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

Dr. Stephen D. Wolf

February 1991

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Introduction

The main objective of this work is to demonstrate the potential of a cryogenic adaptive nozzle to generate Quiet (low disturbance) supersonic flow. The work under this contract can be summarized as follows:

1) Research to find a unique drive system for the Fluid Mechanics Laboratory (FML) Laminar Flow Supersonic Wind Tunnel (LFSWT) using a pilot tunnel.

2) Supportive effort for ongoing Proof of Concept (PoC) research leading to the design of critical components of the LFSWT.

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

4) Developing a supersonic research capability within the FML.

The application of adaptive wall and cryogenic technologies to a supersonic nozzle has yet to be investigated. This new and novel approach to Quiet supersonic nozzle design will be the subject of preliminary investigations in the third year of this contract.

The body of this annual report summarizes the work of the Principal Investigator (PI).

Proof of Concept Supersonic Tunnel

The desire to use the existing FML compressor system (which generates a manifold pressure of about 8 psia) dictates the use of a unique drive system for the LFSWT. Originally, it was hoped that the use of a single stage of relatively large injectors would pull the supersonic flow through the test section at the desired Mach 2.5 with stagnation pressures down to 5 psia. To run in this condition, the compression ratio of the LFSWT must be 0.625:1, which is far below the 2:1 necessary to drive typical Mach 2.5 wind tunnels. Hence, the necessary drive system is completely unproven. It was
for this reason, a decision was made in April 1989 to build an eighth-scale model of the FML supersonic tunnel called the Proof of Concept supersonic wind tunnel (PoC). The PoC test section is 1 inch (2.54 cm) high and 2 inches (5.08 cm) wide.

The PI assisted in design of the PoC with particular reference to the adjustable injector geometry. This work included defining the test envelope of mass flow and injector exit Mach number. Both these injector variables can be independently varied on the PoC.

The test plan consisted of:

- Phase 1 - Investigation of the drive system.
- Phase 2 - Instrumentation development.
- Phase 3 - Investigation of adaptive wall and cryogenic technologies.

PoC drive system analyses were performed on a PC computer to gain a better understanding of what performance to expect when PoC was commissioned. It became clear that the theoretical assumptions of the original design were optimistic.

During the manufacturing of PoC, the PI helped resolve the inevitable engineering problems. Rapid progress allowed the PoC to be commissioned in October 1989, a mere four months after construction was started. Initially, we encountered poor flow quality due to a lack of properly regulated dried air. The initial performance of PoC was well below our requirements.

With a new air dryer system, we achieved Mach 2.5 in the test section with a stagnation pressure (Po) of about 13 psia. Modifications were made to the PoC hardware to increase the injector exit Mach number to reduce the static pressure in the mixing region. With these modifications we found that Mach 2 was the maximum Mach number we could achieve with the fixed compression ratio across the injectors. Nevertheless, this increase in injector Mach number did allow us to drop Po to about 12 psia.

The poor performance of the PoC drive system spurred the need for more detailed research with flow visualization in the test section and mixing region. Consequently, we designed new side plates for the PoC which gave complete optical access for use with schlieren systems described later. PoC was out of operation for
several months until July 1990. The flows visualization gave many clues about the complex flow phenomena that exists in PoC. We noticed a flow separation in the test section exit, leading to premature flow breakdown. We then found that a second throat in the test section exit could significantly delay this separation. By the end of August 1990, we had dropped Po down to 9.9 psia.

We explored the effect of different second throats downstream of the test section using a variety of both fixed and variable geometries. Theoretically predictions of these effects were computed using a Navier-Stokes code for comparison with PoC data. Unfortunately, the comparisons are only good when the flow in the PoC was well established with Po substantially above its minimum value and no flow separations are present.

Use of long second throats, about 5 times the test section height, provided our best operating condition with a Po of about 8 psia. The compression ratio across the test section in this condition is unity. This was a major breakthrough in drive system efficiency. The previous best compression ratio was reported as 1.4. In addition, no overpressure was required to achieve a start.

In this configuration the effects of changing the mass flow ratio between injector and test section flow were investigated. We found that the optimum mass flow ratio between injector and test section flows was about 7.2:1. This ratio is less than half of what we were expecting. So, clearly the full scale LFSWT would not use all the mass flow capability of the FML compressor, if we are to use a scaled up version of this PoC drive system. It was decided to use this excess mass flow to drive a second stage of ambient injectors to see if this would be beneficial. A Mach 2 second injector system was installed in the existing subsonic PoC diffuser during January 1991. The effect was dramatic and has allowed the primary injector Mach number to be raised from 2 to 2.5. This change has allowed Po to be lowered to 5.6 psia with near Mach 2.5 flow in the test section. Interestingly, still no overpressure was required to start the test section flow.

We are now extremely close to our goal of operating PoC with Po = 5 psia, with more adjustments to the second throat and the primary injectors to investigate before our March 31, 1991 project milestone for finalizing the LFSWT drive system.
In parallel with this drive system effort, we started in October 1990 to design a new settling chamber for the PoC wind tunnel. This modification will allow us to investigate important design principles for the LFSWT settling chamber. The new PoC settling chamber is larger than before with a variety of flow smoothing and straightening devices to achieve the low disturbance free stream flow required for Quiet operation at Mach 2.5. The settling chamber is unique and uses a recirculation concept for passive removal of the sidewall boundary layers.

FML Supersonic Wind Tunnel

The PI assisted with the on-going detailed design studies for the FML supersonic wind tunnel. Much still depends on the outcome of the PoC phase I tests which are still incomplete. We have preliminary drawings but we are unable to proceed until we have definite confirmation of the drive system performance. We expect the LFSWT design to be completed by the end of FY 1992.

Data Acquisition Equipment

The FML had no supersonic research capability at the beginning of this contract. It was decided to build a data acquisition system from the ground up. The need for automatic control of the FML supersonic tunnel dictated the need for a PC type computer. This need coupled with other requirements for ease of data presentation and documentation led to the purchase of a 386/25 PC computer and ancillary hardware. This economical system can be used for data acquisition, data reduction and presentation, report and bibliography production and theoretical analyses. The PC computer also provides a user friendly operating system which is ideal for research work.

An assessment of the instrumentation requirements for future supersonic testing were made. We will need hot-wires, temperature and pressure probes and transducers, schlieren and laser systems, and flow control devices. In addition, a unique experimental Focusing Schlieren system, which has the advantage of requiring no mirrors amongst others was brought together. This system is a rediscovery of a 1950 idea by NASA scientists at the Langley Research Center. A visit by the PI to NASA Langley allowed to find out about this very useful system before any details were published. A conventional schlieren system has now been used with PoC to
assess the advantages of different schlieren systems observing the same phenomena.

Initial PoC data acquisition was manual via a Mercury manometer. We now measure static pressures throughout the PoC using a scanivalve pressure measurement system. This system provides automatic data acquisition of static pressures for monitoring and data storage purposes.

**Library Search**

We have 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. This task rapidly took on mammoth proportions and it became necessary to create a PC computer database. This database provides immediate access and sorting of all citations as these are found.

Currently, the *Supersonic Nozzle Bibliography* contains 730 citations dating back to 1926. An extract of the *Supersonic Nozzle Bibliography* has now been published as a NASA contractor's report. This report is focused on nozzle design and contains 298 citations and abstracts.

This library search shows that our work with the PoC drive system is indeed unique. The performance demonstrated so far is a considerable advance over previously published results.

This library search revealed little research activity in the area of supersonic nozzle wall cooling. Most wall cooling work has been directed towards pure boundary layer transition research on flat plates and cones. Therefore, our work at FML will be unique and provide a contribution to the literature in the field of *Quiet* wind tunnel design.

