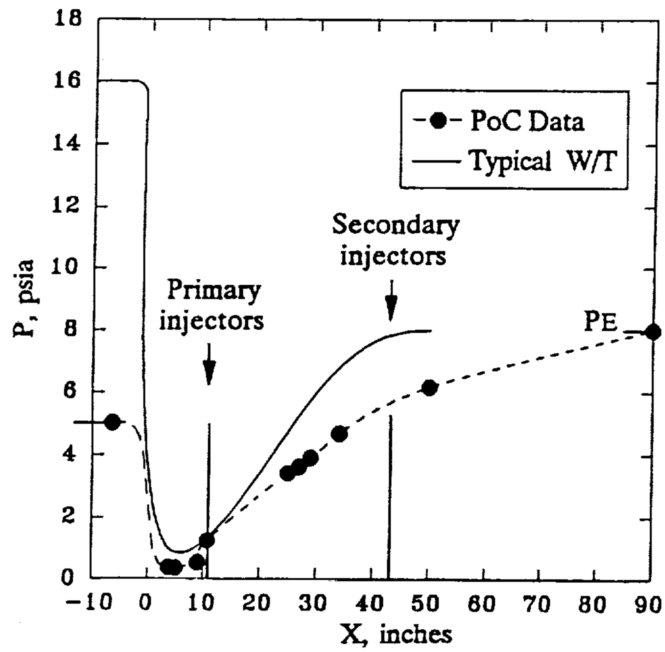

Fig. 11 - Comparison of pressure distributions in PoC operating at a minimum Po *of 5 psia (0.34 bar) with a conventional supersonic wind tunnel operating with the same* PE.

The dual injector system which achieved the unique Po = 5 psia (0.34 bar) operating condition, had the following set up. The primary injector condition (Xn = -3; Si = 0.7) equates to an exit mass flow of 0.807 lb/sec (0.366 kg/sec) at about Mach 2.4. The test section mass flow is 0.088 lb/sec (0.04 kg/sec) at Po = 5 psia (0.34 bar), so the primary injector mass flow ratio is a high 9.17:1. The secondary injector mass flow ratio is 1.84:1. Hence, the total PoC mass flow is 0.088 + 0.807 + 1.648 = 2.543 lbs/sec (1.153 kg/sec).

We found that this Po = 5 psia (0.34 bar) condition can be repeated at will, and requires no special startup or shutdown procedures. For example, either the injector flows or test section flow can be started first for a successful startup. In addition, we have not noticed any effect of changes in the ambient conditions on the performance of the dual injector system. No adjustment of the primary injector geometries is required during operation over the Po range from 5 - 15 psia (0.34 - 1.03 bar).

6. Discussion of Results

The first run of PoC at Mach 2.5 was made with the single-stage injectors set for an exit Mach number of about 1.4 and a mass flow of 1.58 lb/sec (0.717 kg/sec), according to the Spiegel design analysis. The minimum Po in this configuration was disappointingly high (at about 22.65 psia - 1.56 bar) with a correspondingly high compression ratio of 2.83:1. The maximum injector mass flow ratio was a low 3.98:1, proving that the predicted 15:1 mass flow ratio was unattainable. These results show that the Spiegel design analysis is inappropriate for the LFSWT drive system.

Despite the inaccuracy in minimum Po, the results in Figure 5 clearly show the strong influence of increasing the exit Mach number of the single-stage injectors, which allowed the minimum Po to be reduced to 12.57 psia (0.87 bar). The single-stage injector exit Mach number could not be raised above Mach 2. Hence, the local compression ratio across the injectors, P_{∞}/P_E, is theoretically about 1.4, so the effective P_E is 10.6 psia (0.73 bar), when the actual P_E is 8 psia (0.55 bar). It is highly probable that the associated reduction of the convective Mach number in the supersonic shear layer did actually improve mixing. We cannot be more specific because the exit Mach numbers of the supersonic diffuser and injectors could not be measured directly. Also, the reduction of static pressure in the injector exit, Pinj, forces *Pe* to drop, since the shear layer between the test section and injector flows cannot support a pressure imbalance. This drop in *Pe* reduces the static pressure recovery across the supersonic diffuser below the maximum possible for a given Po. Hence, we found that Po could be reduced to restore the maximum static pressure recovery across the supersonic diffuser.

Our efforts to increase the maximum static pressure recovery across the supersonic diffuser were successful as shown in Figure 8. However, we encountered limitations to our CFD analysis of the diffuser. In particular, Figure 7 shows a disagreement between theory and experiment in the diffuser exit. Interestingly, a large effective wedge angle (theoretically 17.8°) would produce the pressure rise measured in the diffuser exit for this case. With optical access, we observed that the original (convergent) supersonic diffuser allowed the static pressure in the mixing region to be fed upstream through the wall boundary layers. This forward feeding of pressure generated flow separations in the diffuser as minimum Po was approached. These separations then led to flow oscillations and premature test section unstart. Clearly, our computational method was incapable of predicting the minimum Po in this situation, so CFD alone could not be used to optimize the diffuser. In fact, the CFD analysis indicates that a short diffuser is beneficial to *Pe* (as shown in Figure 8) if flow separations in the diffuser could be somehow eliminated.

Our experimental study of 2-D diffusers found three important points. Experimentally, we found that the minimum throat area ratio of the diffuser, Adif/Ats, is 0.76, which is the same as predicted by Pope and Goin.[10] Secondly, we confirmed that the smaller the throat, the more efficient the compression due to stronger shocks in the flow. Finally, we found that the length of the diffuser delayed the upstream influences of flow oscillations at low Po, and generated a more stable shock pattern, which is conducive to low-disturbance test section flow. In fact, we confirmed the findings of Patterson[11] that the optimum length of a supersonic diffuser is about five times the test section height in 2-D flows.

The optimum diffuser turned out to be the longest diffuser we tested. The long, effectively parallel, walls appear to give the best flow stability as previously reported.[11] The influence of the thick diffuser trailing edge has not been investigated, but this could conceivably be improving the mixing of the primary injector and diffuser flows and will be retained in the LFSWT. When Po was manually oscillated about unstart, we did not observe any hysteresis in the movement of the shock train in the diffuser. The usual overpressure is not required to start.

Our limited experience with variation of injector mass flow in the single-stage injector system shows that the influence of this variable on the minimum Po is generally weak. However, the injector mass flow ratio must be kept above some threshold. Also, flow stability in the supersonic diffuser was worse at the lower mass flows. It is interesting to see in Figure 6, that the sizable static pressure changes in the mixing region, due to a change in the injector mass flow, had minimal effect on the mean test section flow. However, when testing a different supersonic diffuser within the same range of injector mass flow, a large increase of minimum Po occurred with a reduction in mass flow. Based on our limited experience, the effects of injector mass flow cannot be ignored.

The optimization of the dual injector system utilized the favorable influence of the secondary injectors on the primary injectors. We found that when the primary injectors experienced a reduction in the static pressure in the downstream end of the mixing region of 5.73 psia (0.395 bar), the exit Mach number of the primary injectors could be increased to Mach 2.4. So, in effect, the local compression ratio across the injectors was increased from 1.4:1 to 1.85:1. Hence, the effective PE of the primary injectors became 8 psia (0.55 bar), a drop of 2.6 psia (0.18 bar) from the single injector system. This effective PE would appear to be some measure of the pressure losses in the mixing region.

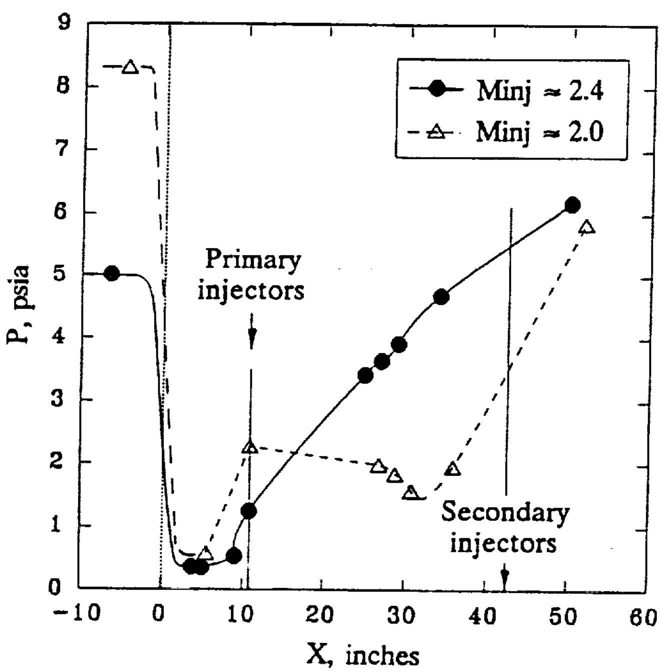

Fig. 12 - Comparison of pressure distributions in the PoC with a dual injector system and different primary injector exit Mach numbers.

In Figure 12, the static pressure in the downstream end of the mixing region is shown to have risen to 3.9 psia (0.269 bar), with the exit Mach number of the primary injectors increased to about Mach 2.4. This increased exit Mach number, for the dual injector system, reduced the convective Mach number in the mixing region shear layers to improve mixing, although this could not be quantified. But more importantly, the increase in exit Mach number reduced Pinj by approximately 0.8 psia (56 mbar) and dropped *P*e by a similar 1.02 psia (70 mbar). So, to restore the maximum static pressure recovery across the supersonic diffuser, we were able to reduce Po down to 5 psia (0.34 bar) from the 8.3 psia (0.57 bar), as shown in Figure 12. This reduction in Po corresponded to a change in Pts of 0.193 psia (13 mbar), which is consistent with maintaining a high static pressure recovery across the diffuser of about 4.4:1.

The introduction of secondary injectors definitely supported our earlier findings, that reducing the static pressure in the mixing region is very important to lowering the minimum Po. Obviously the influence of injector upon injector and injector upon test section involves some complex fluid mechanics. There is clearly an interaction between local static and total pressures in the mixing region due to shock and viscous losses. This interaction provides local compression ratios across components sufficient to support supersonic flow at uniquely low overall compression ratios. Consequently, we were able to run PoC very efficiently at overall compression ratios down to only 0.625:1.

During the course of this investigation, several areas involving complex fluid flow phenomena have been encountered. These include multiple injector flows, supersonic shear layers and mixing of multiple streams, and supersonic diffusers. As the focus of this investigation was on achieving a test Re of 1 million per foot, these phenomena were studied only to the extent necessary to attain our goal. Consequently, there remain fruitful areas of research in this respect. It is quite probable that even more efficient drive systems could be achieved from a more thorough investigation of these complex interacting flows.

7. Future Developments

The efficient PoC drive system described above will now be scaled up eight times for use with the LFSWT. We expect the quality of the drive system components to increase in the scale up, since manufacturing tolerances remain fixed. However, the mass flow of the PoC dual injector system became too large during development. Consequently, we have to reduce the total PoC injector mass flow to less than 1.906 lbs/sec (0.859 kg/sec), which scales up to less than the 122 lbs/sec (55 kg/sec) available for the LFSWT. Also, the streamwise separation between the primary and secondary injector stages may have to be reduced, due to concerns over the total length of the LFSWT. A PoC study will hopefully verify that both the size of the secondary injectors and the current injector separation distance can be reduced.

We do not expect any problems due to Reynolds number effects in the scaling up of the drive system. The relative boundary layer thicknesses will reduce as the size of the tunnel increases. Also, the higher local Reynolds numbers, due to increased run lengths, will ensure that the boundary layers in the injectors and the supersonic diffuser are turbulent. Nevertheless, we are designing some limited adjustment into the LFSWT ambient primary injectors to allow for changes in the injector flow conditions due to the intake filters, etc.

We plan to quantify the stability of the wave patterns in the supersonic diffuser at low Po using high speed photography and hot-wire measurements. Unfortunately, the PoC has no flow conditioning in the settling chamber in its current configuration. We are sure that this has a detrimental effect on flow stability. As part of our development of a low-disturbance settling chamber, we will be introducing flow conditioning into a new PoC settling chamber. We will then be able to study stability of the diffuser wave patterns in a better simulation of the LFSWT environment.

8. Conclusions

1) An uniquely efficient drive system has been developed, which allows a pilot Mach 2.5 tunnel to operate continuously at compression ratios down to 0.625:1.

2) The PoC drive system does not require any overpressure to start and no hysteresis has been observed.

3) The novel focusing schlieren system proved to be very effective. The simplicity of the system was extremely important to the rapid progress of this project.

4) The ability to use non-specialist indraft compressors (available in many aeronautical research centers) with a supersonic wind tunnel may allow more wind tunnels (of the LFSWT type) to be built to meet the challenge of designing the next generation of supersonic transports.

Acknowledgments

The authors wish to acknowledge the important contributions of Mr. Robert Meneely and Dr. Daniel Reda. The decision to use an injector drive system is due to discussions held between S. Davis and N. Riise in May 1988.

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APPENDIX B

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AIAA Paper 92-3909
DEVELOPMENT OF THE NASA-AMES LOW-DISTURBANCE SUPERSONIC WIND TUNNEL FOR TRANSITION RESEARCH UP TO MACH 2.5

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DEVELOPMENT OF THE NASA-AMES LOW-DISTURBANCE SUPERSONIC WIND TUNNEL FOR TRANSITION RESEARCH UP TO MACH 2.5

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Abstract

A unique, low-disturbance supersonic wind tunnel is being developed at NASA-Ames to support supersonic laminar flow control research at cruise Mach numbers of the High Speed Civil Transport (HSCT). The distinctive aerodynamic features of this new quiet tunnel will be a low-disturbance settling chamber, laminar boundary layers on the nozzle walls and steady supersonic diffuser flow. Furthermore, this new wind tunnel will operate continuously at uniquely low compression ratios (less than unity). This feature allows an existing non-specialist compressor to be used as a major part of the drive system. In this paper, we highlight activities associated with drive system development, the establishment of natural laminar flow on the test section walls, and instrumentation development for transition detection. Experimental results from an 1/8th-scale model of the supersonic wind tunnel are presented and discussed in association with theoretical predictions. Plans are progressing to build the full-scale wind tunnel by the end of 1993.

Symbols

Cp	Pressure coefficient ($\gamma PMe^2/2$)
Me	Free stream Mach number
P	Local static pressure
Po	Tunnel stagnation pressure
P_E	Exit (manifold) total pressure
Prms	Pressure measurement rms
Re	Unit Reynolds number per foot
To	Tunnel stagnation temperature
u	Local velocity in boundary layer
Ue	Free stream velocity
X	Streamwise position relative to Mach 2.5 nozzle throat station (positive downstream)
γ	Ratio of specific heats

1. Introduction

Aerodynamicists now consider the use of a low-disturbance or "quiet" wind tunnel as an essential part of meaningful boundary layer transition research at supersonic speeds. This realization is based on many years of experience with old "noisy" supersonic wind tunnels, and a growing respect for the pioneering research of Laufer[1,2] at the Jet Propulsion Laboratory (JPL) from the mid-1950s to the early-1960s, and the work of Pate and Schueler[3] in the late-1960s. This situation has provided the impetus for the development of a new, unique, continuously-operating Laminar Flow Supersonic Wind Tunnel (LFSWT) in the Fluid Mechanics Laboratory (FML) at NASA-Ames. This LFSWT concept is based on the now decommissioned (but soon to be rebuilt) JPL 20-inch supersonic wind tunnel, which is the first documented quiet supersonic wind tunnel.[4] The proposed test envelope for the LFSWT was chosen to cover a significant portion of the HSCT operating envelope with a Re range of 1 to 3 million per foot and a Mach number range from 1.6 to 2.5. Also, the LFSWT test envelope will cover the test conditions flown by NASA F-16XL aircraft in support of Supersonic Laminar Flow Control (SLFC) studies, as shown in Figure 1.

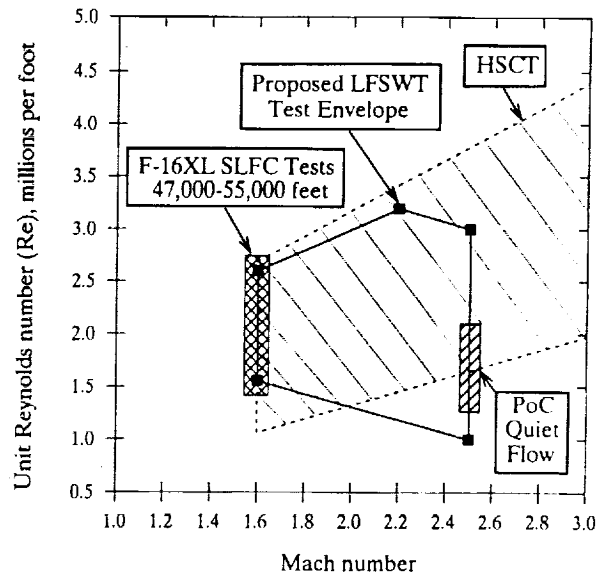

Fig. 1 - Proposed LFSWT test envelope compared with the flight envelopes of the HSCT at cruise and the F-16XL SLFC flight tests.

The LFSWT is currently being designed as a research tunnel with an 8 inch (20.32 cm) high, 16 inch (40.64 cm) wide and 32 inch (81.28 cm) long test section, sized to operate at mass flows up to 21 lbs/sec (9.5 kg/sec). The use of existing support equipment (the FML indraft compressor and the NASA-Ames 3000 psi (207 bar) dry air supply) will significantly reduce the project costs, and will allow the LFSWT to be brought on-line more rapidly to impact the critical technology development phase of the HSCT before 1997.

The decision to use the FML non-specialist indraft compressor to power the LFSWT created several technical concerns. The FML compressor has a measured capacity of 228,000 icfm (about 143 lbs/sec - 65 kg/sec with a minimum P_E of 8 psia - 0.55 bar) and a pressure ratio of 1.8:1. Consequently, to achieve the low end of the Re range, the LFSWT must operate with a Po which is less than the minimum P_E. This means that the LFSWT compression ratios will be uniquely less than unity (Po/P_E down to 0.625:1 with Re = 1 million per foot at Po = 5 psia - 0.34 bar). So, the utilization of the FML compressor precludes the use of a conventional drive system to achieve the desired Re range. Consequently, a novel drive system was developed using an 1/8th-scale model of the LFSWT, which we call the Proof-of-Concept (PoC) supersonic wind tunnel. The initial PoC drive system is described in detail by Wolf et al[5] and requires less than half of the normal run compression ratio. The drive system works by using compressor mass flow capability (which greatly exceeds the mass flow necessary for the test section flow alone) to drive two stages of ambient injectors, which pull the flow through the test section at low Po. Two stages of injectors became necessary so that the primary injectors could operate at a higher Mach number, which then lowered the exit pressure of the test section flow and allowed the PoC to operate at a lower Po.

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This paper contains a brief description of PoC and its recent modifications for drive system tuning and quiet flow studies to aid the LFSWT design process. We describe the ongoing combination of theoretical and experimental research efforts to ensure there is quiet flow in the LFSWT. While we use the PoC for laminar flow studies, we are also developing and gaining experience with the latest instrumentation for transition research. This experience will aid our development of quiet nozzles, improve flight test measurements, and also give FML the tools required for future transition research when the LFSWT comes on-line. This activity is discussed with particular reference to hot-wires, hot-film gages, focusing schlieren, and liquid crystal coatings. We intend that this paper should help others engaged in supersonic transition research by outlining the important aspects of developing a *State-of-the-Art* supersonic transition research facility.

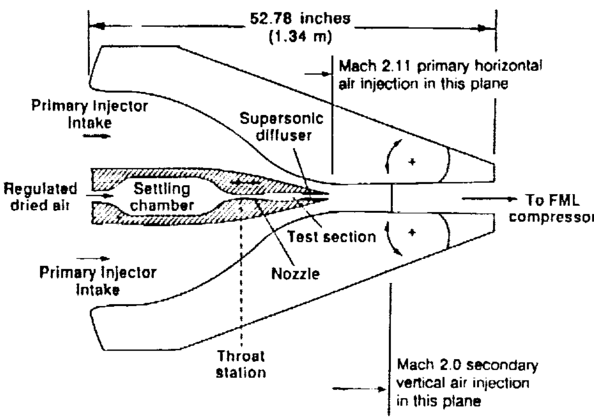

Fig. 2 - A schematic layout of the PoC supersonic wind tunnel.