The size of bibliography shows the enormity of the transition problem. We have concentrated on the low end of the supersonic speed range to make the library search more manageable. The other current work on *Quiet* supersonic flows is being done at NASA Langley. Our approach to *Quiet* supersonic flow is different from that of Langley's and at a lower Mach number more appropriate for High Speed Civil Transport (HSCT) transition research.
Publications

As a part of this research effort, the PI compiled and edited a report entitled **Supersonic Wind Tunnel Nozzles - A Selected, Annotated Bibliography to Aid in the Development of Quiet Wind Tunnel Technology** (See Appendix A). This bibliography was published as NASA CR-4294 in July 1990. The abstract is as follows:

This bibliography, with abstracts, consists of 298 citations arranged in chronological order. We selected the citations to be helpful to persons engaged in the design and development of quiet (low disturbance) nozzles for modern supersonic wind tunnels. We include author, subject, and corporate source indexes to assist with the location of specific information.

A paper was presented entitled **Status of Adaptive Wall Technology for Minimization of Wind Tunnel Boundary Interferences** at the 17th ICAS Congress, Stockholm, Sweden, September 9-14, 1990 (See Appendix B). The abstract is as follows:

This paper reviews the status of adaptive wall technology to improve wind tunnel simulations. This technology relies on making the test section boundaries adjustable with a tunnel/computer system to control the boundary shapes. This paper briefly considers the significant benefits of adaptive wall testing techniques. A brief historical overview covers the disjointed development of these testing techniques from 1938 to present. Currently operational Adaptive Wall Test Sections (AWTSs) are detailed. This review shows a preference for the simplest AWTS design with 2 solid flexible walls. A review of research experience with AWTSs shows the many advances in recent times. We find that quick wall adjustment procedures are available. Requirements for operating AWTSs on a production basis are discussed. Adaptive wall technology is mature enough for general use in 2-D testing, even in cryogenic wind tunnels. In 3-D testing this technology is not so advanced because of low priority development and misconceptions.
Summary of Progress

1) A pilot supersonic wind tunnel has been commissioned establishing a preliminary supersonic research capability in FML.

2) An efficient tunnel drive system has been developed which is extremely close to our initial design goals for the LFSWT.

3) We have carried out unique research on the use of large ambient injectors and supersonic mixing layers in a supersonic wind tunnel.

4) The "State of the Art" in Quiet technology, supersonic transition, supersonic wind tunnel drive systems, and supersonic mixing layers has been established through an extensive library search, part of which has already been published.
Supersonic Wind Tunnel Nozzles

A Selected, Annotated Bibliography to Aid in the Development of Quiet Wind Tunnel Technology

Stephen W. D. Wolf

COOPERATIVE AGREEMENT NCC2-604
JULY 1990

NASA
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### Acknowledgements

The financial support of the Fluid Dynamics Research Branch at NASA Ames Research Center under Cooperative Agreement NCC2-604 is hereby acknowledged. The assistance of the staff at the NASA Ames main library in the locating of literature for this bibliography has been exceptional. The compilation of this bibliography would not have been possible in just 10 months without the tutoring of Mrs. Marie Tuttle (A librarian at NASA Langley Research Center) over many years.
INTRODUCTION

This publication contains 298 citations related to supersonic nozzles, in particular, those used in wind tunnels. The main purpose of this bibliography is to list available publications that may be helpful to persons engaged in the design and development of Quiet supersonic nozzles. The term Quiet requires some explanation here and refers to the minimization of disturbances and nonuniformity in supersonic flows within the confines of a wind tunnel.

We achieve this minimization of tunnel disturbances by striving for uniform free stream flow and laminar boundary layers on the nozzle walls. Consequently, the design of Quiet supersonic nozzles requires understanding of settling chamber/contraction design and high speed transition in addition to the other well known aspects of nozzle design. We have used our interpretation of modern nozzle design requirements to decide which publications to include in this bibliography.

We have included those papers concerned only with the attainment of supersonic Mach numbers ($1 < M_{\infty} < 5$) at the nozzle exit. However, as we learn more about supersonic transition, I am sure we will find that this range is too large and needs to be subdivided. We concentrate this bibliography on the lower end of the supersonic range for two reasons. First, testing in this Mach number range still holds many aerodynamic unknowns and secondly, the next generation of supersonic transports will undoubtedly operate in the Mach 2 to 3 range.

We include some interesting historical material dating back to the early days of supersonic research during the 1920s. Of course, many of the old reports are now unavailable and the bibliography starts with a 1926 paper by Stanton from the National Physical Laboratory in England. The Laval nozzles used in a steam turbine in about 1925 were the first supersonic nozzles ever built. However, the Laval nozzle alone was not adequate for use in supersonic wind tunnels, since the aerodynamicist requires uniform flow in the test section. It was not until after Prandtl and Busemann applied the Method of Characteristics to nozzle design in 1929, that the classical supersonic nozzles, we are familiar with to-day, began to appear. Initially the emphasis on nozzle design was to achieve the desired Mach number at the test section entrance. Work by Laufer in the 1950s and 1960s lead to a change in this initial emphasis. Now we require a supersonic nozzle design to produce a low disturbance flow at the desired Mach number. The era of Quiet supersonic nozzles had arrived and will remain with us.

The design of a Quiet supersonic nozzle is still the subject of considerable research. The most notable work of recent times has come from Beckwith and his co-workers at NASA Langley Research Center. Their approach seems successful for Mach numbers of 3.5 and above. However, widespread acceptance of their approach has yet to occur. We hope to provide a complete overview of supersonic nozzle research and development, past and present, by covering most aspects in this bibliography.

Obviously we have references to Quiet supersonic nozzle design, but we also include references to nozzle design studies for rocket engines and chemical lasers. All the many different types of nozzles are considered although, of course, the axisymmetric and planar (two-dimensional) types are the most popular for wind tunnels. Several computer programs are included in the papers to allow readers to quickly design their own nozzles. We provide citations on engineering studies of different nozzle types which discuss such topics as nozzle contour accuracy and adjustments, and achievable wall smoothness.

We include citations on numerous nozzle flow analyses which consider the three flow regimes within a supersonic nozzle and its contraction. These analyses have allowed numerous test phenomena to be explored perhaps simpler than by experiment. The vast amount of experimental data on supersonic nozzles is well documented in this bibliography. The associated experiments involve advanced topics like nozzle wall cooling and heating as well as the essential calibration procedures. Measurements in supersonic wind tunnels are covered in this bibliography to show what can and has been learnt about a nozzle once it is constructed.
Finally, the design and operation of supersonic wind tunnels is included for both historical and practical reasons. These citations give a feel for the experience gathered on supersonic nozzles in the practical sense. These citations also show the obvious age of most of the supersonic tunnels in operation to-day.

The bibliography arrangement throughout is chronological by date of publication. However there are some exceptions. Firstly, papers presented at conferences or meetings are placed under dates of presentation. Secondly, English translations of papers are placed with the original citation regardless of publication date for clarity. These rules mean this bibliography also serves as a history of the development of supersonic nozzles and wind tunnels. We have included author, subject, and corporate source indexes at the end of the bibliography. In most cases, authors' abstracts are used. However, in some instances we did take license to shorten existing abstracts or write abstracts for those citations we found had no abstract.

We hope that all the citations can be obtained through normal library services. If no abstract accompanies a citation this should be taken as an indication that the associated paper has not yet been located by the compiler. This does not mean that the paper is unavailable. For example, some normally easy to locate NASA/NACA references are currently unavailable to the compiler due to earthquake related library problems. Identifying information, including accession and report numbers when known, is included in the citation in order to simplify library requests. Please note, that some of the citations have restrictions on distribution. These restrictions are stated with the relevant citations.

This bibliography is our first attempt at collating the vast number of citations related to supersonic nozzles and wind tunnels. This bibliography has come about because of the huge wealth of knowledge we have found in the literature. This bibliography reflects the major changes in supersonic nozzle design and uses since the 1950s. Subsequent bibliography updates are planned and we would be most grateful for any assistance with corrections or additions to this bibliography. Please write in the first instance to:

Dr. Stephen Wolf
Mail Stop 260-1
NASA Ames Research Center
Moffett Field
CA 94035-1000, USA
### Ordering Information

To assist with locating the citations in this bibliography, ordering information for the different types of materials is given below:

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<td>Technical Information Service American Institute of Aeronautics and Astronautics (AIAA) 555 West 57th Street, 12th Floor New York, NY 10019 USA</td>
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In a 1916 paper on the discharge of gases under high pressures, the late Lord Rayleigh called attention to the deficiencies in the present state of knowledge of the characteristics of the flow of gas, from a vessel in which it is compressed, through an orifice into the atmosphere or into a receiver at a lower pressure. This paper describes some of the first documented experiments on high speed jets formed by convergent-divergent nozzles. It was concluded that Prandtl's theory was inadequate to account for, at any rate, all the wave characteristics observed.

*Engineering Department, National Physical Laboratory, UK.


*Engineering Department, National Physical Laboratory, UK.


*Langley Memorial Aeronautical Laboratory, Langley Field, Virginia, USA.


*Engineering Department, National Physical Laboratory, UK.


5070 247
(Ames Research Center library no.)

This early work reviews the theory of nozzle flows and compares results with experiment. The effects of condensation are considered. Jet impulse and Carnot shock losses using Laval nozzles are evaluated.


*National Physical Laboratory, Teddington, UK.


*Langley Memorial Aeronautical Laboratory, Langley Field, Virginia, USA.


5341 310
(Ames Research Center library no.)
I have undertaken the task of achieving systematically velocities lying between 1 and 2.5 times the velocity of sound, and of determining the laws governing their development. One of the main obstacles to the attainment of supersonic streams of high order, which are of great interest practically, is the very high power required; I have therefore paid particular attention to the energy side of the question. I have determined by systematic trials, the best shape of funnel (nozzle), I have shown especially the case with which the supersonic stream can diverge and the impossibility of its converging at all rapidly. Avoiding the use of the free stream, which is extravagant in power requirements and liable to irregularities, I have devised practical funnels giving velocities between 1.2 and 2.5 times the velocity of sound. I have then introduced obstacles (both cylindrical and plane) into the funnels, firstly for optical examination of the flow and then for measurements of resistance. I obtained an important result with a hot-wire anemometer that when the velocity increases beyond 0.8 times the velocity of sound, the fluid heats the hot-wire further instead of cooling it.

*Consultant to the Ministry of Air, France.


High-speed wind tunnels are of practical as well as theoretical value for the solution of certain aerodynamic problems. The design of high-speed wind tunnels is reviewed. Different tunnel drive systems are discussed. The supersonic wind tunnel in Zurich is described as an example of current state of the art in high-speed wind tunnel design.


This paper presents the results of tests conducted to determine the effect of the constructional elements of a Laval nozzle on the velocity and pressure distribution and the magnitude of the reaction force of the jet. The effect was studied of the shapes of the entrance section of the nozzle and three types of divergent sections: namely, straight cone, conoidal with cylindrical end piece and diffuser obtained computationally by a graphical method due to Professor F. I. Frankl. The effect of divergence angle of the nozzle on the jet reaction was also investigated. The results of the investigation showed that the shape of the generator of the inner surface of the entrance part of the nozzle essentially has no effect on the character of the flow and on the reaction. The nozzle that was obtained by graphical computation ensured the possibility of obtaining a flow for which the velocity of all the gas particles is parallel to the axis of symmetry of the nozzle, the reaction being on the average of 2 to 3 percent greater than for the usual conical nozzle under the same conditions. For the conical nozzle, the maximum reaction was obtained for a cone angle of 25° to 27°. At the end of this paper a sample computation is given by the graphical method. The tests were started at the beginning of 1936 and this paper was written at the same time.

*Central Aero-Hydrodynamic Institute (TsAGI), Khukovsky, USSR.

Model A-4 (V2 rocket) with tail assembly was to be investigated concerning its aerodynamic properties during flight within the region close to the speed of sound. The investigations within the velocity range of v=14 to 46 m/sec. were conducted by the Zeppelin Aircraft Construction Co. at Friedrichshafen. The results are available in test report 600/164. These investigations were continued from high subsonic velocities to 2.5 times the speed of sound in the Peenemunde supersonic wind tunnel. The measurements provide the aerodynamic coefficients $C_a$ and $C_w$ as basis for the calculation of trajectories, as well as the position of the center of pressure.

*Peenemunde Army Research Station, Germany


The supersonic tunnel of the Heerswaffenamt was constructed to facilitate the development of spin- and fin-stabilized projectiles. The use of a supersonic tunnel for this purpose is a relatively recent undertaking; how much such work can contribute to this development is not generally realized even now. This paper describes the supersonic tunnel, the most powerful and best equipped in Germany, and gives some experimental results. The first part includes a discussion of the various types of supersonic tunnels, and the principles upon which the present tunnel operates are described. The fundamental considerations governing the choice of methods of measurement and the size of equipment are given. There follows a detailed technical description of the installation and the measuring equipment. The second part of this paper contains preliminary experimental results which were obtained in the wind tunnel and concern problems in external ballistics. Fin stabilization, aerodynamic resonance frequencies and aerodynamic damping - all problems that arise in connection with the stabilization of non-rotating projectiles at supersonic velocities - these are clarified by means of Schlieren photographs, three component force measurements, and oscillation records obtained on a model of a fin grenade at $M = 1.4$ and $1.8$. Finally, Schlieren photographs, and the results of measurements of the aerodynamic forces on a non-rotating model of a 15cm Skoda grenade, are presented for the same Mach numbers.


On the basis of Schlieren (refraction) pictures there are treated from the estimates of experience of gas dynamics, those sections which are applicable to the flow in steam turbines. At the mid-point stands the flow in a Laval nozzle. The behavior of supersonic flow has the result that, in the widened part of the Laval nozzle, a jet with curvy velocity distribution is ordinarily formed. Only by the use of particular measures in the shaping of the nozzle, for which a definite minimum length is necessary, is there success, even for supersonic flow, in flattening out the velocity distribution to such a point that it becomes useful for high-velocity canals. The viewpoints which are decisive in this are given. Moreover, the nozzles with oblique sections and the occurrence of compression shocks are treated

no. 66/36g) is North American Report AL-1715, May 1953.

DTIC ATI-1855

This is a three part paper. Part I consists of an exhaustive treatment of the fundamentals of two-dimensional inviscid supersonic flow theory developed by Prandtl and Busemann. Part II discusses the practical application of the Method of Characteristics in two-dimensional supersonic nozzle design and construction. Since the graphical method is an approximate method which, in addition, assumes inviscid flow, the wind tunnel nozzles developed by this method will not provide the required flow pattern. The literature has nothing to say concerning the corrections which are necessary because of these errors. Part III shows how, using the basic concepts of the Prandtl-Busemann method and considering the effect of the wall boundary layers, such a correction may be carried out.

*Wasserbau-Versuchsanstalt, Kochel, Germany.


DTIC ATI-1841

Several types of supersonic nozzles, exit cones, fixed and flexible diffusers were tested to find the most efficient and versatile arrangement for the continuous-operation wind tunnel. A closed chamber would show the least losses, but the models would be hard to reach. To prevent the formation of perpendicular pressure waves, the best exit cone would have to have the same contour as the supersonic nozzle. In this way, most of the speed energy would be recovered and converted into pressure again.

*Wasserbau-Versuchsanstalt, Kochel, Germany.