2. Tunnel Hardware Development

The aerodynamic lines of the LFSWT are being studied with the aid of the PoC. A schematic of the PoC layout is shown in Figure 2 to illustrate the novel dual-stage injector drive system. It should be noted that the two stages of injectors are orientated at right angles to one another, from practical considerations. The PoC test section is 1 inch (2.54 cm) high and 2 inches (5.08 cm) wide. The only nozzle tested so far is a two-dimensional, fixed-block, Mach 2.5 type, designed according to the methodology of Riise[6] used at JPL. The nozzle design is considered long, with the surface curvature minimized. The nozzle has a throat to exit length of 5.114 inches (13 cm), with a throat height of 0.38 inch (9.65 mm). The nozzle and test section are made from 6061-T6 aluminum. The flow surfaces along the nozzle are hand finished to about a 2L standard (roughness height 2 microinches - 0.05 micron). We consider the laminar flow requirements for the nozzle surface finish at low Re to be less stringent than those required for the Mach 3.5 Langley Pilot Quiet Tunnel.[7] A two-dimensional nozzle was chosen to minimize focusing of disturbances, due to shape imperfections, on the tunnel centerline, and also to allow complete optical access to the nozzle and throat for transition studies associated with wind tunnel development. The three-dimensional PoC contraction is 6 inches (15.24 cm) long on the floor and ceiling and 2.5 inches (6.35 cm) long on the sidewalls. The sidewall contractions are shorter to make the sidewalls parallel upstream of the nozzle throat for optical access.

The test section is fed with regulated, dried air which has a dew point of about -50°F (227 K) from the existing NASA-Ames 3000 psi (207 bar) supply. Of course, the dried air is essential to eliminate any condensation effects in the test section, as found in the experimental results discussed later. The PoC dual-stage injectors draw in ambient air from the surrounding room. The exit Mach number of the primary injectors is 2.11, while the secondary injectors operate at Mach 2. The air mass flow ratio between injectors and test section rises to a massive 27:1 at the minimum Po of 5.4 psia (0.37 bar).

The secondary injectors were originally positioned for convenience 31.24 inches (0.79 m) downstream of the primary injectors. To shorten the overall drive system, the secondary injectors were redesigned to allow the separation between injector stages to be reduced to a minimum of 6.46 inches (16.41 cm). The new secondary injectors are shown with minimum stage separation in Figure 3. In addition, a family of secondary injector nozzle blocks was made to study the reduction of injector mass flow from the reported 1.648 lbs/sec to 1.099 lbs/sec (0.747 kg/sec to 0.498 kg/sec respectively), with the exit Mach number fixed at Mach 2, based on previous PoC experience.[5]

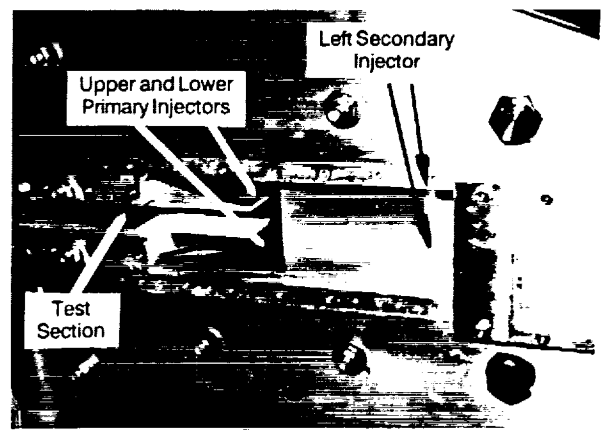

Fig. 3 - The relative position of the primary and secondary stages of ambient injectors in the PoC, with the right-hand secondary injector and window removed.

The PoC was initially fitted with an open two-dimensional settling chamber. This simple settling chamber was only adequate for drive system studies. We have now installed a larger three-dimensional settling chamber equipped with multiple flow straighteners and conditioners and a contraction ratio of 12:1 (based on test section area) for low-disturbance operation. A schematic of the settling chamber is shown in Figure 4, highlighting its modular design, which allows component holder interchangeability. The flow velocity in the settling chamber is 20 fps (6.1 m/sec) with Po = 15 psia

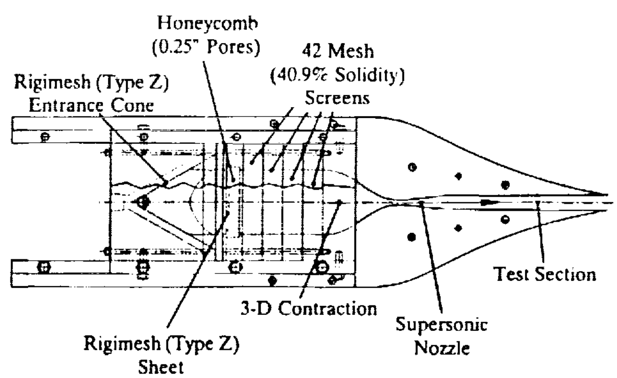

Fig. 4 - Schematic of the new PoC settling chamber.

(1.02 bar) and To = 50°F (283 K). Figure 5a shows the new PoC settling chamber in situ. The settling chamber design is based on knowledge of the literature, in particular the work of Beckwith.[8] The versatile design can accommodate boundary layer suction upstream of the contraction, should this prove necessary.

The associated three-dimensional contraction was made integral with a new Mach 2.5 nozzle (the same shape as the original nozzle[5]) and a new longer test section/supersonic diffuser (see Figure 5b). This design removes all hardware joints on the nozzle floor and ceiling upstream of the test section. The shape of the contraction was calculated using a fifth-order polynomial, with zero surface slope and curvature at the upstream and downstream ends. The new test section is 4 inches (10.16 cm) long (compared to the original length of 0.665 inch - 1.69 cm) with slightly diverging floor and ceiling. The supersonic diffuser[5] is unchanged except the ramp height was increased by 0.019 inch (0.48 mm) to maintain a throat height of 0.76 inch (1.93 cm).

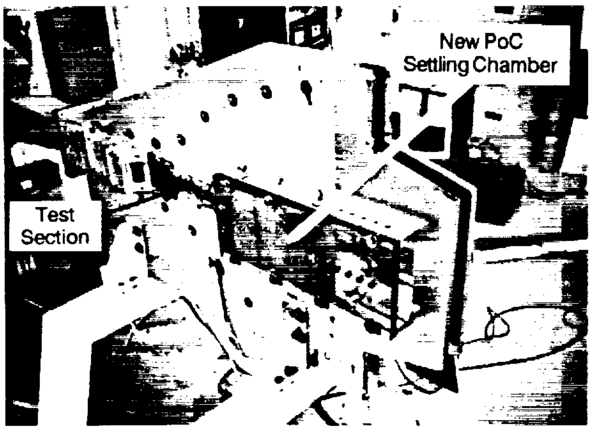

Fig. 5a - The new PoC settling chamber in situ.

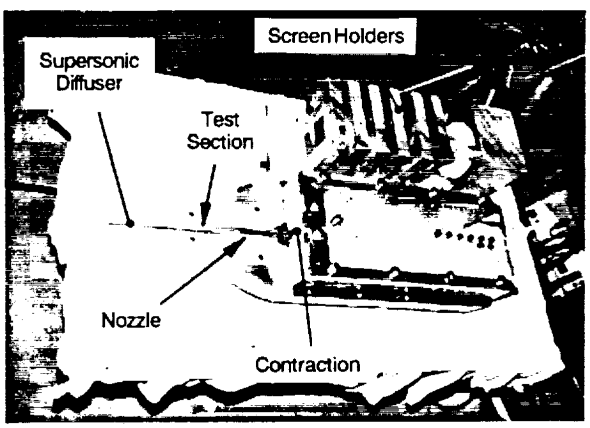

Fig. 5b - A display of new PoC settling chamber components.

We use a porous material in the settling chamber to provide both isolation from upstream air supply noise and turbulence, and a means to spread the inlet pipe flow into the settling chamber with minimum disturbances. To this end, we utilize both a cone and flat sheet of Rigimesh type-Z material, which is 0.009 inch (0.23 mm) thick and has a pore size of approximately 39 microinches (1 micron). The pressure load on the 60° cone is supported by a perforated sheet on the downstream side of the cone. This perforated sheet is sufficiently open to minimize flow blockage. The flat sheet is supported by a 1 inch (2.54 cm) thick honeycomb sheet with a 0.125 inch (3.17mm) cell size.

The honeycomb sheet is followed by 4 screens each made from 42-mesh stainless steel cloth with 40.9% solidity. The screen separation is equivalent to 63 mesh lengths, which is more than the 50 mesh lengths required for small structure turbulence decay according to Groth and Johansson.[9]

As part of the continuing improvement of tunnel controls, a new Po control system was installed along with the connection to the 3000 psia (207 bar) dry air supply. The Po control system is based on a Fisher DPR-900 integral controller which monitors Po and drives the PoC air regulator. The system allows Po to be set rapidly and held within an accuracy of 0.05 psia (0.0034 bar).

The instrumentation used in the PoC includes pressure taps for steady-state measurements, and hot-wires (single 4 and 5 micron Tungsten wire types), Kulite (XCS-093) pressure transducers, and TSI (Model 1237) platinum hot-film gages for dynamic measurements. The static pressures are measured using a scanivalve system connected to a standard PC A/D converter card. The hot-wires are powered by FML's own constant-temperature bridge circuit with the output signal fed to a Tektronix 2642A Fourier Analyzer system, as are all the dynamic measurements. The Kulites are powered by high frequency response signal conditioners (Dynamic 8000s with a 3dB dropoff at 500KHz). The hot-film gage is powered by a constant-current bridge devised by Demetriades at Montana State University. The Tektronix 2642A Fourier Analyzer system can sample an input signal at up to 512KHz with 16-bit resolution, and provide 4096-point real-time FFTs, data capture and display. All data is then collected on to a PC computer for data archiving, post processing and data presentation.

Dynamic measurements can be made in either the test section or in the settling chamber. In the test section, the hot-wire is buried in the supersonic diffuser molding to minimize blockage, as shown in Figure 6. The hot-wire probe protrudes 0.625 inch (15.9 mm) upstream into the test section, at an X location of 8.375 inches (21.27 cm), and sits about 0.069 inch (1.75 mm) above the test section floor. A Preston tube with a 0.029 inch (0.73 mm) outside diameter was fitted in place of the test section hot-wire for some tests. The hot-film gage was flush mounted in the left sidewall, on the test section centerline, at an X location of 6.69 inches (16.99 cm).

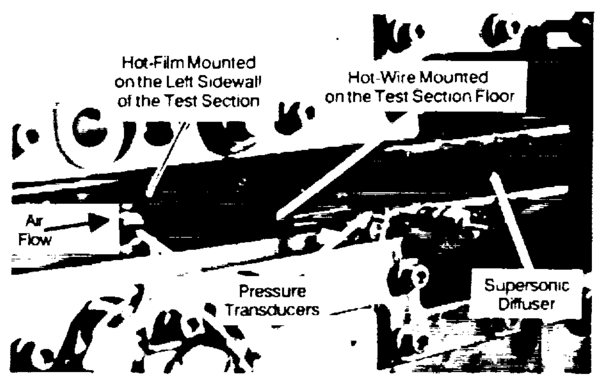

Fig. 6 - Hot-wire and hot-film instrumentation mounted in the PoC test section.

In the settling chamber, a special instrumentation holder block allows two probes to be mounted side-by-side and inserted in any holder location. Three interchangeable traversable probes are available: pitot pressure probe fitted with a Kulite; a temperature probe fitted with a type-T thermocouple; and a hot-wire probe fitted with a 4 micron Tungsten wire. These probes can be used for detailed mapping of the flow field at any location in the settling chamber.

The PoC polycarbonate Lexan-type windows have been used with a focusing schlieren system[10] to observe wave patterns in the supersonic diffuser and mixing region. Alternatively, one window can be replaced by an aluminum blank for use of shear-stress-sensitive liquid crystal coatings (discussed later).

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3. Numerical Research

In attaining a quiet environment necessary for transition studies in a supersonic wind tunnel, there are two main sources of disturbances which need to be carefully addressed and minimized to the extent possible. One is free stream turbulence arising from the settling chamber and upstream piping. Another significant source is the sound field radiated by turbulent tunnel-wall boundary layers. Pate and Schueler[3] and others have shown the adverse effect of radiated noise on transition Reynolds numbers at supersonic speeds. For the LFSWT, it is therefore desirable that the boundary layers remain laminar within the nozzle and test section as far as possible. Malik[11] and others have shown that compressible stability theory with the e^N method predicts boundary layer transition onset arising from Tollmein-Schlichting (TS) waves and Görtler vortices. For sufficiently small free stream turbulence levels in the tunnel, the value of N may approach 10, which is the value associated with high altitude flight in the quiescent atmosphere. Stability calculations within the present context may then serve two purposes: (1) as a predictive tool in designing the nozzle and test section; and (2) as a diagnostic tool in analyzing the experimental results.

The flow through a two-dimensional nozzle, test section, and supersonic diffuser is analyzed computationally with three different codes in order to predict both the mean flow and boundary layer stability and transition. A Navier-Stokes (NS) code, previously described by Wolf et al[5], is used to predict the mean flow quantities in the tunnel. For purposes of analyzing the stability characteristics of the wall boundary layers, the mean flow is assumed laminar in the nozzle and test section, but with turbulent boundary layers in the supersonic diffuser. A boundary layer code by Harris and Blanchard[12] is next employed to provide detailed boundary layer quantities and derivatives for use by the stability code of Malik,[11] since the resolution requirements to accurately obtain first and second derivatives in the boundary layer are not easily met with a NS code. The Malik code uses linear spatial stability theory to analyze the stability of two-dimensional and axisymmetric, compressible wall-bounded flows. Wall curvature is accounted for, so the analysis considers both TS waves (1st, 2nd, etc. modes) and Görtler vortices. Transition onset is predicted with the e^N method.

The PoC/LFSWT nozzle in the present study was intentionally made long so that instabilities arising from curvature effects would not cause transition. This decision was supported by the study of Wolf.[13] Calculations indicate that this approach was successful, in that the maximum N factor due to Görtler vortices thus far computed is less than 4. No significant TS instabilities at the PoC operating conditions have yet been found numerically.

4. Experimental Program

LFSWT drive system tuning has now continued beyond the initial drive system design studies, which successfully demonstrated that Mach 2.5 flow could be achieved over the desired Re range.[5] This additional tuning became necessary to address concerns over the drive system length and the ability of the FML compressor to provide sufficient mass flow. The PoC was used to carry out the necessary drive system tuning. For this purpose, the PoC was modified to allow the separation between the two injector stages to be varied and the mass flow of the secondary injectors to be reduced. Both these parameters were previously fixed on the PoC.[5]

Following the drive system tuning, the experimental program has focused on studying quiet flow in the PoC. Preliminary flow measurements were made in the settling chamber and the extent of natural laminar flow that exists along the PoC test section walls has been documented at Mach 2.5. Of course, the existence of laminar flow on the nozzle walls is a critical element of a quiet supersonic wind tunnel. Our intent with the LFSWT is to go beyond this requirement and obtain laminar flow throughout the test section. This situation will eliminate the existence of a test rhombus bounding the quiet flow, which will allow testing anywhere in the test section. This means that the model will not have to be positioned in a variable test rhombus, which greatly simplifies the method of model support.

Initially, we are concerned with obtaining natural laminar flow on the nozzle and test section walls using passive laminar flow control. These passive means are a low-disturbance free stream, a low curvature, long nozzle and a smooth wall finish. The documentation of natural laminar flow, using the solid block Mach 2.5 nozzle, is the first stage of an ongoing verification of the LFSWT test envelope.

For the quiet flow studies, the PoC was fitted with a new low-disturbance settling chamber/nozzle/test section, instrumentation for dynamic measurements, and a closed-loop control system for setting and maintaining Po. Dynamic flow measurements in the test section and settling chamber were then made to document the flow quality in PoC over the entire Re range. To assist with verification of our instrumentation, the settling chamber was degraded and the associated effects on laminar flow in the PoC test section were documented and are discussed later.

5. Instrumentation Development

5.1 Hot-Wire

The use of hot-wires is well documented but still requires considerable operator interpretation, particularly at supersonic speeds.[14] We use a 5 micron Tungsten wire built at NASA-Ames in our supersonic testing. This wire type is durable and has a typical calibrated response rate of 15KHz, using a square wave with the wind off. During tunnel operation, the probe is in the outer portions of the floor boundary layer and can only be calibrated when laminar flow is present. However, the response calibration does not change from wind off to wind on in this situation. Nevertheless, we are currently unable to calibrate the output of the hot-wire to aerodynamic parameters, so our data are only qualitative at present.

As PoC testing has progressed, we have gained experience with the use of hot-wire instrumentation. The new FML constant-temperature anemometer has worked flawlessly and provides a high level of adjustability. Wind-off signal noise is extremely low. By experience, we have found that the signal rms can be best recorded as an average of 20 samples taken without interruption. Our waveform analyzer requires less than a second to perform this average of 20 4096-point FFTs under PC software control. The signal spectrum is then available for storage and printing.

5.2 Hot-Film

Hot-films are well know detectors of shear stress. We employed a commercially available hot-film gage mounted on a cylindrical glass substrate. The heat-sink effect associated with this configuration (run as a constant temperature sensor) was found to be very large. This finding necessitated the building of a specialist constant-current circuit to drive the sensor and maintain a low output signal DC voltage for ease of measurement.

Concern over the repeatability of the hot-film data from the PoC led to an independent transition-detection calibration of the hot-film in another quiet supersonic wind tunnel at Mach 3. This calibration was undertaken by the Montana State University and involved the hot-film being exposed to laminar, transitional and turbulent boundary layers. However, this calibration only allows us to qualitatively assess the hot-film data from the PoC.

5.3 Focusing Schlieren System

Based on the pioneering work of Weinstein[10] at NASA-Langley, a focusing schlieren system has been developed for use with the PoC. The main features of this system are:

1) The windows do not have to be made of schlieren-quality glass, any transparent material is good and in this application polycarbonate windows are used.

2) Thin slices of the flow can be observed with similar resolution to conventional schlieren systems.

3) Mirrors are not required.

4) Simple setup allows view changes at will.

5) A point light source is not required.

These features have proven to be very important to this project and have allowed flow visualization to occur in a timely manner and to change rapidly with research needs.

The concept was developed back in the late 1940s and provides a very versatile system ideal for research. The PoC system has been used to observe the drive system performance in the supersonic diffuser[5] and in the mixing region. We are currently attempting to use this schlieren system to observe boundary layers and to detect transition to turbulence. For this purpose, the focusing schlieren system is being enhanced with the addition of a high intensity spark illumination and cylindrical lenses for boundary layer magnification.

5.4 Liquid Crystal Coatings

The liquid crystal coating technique is a method for visualization of surface shear stress patterns in both steady and transient flows, as reported by Smith[15] and Reda.[16] In the present application, one of the PoC windows was replaced by an aluminum (black) insert and the flow surface was coated with a shear-stress-sensitive/temperature-insensitive liquid crystal film. The coated areas (in the supersonic diffuser and mixing region) were obliquely illuminated by white light through the opposite window. Then the color-change response of the liquid crystal film to surface shear stress events was photographed on video and movie film. Framing rates from 30 to 1000 images/sec were utilized.

We have tested the frequency response of the newly-formulated liquid crystal compound (Hallcrest BCN/192) by using the PoC startup and off-design operation to create highly transient flows. During these tests, all boundary layers on the nozzle and sidewall surfaces were turbulent because the low-disturbance settling chamber had not yet been installed. These observations showed the liquid crystal coating response time to be less than, or equal to, the time between sequential images taken at 1000/sec (i.e., one millisecond).