The flow of air at speeds above sound is essentially different from that below sound and requires special supersonic wind tunnels for investigation. Whereas the older installations could be operated intermittently only, there are to-day plants which can be operated continuously, and which are bringing a valuable contribution to the science of ballistics, aeronautics and turbo-machinery design.


Methods developed in Parts III and V of this series are here extended to the more difficult problem of compressible fluid moving at high speed through a converging-diverging nozzle. Solutions of sufficient accuracy for practical purposes can be obtained for a nozzle of any specified shape, provide that the velocity of the fluid nowhere exceeds the local speed of sound. Otherwise the computed velocities fail to converge - a result similar to what was obtained by Taylor and Sharman using an electrical tank. The reason for this failure is discussed, and an alternative method (not in itself entailing the 'relaxation' technique) is proposed to meet this difficulty. In a subsequent paper this will be applied to determine the supersonic regime.


*Institute for Mathematics and Mechanics, New York University, New York, USA.

Flow patterns for compressible fluids at supersonic velocities are discussed, and it is shown that shock fronts form when neighboring Mach lines (envelopes of wave fronts originating from point disturbances) intersect. A criterion for divergence of Mach lines is developed for cases in which the passage is symmetrical in two or three dimensions and has a straight axis. This criterion is used as the basis for designing supersonic nozzles and diffusers. The analysis indicates that only a nozzle of infinite length can discharge a parallel stream into a tube of constant cross section without the formation of shock fronts. Methods are presented for designing nozzles of finite length, with the intensity of shock fronts reduced to as small a value as possible, and it is shown that nozzles of reasonable length may be designed so that shock fronts are insignificant. Experimental observations indicate that the proposed method of nozzle design is a practical one. With regard to supersonic diffusers having a straight axis, it is shown that shock fronts cannot be avoided, even though the diffuser is of infinite length. However, the methods of this paper may be used as an aid in determining the best diffuser design.

*Department of Mechanical Engineering, Massachusetts Institute of Technology, USA.


5341 CIT/15
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A small-scale supersonic tunnel has been designed, constructed and operated with a working section of 2.50 x 2.56 in. Its characteristics have been investigated completely at Mach numbers of 2.0 and 3.2, at which speeds the phenomena are representative of the operating range. Preliminary tests have been made with a nozzle designed for a Mach number of 4.4, indicating that Mach numbers of this order may be used for model testing. Data necessary for the engineering design of a large-scale supersonic tunnel have been obtained and are given in detail. An optical system has been developed with which high-speed schlieren photographs may be made of the flow in the tunnel. A model support system and balance system have been developed which will permit force and pressure measurements to be made on a model supported in the tunnel at the same time that the flow is being observed optically.

*California Institute of Technology, Pasadena, California, USA.


An iterative method outlined by Green & Southwell (1944), but not applied in their paper for the reason that it makes no use of 'relaxation methods', is here applied to determine the supersonic regime for gas flowing irrotationally through a convergent-divergent nozzle, i.e. that (unique) regime in which the pressure and density of the gas decrease continuously in its passage from end to end. Osbourne Reynolds's approximate treatment of the problem (1886) assumed the velocity to be distributed uniformly over each cross-section, and in consequence found a unique value for the limiting mass-flow, whether the velocity be subsonic or supersonic downstream of the 'throat' (upstream it is always subsonic). Here, a more exact treatment shows that the supersonic value is very slightly (0.045%) greater than the subsonic value, which Reynolds's theory overestimates by 0.083%. The two regimes of course imply two different values of the pressure at exit, and for intermediate pressures (on the assumptions of this paper) there is no solution of the problem. Even in the subsonic regime (when the mass flow is critical) velocities exceeding the local speed of sound are attained in regions adjoining the nozzle walls near the throat.


*Institute for Mathematics and Mechanics, New York University, New York, USA.

This lecture considers the position of aerodynamics in the general aero-thermo-hydrodynamic field and briefly reviews some of the early high speed research and wind tunnels. Aeronautical engineering problems and developments are discussed, in particular the effect of compressibility on control surface effectiveness. The lecture concludes, "Much remains ahead as the problem of increasing speed is solved to the achievement of clock-stopping speeds in east-to-west flight."

*Langley Memorial Aeronautical Laboratory, Langley Field, Virginia, USA.


DTIC ATI-1962

The report describes the planned construction of a supersonic wind tunnel. Special attention is given to the following subjects: layout of the wind tunnel, arrangement of the installation views of the models, of the range of measurements, theoretical pump performance in comparison to the theoretical tunnel performance, distribution of blowers in the exhaust installation, data of the airdry unit, data for the engine, and assembly drawings of the test room.

*Wasserbau-Versuchsanstalt Kochelsee G.m.b.h., Kochel, Germany.


5341 238a
(Ames Research Center library no.)

A method of designing channels to convert uniform supersonic flow of any Mach number into uniform supersonic flow of any other Mach number is given. The theory is exact, neglecting the boundary layer, and tables are given enabling all parts of the channels to be calculated quickly with high accuracy. Use is made of construction of transforming divergent radial flow into a uniform stream, first given by Busemann in 1931; an exact analytic expression has been found for this method. The method of "analytical characteristics" used in the present paper can be applied to any type of supersonic flow and is not restricted to purely radial flow. Finally in the Appendix details of the calculations involved are given. Symmetrical channels only have been considered but any two walls which give the same velocity along the axis can be used.

*Aerodynamics Division, National Physical Laboratory, Teddington, UK.


This paper describes the Daingerfield wind tunnel facilities, including the design and construction of eight nozzles. The test section size is 15" by 20" for one-blower operation and 19" by 27 1/2" for two-blower operation. The methods of calibrating the nozzles are discussed.

*Lone Star Laboratory, Daingerfield, Texas, USA.


A system for calculating the physical properties of supersonic rotational flow with axial symmetry and supersonic rotational flow in two-dimensional field was determined by use of the characteristics method. The system was applied to the study of external and internal flow for supersonic inlets with axial symmetry. For a circular conical inlet the shock that occurred at the lip of the inlet and became a normal shock at the axis. The region in which strong shock occurred increased with the increase of the angle of the internal cone at the lip of the inlet. For an inlet with a central body, the method of characteristics was applied to the design of an internal-channel shape that theoretically, results in a very efficient recompression in the inlet; it was shown that if an effuser is connected with the diffuser a body of revolution with very small shock-wave drag can be determined.

*Flight Propulsion Research Laboratory, Cleveland, Ohio, USA.


DTIC ATI-102029


DTIC ATI-3734

On May 13, 1946, a group of Packard engineers interviewed German scientists from Peenemunde in regard to high speed flow problems and problems associated with experimental flow research. The problems connected with the design and manufacture of supersonic nozzles were discussed with Dr. Hermann and other members of his old research group. In addition effects of condensation and the development of non-intrusive measurement techniques (in particular X-ray and sonic wave techniques) were discussed. Of immediate interest were for instance, Dr. Hermann's statements concerning the required accuracy limits of supersonic test nozzles and Dr. Goethert's comments on the significance of boundary layer removal in the transonic range of flight.

*Packard Motor Car Company, Aircraft Engine Division, Toledo, Ohio, USA.


6410 1071
(Ames Research Center library no.)

A graphical method for the determination of the contours of a two-dimensional nozzle for effecting ideal expansion of gases to a uniform parallel jet was developed by Prandtl and Busemann. The present report develops an analytical solution to the problem to eliminate the labourious graphical construction and the graphical error inherent in the former method. Mathematical equations are developed which permit the immediate determination of the coordinates of any streamline in point source flow. The equations are extended to permit solution of the flow from the parallel throat section of the Laval nozzle.

*Engineering Department, North American Aviation, Inc., Inglewood, California, USA.

A series of solutions for the two-dimensional flow of a frictionless, adiabatic, perfect gas inside of a hyperbolic nozzle have been presented. These solutions show an almost continuous transformation of known subsonic solutions to the known subsonic-supersonic solution. From this report it can be concluded that for the corresponding two-dimensional problem, a perfect fluid theory supplemented with shock waves is still not enough. For adequate check with experiment, a theory must be based upon a fluid with friction (at least in the regions near the walls; in other regions friction would have no effect). Solutions with this imperfect fluid involve a prohibitive amount of labor with present computing techniques.

*Harvard University, Cambridge, Massachusetts, USA.

42. Bibliography on German Supersonic Research. USAF AMC Bibliography-1, June 1946, 118 pp, 784 refs (approx.).

DTIC AD-B953385

This Bibliography supersedes the preprint published in March 1946, listing captured German documents on supersonics and closely related data which are available at the Air Documents Division of Intelligence, T-2, at Wright Field, Dayton, Ohio. Topics covered in this extensive Bibliography include supersonic aerodynamics (theory, experiments, heat transfer, wind tunnels), transonic aerodynamics (theory, experiments, testing techniques), power plants (including Laval nozzles), development of jet and rocket propelled planes and missiles, and boundary layer theory and experiments. No abstracts are given. Many of the titles are in German.


A two-dimensional flow field in which the velocity is everywhere supersonic can always be represented approximately by a number of small adjacent quadrilateral flow fields in each of which the velocity and pressure is constant. These quadrilaterals must be separated by lines representing waves in the flow; changes in velocity and pressure through any wave can be computed. By increasing the number of small area into which the complete flowfield is divided, the accuracy of this approximation may be increased without limit. This constitutes the "method of characteristics" solution, which has been known for many years. This method may be applied to the graphical computation of flow in a supersonic nozzle, with the particular aim of producing uniform supersonic flow at the end of the nozzle. It is pointed out that such a computation is essentially simple and rapid, and its essential features are presented in a form which, it is hoped, may be easily applied to engineering problems.
This chapter examines first compressible flow along a single wall and then progresses to the flow between two walls. A section on supersonic nozzle design discusses the application of the method of characteristics.

An approximate method for three-dimensional axially symmetrical supersonic flows is developed; it is based on the characteristics theory (represented partly graphically, partly analytically). Thereafter this method is applied to the construction of rotationally symmetrical nozzles.

The object of this report is to review the methods used in designing two-dimensional shock free supersonic tunnels. The report concentrates on the method of characteristics.

Two numerical-graphical methods of solution of supersonic flow problems are developed from the basic theory of characteristics. Both are restricted to plane potential shock-free flow. The "lattice-point" method, recommended for flows with given boundaries, is applied to flow in a hyperbolic channel. The "field" method, suitable for problems in which flow boundaries are to be determined, as in the case of supersonic nozzle design, is used for a problem of parallel routing of an initially expanded flow. Complete rules of procedure are presented for each method.
method, a table of useful functions is included.

*Department of Mechanical Engineering, Massachusetts Institute of Technology, USA.


Approximation formulas were developed for the position and form of the critical curve for transition through the critical velocity in the neighborhood of the narrowest cross-section of flat and rounded Laval nozzles. In comparison with nozzle flows calculated by Oswatitsch and Rothstein (1942), they showed satisfactory agreement. In addition, corresponding approximation formulas were deduced for the flow over profiles with local supersonic regions.


*Research Dept., United Aircraft Corp., USA.


6410 1351
(Langley Research Center library no.)

An analytical method is presented for the construction for an inlet of a two-dimensional supersonic shock free nozzle. This construction is an adaptation to two dimensions of the method developed recently (Freidrichs - 1944) for three-dimensional nozzles. It affords an analytical description of the flow of a non-viscous fluid through a two-dimensional Laval nozzle from the subsonic to the supersonic regime, and in particular permits a study of the transition through the speed of sound. The construction is carried out for a nozzle designed to produce exhaust flow of constant speed and direction at Mach number 1.5. Compensation for boundary layer is not included in this construction, and subsequent adjustment of the constructed contour for boundary layer displacement thickness is necessary.

*Research Dept., United Aircraft Corp., USA.


*Ames Aeronautical Laboratory, Moffett Field, California, USA.


*Research Dept., United Aircraft Corp., USA.


*University of Michigan, Ann Arbor, Michigan, USA.


*Princeton University, Princeton, New Jersey, USA.

This chapter considers flow through different nozzles including the De Laval nozzle. Shock patterns in nozzles are considered. The "perfect nozzle" is discussed.

*Institute for Mathematics and Mechanics, New York University, New York, USA.


Equations are presented for obtaining the wall coordinates of two-dimensional supersonic nozzles. The equations are based on the application of the method of characteristics to irrotational flow of perfect gases in channels. Curves and tables are included for obtaining the parameters required by the equations for the wall coordinates. A brief discussion of characteristics as applied to nozzle design is given to assist in understanding and using the nozzle-design method of this report. A sample design is shown.

*Flight Propulsion Research Laboratory, Cleveland, Ohio, USA.


A graphical-numerical method of characteristics, based on the Tollmien method, is developed for axially symmetric isentropic flow problems. Complete rules of procedure are presented for the application of the method. Similar methods have been given by Ferrari and Ferri. As an illustration, the flow around a conical-nosed body is calculated. In this problem a procedure is given for approximating the curved shock wave, assuming constant entropy behind the shock wave.

*The John Hopkins University, Silver Springs, Maryland, USA.


The paper gives a review of present state of development of the method of characteristics for the solution of problems of compressible flow. The theory applies when the flow depends on two independent variables only (one of which may be time); heat radiation, heat conduction, and viscosity are neglected, and the theory of this form appears to give a good approximation in a large number of practical applications. The flow, if steady, is taken to be supersonic.

*Department of Mathematics, University of Manchester, UK.


The theory of supersonic flow in nozzles is discussed, emphasis being placed on the physical rather than the mathematical point of view. Methods, both graphic and analytic, for designing nozzles are described together with a discussion of design factors. In addition, the analysis of a given nozzle shapes to determine velocity distribution and possible existence of shock waves is considered. A description of a supersonic protractor is included in conjunction with a discussion of its application to nozzle analysis and design.

*Ames Aeronautical Laboratory, Moffett Field, California, USA.


5391/4
(Ames Research Center Library no.)

Following a discussion of difficulties in connection with designing nozzles of conventional type for use at low pressures, the possibilities of two new designs, respectively
called a slip plate and a molecular plate, are investigated. The theoretical behavior expected of each is outlined and experimental data for the slip plate nozzle are discussed.

*Department of Engineering, University of California, Berkeley, California, USA.


5341 CIT/14
(Ames Research Center library no.)

A small supersonic wind tunnel, with test section approximately 12" square, has recently been put into operation at the Jet propulsion Laboratory, at the California Institute of Technology. Some of the newer design and performance features of this wind tunnel will be discussed in the present paper. The tunnel has been operated satisfactorily up to Mach number 3.0, and preliminary calibration in this range is in progress.

*Jet Propulsion Laboratory, Pasadena, California, USA.


*US Naval Ordnance Laboratory, White Oak, Maryland, USA.


In this paper attention is confined to the steady irrotational isentropic flow of a perfect gas at supersonic speeds, in particular, flow in which one characteristic is straight. In two dimensions any characteristic along which the velocity is constant in magnitude and direction is straight and, conversely, if a characteristic is straight the velocity must be constant along it; in three-dimensional flow with axial symmetry the velocity along a straight characteristic is not generally constant but varies according to a definite relation with the radial distance. We establish this relation and describe the types of flow corresponding to it.

*Department of Mathematics, University of Manchester, UK.