6. Experimental Results

6.1 Drive System Tuning

Since the last report on the PoC drive system[5], measurements in the primary injector exits show that the actual Mach number of the primary injectors is 2.11. This is significantly less than the previously estimated Mach 2.4 and shows that the influence of the second stage of injectors is much smaller than previously thought. Nevertheless, the PoC drive system continues to operate over the desired Po range with a P_E of 8 psia (0.55 bar).

The movement of the secondary injectors upstream towards the primary injectors had no noticeable effect on the performance of the PoC. The reduced separation distance of 6.46 inches (16.41 cm) was sufficiently long to allow 2 wave reflections in each of the primary injector flows, above and below the test section flow, as shown in Figure 7. The comparison of static pressures (shown in Figure 8) indicates that the test section flow was not affected by the secondary injector movement. This shortening of the PoC drive system will result in a 198 inch (5.03 m) reduction in the length of the LFSWT. Unfortunately, both sets of PoC data indicate that the test section Mach number was reduced below 2.5.

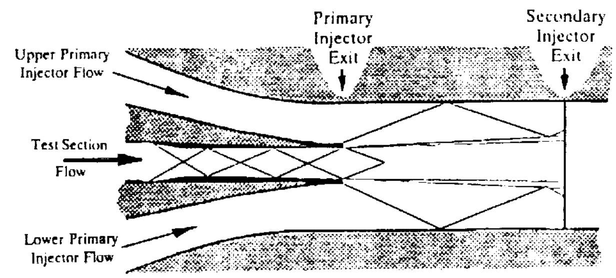

Fig. 7 - Schematic of the shock patterns in the mixing region between the PoC injector stages.

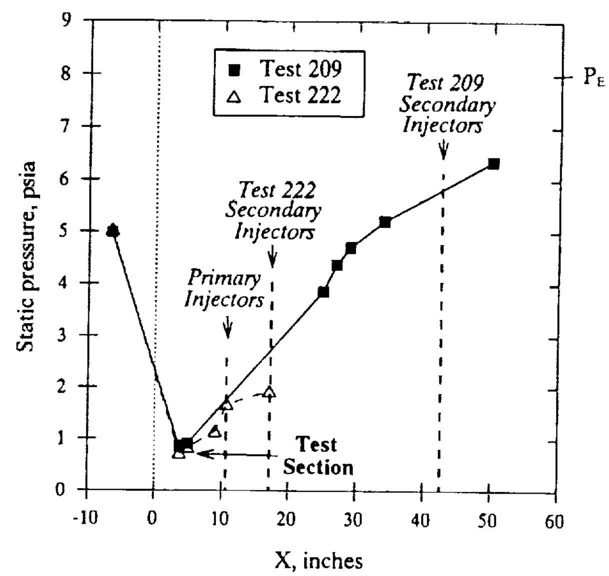

Fig. 8 - Comparison of PoC pressures with different secondary injector locations, at a minimum Po of 5 psia (0.34 bar).

This loss of desired test Mach number was traced to the degradation of our temporary air drier prior to these tests. The resulting condensation effects in the nozzle (which were not visible to the operators) actually caused the test Mach number to go down. Once the PoC was connected to the NASA-Ames 3000 psi (207 bar) dry air supply, good air quality was restored and the test Mach number returned to 2.5. Furthermore, we now use a hydrometer to continuously monitor the dew point of the inlet air to check for sufficient dryness, which we define as a dew point of less than -15° F (247 K).

The drive system tuning continued with a study of the effects of reducing the mass flow of the secondary injectors. This was an attempt to lower the overall mass flow requirement of the LFSWT drive system. We reduced the PoC secondary injector mass flows in stages (by 11%, 22% and 33%) and found that the minimum Po for Mach 2.5 operation had risen for each reduction in mass flow. Adjustment of the primary injectors failed to produce any significant improvement in the minimum Po. This effort confirmed that the LFSWT drive system for Mach 2.5 operation requires up to 184 lbs/sec mass flow at a maximum Po of 15 psia (1.02 bar), if the Po range from 5 to 15 psia (0.34 to 1.02 bar) is to be preserved with P_E = 8 psia (0.54 bar).

6.2 Quiet Flow Studies

6.2.1 Settling Chamber

The new PoC low-disturbance settling chamber (previously described) has been operated over a Po range from 5 to 15 psia (0.34 to 1.02 bar). This Po range corresponds to a mass flow range of 0.097 lbs/sec (0.044 kg/sec) to 0.358 lbs/sec (0.162 kg/sec) for To = 50°F (283 K). The static pressure distributions across the components of the settling chamber are shown in Figure 9 for different Po. It can be seen that the maximum pressure drop of about 2.5 psia (0.17 bar) occurs across the flat sheet of Rigimesh. The Rigimesh cone supports minimal pressure load, which simplifies the necessary support structure for the full-scale LFSWT cone.

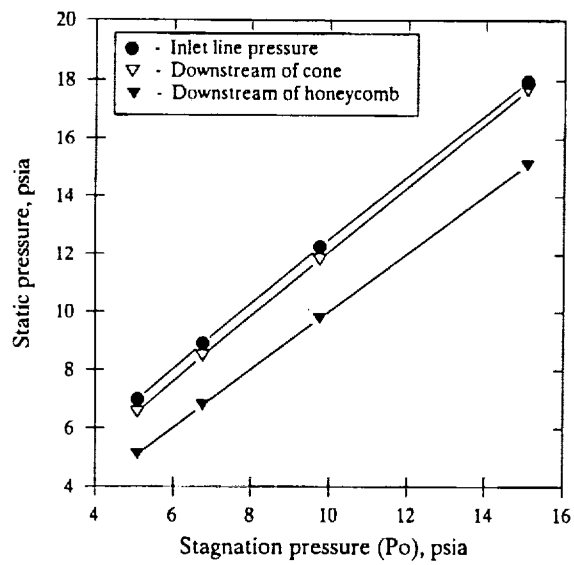

Fig. 9 - Pressure distributions through the PoC settling chamber.

Preliminary flow disturbance measurements were made in the plane of the settling chamber exit (at a single location on the tunnel centerline) using a Kulite total pressure probe and a 4 micron Tungsten hot-wire. The Kulite data are shown in Figure 10 over the Po range for two settling chamber configurations, with and without the honeycomb and Rigimesh sheet installed. The ratio of the Prms with Po shows a significant rise with the honeycomb and Rigimesh sheet removed. This pressure ratio drops with increasing Po. With all the settling chamber components in place, the pressure fluctuations are of the order 0.1%. The sharp increase in pressure ratio at low Po has been traced to tunnel leaks which caused unstarting of the nozzle flow.

The hot-wire data from the settling chamber are shown in Figure 11. Again, about a fourfold increase of signal rms is associated with the removal of the honeycomb and Rigimesh sheet. The signal levels, with all the settling chamber components in place, are reasonably low compared to the 0.7 mV wind off noise level. However, in the absence of a hot-wire calibration of volts-vs-velocity, these data can only be discussed qualitatively.

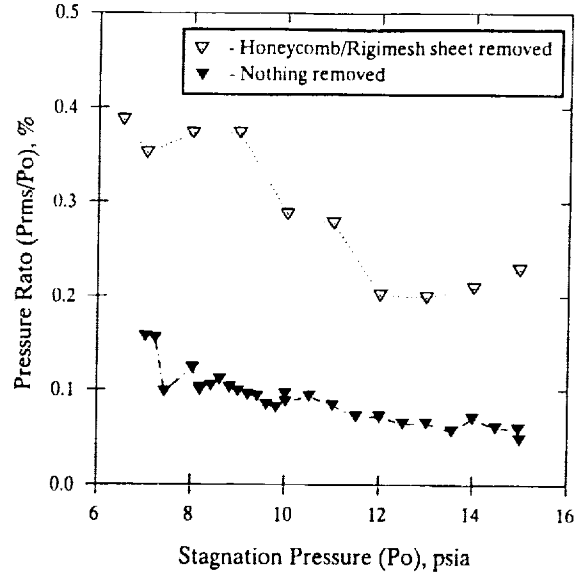

Fig. 10 - Summary of Kulite pressure data from the PoC settling chamber.

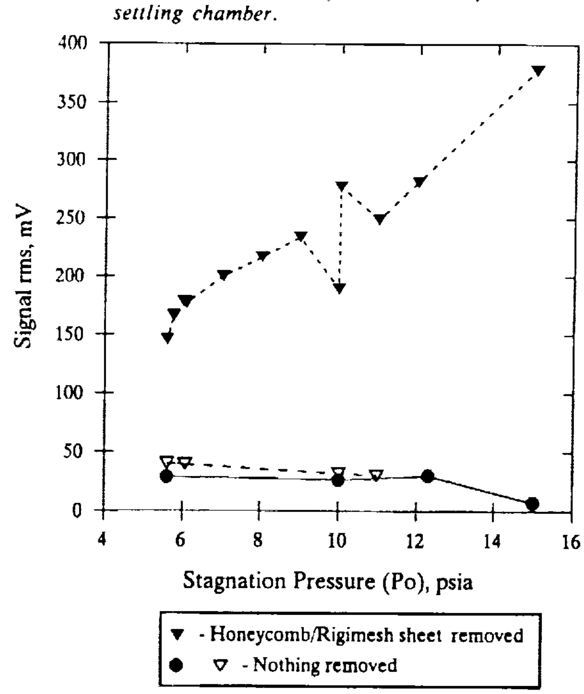

Fig. 11 - Summary of the uncalibrated hot-wire data from the PoC settling chamber.

6.2.2. Laminar Flow

Our laminar flow studies involve the use of different types of instrumentation to confirm the state of the test section boundary layer. The detection of boundary layer transition tends to be qualitative and our goal was to find at least 2 measurement techniques which agreed about the location of transition.

We found that the hot-wire measurements made above the PoC test section floor, in the outer portions of the boundary layer (see Figure 12), show a sharp rise in signal rms when Po is about 9 psia (0.61 bar). The hot-wire signals for Po = 8.02 psia (0.54 bar) and 9 psia (0.61 bar) are shown in Figure 13a. The difference in the signals is indicative of

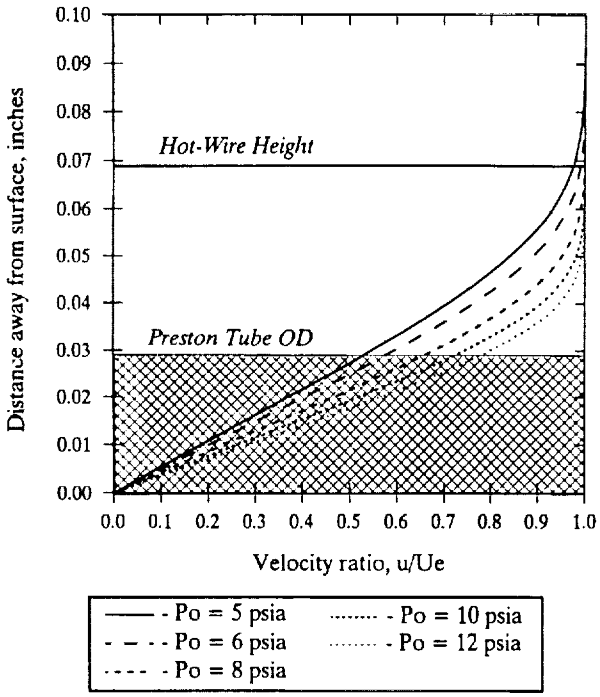

Fig. 12 - Calculated laminar boundary layer profiles in the PoC test section (X = 8.375 inches - 21.27 cm).

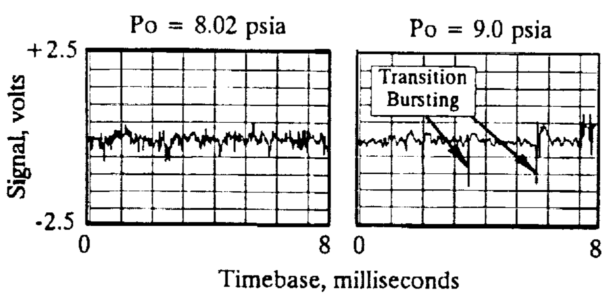

Fig. 13a - Comparison of hot-wire signals from the PoC test section at different Po, near transition onset.

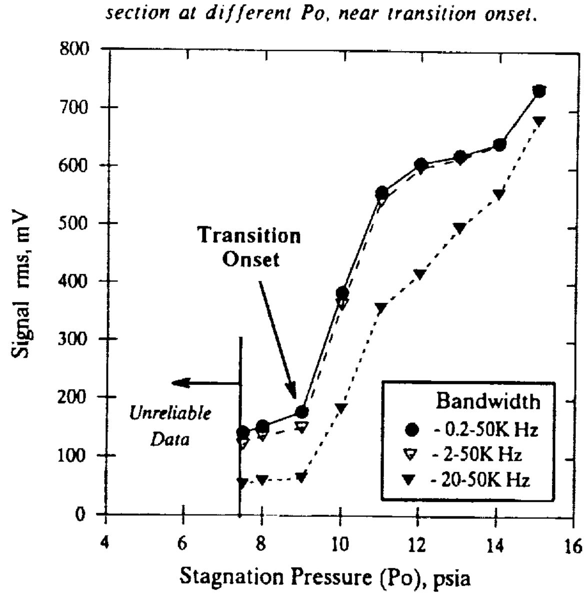

Fig. 13b - Summary of hot-wire data from the PoC test section at Mach 2.5 (X = 8.375 inches - 21.27 cm).

transition bursting. The signal spectrums are broadband with no discrete frequencies.

The associated rise in signal rms is independent of the signal bandwidth, as shown in Figure 13b. In fact, the hot-wire signals follow a pattern over the Po range which is associated with a familiar non-bypass transition process[17], where the transition bursting reaches a maximum frequency. Unfortunately, the uncalibrated hot-wire data can only be used qualitatively. The hot-wire data at lower Po in this test series were unreliable due to intermittent tunnel leaks, but low signal rms was observed down to a Po of 5.4 psia (0.37 bar).

To check the reliability of the hot-wire data from the PoC test section, the honeycomb and Rigimesh sheet were removed from the settling chamber. The uncalibrated hot-wire data taken with and without the honeycomb and Rigimesh sheet installed, are shown in Figure 14. Clearly, the increase of free stream turbulence (previously documented) had the effect of initiating transition onset, at the same location, at a lower Po of about 6 psia (0.41 bar) and hence a lower Re. Note, in this data set that a low signal rms was achieved down to a Po of 6 psia (0.41 bar) before tunnel leaks occurred.

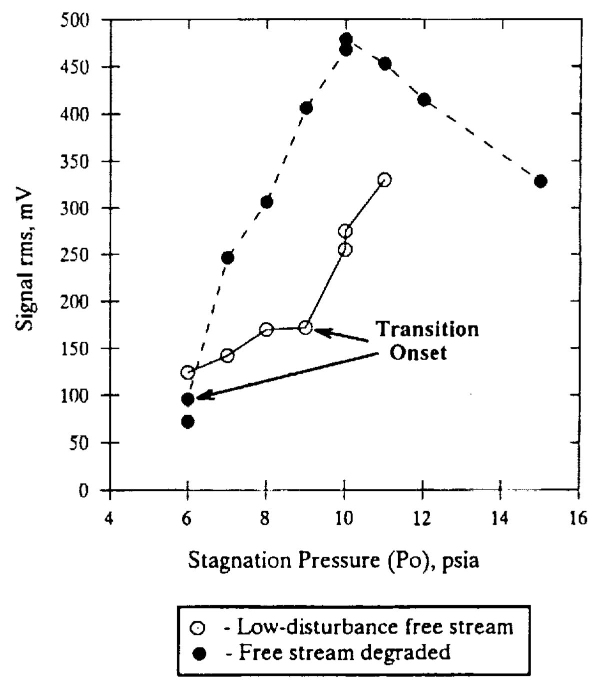

Fig. 14 - Effect of different free stream disturbance levels on hot-wire data from the PoC test section at Mach 2.5.

In another series of tests, the hot-wire was replaced by a Preston tube. This tube was sized to fit in the lower half of the floor boundary layer (See Figure 12). The data from the Preston tube are shown in Figure 15, over an extended Po range from 5.4 psia (0.37 bar) to 20 psia (1.36 bar). This Po range corresponds to an Re range from 1.25 to 4.64 million per foot. It is clear that there is a significant rise in the probe Cp at a Po of about 8.5 psia (0.58 bar). This rise is associated with transition onset where the boundary layer profile starts changing from a laminar type to a turbulent type.[18] The probe Cp reaches a plateau at about 16 psia (1.09 bar).

The sidewall boundary layers were studied with a flush-surface-mounted hot-film. The hot-film data are shown in Figure 16 over an extended Po range up to 20 psia (1.37 bar). The calibration of the hot-film is only qualitative[19] as indicated on Figure 16. Nevertheless, the hot-film data show that the boundary layer on the sidewall remained laminar over

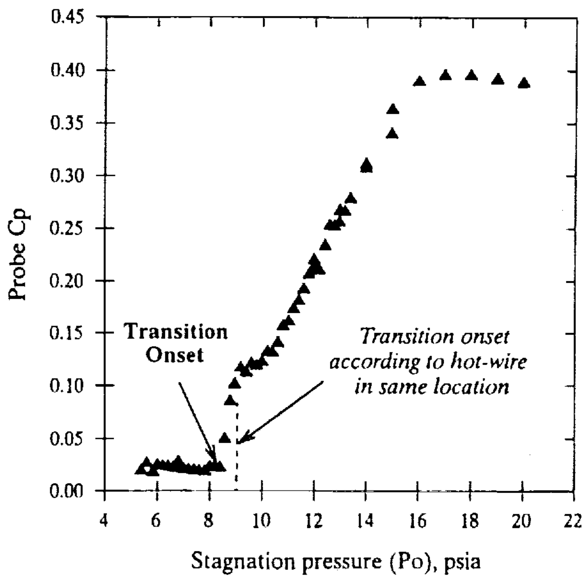

Fig. 15 - Summary of Preston tube data from the PoC test section at Mach 2.5 (X = 8.375 inches - 21.27 cm).

the entire Re range, at an X location of 6.69 inches (17 cm), with no tunnel leaks. The hot-film signal rms is seen to jump to expected levels for turbulent flow only when tunnel leaks caused the nozzle flow to unstart. This flow break down caused transition bypass to occur on the sidewall, as shown in Figure 16, where hot-film data with and without tunnel leaks are compared. In addition, the same leaks cause transition bypass to occur on the test section floor and ceiling, as measured by the hot-wire probe in the test section.

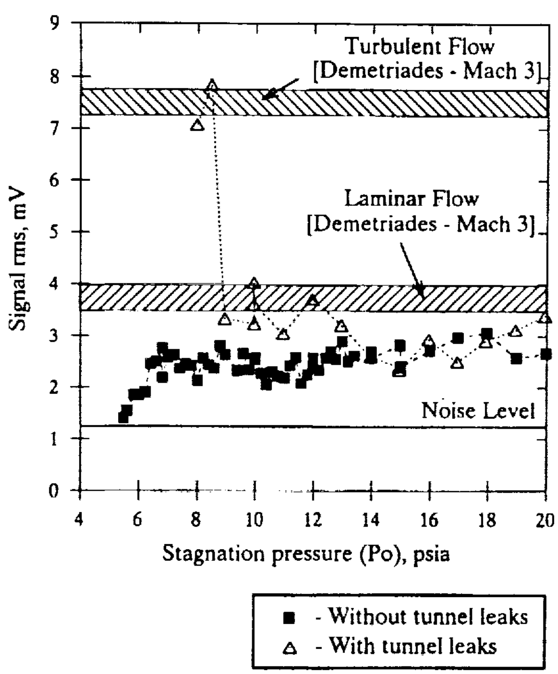

Fig. 16 - Summary of hot-film data from the PoC test section sidewall at Mach 2.5 (X = 6.69 inches - 17 cm).