The problems of experimentation of rarefied gas are discussed. First, the extremely large viscous effects in a wind-tunnel nozzle are shown. Then the difficulties of flow measurement are surveyed, pointing out particularly the unconventional behavior of the Pitot tube in rarefied gas. The performance of a hot-wire anemometer is then studied in some detail to show its feasibility. Finally, the rules of achieving complete flow simulation of rarefied gas flow are formulated.

*Massachusetts Institute of Technology, Massachusetts, USA.


A method is presented for the design of supersonic-wind-tunnel nozzles that produce uniform flow over a continuously variable Mach number range without the use of flexible walls. Experimental results obtained from a preliminary investigation of such a tunnel are included. Overexpansion of the flow in the neighborhood of the throat was observed at all Mach numbers. Boundary-layer effects were noticeable above the design Mach number of 2.0. Mach numbers were obtained in the range of 1.76 to 2.65 with the limits imposed by either tunnel choking or the structural design of the tunnel. At any given flow-turning angle, less than 1-percent variation in Mach number was observed in the test-section flow downstream of the overexpansion region.

The paper deals with the growth and decay of disturbances along Mach lines in isentropic, irrotational, steady, two-dimensional or axially symmetric, supersonic flow; in particular, the distribution of disturbances is investigated along a Mach line in axially symmetrical flow on which the velocity is constant. As an example, the field of flow in the entry of the contractor of circular cross-section is calculated from the focusing laws, and the analytical expressions are compared with the results of the numerical methods of Massau and Tupper. The disturbance generated by the diffuser entry leads to a singularity of the flow pattern on the axis, the nature of which is investigated within the framework of the linear theory.

*Department of Mathematics, University of Manchester, UK.


5341.1 48 (Ames Research Center library no.)

*Case School of Applied Science, Cleveland, Ohio, USA.


This paper concentrates on a few interesting points of future problems in aeronautics and ballistics. The size of future supersonic wind-tunnels is discussed. The use of grating diffusers is proposed to reduce costs in large wind tunnel construction. Grating Nozzle designs are also presented as a viable alternative to conventional designs. Finally, testing techniques for heat-transfer experiments are discussed. It is concluded that temperature control of the tunnel walls in a wide temperature range must be provided for.

*US Naval Ordnance Laboratory, White Oak, Maryland, USA.


QA 930 F4 (Ames Research Center library no.)

This book section considers the theoretical design of the supersonic part of a two-dimensional effuser. The case of supersonic two-dimensional jets is analyzed in detail with flow visualization.

*Langley Memorial Aeronautical Laboratory, Langley Field, Virginia, USA.
The equations for the nozzle's contours are derived by integration of the characteristic equation of the axially symmetric flow. Since it is not possible to integrate these equations mathematically in an exact form, it was necessary to find a way to approximate the calculations. The approximation offers itself by considering and comparing the conditions of the flow in a cone with those in a nozzle, as a linearization of the characteristic equations. The first part of the report deals with equations for the transition curve by which the conical source flow is converted into a parallel stream of uniform velocity. The equations are derived by integration along a Mach line of the flow in the region where the conversion takes place. A factor $f$ is introduced expressing a relation between the direction and the velocity of the flow along a certain Mach line. $f$ remains undetermined and is not involved in the final equations. In the second part of the report, the spherical sonic flow section is converted into a plane circular section of the throat. The nozzle's contour adjacent to the throat is formed by the arc of a circle connected with the transition curve by a straight line. The gas dynamic properties of the boundary Mach line are calculated in Table I, the use of which shortens the calculations considerably.

*Engineering Department, North American Aviation, Inc., Inglewood, California, USA.


The present report describes a new method for the prediction of the flow pattern of a gas in the two-dimensional and axially symmetrical case. It is assumed that the expansion of the gas is adiabatic and the flow stationary. The several assumptions necessary on the nozzle shape effect, in general, no essential limitation on the conventional nozzles. The method is applicable throughout the entire speed range; the velocity of sound itself plays no singular part. The principle weight is placed on the treatment of the flow near the throat of a converging-diverging nozzle. For slender nozzles, formulas are derived for the calculation of the velocity components as functions of the location.

*Palmer Physical Laboratory, Princeton University, Princeton, New Jersey, USA.


5371 18 (Ames Research Center library no.)
In this report the ordinates are determined for an axially-symmetric supersonic nozzle to be used at pressures in the range 50-100 microns Hg at the test section (nozzle exit). The nozzle is designed on the basis of an isentropic core corrected for boundary layer growth to produce a parallel flow at Mach number of 4.17 at the test section. Preliminary tests on a similar nozzle which is not corrected for boundary layer growth are discussed.

*Department of Engineering, University of California, Berkeley, California, USA


5341 NOL/3
(Ames Research Center library no.)

The 18 x 18 cm tunnel is a versatile research tool designed to make possible detailed experimental study of primary problems such as; (1) how the design of the various parts of supersonic wind tunnels can be improved (the diffuser, for example) and (2) how such instruments as the interferometer can be used to obtain aerodynamic data. This report covers in detail the mechanical features of the 18 x 18 cm tunnel and is intended as a guide for the operation of the tunnel and the planning of the experimental work with it. The design criteria used in arriving at the present design and some of the types of experimental work planned for the tunnel are indicated.

*US Naval Ordnance Laboratory, White Oak, Maryland, USA.


Part I of this report contains a description of general tunnel characteristics and measuring facilities available without regard to a specific Mach number. The subsequent sections present the aerodynamic flow calibration of the tunnel at specific Mach numbers. At present this will include only the Mach 1.90 configuration. As additional nozzle blocks are calibrated, supplementary sections will be added to this basic report. A description and analysis of the force measuring facilities available will be the subject of a separate report.

*Engineering Research Institute, University of Michigan, Ann Arbor, Michigan, USA.


TL 573 B58
(Ames Research Center library no.)

This chapter examines one-dimensional compressible flow. The basics are reviewed followed by discussion of the problems of supersonic nozzle design. Some flow visualization is presented to emphasis their role in obtaining a De Laval nozzle

*Applied Physics Laboratory, The John Hopkins University, Silver Springs, Maryland, USA.


TL 573 P6
(Ames Research Center library no.)

This chapter provides an overview of the current state of the art in supersonic wind tunnels. The design of supersonic tunnels is discussed and a list of operational supersonic tunnels presented. Simple supersonic nozzle design is considered. Also apparatus for flow visualization in supersonic tunnels is considered

*Daniel Guggenheim School of Aeronautics, Georgia Institute of Technology, Georgia, USA.

*University of Michigan, Ann Arbor, Michigan, USA.


*University of Michigan, Ann Arbor, Michigan, USA.


The theory of the flow through a throat near sonic velocity is developed, and is followed by a discussion of the conventional method of designing supersonic nozzles using the method of characteristics. A method of improving the Mach number distribution of the nozzle using the experimental results is developed. The nozzles designed were tested in a 3-in. square wind tunnel in which the Mach number distribution was obtained by shaping the top wall of the working-section. Considerable difficulty was found in improving the distribution; this was considered to be due to the discontinuity in curvature at the point of inflexion and the influence on the boundary layer of the sudden relaxation of the pressure gradient along the wall. An alternative method of design was developed which avoided this discontinuity in curvature, and considerably better results were obtained when attempts were made to improve the experimental Mach number distribution. The flow through the throat of the liners was determined experimentally and compared with the theory. The agreement was good on the whole, although there were differences in the subsonic entry region because the bottom wall only became flat a short distance before the throat. In addition, tests were made which showed that the assumption of two-dimensional flow through the throat was justified. The method developed to improve the distribution in the nozzle was extended to derive liner shapes for Mach numbers differing by 0.10 from the design Mach number. It was found that changes of this order could be made fairly successfully but further modification was necessary to reach the standard required for tunnel use. The necessity for a smooth and accurately constructed liner surface is stressed. The limitations of the methods used for designing supersonic nozzles are discussed and several problems are mentioned which is thought will need further consideration.