During these laminar flow studies there was some concern about the drift in temperature of the inlet air and the PoC nozzle/test section structure. The air supply to the PoC is not heated and the inlet air temperature is always lower than ambient due to the expansion across a single air regulator. We monitor the inlet air temperature on a regular basis to check for repeatability of test conditions. The thermal mass of the PoC is large compared to the heat transfer associated with the nozzle/test section flow. We have observed the PoC structure reaching near temperature equilibrium within about the first 5 minutes of running. This temperature equilibrium is affected only slightly by changes of inlet air mass flow, despite noticeable changes in the inlet air temperature. To assess the long term effects of temperature drift, we operated the PoC for 2 1/2 hours continuously and monitored our hot-wire and hot-film instrumentation. No significant changes in the test section flow were observed during this test.

7. Discussion of Results

The latest LFSWT drive system tuning has defined both the maximum mass flow required and the dual injector separation for Mach 2.5 operation. These parameters are 184 lbs/sec (83.46 kg/sec) for the mass flow and 51.68 inches (1.31 m) for the injector separation. Unfortunately, recent investigations of the FML compressor have revealed that a P_E of 8 psia (0.34 bar) cannot be maintained at the high mass flows now required for Mach 2.5 operation. In fact, P_E rises to 8.8 psia (0.6 bar) at high mass flows precluding PoC operation below a Po of 15 psia (1.02 bar). A decision has therefore been made to concentrate the initial LFSWT operating envelope on a lower Mach number. We have chosen Mach 1.6 in order to support F-16XL SLFC flight testing.

Preliminary measurements in the PoC settling chamber show that the free stream flow entering the nozzle/contraction is low-disturbance, according to Beckwith et al.[20] Of course, the flow entering the LFSWT settling chamber will have to pass through a different array of valves and pressure reducers at 64 times larger mass flows. However, we know that the noise and turbulence entering the LFSWT settling chamber will be less than that of a blowdown wind tunnel.[7] Nevertheless, the modular design of the PoC settling chamber is the best currently available for the LFSWT and the design will proceed accordingly.

In fact, the settling chamber effectiveness has been verified by the existence of laminar flow in the PoC test section. Two transition measurements (hot-wire and Preston tube) agree that transition occurs 84% along the test section floor at a Po of about 8.5 psia (0.58 bar) which corresponds to a Re of about 2 million per foot. Furthermore, when the settling chamber effectiveness is reduced by removing the honeycomb and Rigimesh sheet, transition occurs at a lower Po. This result is actually a repeat of Laufer's work[2] at JPL, which highlighted the strong effect of free stream turbulence on transition, particularly at Mach numbers less than 2.5. This result is also further proof that the complete settling chamber is producing low-disturbance flow to sustain laminar flow to a higher Re.

The steadiness of the supersonic diffuser flow has also been verified by the existence of laminar flow in the test section. The new PoC test section is 3.335 inches (8.47 cm) longer than before, so the PoC can better simulate the LFSWT test section flow. This improvement, combined with dynamic instrumentation has allowed us to document the extent of steady flow at the inlet of the supersonic diffuser, as part of our laminar flow studies. The minimum Po at which Mach 2.5 could be maintained steadily was 5.4 psia (0.37 bar) without tunnel leaks. Below this Po, the average test section Mach number dropped and the hot-wire probe in the test section experienced significant velocity fluctuations. This onset of unstart has previously been observed (with the aid of our focusing schlieren system) as the entire supersonic diffuser flow becoming oscillatory and highly unstable. It is clear that once the inlet flow to the supersonic diffuser becomes oscillatory that laminar flow is lost.

The absence of transition on the PoC sidewall was expected, because of the short run lengths coupled with favourable pressure gradients and the absence of curvature. Consequently, the extent of quiet flow in the PoC is determined by the transition location on the floor and ceiling of the test section. In the LFSWT, transition may occur first on the sidewalls (as occurs in the Mach 3.5 Langley Pilot Quiet Tunnel[7]) and this is one of the reasons for making the test section cross-section rectangular. By placing the sidewalls further from the tunnel centerline than the floor and ceiling, we can potentially maintain a quiet test core to higher Re. Also, the rectangular shape of the test section and supersonic diffuser means the primary injectors need only be mounted on the long floor and ceiling of the test section/supersonic diffuser, leaving the test section sidewalls clear of ducting.

Obviously, the tunnel leaks in the PoC (referred to earlier) have severely hampered research at low Po. The problem is peculiar to the small-scale of the PoC and has been traced to internal leak paths around the PoC windows. This is a legacy of using the PoC for much longer than originally planned. A solution to the problem has now been found by potting the windows in a silicone-based sealer instead of vacuum grease.

The existence of laminar flow in a small wind tunnel like PoC (with short flow lengths) does not guarantee long lengths of laminar flow in a larger wind tunnel like the LFSWT. Preliminary CFD analyses predicted that transition would not occur along the PoC Mach 2.5 nozzle or test section. Unfortunately, this prediction has been disproved by the PoC experiments. Nevertheless, this information should help improve future transition predictions for the PoC and hence for the LFSWT. Presently, we can confirm that laminar flow can exist at a location 84% along the PoC test section floor from Po = 5.4 psia (0.37 bar) to Po = 8.5 psia (0.59 bar), which corresponds to an Re range of 1.25 to 1.97 million per foot with a To of about 50°F (283 K), as shown in Figure 1.

8. Future Plans

Based on the inability of the FML compressor alone to drive the LFSWT at Mach 2.5, the validation of the LFSWT test envelope will continue by operating PoC at Mach 1.6 in the near future. We hope to study and document quiet flow and LFSWT drive system parameters for Mach 1.6 before the end of June 1992, to impact the LFSWT design process. At the same time, further flow measurements will be made in the settling chamber with different configurations.

Instrumentation development will continue using commercially available hot-film arrays, which span the entire length of one wall of the contraction/nozzle/test section. This measurement technique should allow documentation of where transition occurs at a given Re. In addition, work with the focusing schlieren and liquid crystal coatings will continue to document PoC transition. New hot-wire mounts will hopefully allow hot-wire calibration in the free stream, so we can relate the hot-wire data to flow velocity. Also, the X location of the test section hot-wire probe will be varied to study the PoC flow at different streamwise locations.

Quiet wind tunnel development work will continue with CFD analyses directed at active control of supersonic transition using nozzle wall heating and cooling together with nozzle contour and length changes. This effort will support the eventual expansion of the actual LFSWT test envelope for quiet flow to the proposed envelope shown in Figure 1.

9. Conclusions

1) Preliminary flow studies in the new PoC settling chamber indicate that the free stream is low-disturbance.

2) Natural laminar flow has been documented along at least 84% of the PoC test section at Re from 1.25 to 1.97 million per foot.

3) A linear stability analysis (e^N method) is now available at NASA-Ames to assist our nozzle design studies and quiet wind tunnel development.

4) The uniquely efficient Mach 2.5 PoC drive system has been successfully shortened by 24.78 inches (62.94 cm), which is equivalent to reducing the length of the LFSWT by 16.5 feet (5.03 m).

5) The maximum mass flow required for the LFSWT Mach 2.5 drive system is 184 lbs/sec (83.46 kg/sec) with a P_E of 8 psia (0.54 bar) , which exceeds the capabilities of the FML compressor alone.

6) Design of the LFSWT is now proceeding with an emphasis on Mach 1.6 operation.

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APPENDIX C

The FML Compressor as a Drive System for the LFSWT

- A Narrative -

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James Laub and Stephen Wolf
March 30, 1992

MPCC9700 P-10

Introduction

During 1988, a feasibility study was commissioned by Dr. Sanford Davis to decide how best to build a Laminar Flow Supersonic Wind Tunnel (LFSWT) at the NASA-Ames Fluid Mechanics Laboratory (FML) for boundary layer transition research. A full report by Riise was received on this study in June 1988.[1] The report was based on the following guidelines: Continuous operation at Mach numbers within the range from 1.8-4.0 was required to impact the NASA "HiStar" project; Low disturbance flow is essential for transition research; Minimum test section height is 8 inches; Tunnel must fit in an FML test cell; Tunnel design should be simple; Construction costs should be minimized. The study considered 3 drive system options: 1) Use of surplus JPL compressors and support equipment; 2) Combined use of the FML compressor with ambient injectors; 3) A combination of the FML compressor with injectors driven by the NASA-Ames 3000 psia air supply.

Riise Findings

The Riise report concluded that option 2 was the most viable drive system for the LFSWT at Mach numbers below 2.5. This conclusion was based on the assumption that the FML compressor would provide suction up to a specification rate of 160 lbs/sec (235,000 ICFM at 70^0 F) with a manifold total pressure (Pm) equal to 8 psia. It appears that Riise based his calculations on a corrected inlet static pressure to the compressor, provided by the manufacturer, Allis-Chalmers (see attached figure). These pressures are different (by roughly 0.1 psia) from Pm calculated by the manufacturer (shown by the open squares on the attached figure). While the effect of this oversight appears small, a precedent was set to use static pressure taps to measure Pm rather than a total pressure probe. In addition, Riise was assumed to have summarized the manufacturer's data, which appears not to be the case. Only recently have FML personnel gained a full understanding of the manufacturer's performance tests and know the location and type of pressure measurements made.

Mass Flow Studies

At the outset of the LFSWT project, concerns were expressed about the effect on compressor performance of the significant differences between the manufacturer's test arrangement and the FML manifold arrangement. A series of compressor test were performed in May 1989 by Laub (RFR) and Meneely (EEF) to measure the mass flow capability and detect actual compressor surge. In these tests, Pm was measured as a static pressure in a non-flow section of the manifold (see attached figure). However, it was decided to use an inlet static pressure (Pi) (measured at the location shown on the attached figure) as the set point for the compressor control system. Unfortunately, this Pi was incorrectly assumed to be the same as the corrected inlet static pressure shown in the manufacturer's performance curve.

The mass flow was calculated from a static pressure difference measured across the compressor inlet contraction (shown on the attached figure) using factory installed pressure instrumentation. This method of determining mass flow is only accurate within the order of 40,000 ICFM at high mass flows, but nevertheless the same method was used for both the Allis-Chalmers and Laub/Meneely tests. For the Laub/Meneely tests, all the flow through the compressor at high mass flows had to come through the manifold Surge Control Valve (SCV) because suitable tunnels were not available. The tests indicated that 157 lbs/sec (228,000 ICFM) was achievable with a Pi of 7.38 psia, which was assumed to collaborate Riise's findings. In fact, the 228,000

ICFM mass flow was achieved with a Pm of 8.1 psia (as shown by the solid squares on the attached figure), which disproved Riise's assumption that Pm of 8 psia could be maintained up to 235,000 ICFM.

In fact, at the time of the test, Laub and Meneely considered Pi to be Pm, following the guidance of Riise. A misunderstanding that was further compounded by unknowingly measuring Pi at a different location from the manufacturer. This difference caused Pi to always be lower than that reported by the manufacturer by the order of 0.17 psia. Consequently, the perceived Pm was order of 0.27 psia lower than the actual Pm, adding in the 0.1 psia from Riise's assumption. The FML compressor was therefore assumed to be producing a lower Pm than it actually was, which gave the false impression that there was a pad for LFSWT operation. In fact, it was thought that the FML compressor was performing much better than in the manufacturer's tests, when in fact there was no significant difference at high mass flows (see data depicted as open and closed squares on the attached graph).

In addition, these tests were carried out with various Pi set points (as done in the manufacturer's tests) down to 7.2 psia, where increasing compressor vibrations were measured. In the interests of compressor safety, it was concluded that a set point Pi of 7.8 psia was the lowest set point appropriate for normal compressor operation. At this time, a set point of 7.8 psia was incorrectly assumed to be more than sufficient to generate a Pm of 8 psia for LFSWT operation.

Probe Measurements

As the development of the LFSWT drive system progressed, using an eighth-scale model (called the Proof of Concept (PoC) supersonic wind tunnel) the mass flow requirement increased. In September 1991, Wolf et al[2] reported the first concerns about whether or not the FML compressor mass flow would be adequate for LFSWT operations at Mach 2.5. They stated that 163 lb/sec (240,000 ICFM at 70^0 F) mass flow would be required for the LFSWT. Since then, the mass flow requirement has risen to 173 lbs/sec (254,000 ICFM at 70^0 F), despite attempts to reduce the mass flow of the ambient injectors, making the situation even worse. Consequently, another compressor test was performed in February 1992 by Laub and Wolf to establish if the compressor could provide the extra mass flow needed and at what Pm. The Pi set point was fixed at 7.8 psia for this series of tests. In addition, it was decided to install a traversing pitot-static probe in the FML manifold just upstream of the compressor inlet contraction (see attached figure) to directly measure mass flow and manifold pressure. The probe was regarded as essential to dispel a growing number of concerns about the 5/89 mass flow studies. This use of a pitot-static probe is accurate to about 5,000 ICFM and therefore considerably better than the pressure difference method used previously to calculate mass flow.

For the probe measurements, the maximum compressor mass flow was provided by opening the SCV and running the tunnels in test cells #2 and #3 (TC2 and TC3), with a Pi of 8 psia actually achieved in the test. By integrating pitot-static traverse data acquired with the compressor operating at maximum power conditions, we found a maximum mass flow of 206 lbs/sec (271,000 ICFM) through the compressor. However, the Pm (again measured at the same non-flow section of the manifold) was found to be a high 8.84 psia, which, as expected, matched the total pressure measured at the probe.

Since the LFSWT Mach 2.5 drive system requires less than the maximum measured mass flow of the compressor, further tests were carried out to find the mass flow at which Pm begins to rise above its minimum value. By running the tunnels in test cells #2 and #4 (TC2 and TC4), we found that at 109 lbs/sec (158,000 ICFM) the minimum Pm was 8.3 psia (measured at the probe), with a Pi set point of 7.8 psia. A small increase of mass flow through the compressor from this condition, produced a rise in Pm (see the open triangles on the attached figure).

The realization that the minimum Pm is 8.3 psia was an unpleasant surprise. Furthermore, the mass flow at which Pm begins to climb is much lower than Riise reported.[1] This situation is further complicated by the expected 0.2 psia rise in Pm due to the two 90^0 turns the LFSWT flow must make to reach the compressor inlet from test cell #1 (TC1). Consequently, for a Pi

of 7.8 psia, future LFSWT drive systems (below 158,000 ICFM mass flow) will be designed for a Pm of 8.5 psia, instead of 8.0 psia, to provide some pad for unexpected pressure losses. Of course, at the beginning of the LFSWT project, the Pm pad incorrectly associated with a 7.8 psia set point, was assumed adequate to accommodate this 0.2 psia pressure loss.

Discussion and Conclusions

It is clear that a chain of incorrect assumptions has evolved due to 3 factors: 1) Riise mistakenly used a corrected static manifold pressure as Pm; 2) The differences between the inlet static pressures measured by Allis-Chalmers and Laub/Meneely were not known in 1989; 3) The effect of Pi set point on Pm was not known during either the manufacturer's tests or the mass flow studies. These factors led to a false impression that the FML compressor was adequate to drive the LFSWT at mass flows up to 240,000 ICFM with a Pi set point of 7.8 psia.

By careful measurements using a pitot static probe, we have dispelled any misconceptions about the location and type of pressure measurements used in all the compressor tests. We have now identified the true operating regime of the FML compressor and manifold, in terms of mass flow and manifold pressure. Furthermore, the simulation of a flow in the FML manifold, which was representative of LFSWT operation, identified an injector effect of the SCV. An effect that has allowed the PoC to operate at very low mass flows (1/64th of LFSWT mass flows) with a Pm of 8 psia. A situation which further compounded the ill founded confidence we had in the ability of the FML compressor to drive the LFSWT at Mach 2.5 with a set point of 7.8 psia.

Because of the adverse findings of the probe measurements described earlier, a study of the manufacturer's data was undertaken. This study has found, from comparisons of existing similar pressure measurements, that Pm can be reduced by lowering the Pi set point. For example, it should be possible to operate the FML compressor with a Pm of 8 psia over a mass flow range from at least 120,000 ICFM to 220,000 ICFM, by lowering the Pi set point to 7.6 psia. Alternatively, we could drop the reliance on Pi for compressor control, since Pi is an abstract pressure found from a single measurement in a very unreliable location, i. e. located near the inlet guide vanes and a duct geometry change. A true averaged inlet static pressure measurement, as used by the manufacturer, would provide a more reliable measurement, which could easily be transformed into a direct Pm value. It would therefore be possible to set Pm directly to avoid operator/user confusion and provide FML researchers with the actual value of Pm they require to test within surge/safety limits.

We can conclude that the LFSWT Mach 2.5 drive system will only function with an auxiliary compressor to assist the FML compressor. This auxiliary compressor is essential to maintaining Pm at 8 psia at high mass flow (254,000 ICFM), regardless of what FML compressor set point is used. If we continue to use the FML compressor alone at a Pi set point of 7.8 psia, a Mach 1.6 drive system for LFSWT is feasible with a mass flow less than 158,000 ICFM to provide maximum primary injector Mach number. However, by lowering the Pi set point below 7.8, it would be possible to drop Pm over a useful mass flow range up to 220,000 ICFM. This would have two effects: 1) Allow the primary injector Mach number to be raised further; 2) Allow the primary injector mass flow to be significantly larger for drive system tuning. Both these effects will enhance the ability of a Mach 1.6 drive system to operate at low stagnation pressures and the ability of other LFSWT drive systems to operate at other higher Mach numbers up to 2.5.

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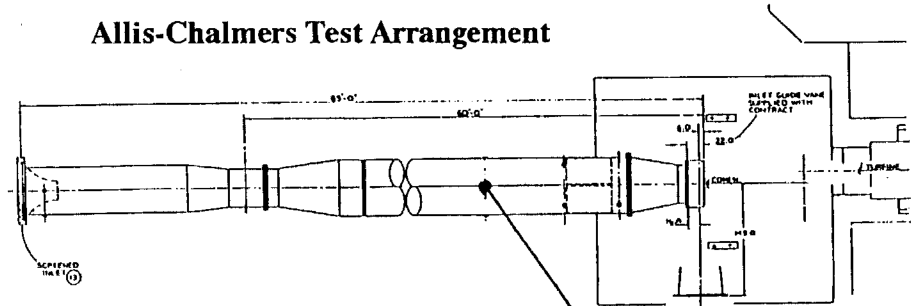
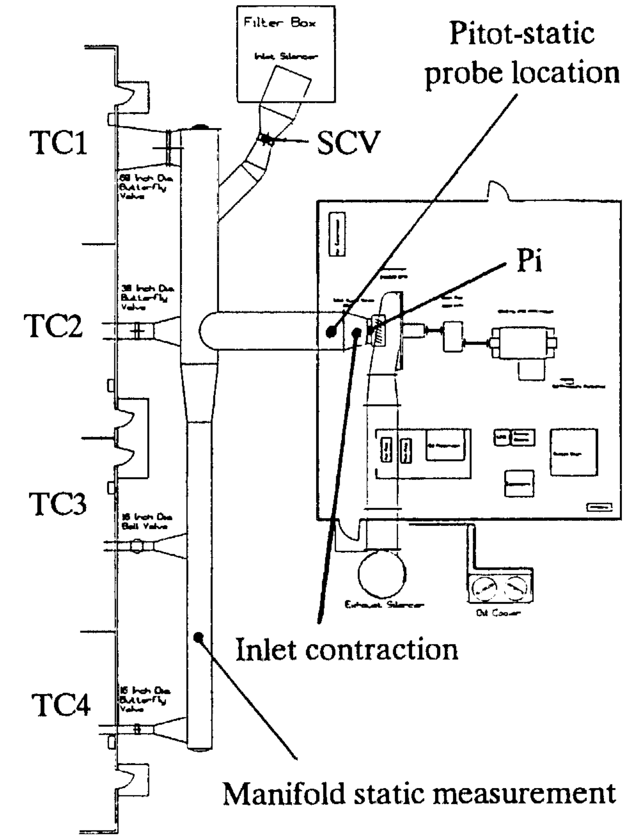
FML Compressor Performance Summary

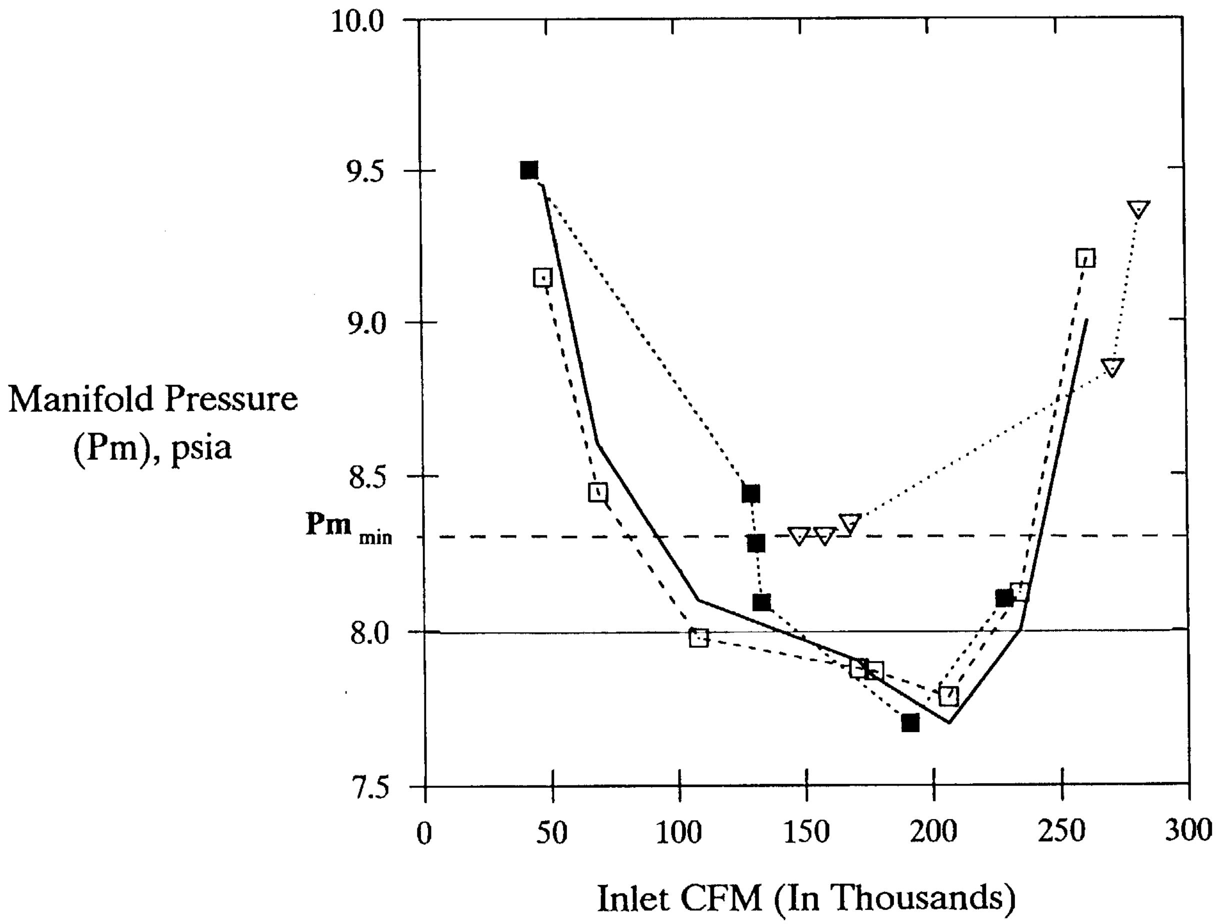
Allis-Chalmers Test Arrangement



Inlet static measurement for manufacturer's performance curve

FML Manifold Arrangement





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APPENDIX D

LFSWT Design Packages

The LFSWT design project can be conveniently divided into the following logical packages. Concerns are considered as current obstacles to final design of each package or sub-package.

1. Drive System

The drive system consists of the primary and secondary stages of injectors and the supersonic and subsonic diffusers. There are two special requirements: Structural vibrations due to the drive system must be isolated from other components of the wind tunnel; and the drive system should not compromise all round access to the test section.

1.1 Primary Injectors
The primary injectors will require tuning in-situ and must be designed accordingly.

Concerns:

What is the baseline operating conditions of the primary injectors?
What in-situ adjustment is required?

1.2 Secondary Injectors
The secondary injectors will suck air from outside building 260.

Concerns:

Will the secondary injectors be closed off for high Po operation?
Can the secondary injectors operate at low enough mass flows?

1.3 Subsonic Diffuser
The subsonic diffuser must decelerate the injector and test section flows to subsonic speeds before reaching the manifold 60 inch valve. The diffuser cross-section will change from rectangular to round and pierce the test cell wall.

Concerns:

How long do we make the diffuser?

1.4 Supersonic Diffuser
The two-dimensional supersonic diffuser will need to be adjustable for model blockage compensation. Optical access will be required through both sidewalls using Zelux-W polymer material for the windows.

Concerns:

What adjustments are required for model compensation?

2. Nozzle

The nozzle consists of the three-dimensional contraction, two-dimensional throat and the supersonic nozzle itself. The nozzle should be mechanically isolated from the settling chamber/drive system. Two nozzles are required with contours for Mach 1.6 and 2.5 flow, which should be quickly interchangeable. The nozzles will have no adjustment and will be fixed blocks. However, provision needs to be made in the support structure to accommodate a sophisticated flexible plate nozzle requiring top and bottom wall access. The flow should see no joint lines on the top and bottom walls of the nozzle. Optical access is required through both sidewalls using Zelux-W polymer material for the windows. Instrumentation and probe access will be required on the sidewalls.

Concerns:

What surface finish is required?
Is boundary layer removal required immediately upstream of the throat?
What nozzle contours produce natural laminar flow?

3. Settling Chamber

The settling chamber will contain numerous flow conditioning components which may change from time to time and will require inspection. The settling chamber must be modular and be designed in such a way as to allow simple access to the individual components. The cross section of the settling chamber is ideally square.

Concerns:

What is the maximum length of the settling chamber?
What are the cone flow requirements?
What pressure drop can be achieved upstream of the settling chamber?
Is boundary layer bleed required upstream of the contraction?
What are the optimum components for low-disturbance operation?

4. Test Section

The test section needs to provide all-round accessibility for flow visualization and instrumentation. Quick removal/replacement of the test section would be an advantage. The test section support structure should have a minimum impact on research needs, by providing sufficient support flexibility to accommodate future requirements. Two test sections are required to allow model installation independent of tunnel operation/research.

Concerns:

Do the test section walls need active control of temperature or shape?
What form of probe mount do we need for nozzle research?
What type of model supports should we design for?

Stephen Wolf
November 25, 1991

LFSWT Design Outline

6th March 1992

Settling Chamber:

Length of *Black Box* pressure reducer . . . . . 25 inches

Maximum cross-section - 39.24 inches square

Cone housing length . . . . 56.69 inches

Honeycomb housing length . . . . . . 12 inches

Total length of screen holders. . . . . 48 inches maximum
24 inches minimum

Adjustable length for recirculation chamber etc . . . 48 inches

Settling chamber centerline height - 72 inches above floor

Nozzle/Contraction:

Inlet size - 39.24 inches square

Overall contraction length to nozzle throat . . . . 48 inches

Nozzle width - 16 inches

Nozzle length (Mach 1.6 operation) . . . . . 27.376 inches

Nozzle exit height - 8 inches

Test/Section:

Inlet size - 8 inches high; 16 inches wide

Test section length . . . . 32 inches

Test section wall divergence - 0.25 degree

Test section centerline height - 72 inches above floor

Exit size - 8.280 inches high; 16 inches wide

Supersonic diffuser:

Supersonic diffuser length . . . . . . 41 inches

Minimum throat height - 6.056 inches

Maximum ramp height - 1.112 inches

Ramp length - 8 inches

Floor and ceiling wall divergence - 0.25 degree

Variable exit cross-section - 6.344 - 8.566 inches high; 16 inches wide

Primary Injectors (2 off):

Variable throat area - 90 - 180 square inches

Varaible exit area - 169 - 338 square inches

Variable exit Mach number - 1.8 to 2.2

Variable exit cross-section - 16 inches wide; 10.56 - 21.12 inches high

Throat to exit length - 32 inches

Injection angle (relative to centerline) - 10 degrees

Total mass flow range - 62 - 124 lbs/sec

Mixing Region:

Mixing region length . . . . **51.68** inches

Exit cross-section - 41.246 inches high; 16 inches wide

Secondary Injectors (2 off):

Throat area (Mass flow = 34.65 lbs/sec) - 106 square inches

Exit area (Mach 1.8) - 152.53 square inches

Exit cross-section - 41.246 inches high; 3.698 inches wide

Throat to exit length - 20 inches

Injection angle (relative to centerline) - 10 degrees

Total mass flow - 69.3 lbs/sec

Length of sidewall flare section . . . . . **12** inches

Subsonic diffuser:

Inlet cross-section - 41.246 inches high; 36.758 inches wide

Outlet cross-section - 60 inch diameter

Length with 7 degree total angle . . . . . **190** inches

Inlet centerline height - 72 inches above floor

Outlet centerline height - 92 inches above ground level

Centerline inclination - 6 degrees

Maximum total length. . . . . **591.754** inches (49.312 feet)

Test Cell #1 length - 38 feet; Distance between FML manifold and test cell wall - 10 feet
Maximum Length of LFSWT in High Bay - (49.312-10-38) = 1.312 feet

Stephen Wolf 3/6/92

APPENDIX E

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Adaptive Wall Technology for Minimization of Wind Tunnel Boundary Interferences - A Review

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Preprint of paper for the International Conference
on
**Adaptive Wall Wind Tunnel Research
and
Wall Interference Correction**

Xian, Shaanxi Province, China
10-14 June 1991

ADAPTIVE WALL TECHNOLOGY FOR MINIMIZATION OF WIND TUNNEL BOUNDARY INTERFERENCES - A REVIEW

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ABSTRACT - This paper reviews adaptive wall technology for improving wind tunnel flow simulations. The technology relies on a tunnel/computer system to control the shapes of the test section boundaries. This powerful marriage of experiment and theory is used to minimize boundary interferences at the very source of the disturbances. The significant benefits of adaptive wall testing techniques are briefly discussed. A short historical overview describes the disjointed development and the status of these testing techniques from 1938 to present. Some of the currently operational Adaptive Wall Test Sections (AWTSs) for aerofoil and turbomachinery research are described. Some observations on the achievements and future directions of adaptive wall research are presented to stimulate round table discussion.

1. Introduction

The need to improve the operating efficiencies of flight vehicles continues to motivate the quest for better and better ground-based simulations of "real" flow fields. Unfortunately, to-day's wind tunnel data still suffers from significant boundary interference effects, particularly at transonic speeds. (In this paper, boundary interferences are unwanted flow disturbances generated by the test section walls, excluding sidewall disturbances in 2-D testing.) These significant effects exist despite considerable efforts to remove simulation problems over the last 45 years. Commonly used techniques for minimizing boundary interferences (i.e. use of small models, ventilated test sections and post-test linear corrections) are now sadly out of date and fail to provide the high levels of accuracy we must now demand from wind tunnel data.

A potential solution to this dilemma has existed, in a conceptual form, for about 53 years. It involves using testing techniques which minimize boundary interferences at the very source of these disturbances. These techniques adapt the test section boundaries to streamline shapes so the test section walls can become invisible to the model. We know this concept as the **Principle of Wall Streamlining** which was first used back in 1938, and has been described many times before.[1] In fact, the adaptive wall concept offers an elegant way to simplify the problem of boundary interferences. Wall adjustment strategies need only compute the relatively simple flow field over the test section boundaries (in the farfield). The complex flow field round the model need never be calculated. The adaptive wall concept actually splits the test flow field into a real part which is contained inside the AWTS, and an imaginary computed part which flows over the outside of the AWTS and extends to infinity. Hence, this concept allows us to simplify the CFD task by increasing the complexity of the test section hardware.

This paper briefly reviews the disjointed development and status of adaptive wall testing techniques and considers the significant benefits of using such testing techniques. Operational AWTSs (not detailed in other ICAW papers) used in both aerofoil and turbomachinery research are described. These descriptions are intended to give a balanced view of the *State of the Art* at this conference. Observations on the achievements and future directions of adaptive wall research are presented as topics for round-table discussion. Such important topics as 3-D testing potential, AWTS design, wall adjustment

strategies, research directions and accomplishments are included. However, these observations should not be considered as a complete synopsis of each topic. Unfortunately, the need for conciseness precludes a full analysis of these complex topics in this paper.

2. An Historical Overview of Adaptive Wall Research

The adaptive wall testing techniques we know today are a rediscovery of the first solution to severe transonic wall interferences (i.e. choking). Figure 1 illustrates the disjointed development of adaptive wall technology. The National Physical Laboratory (NPL), England, built the first AWTS in 1938, under the direction of Dr. H. J. Gough.[2] Their pioneering adaptive wall research developed the first viable testing technique for achieving high speed (transonic) flows in the confines of a wind tunnel, using a flexible walled AWTS. They opted for minimum mechanical complexity in their AWTS and used only two flexible impervious walls. The absence of computers made wall streamlining a labour intensive process which was surprisingly fast, of the order 20 minutes. Sir G. I. Taylor developed the first wall adjustment strategy, which is still valid to-day.[3] NPL researchers went on to successfully use flexible walled AWTSs for general testing up until the early 1950s. They generated an extraordinary amount of 2- and 3-D transonic data[4] during this 14-year period, which is more than half of all the AWTS data produced to date. The NPL 20 x 8 inch (50.8 x 20.3cm) High Speed Tunnel even became the first adaptive wall wind tunnel to operate at low supersonic Mach numbers. Some early adaptive wall work was also carried out in Germany during the 1940s, but this effort came to nothing, despite the building of the largest ever AWTS with a 3m (118 inch) square cross-section.

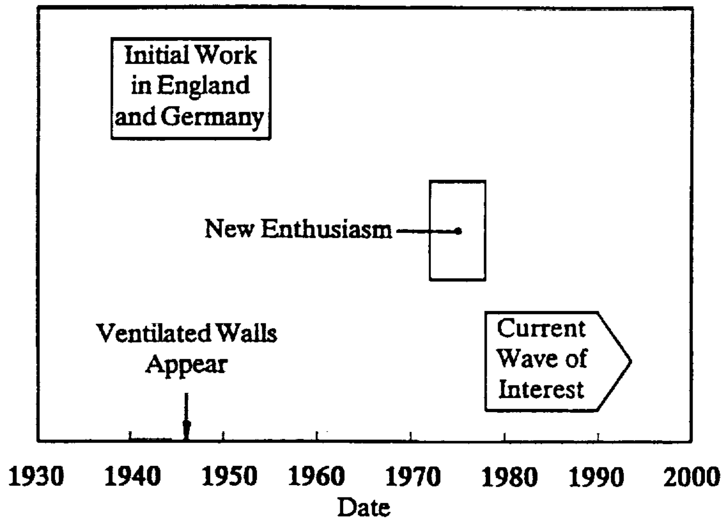


Fig 1 - The disjointed development of adaptive wall technology from 1938 to present.

In 1946, at NACA Langley, the dynasty of ventilated test sections began, offering a "simpler" approach to high speed testing. The passive uncontrolled boundary adjustments of ventilated test sections fostered the political obsolescence of NPL's AWTSs. Furthermore, serious misconceptions caused the benefits of adaptive wall technology to be overlooked and then simply forgotten. Ironically, this "obsolescence" did not occur before AWTS data were used as standards against which to judge ventilated test sections.[5]

After about 20 years of complete inactivity, interest in adaptive walls was rekindled because this was an "obvious" concept to try. Around 1972, several researchers, in Europe and the USA, independently rediscovered the concept of adaptive wall testing techniques.[6] This new wave of interest was spawned by an strong need for better wind tunnel data at transonic speeds in both 2-D and 3-D testing.

This renewed interest produced 5 adaptive wall research groups around the world. Initial adaptive wall research, in the mid-1970s, was directed towards low speed 2-D testing, because this was a relatively quick way to reprove the adaptive wall concept. In this early phase, notably work at Southampton University (England) demonstrated the AWTS versatility to create 6 flow field simulations and produced the first predictive wall adjustment strategy of Judd et al.[7] In addition, the intuitive design principles for 2-D flexible walled AWTSs were quantified and optimized.

In the mid to late 1970s, the successful low-speed research effort paved the way for transonic 2-D research, which introduced fast automatic wall adjustments (taking only seconds in some cases) with notable work at Southampton University, ONERA/CERT (France) and TU-Berlin (Germany). Rapid progress in the development of subsonic and transonic 2-D adaptive wall testing techniques was made, helped greatly by the growing availability of computers. During the 1980s, this progress led to successful 2-D tests at high lift and high blockage conditions, and the use of AWTSs in cryogenic wind tunnels (i.e. in the NASA Langley 0.3-m Transonic Cryogenic Tunnel and the ONERA/CERT T2 tunnel). All these successes were achieved in flexible walled AWTSs. Parallel research using ventilated walled AWTSs was as vigorous but alas this approach encountered some fundamental limitations. In particular, researchers found it was impossible to achieve sufficient control of ventilated walls if relatively large model disturbances were present.[8]

Initial research using AWTSs for 3-D testing began in the late 1970s, and concentrated on exotic AWTS designs like the rod-wall tunnel at AFFDL at Dayton, Ohio (USA),[9] the rubber-tube DAM AWTS at DFVLR Göttingen (Germany)[10] and the octagonal AWTS at TU-Berlin.[11] These complex designs proved that 3-D AWTSs could minimize boundary interferences but were impractical. Fortunately, the rod-wall tunnel did show that perhaps a 2-D AWTS could be successfully used in 3-D testing. This important finding resulted in numerous subsonic and low transonic tests in the 1980s. Research at the Von Karman Institute (Belgium) and DFVLR Göttingen produced the first predictive wall adjustment strategy for 2-D AWTSs used in 3-D tests devised by Wedemeyer and Lamarche.[12] Meanwhile, studies of the residual interferences at Southampton University, established the usefulness of 2-D AWTSs in 3-D testing. Other notable work at DFVLR Göttingen and NPU (China) involved preliminary 3-D tests at Mach 1.2, in which oblique shock wave reflections were successfully attenuated by local flexible wall bending. Again all these advances were achieved in flexible walled AWTSs. Parallel research with ventilated walled AWTSs for 3-D tests occurred at NASA Ames and AEDC, both in the United States. This work was eventually abandoned with the failure to develop fast wall adjustment strategies and to achieve adequate boundary control. However, researchers at TsAGI (USSR) began operating a large, 2.75m (9 foot) square, ventilated walled AWTS in the TsAGI T-128 tunnel during 1986. This effort continues with some success, albeit with relatively small models.[13]

Unfortunately, adaptive wall research has been significantly slowed in recent years by low priority funding and the demise of active research groups at NASA Langley and TU-Berlin. The development of 3-D transonic adaptive wall testing techniques in the 1990s currently rests with 3 organizations: DLR Göttingen, NPU, and Southampton University.

Research has continued into advanced 2-D testing techniques with the goal of extending the useful speed range up to low supersonic Mach numbers and improving cascade simulations. Initial supersonic 2-D testing at NPL was followed some 40 years later by preliminary research at Mach 1.2 performed at TU-Berlin and ONERA during the mid-1980. More recently there have been significant strides at Southampton University[14] again at Mach 1.2. Cascade simulations pioneered at Southampton University at low speeds in 1974, are now used only at Genoa University (Italy) at transonic speeds.