*Royal Aircraft Establishment, UK.


When a uniform, two-dimensional supersonic flow expands suddenly round a corner in a wall it forms a pattern known as a Prandtl-Meyer expansion or centered simple wave. If the flow is two-dimensional but not initially uniform, or if it is axially-symmetrical, the expansion is still centered, but is not a simple wave. An approximate solution is given in this paper for the isentropic, irrotational, steady two-dimensional or axially-symmetrical flow of a perfect gas in the neighbourhood of the center of such an expansion. The solution is designed to replace the conventional method of characteristics in such a region. The main application is to a jet issuing from a nozzle that discharges into a container with pressure lower than that in the nozzle; in particular, a formula is derived for initial curvature, at the lip of the nozzle, of the boundary of the jet. The solution also applies to the flow near an edge in a boundary wall, and a formula is derived for the velocity gradient on the wall immediately downstream of the edge.

*Mechanical Engineering Laboratory, The Technical University of Denmark, Denmark.

97. *Beavan, J.A.; and Holder, D.W.: Recent Developments in High Speed Research - In the Aerodynamics Division of the National
This lecture reviews the vast amount of high speed research carried out at NPL since 1935. The lecture is divided into sections. First, the NPL high speed tunnels are described including the latest 18 x 14 inch transonic tunnel. Next, the measuring instruments and experimental techniques are reviewed in detail. Then recent experiments are reviewed including fundamental and applied research at supersonic speeds. Finally, the scope of future work is discussed.

*National Physical Lab., Teddington, UK.


The design, development, and performance of equipment suitable for use by educational institutions for student training and basic compressible-flow research are described. The equipment consists of an induction tunnel having a 4- by 16-inch test section and capable of operating at Mach numbers ranging from about 0.4 to 1.4 and a blowdown tunnel having a 4- by 4-inch test section for supersonic Mach numbers up to about 4.0. Nozzle pressure distribution tests made in the induction tunnel showed satisfactory performance at subsonic speeds. At low supersonic speeds (Mach numbers 1.2 and 1.4), however, adverse condensation effects were encountered when test were made under high-humidity conditions. The supersonic nozzles of the blowdown tunnel produced average Mach numbers close to the design values and sufficient uniform velocity distributions for most of the intended uses of this equipment.

* * Lewis Flight Propulsion Laboratory, Cleveland, Ohio, USA.


Boundary-layer measurements were made in the transonic and supersonic regions of a channel having maximum cross-sectional dimensions 3.84 by 10 inches and designed by potential-flow methods for a uniform Mach number of 2.08 in the test section. At inlet pressures from 37 to 13 inches of mercury absolute, turbulent boundary layers were observed throughout the channel; at an inlet pressure of 5 inches, laminar boundary layers were observed near the channel entrance with turbulent layers downstream. A comparison of the experimental and theoretically computed boundary layers at the high inlet pressures showed good agreement when empirical friction coefficients were evaluated from Reynolds numbers based on the kinematic viscosity of the air at the wall. Despite this agreement between experiment and theory, local differences in rates of boundary-layer growth still existed that are attributed to secondary flows in the boundary layer. At low inlet pressures, substantial increases in the boundary-layer rates of growth with an uneven development of the boundary layer along the bottom wall of the channel were observed. Experimental and empirical skin-friction coefficients were in poor agreement at all inlet pressures. Secondary flows in the boundary layer caused by static-pressure gradients transverse to the upstream direction are believed to be the reasons for the poor agreement.

* * Lewis Flight Propulsion Laboratory, Cleveland, Ohio, USA.


The paper is concerned with the two-dimensional, steady, irrotational, isentropic flow of a perfect gas in a wind tunnel nozzle which is found to produce a flow in the test rhombus deviating slightly from the desired uniform flow. The minimum corrections are derived that must be applied to the liners in order to produce a uniform flow in the test rhombus. If the uncorrected nozzle produces a flow of uniform direction, measurement of the pressure on the axis, in the test rhombus, suffices to determine these corrections. If not,
further pressure measurements are required. A simple test is indicated for checking whether the flow stream direction is uniform. The method cannot be used to correct for deviations from a two-dimensional flow.

*Department of Mathematics, University of Manchester, UK.


An investigation to determine the effect of distributed boundary layer suction on the pressure recovery of a supersonic wind tunnel has been conducted in a 3.84- by 10-inch tunnel operating at a Mach number of 2.0. With suction applied to two walls of a constant area section in the vicinity of the normal shock, a reduction of 4 percent of the operating pressure ratio was obtained. This reduction was attributed to an improvement (reduction in Mach number) in the flow characteristics at the subsonic-diffuser inlet. The normal shock predicted by one-dimensional theory was, in practice, replaced by a multiple-branch shock configuration. The change in static pressure, total pressure, and Mach number occurred gradually in the streamwise direction and finally approached the predicted Rankine-Hugoniot values.

*Lewis Flight Propulsion Laboratory, Cleveland, Ohio, USA.


The second section deals with high-speed tunnel apparatus and techniques. This section covers such topics as measurements in high-speed wind tunnels and flow visualization.

*National Physical Laboratory, Teddington, UK


*Jet Propulsion Laboratory, Pasadena, California, USA.


*Princeton University, Princeton, New Jersey, USA.


The design of a small supersonic wind-tunnel test section (4 by 10 in.) incorporating a flexible nozzle is outlined. The flexible nozzle consists of a high strength stepped steel plate. Two screw jacks provide an easy means of continuously changing the nozzle's shape according to the aerodynamic requirements. The boundary-layer compensation can also be varied during operation. Pressure surveys, together with schlieren and interferometric analysis of the test section, show the flow to be uniform over the operating range (M = 1.1 to 1.5).

*Guggenheim Aeronautical Laboratory, California Institute of Technology, Pasadena, California, USA.

The idea is advanced of making a supersonic nozzle by producing one, two, or three successive turns of the whole flow; with the result that the wall contour can be calculated exactly by means of the Prandtl-Meyer "Lost Solution."


N-12619
(Langley Research Center library no.)

In this report certain aspects of the design of a nozzle for a supersonic wind tunnel are examined. Modifications to the standard Prandtl-Busemann method are suggested which should improve the agreement between theoretical flow patterns and that actually achieved in practice. Consideration is given to the questions of the subsonic and transonic portions of the nozzle, the continuity of curvature on the boundary of the supersonic part, the taking of the mean values on the characteristics and the boundary layer correction. It is believed that the suggestions here made constitute an improvement on the Prandtl-Busemann method in its simplest form, without unduly complicating the actual computation. Part II of the report, to be published shortly (Unfortunately, Part II appears never to have been published - SW), will deal with the numerical application of the theory.

*Physical Research Division, Armament Research Establishment, Ministry of Supply, UK.


*Naval Supersonic Laboratory, Massachusetts Institute of Technology, Massachusetts, USA.


N-21010
(Ames Research Center library no.)


Auxiliary boundary conditions are derived to assure continuity of wall curvature in applying the method of characteristics to the design of two-dimensional symmetrical supersonic nozzles. An illustrative example is included.

*Lewis Flight Propulsion Laboratory, Cleveland, Ohio.


Since Prandtl's 1932 discussion on the attainment of uniform flow in a low-speed constant density, wind tunnel, designers have been acutely conscious of the benefits of high contraction ratio as a means of reducing the dimensionless measure of steady-flow irregularity in the working section. Two simple generalizations are of interest: (a) the effect of upstream density (temperature) irregularities on the working section velocity irregularities in
a low-speed tunnel; and (b) extension to compressible flow of Prandtl's rough estimate for the effect of upstream irregularities, now in both mean velocity and stagnation temperature.