We find that AWTSs are now available for commercial use at NASA Langley (2-D only), ONERA/CERT (2- and 3-D), and TsAGI (3-D only). There are plans to build new transonic cascade AWTSs at Genoa University and DLR Göttingen, and new transonic flexible walled AWTSs in China (for the CARDC 0.6m high speed tunnel), Germany (for the DLR TWG 1m transonic wind tunnel) , France (for the ONERA S3Ch tunnel) and Russia (at TsAGI but not for the T-128 tunnel). Currently, we find that AWTSs are not incorporated in any of the major wind tunnel construction projects underway worldwide. However, many of these projects do include provisions in the test section design for the eventual use of adaptive wall technology during the life of the wind tunnel. (The European Transonic Windtunnel - ETW is a prime example of this practice.) This shows that the development of adaptive wall technology is expected to continue to a successful conclusion.

3. Adaptive Wall Benefits

This section considers all the benefits of we can reasonably expect from using adaptive walls. All the benefits have to some degree been researched and are known to be real. The descriptions are necessarily concise and more details can be found in the literature.[15] The benefits of adaptive walls can be summarized as follows:

1) Minimization or elimination of boundary interferences.
2) Test Reynolds numbers increased for a given test section size.
3) Flow disturbances minimized with solid walls.
4) Boundary measurements routine with solid walls.
5) Tunnel operating costs reduced.
6) Test programme efficiency improved.
7) Sidewall problem minimized by use of large models.
8) Controlled and versatile test section environment with solid walls.

The primary advantage of adaptive wall testing techniques is the capability to minimize and in some cases eliminate boundary interferences at the source of the disturbances. With the boundary interferences minimized or eliminated, we are free to increase the size of the model for a given test section size. We can typically double the test Reynolds number. The larger model also allows simpler instrumentation and more detailed measurements. Alternatively, we can keep the same model size and shrink the test section, thereby reducing the tunnel size and operating costs. Interestingly, the task of magnetically suspending models (which eliminates support interferences) becomes simpler in an AWTS because the supporting coils can be positioned closer to the model.

With solid adaptive walls (called flexible walls), the test section boundaries are simple and smooth compared to the complexity of porous or slotted ventilated walls. The simplicity provides easy boundary measurements on the actual walls, for both residual interference assessment (to account for minimized but correctable boundary interferences) and wall streamlining. While the smoothness of the flexible walls minimizes disturbances to the tunnel free stream, significantly improving flow quality. (An improvement which is very important in transonic boundary layer transition research at full scale Reynolds numbers.) In addition, smooth walls reduce the tunnel drive power necessary to achieve a given test condition, with the model and test section size fixed. The elimination of the plenum volume, when a closed flexible walled AWTS is used for transonic testing, reduces settling times and minimizes flow resonance, which is particularly important for blowdown tunnels.

Adaptive wall technology can provide the aerodynamicist with real-time "corrected" data, even in the transonic regime. This situation represents another significant advantage to the wind tunnel user. Since, the final results are known real-time, test programmes can be much more productive, with considerable reduction in the number of data points and tunnel entries necessary to achieve any given test objectives. This increased productivity will, of course, provide significant cost savings.

Since flexible walled AWTSs provide a more controlled environment, good data repeatability is easier to achieve. Furthermore, the bigger the AWTS, the more accurately the AWTS environment can be controlled. Also, measurement of the test Mach number is made directly at the entrance of the AWTS and can be easily repeated. In 2-D testing, acceptable repeatability of the order 0.001 in normal force coefficient and 5 counts in drag coefficient is reported in a 0.33m (13 inch) square AWTS.[16] Considering the difficulty of achieving good repeatability in conventional ventilated test sections, the prospect of excellent repeatability in flexible walled AWTSs is very good.

In 2-D testing, the influence of sidewall boundary layers can be significantly reduced in an AWTS producing a near-ideal 2-D environment. This advantage is related to the use of relatively large models, particularly at high unit Reynolds numbers. If sidewall Boundary

Layer Control (BLC) is necessary, this can be accomplished by conventional means. Flexible walled AWTSs have been shown to routinely streamline around the model with BLC and automatically accommodate any mass flow changes in the tunnel at the same time.

Finally, a flexible walled AWTS has the advantage of being capable of generating 6 different flow field simulations with the same model if desired.[17] These 6 flow field simulations are free air, open jet, closed tunnel, ground effect, steady pitching and cascade. The imparted tunnel versatility is excellent for CFD code validation testing, since this capability allows different known boundary conditions to be setup in the AWTS.

4. Some Currently Operational AWTSs

This section contains descriptions of 8 currently operational AWTSs which do not form part of other papers in this conference. The descriptions are presented by organization in alphabetical order. A complete detailed listing of the 22 AWTSs currently thought to be in use around the world for aerofoil and turbomachinery simulations, is shown on Table 1.

4.1 Aerodynamic Institute, RWTH Aachen, Germany

The test section of the Transonic- and Supersonic Tunnel (TST) at RWTH Aachen was equipped with flexible walls in 1985/6. The AWTS is 40cm (15.75 inches) square and 1.414m (4.64 feet) long. The top and bottom walls are flexible and mounted between two parallel sidewalls. The flexible walls are made from 1.3mm (0.051 inch) thick spring steel. Each wall is supported by 24 motorized jacks (See Figure 2).

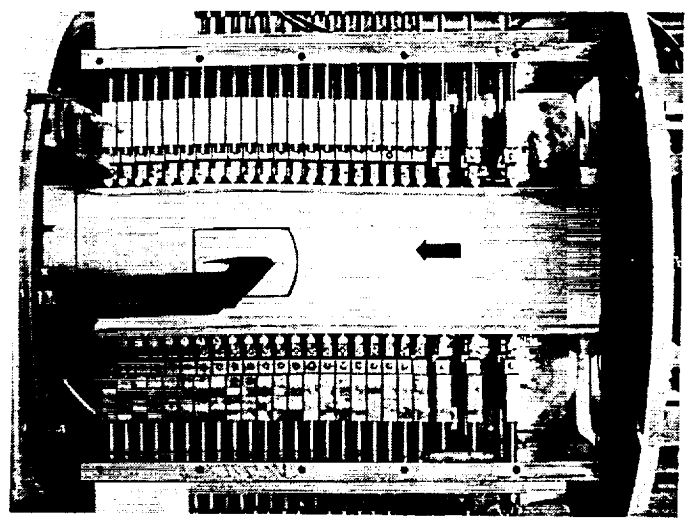


Fig. 2 - The TST AWTS with one sidewall removed.

The TST is an intermittent tunnel capable of operation at Mach numbers between 0.2 and 4, with run times between 3 to 10 seconds. The AWTS has only been used for 2-D testing up to about Mach 0.8, with a test section height to model chord ratio down to 2:1. Usually 3 or 4 tunnel runs are required for each data point at low transonic Mach numbers. Boundary measurements are static pressures measured along the flexible walls at the jacking points. Wall adaptation calculations of the Cauchy type and automatic wall adjustments are made between tunnel runs.[18]

Empty test section calibrations reveal Mach number discrepancies less than 2%, where the model is usually mounted, at Mach 0.82. Lower Mach numbers produce lower discrepancies. Mach number is controlled, up to low transonic Mach numbers, by a downstream sonic throat. The average accuracy of the wall contours, measured by potentiometers at each wall jack, is ±0.1mm (±0.004 inch).

4.2 CAE, Harbin, China

The Chinese Aeronautical Establishment, within the Harbin Aerodynamics Research Institute, installed adaptive walls in the FL-7 transonic tunnel during 1989. The AWTS measures 0.52m (20.47 inches) high, 0.64m (25.2 inches) wide, and 1.75m (68.9 inches) long. The AWTS is equipped with 2 uniform but variable porosity walls, with holes slanted 60° from the vertical, and 2 solid sidewalls. Each perforated wall consists of 11 equal length segments. The porosity of each segment can be independently varied between 0% and 11%.

Table 1 - Currently Used Adaptive Walled Test Sections

Organization	Tunnel	X-Section (h x w) m	Length, m	Approx. Max. Mach No.	Approx. Max. R_c (millions)	Walls	Adaptation Control	Remarks
Aachen, Aero. Institute[2]	TST	0.4 Square	1.414	4.0	2.8	2 Flexible 2 Solid	24 Jacks/Wall	Issue 10
Arizona University[3]	HLAT	0.51 Square	0.914	0.2	...	2 Arrays of Venetian Blinds 2 Solid	16 Panels of Vanes and a Variable Angle Nozzle	Issue 3
CAE[2] Harbin, China	FL-7	0.52 x 0.64 Rectangular	1.75	>0.8	...	2 Porous 2 Solid	11 PCCs/Wall	Issue 10
DLR[3]	HKG	0.67 x 0.725 Rectangular	4.0	>1.2	...	2 Flexible 2 Solid	17 Jacks/Wall	Issues 7,14
FFA Sweden	L2	2.0 x 1.025 Rectangular	4.5	0.27	10	2 Flexible 2 Solid	16 Jacks/Wall	
Genova University[2]	Low Defl. Cascade	0.2 x 0.05 Rectangular	1.58	2.0	1	2 Flexible 2 Soild	36 Jacks/Wall	Issue 7
Genova University[2]	High Defl. Cascade	0.2 x 0.05 Rectangular	1.6	>1.18	1	2 Flexible 2 Solid	13 Jacks-Ceiling 26 Jacks-Floor	Issue 7
NASA Ames[2+3]	HRC-2 AWTS1	0.61 x 0.41 Rectangular	2.79	>0.8	30	2 Flexible 2 Solid	7 Jacks/Wall	
NASA Ames[2]	HRC-2 AWTS2	0.61 x 0.41 Rectangular	2.79	>0.8	30	2 Flexible 2 Solid	11 Jacks/Wall	Issue 10
NASA Langley[2+3]	0.3-m TCT	0.33 Square	1.417	>1.3	120	2 Flexible 2 Solid	18 Jacks/Wall	Issues 1-5,7, 8,13
N P Univ.[2+3] Xian, China	WT52	0.3 Square	1.08	1.2	...	2 Flexible 2 Solid	16 Jacks/Wall	Issue 14
N P Univ.[2+3] Xian, China	Low Speed	0.256 x 0.238 Rectangular	1.3	0.12	0.50	2 Flexible 2 Solid	19 Jacks/Wall	Issues 2,5,9
ONERA/CERT[2+3]	T2	0.37 x 0.39 Rectangular	1.32	>1.0	30	2 Flexible 2 Solid	16 Jacks/Wall	Issue 2
ONERA[2+3]	S5Ch	0.22 x 0.18 Rectangular	0.3	1.2	...	2 Multiplate 2 Solid	302 Transverse Sliding Plates	Issue 9
RPI[2] Troy, NY	3 x 8	0.20 x 0.07 Rectangular	0.6	0.86	...	1 Flexible 3 Solid	6 Jacks	
RPI[2] Troy, NY	3 x 15	0.39 x 0.07 Rectangular	0.6	0.8	...	4 Solid	Multiple Top Wall Inserts	
Southampton University[2+3]	SSWT	0.152 x 0.305 Rectangular	0.914	0.1	0.38	2 Flexible 2 Solid	17 Jacks/Wall	Variable T.S. Height
Southampton University[2+3]	TSWT	0.15 Square	1.12	>1.0	2.5	2 Flexible 2 Solid	19 Jacks/Wall	Issue 1
Sverdrup Technology[3]	AWAT	0.305 x 0.61 Rectangular	2.438	0.2	...	3 Multi-Flexible Slats 1 Solid	102 Jacks-Ceiling 15 Jacks/Sidewall	Issue 4
Tech. University Berlin[3]	III	0.15 x 0.18 Octagonal	0.83	>1.0	...	8 Flexible	78 Jacks Total	Issue 6
TsAGI[2+3] U.S.S.R.	T-128	2.75 Square	8.0	1.7	9	4 Porous	32 Control Panels per Wall	Issues 11,13
Umberto Nobile[2]	FWWT	0.2 Square	1.0	0.6	3.5	2 Flexible 2 Solid	18 Jacks/Wall	

[2] - 2D Testing Capability
[3] - 3D Testing Capability
[2+3] - 2D and 3D Testing Capability
PCC - Plenum Chamber Compartments

December 1990
SWDW

Note: The Remarks column refers to issues of the **Adaptive Wall Newsletter** which contain related articles.

Researchers have carried out several 2-D tests at Mach numbers up to 0.8 at zero lift conditions, with a test section height to model chord ratio of 2.6:1. The wall adjustment strategy is based on influence coefficients. Boundary measurements are made at two control surfaces/lines near one of the porous walls, probably using Calspan pipes.

4.3 The Aeronautical Research Institute of Sweden (FFA), Stockholm

The FFA has recently converted its low-speed L2 wind tunnel to accommodate a large flexible walled AWTS for high Reynolds number laminar flow experiments. The AWTS is 2m (79 inches) high, 1.025m (40.35 inches) wide, and 4.5m (177 inches) long with 2 flexible walls each controlled by 16 jacks. Its novel feature is that the AWTS is rotated through 90 degrees so the flexible walls are vertical and the solid sidewalls horizontal. This arrangement allows the large flexible walls, which are 4.5m (177 inches) long and 1.025m (40.35 inches) wide, to be freely suspended from above. Hence, the jacks to control the shape of each flexible wall, do not support the weight of the walls themselves and can therefore be made smaller. The flexible walls are made of 9mm (0.35 inch) thick plywood with plexiglass window inserts for optical access to the model.

Tests successfully completed in 1988 involved the use of a large aerofoil with a 1.6m (63 inch) chord at Mach 0.27.[19] The model blockage was 11.4% and the test section height to model chord ratio was just 1.25:1. The capability to use a large model in this AWTS has allowed flight Reynolds numbers of 10 million to be achieved. Flow quality for these laminar flow experiments is quoted as less than 0.1% variation in dynamic pressure. The well proven wall adjustment strategy of Judd et al[7] is used with this AWTS.

4.4 Genoa University, Italy

The Department of Energy Engineering at Genoa University operates two adaptive wall cascade tunnels. Both tunnels have a cross-section of 20cm (7.87 inches) high and 5cm (1.97 inches) wide. One is the Low Deflection Blade Cascade Tunnel (LDBCT), which became operational in 1982. The other is the High Deflection Blade Cascade Tunnel (HDBCT) which became operational in about 1985.

The LDBCT can test up to 12 blades, at Mach numbers up to 2.0, with flow deflections up to about 35°. The AWTS has 2 flexible walls and 2 solid transparent sidewalls. The flexible walls are 1.58m (5.18 feet) long and each is shaped by 36 manual jacks. Wall streamlining is performed upstream and downstream of the cascade.

The HDBCT has a similar configuration except the AWTS is 1.6m (5.25 feet) long and wall adaptation is performed only downstream of the cascade. The top flexible wall is supported by 13 manual jacks and the bottom flexible wall by 26 manual jacks (as shown in Figure 3). The AWTS can accommodate flow deflections up to 140°. Up to 13 blades can be fitted in the cascade, with test Mach numbers up to 1.18 reported.[20]

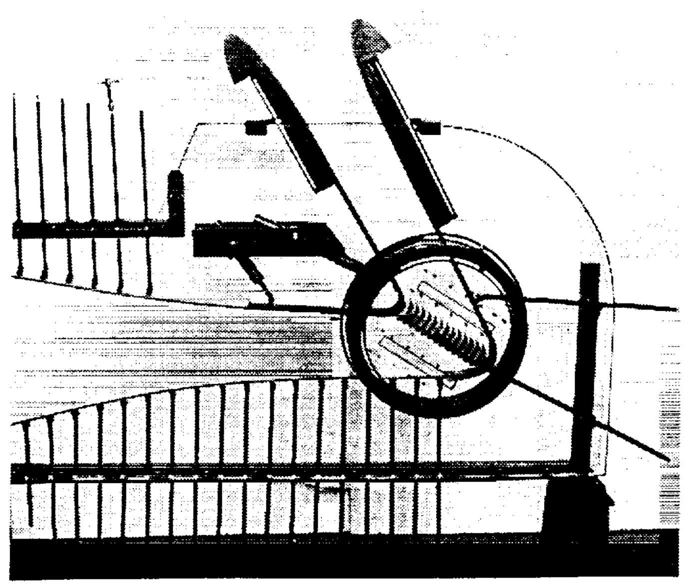


Fig. 3 - The Genoa HDBCT test section as seen through the transparent sidewalls.

Both AWTS need only approximate wall adjustment strategies due to the large

number of blades used in the cascade. The smooth flexible walls have provided remarkably good flow quality for cascade research. The LDBCT is also used for probe calibration.

4.5 NASA Ames Research Center, California, USA

The Thermo-Physics Facilities Branch at NASA Ames possesses 2 AWTSs for use in their blowdown High Reynolds Number Channel-2 (HRC-2) facility. AWTS (#1) was constructed in 1981 and AWTS (#2) followed in 1988. Both AWTS are fitted with 2 flexible walls and 2 parallel solid sidewalls. Both AWTSs are 0.61m (24 inches) high, 0.41m (16 inches) wide, and 2.79m (109.8 inches) long.

AWTS (#1) has 7 manually adjusted jacks supporting each flexible wall, while AWTS (#2) has 11 jacks powered by stepper motors. This is the major difference between the two AWTSs. AWTS (#2) is intended as an automated replacement of AWTS (#1) with improved control of the flexible wall shapes. The wall jacks on AWTS #2 are fast moving because of the short duration tunnel runs. (Wall movement speed is about 5mm (0.2 inch) per second.)

The flexible walls are made of 17-4 PH stainless steel plates and are 2.53m (8.32 feet) long. In AWTS (#1), the flexible walls are 15.9mm (0.625 inch) at the ends tapering to 3.17mm (0.125 inch) in the middle. In AWTS (#2), the flexible walls taper down to 2.39mm (0.094 inch) in the middle for increased flexibility. The downstream ends of the flexible walls each house a pivot joint which attaches to a variable sonic throat for Mach number control. Sidewall Boundary Layer Control (BLC) is available by installing porous plates in the sidewall, upstream of the model location. Mach number variations along the test section due to BLC suction are removed by wall streamlining based on simple influence coefficients.[21]

AWTS #1 has been used for validation of 2-D and 3-D CFD codes. No wall adjustment strategy is used. The flexible walls are simply set to predetermined shapes to provide a known environment for the investigation underway. Studies of LDA wake measurements behind 2-D aerofoils have also been carried out. Extensive 3-D tests of a large sidewall mounted half-model (with 4% blockage) are ongoing, with the flexible walls set straight as part of a CFD code validation experiment. The AWTS (#2) has yet to be installed in HRC-2 and used for aerodynamic testing.

4.6 NASA Langley Research Center, Virginia, USA

The NASA Langley 0.3-m Transonic Cryogenic Tunnel (TCT) was fitted with an AWTS during 1985. The AWTS has 2 flexible walls mounted between 2 parallel sidewalls. The flexible walls are made of 304 stainless steel. The walls are 3.17mm (0.125 inch) thick

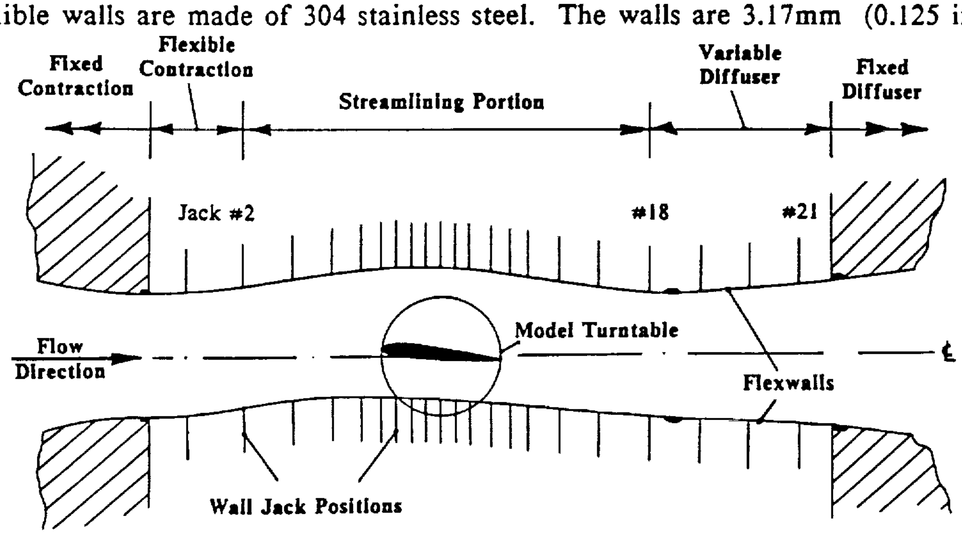
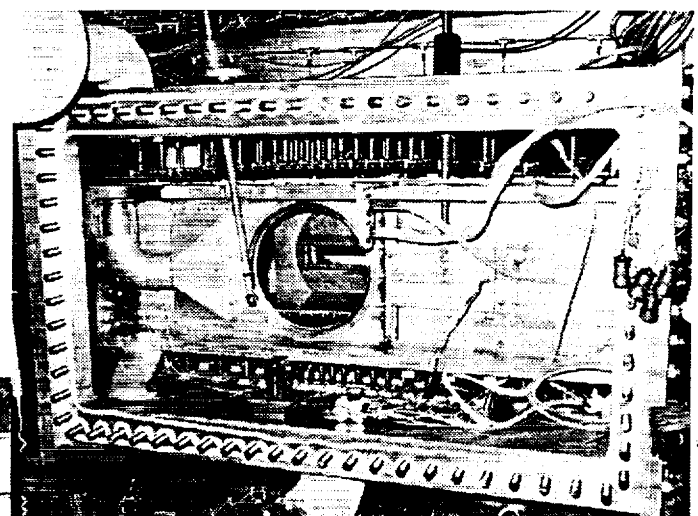


Fig. 4 - A schematic drawing of the NASA Langley 0.3-m TCT AWTS.

at the ends and thin down to 1.57mm (0.062 inch) thick in the middle region as a compromise between flexibility and resistance to bending under pressure load.

The cross-section of the AWTS is 0.33m (13 inches) square and the AWTS is 1.417m (55.8 inches) long. The flexible walls are 1.417m (55.8 inches) long and are shaped by 18 motorized jacks per wall. The grouping of the jacks is shown in Figure 4 for optimum wall control, although the most upstream jack (#1) was found to be redundant. The downstream ends of the flexible walls are attached, by sliding joints, to a 2-D variable diffuser (formed by flexible wall extensions) which connects the AWTS to the rigid tunnel circuit. The shape of the variable diffuser is controlled by 6 motorized jacks (see Figure 4). The wall jacks have insufficient stepper motor power to permanently damage the flexible walls.

The AWTS functions over the complete operating envelope of the continuous running cryogenic tunnel (TCT).[16] The test gas is nitrogen. The AWTS can operate continuously over an 8 hour work shift at temperatures below 120 K. In addition, the AWTS is contained in a pressure vessel for operation at stagnation pressures up to 90 psia (6 bars). The jack motors and position sensors are located outside the pressure shell in a near ambient environment. Sidewall BLC is achieved by fitting porous plates in the sidewalls, upstream of the model position (See Figure 5). Sidewall wall boundary layer suction was found to have little effect on 2-D aerofoil data. Also, routine wall streamlining removed the effects of mass flow changes in the test section.



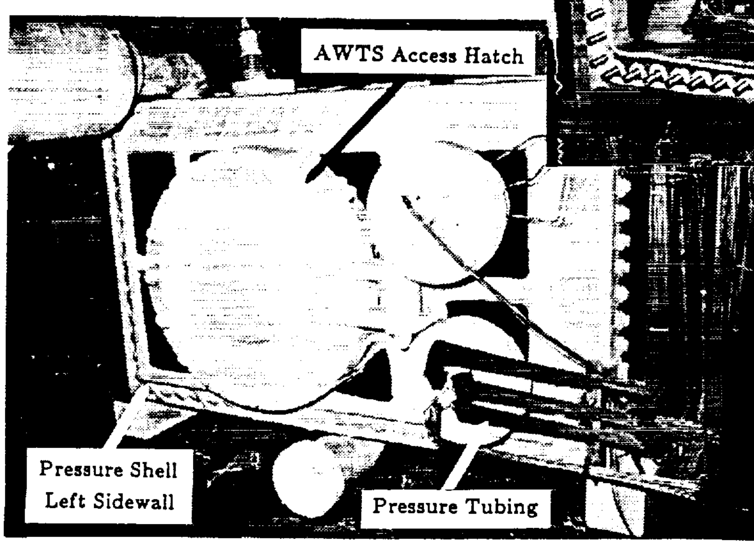


Fig. 5 - NASA Langley 0.3-m TCT AWTS: view above shows the AWTS with the left sidewall of its pressure shell removed; and view left shows the bare sections of the AWTS covered in ice, after a day of operating at cryogenic temperatures.

The wall adjustment strategy of Judd et al[7] is used for 2-D testing. Two residual interference methods (interface discontinuity and two variable) are used to decide when wall streamlining for 2-D testing is complete. Any mismatch between the two methods is a sign of AWTS instrumentation errors. The 2-D test envelope includes normal force coefficients up to 1.54 and Mach numbers up to 0.82, with a model blockage of 12% and a test section height to model chord ratio of 1:1. Record test Reynolds numbers up to 70 million were achieved. Wall movements up to 5.1cm (2 inches) have been experienced during wall streamlining. Boundary measurements are static pressures measured along the centerline of the flexible walls at the recorded jack positions. Wall streamlining takes less than 2 minutes on average and is paced by slow wall movements. A generalized and documented non-expert system[22] is used for AWTS operation within known 2-D test envelopes. We have demonstrated the taking of up to 50 data points (each with wall streamlining) during a 6

Researchers have carried out tests at Mach numbers up to 1.3, using sidewall mounted 3-D wings. For 3-D testing at Mach numbers below 0.8, we have used the wall adjustment strategy of Rebstock[23] to minimize interferences along a predetermined target line anywhere in the test section. Boundary measurements are static pressures from 3 rows of pressure taps on each flexible wall and a row of taps on the centerline of one sidewall. Flexible wall curvature downstream of the 3-D model is automatically minimized by rotating the tunnel centerline, which introduces a known angle of attack correction. For supersonic tests, the adapted wall shapes come from wave theory and form a nozzle ahead of the model.

The flexible walls are set to a nominal accuracy of ±0.127mm (±0.005 inch). No aerodynamic effect of AWTS shrinkage, due to cryogenic operation, has been reported. Mach number is controlled by a closed loop fan drive system (designed around a personal computer) to better than ±0.002, even during wall streamlining.

5. Some Observations on Adaptive Wall Research

We are now approaching 20 years of renewed interest in adaptive wall technology and still no real production-type adaptive wall testing techniques exist for 3-D testing. When compared to the roughly 10 year development period for ventilated test sections, AWTS development appears very slow, and this situation has only helped to strengthened many misconceptions. Clearly, there are many complex reasons for this long development period and a direct comparison with ventilated test section development is very misleading. Some of the observations discussed herein may offer some assistance in unraveling the complicated environment in which new technology must develop to-day.

There are still many unanswered questions about the use of AWTSs in 3-D testing, such as:

- What is the maximum blockage for 3-D models?
- What is the effect of different model configurations?
- When can wall streamlining be stopped and corrections applied with confidence?
- Are all the benefits of adaptive walls achievable in 3-D testing?

These are only a few examples of the more fundamental questions which remain unanswered. Perhaps, this current lack of knowledge could have been adverted if adaptive wall research had received more industrial support. Industrial contracts would have certainly directed this research towards particular testing problems of immediate concern. Left to their own devices, researchers tend to pursue academic goals which causes the research objectives to become diversified. Of course, I am not suggesting all research should be conducted with commercial goals, but in the case of an applied technology like adaptive walls, there must be some direct contact with the "users."

An example of diversification of adaptive wall research is the preoccupation of many researchers to disprove again and again misconceptions about the basic adaptive wall concept. This preoccupation has occurred whenever and wherever new interest in adaptive wall technology has arisen in the last 10 years. Inadvertently, researchers have repeated adaptive wall experiments over and over again without actually making any technical progress. It was in this context that the **Adaptive Wall Newsletter** was conceived as a way of promoting a better, more timely, flow of information between active research groups. Perhaps a more coordinated research effort in the future is a way of encouraging more progress. At present, adaptive wall researchers appear to be working in a technological vacuum, isolated from the aerospace industry they are trying to help.

In contrast to the 3-D testing situation, adaptive wall technology is well established in 2-D free-air simulations and is available for production use. Defined and well documented testing techniques are available and have even been used by non-experts. But even with

this strong technical position, there has been a failure to attract commercial support for 2-D testing in AWTSs. Alas, the idea of 2-D testing being outdated and subservient to CFD predictions is a widely accepted misconception. Surely, the "Quiet" controlled environment of AWTSs is ideal for validation of CFD predictions of such complex and illusive aerodynamic phenomena as transition to turbulence. Furthermore, there are studies of numerous "real" flow fields that can benefit from 2-D testing. In fact, at least 7 currently operated AWTSs are used for 2-D aerofoil research up to drag rise Mach numbers, in addition to adaptive wall research. Now, if the current adaptive wall research into low supersonic testing continues to be successful, then we may soon have available 2-D testing techniques throughout the entire transonic speed regime. The benefits of adaptive wall testing techniques for 2-D testing are clearly there for the taking.

In 2-D cascade testing, adaptive wall research has been limited. In particular, the potential benefits of using only a pair of relatively large blades in a transonic AWTS have yet to be realized. Nevertheless, those who do use cascade AWTSs do so routinely with apparent success. Once again, the industry seems surprisingly oblivious to this effort.

Published data[6] clearly shows that flexible walled AWTSs provide testing capabilities superior to that of ventilated walled AWTSs. I can briefly summarize the effectiveness of flexible walls as follows:

a) Flexible walls can be rapidly streamlined.
b) Flexible walls provide more powerful and direct adaptation control of the test section boundaries, necessary for large models and high lift conditions.
c) Flexible walls provide simple test section boundaries for adaptation measurements and residual wall interference assessments.
d) Flexible walls reduce flow disturbances.
e) No plenum is required around the test section.
f) Flexible walls provide a versatile controlled test section environment.
g) Flexible walls significantly reduce tunnel drive power requirements.

Furthermore, the ventilated walled AWTSs can no longer be considered as a more practical design, because experience has shown that these AWTSs cannot be built from simply modified conventional ventilated test sections. Hence, from a technical standpoint, the use of ventilated walled AWTSs cannot be recommended for any future adaptive wall research. In addition, the continuation of parallel research with ventilated walled AWTSs can only fragment already limited resources.

The use of flexible wall testing techniques for production-type 2-D testing can now be routine. However, experience with production-type testing has identified some special requirements. Firstly, the ability to accurately set the flexible walls to a datum position, usually straight, is very important. Secondly, the calibration of wall position transducers should have equal priority to the calibration of pressure transducers. Thirdly, the AWTS control system should only allow flexible wall pressures to be acquired when the test Mach number is stable. Fourthly, the AWTS control software must be made robust to wall data corruption and most important, be incapable of perpetuating measurement errors. Finally, a good selection process for the initial wall shapes will significantly reduce the time associated with wall adjustments. The order of these requirements is not significant.

Progress in adaptive wall technology will continue to occur in two directions (AWTS design and CFD computations) as a consequence of this technology's unique marriage of tunnel and computer. In 2-D testing, rapid progress was achieved because only two walls need to be adaptable and a simple AWTS is sufficient. Furthermore, wall adjustment strategies based on linear theory (Cauchy type calculations) were acceptable for the majority of 2-D testing, with the method of Judd et al[7] the most widely used. In contrast, the complexity of 3-D boundary adjustments has led to a variety of AWTS designs for 3-D testing, which has slowed progress albeit a necessary part of the learning process.

It is surprising to realize that only in the last 5 years have researchers grasped the fact that, from practical considerations, the optimum AWTS design for 3-D testing is a compromise between size/correctability of residual wall interferences (after wall streamlining), hardware complexity, model accessibility, and the existence of fast wall adjustment strategies. Consequently, in the early days of 3-D testing in AWTSs, when unusual test section designs were fashionable, the residual interference question was never really addressed. Boundary interferences were assumed to be eliminated when the walls of complex 3-D AWTSs were streamlined. This optimistic assumption did much to tarnish the adaptive wall concept in the eyes of the aerospace community and of course created some more misconceptions. Fortunately, the situation has now changed and we find researchers taking a more practical approach to 3-D testing in AWTSs. This change can only help break down some of the obstacles to a more general use of adaptive walls in 3-D testing.

There is now good reason to believe that the simplest AWTS design with just 2 flexible walls is the best for 3-D testing. This simple design reduces both the complexity of calculating the residual wall interferences and the complexity of the tunnel hardware, which allows good model access as a bonus. This important finding has, of course, led to the use of existing 2-D AWTSs for 3-D testing. However, these 2-D AWTSs have wall movement capabilities, which are adequate for 2-D testing, but severely restrict the downwash (and hence lift) of 3-D models around which the walls can streamline. Also, a square cross-section is optimum for 2-D testing. While a rectangular cross-section, which is wider than it is tall, seems better for minimizing 3-D wall interferences with 2-D wall adaptation.[12] The rectangular cross-section will also allow the blockage of moderate aspect ratio models to be increased beyond the current maximum of 1.3%. With limited funds, researchers have done what they can with existing 2-D AWTSs. In fact, NPU now has the first purpose built supersonic 2-D AWTS for 3-D testing, which was commissioned only recently, in 1990.

The variety of 3-D AWTS designs, in many cases, required complex wall adjustment strategies to adjust up to 8 boundaries! The development of these complex strategies consumed considerable research effort. Now with the emphasis on using 2-D AWTSs for 3-D testing, we find only one wall adjustment strategy has received limited international acceptance. This is the strategy of Wedemeyer and Lamarche[12] which minimizes wall interferences only along the centerline of the model. The significance of this limitation has not been fully investigated and remains in doubt. In fact, the assumptions of the Wedemeyer/Lamarche strategy have yet to be really tested. Other strategies have been used but only by their creators. These strategies are the method of Goodyer et al[24] and the method of Rebstock,[23] which have the advantage of allowing the boundary interference to be eliminated along a selectable streamwise line anywhere in the test section. These 3 strategies have not been tested against each other and the use of one or the other appears to be a matter of convenience. There is much work to be done with these existing strategies. Furthermore, the continual advances in computer technology now make it practical (by virtue of reducing execution time) for researchers to use complex non-linear CFD codes in new wall adjustment strategies. There is much to be done in this area if we are to develop adaptive wall testing techniques for 3-D testing throughout the transonic regime.

The lack of residual interference assessment which occurred in the early years of 3-D testing in AWTSs, also occurred in the early days of 2-D testing in AWTSs. In fact, the work at Southampton University took a lead in the development and use of residual interference codes. Back in 1976, the wall adjustment strategy of Judd et al[7] became the first strategy linked to a residual interference code (an interface discontinuity method), which determined when wall streamlining was complete. More recently, various residual interference codes have been developed at Southampton for 3-D testing, including a modified method of Ashill and Weeks. Other centers of adaptive wall research have been surprisingly slow to develop residual interference codes. However, in the last few years the fundamental importance of residual interference assessment in adaptive wall testing techniques has been realized and significant progress can now be expected. The link between wall adjustment strategies and residual interference codes is clearly indispensable.

between wall adjustment strategies and residual interference codes is clearly indispensable. In addition, different residual interference codes could be used together as a monitor of AWTS instrumentation degradation. Unfortunately, the slow development of residual interference codes has also helped hold back the progress of adaptive wall research.

The complexity of AWTSs remains one of the main misconceptions standing in the way of progress. Some operators of conventional wind tunnels are simply horrified at the thought of the model being surrounded by wall jacks and sensors. But this narrow viewpoint only looks at the complexity of the AWTS hardware. If one looks at the overall impact of an AWTS system on normal use of a wind tunnel, the complexity of an AWTS is considerably reduced in this age of computers. The control system for an AWTS is straightforward and similar to other automated processes found in wind tunnels. With user friendly and robust software, it is possible to make adaptive wall testing techniques invisible to the operator. A tunnel operator should be able to have the same contact with an AWTS as with a typical sting support system, which also requires continually resetting for different test conditions. Furthermore, conventional wind tunnels have a hidden complexity in the form of post-test corrections, which normally require considerable amounts of manpower and CPU time. This complexity is of course removed when an AWTS is used, because the real-time data is the final data. So, an AWTS system is not really any more complex because there is a trade-off between hardware complexity and analysis complexity. A trade-off that crops up time and time again in adaptive wall testing techniques.

It is logical to assume that adaptive wall research has the following goals:

1) To define fast adaptive wall testing techniques for different test regimes.
2) To identify acceptable measurement tolerances.
3) To find the optimum AWTS design for different applications.
4) To find if any fundamental limitations to the adaptive wall concept exist.

To some extent adaptive wall research has achieved all these goals in 2-D testing, particularly up to drag rise Mach numbers. However, the situation is quite different in 3-D testing and there is still plenty of work to be done. The lack of interest in boundary interference problems in several major countries of the world, should not deter us from achieving these goals. In fact, developing countries see this situation as an opportunity to catch up or even race ahead in aeronautical research. Actually, adaptive wall technology is very attractive to developing countries because it offers equal, if not better, testing capabilities for a fraction of the normal cost. The list of adaptive wall benefits, described earlier, clearly support this attraction. We can take solace in the fact that no fundamental limitations to the adaptive wall concept have been found over the transonic regime up to Mach 1.2. Hence, adaptive wall testing techniques could be developed for all transonic testing, if the time and money were made available. What keeps driving us on is the thought that the benefits of adaptive walls are too good to be forgotten once again. Indeed, these observations are born out of a desire that adaptive wall research should be properly presented to industry, so that the many benefits are not overlooked.

It is good to remember that the renewed interest in adaptive walls back in 1972 occurred because it was 'obvious' to shape the wind tunnel boundaries to remove interferences. We need to go beyond this thought and find a way to give adaptive wall research credibility in general 3-D testing. Perhaps, we require a clear research objective focussed on some practical ground simulation problem occurring right now. The successful outcome would surely attract financial support from industry which would lead to more and more research. Perhaps, another approach would be to encourage more CFD code validation to develop better design tools. Perhaps, transition research will demand solid boundaries in transonic test sections. What is required is a springboard upon which to propel adaptive wall technology forward from the current doldrums.

The logic of adaptive wall technology will always be strong. The combination of both

experimental aerodynamics and CFD computations to improve the accuracy of wind tunnel data is very powerful. The marriage of tunnel and computer forms an excellent basis upon which to tackle the persistent boundary interference problem. In fact, the polarization of experimental and theoretical aerodynamic research has hampered efforts to solve the boundary interference problem over the last 45 years. It is shocking to find that researchers are still studying the complex flows through slots and holes in the test section walls, when their efforts should really be concentrated on the complex flows round the model.

Now that the expectations of CFD have become more realistic (wind tunnels will not be replaced completely by computers), the relationship between wind tunnel and computer has become much stronger. In my opinion, the AWTS provides the near perfect combination of experimental and theoretical aerodynamics to improve our understanding of aerodynamics in the future. Clearly, perfection can only be achieved by making full use of all advanced technologies available to us.

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I had thought that truth eventually must prevail, but I found silent truth cannot withstand error aided by continual propaganda.
Orville Wright - March 1928

APPENDIX F

Adaptive Wall Technology
for
Improved Free Air Simulations in Wind Tunnels

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ADAPTIVE WALL TECHNOLOGY FOR IMPROVED FREE AIR SIMULATIONS IN WIND TUNNELS

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ABSTRACT

This paper reviews adaptive wall technology for improving wind tunnel free air simulations. This technology uses a powerful marriage of experiment and theory to minimize wall interferences at the very source of the disturbances. The significant benefits of adaptive wall testing techniques are briefly discussed. An overview of Adaptive Wall Test Section (AWTS) design is presented to show the preference for 2 flexible walls for both 2-D and 3-D testing. The status of adaptive wall technology is discussed and future directions for research in 3-D testing proposed.

INTRODUCTION

The desire to achieve higher levels of aerodynamic efficiency from flight vehicles, means we must strive for better and better free air simulations in our wind tunnels. Traditionally, the wind tunnel community uses several well-known techniques to minimize wall interferences. Models are kept small compared with the test section size (sacrificing the test Reynolds number). Ventilated test sections have become an accepted way to relieve transonic blockage and prevent choking (but then other complex wall interferences are introduced). Linearized wall interference corrections are applied post-test to the tunnel data. Usually, all 3 of these techniques are used together in transonic testing. Nevertheless, after 45 years of research, significant wall interference effects still corrupt tunnel data, particularly at transonic speeds. This is despite over 40 years of effort to eradicate the wall interference problem. Consequently, these conventional testing techniques must be considered inadequate for the high levels of accuracy now required from wind tunnel tests.

A potential solution to this dilemma has existed, in a conceptual form, for about 53 years. It involves using testing techniques which minimize wall interferences at the very source of these disturbances. These techniques adapt the test section walls to streamline shapes so the test section walls can become invisible to the model. We know this concept as the **Principle of Wall Streamlining** which was first used back in 1938 at the National Physical Laboratory (NPL), England.[1] The adaptive wall concept is well documented[2] and will not be described again here.

The adaptive wall concept may be viewed as an elegant way to simplify the problem of wall interferences. The adaptive wall concept actually splits the test flow field into a real part which is contained inside the AWTS, and an imaginary computed part which flows over the outside of the AWTS and extends to infinity. Hence, wall adjustment strategies need only compute the relatively simple flow field over the test section boundaries (in the farfield). The complex flow field round the model need never be calculated. So, the Computational Fluid Dynamics (CFD) task becomes almost trivial by increasing the complexity of the test section hardware.

The basic adaptive wall principle is quite simple but its application is not. This complexity arises from the need to adjust the test section walls, for each test condition, to match the real and imaginary flow fields at the interface. However, the use of an AWTS with a computer in a closed loop control system does much to hide this complexity, as discussed later. Also, design principles for AWTSs are available to reduce design complexity and these principles are reviewed briefly in this paper.

The benefits of adaptive walls are presented to show the importance of this technology to the field of

experimental fluid mechanics. The status of adaptive wall technology is briefly discussed with the object of presenting possible directions for future research.

ADAPTIVE WALL BENEFITS

The benefits described in this section have all to some degree been researched and are known to be real advantages in experimental fluid mechanics. The descriptions are necessarily concise and more details can be found in the literature.[3]

Other than the major benefit of minimizing or in some cases eliminating wall interferences, AWTSs offer other important advantages. With wall interferences minimized, we are free to increase the size of the model for a given test section (up to typically a test section height to chord ratio of 1:1). The test Reynolds number can be doubled with model aspect ratio held constant (see Figure 1), perhaps allowing testing at full scale Reynolds numbers particularly in a cryogenic wind tunnel. Larger models are also important for high dynamic pressure tests and provide increased dimensions for more detailing and more volume for instrumentation. We can also expect simpler magnetic suspension of models in an AWTS because the supporting coils can be positioned much closer to the model.

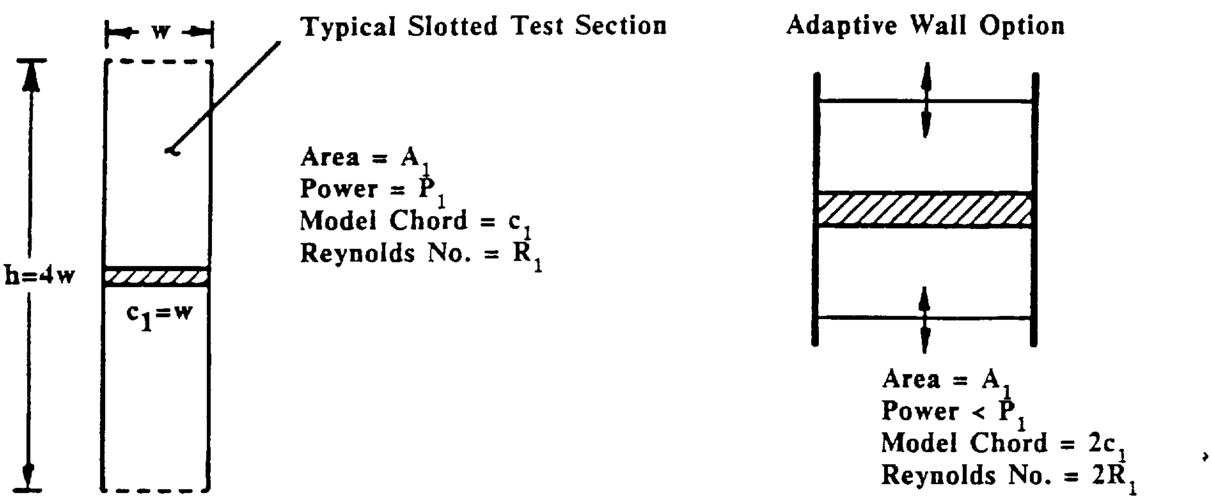

Fig. 1 - Typical Reynolds number benefit of using adaptive walls.

With solid adaptive walls (called flexible walls), the test section boundaries are smooth compared with ventilated walls. This smoothness reduces the tunnel drive power required for a given test condition, with the model and test section size fixed. In addition, the removal of slots and holes reduces tunnel noise and turbulence levels improving flow quality (giving better free air simulations for transition research). Also, the elimination of the plenum volume from the tunnel circuit reduces settling times and minimizes flow resonance, which is particularly important for blowdown tunnels.

Adaptive wall technology can provide the aerodynamicist with real-time "corrected" data, even in the transonic regime. This situation represents another significant advantage to the wind tunnel user. Since, the final results are known real-time, test programmes can be much more productive, with considerable reduction in the number of data points and tunnel entries necessary to achieve any given test objectives. This increased productivity will, of course, provide significant cost savings.

Since flexible walled AWTSs provide a more controlled environment, good data repeatability is easier to achieve. Furthermore, the bigger the AWTS, the more accurately the AWTS environment can be controlled. Also, measurement of the test Mach number is made directly at the entrance of the AWTS and can be easily repeated.

In 2-D testing, the influence of sidewall boundary layers can be significantly reduced in an AWTS, producing a near-ideal 2-D environment. This advantage is related to the use of relatively large models, particularly at high unit Reynolds numbers. If sidewall Boundary Layer Control (BLC) is necessary, this can be accomplished by conventional means. Flexible walled AWTSs have been shown to routinely streamline around the model with BLC, and automatically accommodate any mass flow changes in the tunnel at the same time.

AWTS DESIGN OVERVIEW

The modern interest in adaptive wall testing techniques encompasses two approaches using either ventilated or solid walls. We have observed many interesting designs during the modern era of AWTS development since 1972.[3] In 2-D testing, only 2 walls need to be adaptable and researchers have tested both flexible wall and ventilated wall designs. The complexity of controlling a 3-D boundary for 3-D testing has led to a variety of AWTS designs.[3] Moreover, some approximation in the shape of the test section boundaries is inevitable in 3-D testing. The magnitude of this approximation has at last been recognized as a major research topic. From practical considerations, the design of a 3-D AWTS must be a compromise between the magnitudes of residual wall interferences (after wall streamlining), hardware complexity, model accessibility, and the existence of a rapid wall adjustment procedure.

There is now good reason to believe that the simplest AWTS design, with just 2 flexible walls, is the best for 3-D testing.[4] This simple design reduces both the complexity of calculating the residual wall interferences and the complexity of the tunnel hardware, which allows good model access as a bonus. This important discovery has, of course, led to the use of existing 2-D AWTSs for 3-D testing. However, these 2-D AWTSs have wall movement capabilities which severely restrict the downwash (and hence lift) of 3-D models. Alas, most 2-D AWTSs used for 3-D testing have near square cross-sections which is only optimum for 2-D testing. A rectangular cross-section, which is wider than it is tall, seems better for minimizing 3-D wall interferences with 2-D wall adaptation.[4] The rectangular cross-section will also allow the blockage of moderate aspect ratio models to be increased beyond the current maximum of 1.3%. With limited funds, researchers have done what they can with existing 2-D AWTSs. In fact, Northwestern Polytechnical University (NPU), Xian, China, now has the first purpose built supersonic 2-D AWTS for 3-D testing, which was commissioned only recently in 1990.

There are 22 AWTSs currently thought to be in use around the world, of which 2 are use in cryogenic wind tunnels. There are 17 AWTSs used for free air simulations from low speeds through to low supersonic speeds. Interestingly, all but 3 of the AWTSs have solid/flexible walls.

The layout of a typical transonic flexible walled AWTS for 2-D testing is shown on Figure 2. The close grouping of wall jacks in the vicinity of the model is considered near optimum for wall shape control. The variable diffuser is also an important feature for power and noise reduction.

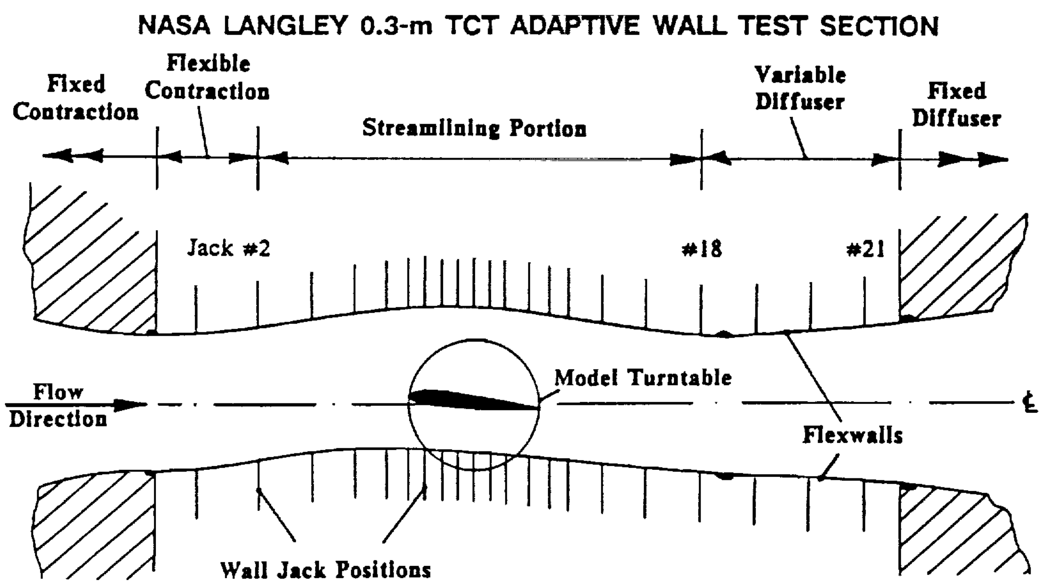

Fig. 2 - Typical transonic flexible walled AWTS for 2-D testing.

Published data[3] clearly shows that flexible walled AWTSs provide testing capabilities superior to that of ventilated walled AWTSs. I can briefly summarize the effectiveness of flexible walls as follows:

a) Flexible walls can be rapidly streamlined, in as little as 10 seconds.
b) Flexible walls provide more powerful and direct adaptation control of the test section boundaries, necessary for large models (12% blockage) and high lift conditions (with lift coefficients above 1.5).

c) Flexible walls provide simple test section boundaries for adaptation measurements and residual wall interference assessments.
d) Flexible walls reduce flow disturbances in the test section.
e) Flexible walls remove the need for a plenum around the test section.
f) Flexible walls provide a versatile controlled test section environment.
g) Flexible walls significantly reduce tunnel drive power requirements.

Furthermore, the ventilated walled AWTS can no longer be considered more practical, because experience has shown that this AWTS design cannot be built from a modified conventional ventilated test section. Hence, from a technical standpoint, the use of ventilated walled AWTSs cannot be recommended for any future adaptive wall research. In addition, the continuation of parallel research with ventilated walled AWTSs can only fragment already limited resources.

TECHNOLOGY STATUS

Adaptive wall technology is well established in 2-D free air simulations and is available for production use. Defined and well documented testing techniques are available and have even been used by non-experts in cryogenic wind tunnels.[5] But even with this strong technical position, there has been a failure to attract commercial support for 2-D testing in AWTSs. Alas, the idea of 2-D testing being outdated and subservient to CFD predictions is a widely accepted misconception. Surely, the "Quiet" controlled environment of AWTSs is ideal for validation of CFD predictions of such complex and illusive aerodynamic phenomena as transition to turbulence. Furthermore, there are studies of numerous "real" flow fields that can still benefit from 2-D testing. In fact, at least 7 currently operated AWTSs are used for 2-D aerofoil research up to drag rise Mach numbers, in addition to adaptive wall research. Now, if the research into low supersonic testing[6] continues to be successful, then we may soon have 2-D testing techniques available over the entire transonic speed regime. The benefits of adaptive wall testing techniques for 2-D testing are simply there for the taking.

The situation is not so good in 3-D testing.[7] We are now approaching 20 years of renewed interest in adaptive wall technology and still no real production-type adaptive wall testing techniques exist for 3-D testing. When compared to the roughly 10 year development period for ventilated test sections, AWTS development appears very slow, and this situation has only helped to strengthened the many misconceptions about adaptive walls. Clearly, there are many complex reasons for this long development period . A major factor is the complicated environment in which new technology must develop to-day, as a means to improve what already exists.

There are still many unanswered questions about the use of AWTSs in 3-D testing, such as:

- What is the maximum blockage for 3-D models?
- What are the effects of different model configurations and varying the position of the zero wall interference line in an AWTS?
- When can wall streamlining be stopped and corrections applied with confidence?
- Are all the benefits of adaptive walls achievable in 3-D testing?

It is reasonable to expect that the general acceptance of adaptive wall technology in 3-D testing will rely on partial answers to all these questions.

Progress in adaptive wall technology will continue to occur in two directions (AWTS design and CFD computations) as a consequence of this technology's unique marriage of wind tunnel and computer. In 2-D testing, rapid progress was achieved because only 2 walls need to be adaptable and a simple AWTS design is sufficient. Furthermore, wall adjustment strategies and residual interference assessments based on linear theory (Cauchy type calculations) are acceptable for the majority of 2-D testing. The method of Judd et al[8] has proved to be the most widely used wall adjustment strategy in 2-D testing up to drag rise Mach numbers. In contrast, the complexity of 3-D testing has led to research into a variety of AWTS designs which of course has consumed considerable research effort, not least in the development of complex wall adjustment strategies for up to 8 walls.

Now there is an emphasis on using 2-D AWTSs for 3-D testing, we find only one wall adjustment strategy has received limited international acceptance. This is the strategy of Wedemeyer and Lamarche[4] which minimizes wall interferences only along the centerline of the model. The significance

of this limitation has not been fully investigated and remains in doubt. In fact, the assumptions of the Wedemeyer/Lamarche strategy have yet to be fully probed. Other strategies have been used but only by their creators. These strategies are the method of Goodyer et al[9] and the method of Rebstock[10], which have the advantage of allowing the wall interferences to be eliminated along a selectable streamwise line anywhere in the test section. These 3 strategies have not been tested against each other and the use of one or the other appears to be a matter of convenience. There is obviously plenty of work to be done in developing both existing and new wall adjustment strategies and associated residual interference codes. Furthermore, the continual advances in computer technology now make it practical (by virtue of reducing execution time) for researchers to use complex non-linear CFD codes in new wall adjustment strategies for 3-D testing throughout the transonic regime.

The complexity of AWTSs remains one of the main misconceptions standing in the way of progress. Some operators of conventional wind tunnels are simply horrified at the thought of the model being surrounded by wall jacks and sensors. But this narrow viewpoint only looks at the complexity of the AWTS hardware. If one looks at the overall impact of an AWTS system on the normal operation of a wind tunnel, the complexity of an AWTS is considerably reduced in this age of computers. The control system for an AWTS is straightforward and similar to other automated processes found in wind tunnels. With user friendly and robust software, it is possible to make adaptive wall testing techniques invisible to the operator. A tunnel operator should have the same contact with an AWTS as with a typical sting support system, which also requires continually resetting for different test conditions. Furthermore, conventional wind tunnels have a hidden complexity in the form of post-test corrections, which normally require considerable amounts of manpower and CPU time. This complexity is of course removed when an AWTS is used, because the real-time data is the final data. So, an AWTS system does not really complicate the gathering of good aerodynamic data, because there is a trade-off between the complexity of the test section hardware and the quality of the real-time data. A trade-off that occurs time and time again in adaptive wall testing techniques.

It is logical to assume that adaptive wall research has the following goals:

1) To define fast adaptive wall testing techniques for different test regimes.
2) To identify acceptable measurement tolerances.
3) To find the optimum AWTS design for different applications.
4) To find if any fundamental limitations to the adaptive wall concept exist.

To some extent adaptive wall research has achieved all these goals in 2-D testing, particularly up to drag rise Mach numbers. However, in 3-D testing, much more research is required to achieve these goals. The current lack of interest in boundary interference problems in several major countries of the world, should not deter us from achieving these goals. These problems will not disappear on their own. In fact, developing countries see this situation as an opportunity to catch up or even race ahead in aeronautical research. Actually, adaptive wall technology is very attractive to developing countries because it offers equal, if not better, testing capabilities for a fraction of the normal cost. The list of adaptive wall benefits, described earlier, clearly support this attraction. We can take solace in the fact that no fundamental limitations to the adaptive wall concept have been found over the transonic regime up to Mach 1.2. Hence, adaptive wall testing techniques could be developed for all transonic testing, if the time and money were made available. Clearly, the benefits of adaptive walls should not be overlooked and forgotten a second time.

It is good to remember that the renewed interest in adaptive walls back in 1972 occurred because it was 'obvious' to shape the wind tunnel boundaries to remove interferences. We need to go beyond this thought and find a way to give adaptive wall research credibility in general 3-D testing. Perhaps, we require a clear research objective focused on some practical ground simulation problem occurring right now. The successful outcome would surely attract financial support from industry which would lead to more and more research. Perhaps, another approach would be to encourage more CFD code validation to develop better design tools. Perhaps, transition research will demand solid boundaries in transonic test sections. What is required is a springboard upon which to propel adaptive wall technology forward from the current doldrums.

The logic of adaptive wall technology will always be strong. The combination of both experimental aerodynamics and CFD computations to improve the accuracy of wind tunnel data is very powerful. The marriage of tunnel and computer forms an excellent basis upon which to tackle the persistent wall

interference problem. In fact, the polarization of experimental and theoretical aerodynamic research has hampered efforts to solve the wall interference problem over the last 45 years. It is shocking to find that researchers are still studying the complex flows through slots and holes in the test section walls, when their efforts should really be concentrated on the complex flows round the model.

Now that the expectations of CFD have become more realistic (wind tunnels will not be replaced completely by computers), the relationship between wind tunnel and computer has become much stronger. In my opinion, the AWTS provides the near perfect combination of experiment and theory to improve our understanding of aerodynamics in the future. Clearly, perfection can only be achieved by making full use of all advanced technologies available to us.

SUGGESTED FUTURE DIRECTIONS FOR RESEARCH

1) Need to build more AWTSs with 2 flexible walls specially for 3-D testing, to probe design principles and 3-D testing technique limitations.

2) Need to focus adaptive wall research on current problems in wind tunnel testing to bridge the gap between academic and industrial interests.

3) Need to emphasis the importance of AWTSs in transition research and CFD code validation to gain popular support for the many advantages.

4) Need to recognize that ventilated walled AWTSs are impractical and consolidate limited resources on developing flexible wall testing techniques.

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