*Aeronautics Department, The Johns Hopkins University, Baltimore, Maryland, USA.


A method is presented for obtaining three-dimensional unsymmetric supersonic nozzles and inlets from known axisymmetric flows. Streamlines bounding the desired exit shape are traced through the known basic flow solution to give the required unsymmetrical wall contours. Several examples are given.

*Lewis Flight Propulsion Laboratory, Cleveland, Ohio, USA.


Many nozzles for supersonic wind tunnels are now designed using the Foelsch method with a cubic initial curve. Nozzle contours obtained from the Foelsch method have inherently a discontinuity in curvature at the "inflexion" point. Since a flexible plate cannot be bent to follow a nozzle contour having a discontinuity in curvature, it seems desirable to develop a method for designing nozzles having continuous wall curvature at all points. The basic formulas for designing such a nozzle are presented herein.

*Naval Ordnance Laboratory, White Oak, Maryland, USA.


The purpose of this review, which was started in January 1951, in conjunction with the author's work on the design of the UTIA 5- by 7-inch supersonic wind tunnel, is to present a treatment of some of the fundamental supersonic wind tunnel design and performance problems. Unfortunately, due to the limited time available, many important details have only been briefly mentioned. However, the author feels that the selected topics which have been reviewed in detail are at least among the important fundamentals of supersonic wind tunnel design and performance.

*Institute of Aerophysics, University of Tennessee, USA.


An investigation of the use of solid fences installed on the side walls of a supersonic wind tunnel to retard the development of transverse flow and to increase the uniformity of the side-wall boundary layer is reported. Beneficial results were obtained with fences which had depths of the order of the boundary-layer displacement thickness and which followed potential-flow streamlines through the nozzle. Reduction of the number of fences on each side wall from four to two eliminated their effectiveness.

*Lewis Flt. Prop. Lab., Cleveland, Ohio, USA.


This note discusses an alternative to the Crown method for the design of supersonic nozzles with continuous wall curvature. The alternative method uses a physically desirable Mach number distribution along the centerline of the nozzle. In this case, the point of inflection must be upstream of the end point of the patching Mach line. This patching Mach line divides the region of curved intersecting characteristics from the simple wave region with straight nonintersecting characteristics.

The flow characteristics of three axially symmetric multi-nozzles of nominal design Mach numbers 2.46, 3.07, and 7.01 and a two-dimensional multi-nozzle of nominal design Mach number 3.07 were investigated. Three types of disturbance, classified as oblique shock waves, corner shock waves, and wakes, were observed in the flow fields. The effect of the geometry of the multi-nozzles on these disturbances and on the actual Mach number and pressure recovery obtained with the multi-nozzle is discussed herein. The magnitude of the disturbances was found to depend on the exit turning angle of the individual nozzles. Generally, Mach number variations of 4.5 percent or less existed in a region which could be used as a test section. The use of multi-nozzles, however, appears to be restricted to experiments in which nonuniformity of flow and large pressure losses can be tolerated and for which simple fabrication and quick interchange of nozzles are desired.


121. British Development of Variable Mach Number Supersonic Wind Tunnels. RAE Aero 344, 1953.


The method of characteristics is described. Then the method is developed in two ways: first by an extension of the linearized theory, in which physical concepts are emphasized; and second, by formal mathematical methods. The latter development is more rigorous and leads to a better appreciation of the concept of characteristic curves, an appreciation which will be useful in attacking other types of problems involving the method of characteristics. A section is devoted to nozzle design which considers amongst other types, sharp-corner nozzles.


The idea is advanced of making a supersonic nozzle by producing one, two, or three successive turns of the whole flow; with the result that the wall contour can be calculated exactly by means of the Prandtl-Meyer “Lost Solution.”


DTIC ATI-1836

To obtain a test section which will be free of structure wave and have a constant velocity along its axis, corrections have to be made on the initial design of the contour of wind tunnel nozzles. The first correction deals with failures which are caused by small errors in design and manufacture only. The second correction considers the fact that air is not an ideal gas but, owing friction on the enclosed walls, produces a boundary layer. To illustrate the described methods, the correction of a nozzle designed for $Ma = 1.57$ is calculated.


N-24357 (Ames Research Center library no.)

This paper briefly describes common methods for establishing supersonic flow for wind tunnels, and shows the range of requirements best met by flexible nozzles. A method of designing nozzle shapes well adapted to use in conjunction with flexible nozzles is presented along with some experimental results. Mechanical and structural design problems are discussed, and some recommendations made.


N-24360
(Ames Research Center library no.)

A new type variable supersonic nozzle is described. This nozzle has rigid upstream walls which are joined by flexible plates at or downstream of the inflection point. Cancellation boundaries are obtained by controlling the flexible plate and conditions only (i.e., without intermediate jacks). Variation of Mach number is achieved solely by symmetrical rotation of the rigid walls about fixed centers.


N-32079
(Ames Research Center library no.)

This report is a tabulation of Mach number, Mach angle, and area ratio vs Prandtl-Meyer angle. The tabulation was made to facilitate design of supersonic nozzles up to Mach number = 10 and was computed in increments of 0.01° in Prandtl-Meyer angle through the range $\psi=0$ to $\psi=102.34$.


N-31301
(Langley Research Center library no.)

Method of obtaining the contour corrections necessary for the production of a satisfactory
flow uniformity at the exit of a flexible plate supersonic nozzle are reviewed. Basic requirements of the methods are considered, and in particular it is found that contour corrections, small in comparison with the boundary-layer thickness, produce flow changes which cannot be described by the simple theory of characteristics. A procedure involving use of experimentally determined jack influences was successfully applied to correction of nozzle profiles in a 12-in. supersonic wind tunnel. Considerations of further application of this method are discussed.

*ARO Inc, Arnold AFS, Tullahoma, Tennessee, USA.


QA 913 S497 (Ames Research Center library no.)

These sections review the methods available for studying mixed subsonic and supersonic flow. First the solution by power series is considered for the flow in a supersonic nozzle throat. The shape of the sonic line is discussed for two-dimensional and axisymmetric nozzles. Finally, the relaxation method is applied to two-dimensional nozzle with hyperbolic walls.

*Department of Mechanical Engineering, Massachusetts Institute of Technology, USA.


N-32646 (Ames Research Center library no.)

A manufacturing method is described that eliminates the tedious and time-consuming process of correcting tunnel liners during the calibration process. It is found that, since no error is introduced in machining the liner profile from a master template, a satisfactory flow is obtained provided a suitably smooth template is used. The manufacturing technique described has become the standard method for producing nozzles for both the NAE 10-inch and 30-inch tunnels. Experiments in the NAE tunnels demonstrate a flow non-uniformity of not more than ±1% about the mean measured Mach number. To-date the work has been confined to two-dimensional nozzles but is now being extended to axi-symmetric nozzles.

*National Aeronautical Establishment, Ottawa, Canada.


TL 500 N6 (Ames Research Center library no.)

A study has been made of the operating characteristics of intermittent wind tunnels, particularly of the blow-down type. For the same test Mach and Reynolds numbers, the blow-down tunnel is considerably cheaper to build and to operate than the comparable continuous tunnel. For tests of many different set-ups, as required for research and preliminary development, the flexibility of the blow-down jet may be more important than long running time. Moderate size equipment of the blow-down type can reach full scale Reynolds numbers with full scale airplane deflections simulated in the model tests. The detailed performance of the components of such equipment have been obtained from calculations and experience. Typical construction, test set-up, instrumentation, and extensions to the transonic and hypersonic region have been discussed. Some examples of operating intermittent tunnels are also given.

*Polytechnic Institute of Brooklyn, New York, New York, USA.


AD-55902
The design procedure for the family of supersonic nozzles in use at the Naval Supersonic Laboratory of the Massachusetts Institute of Technology is herein presented. The basis for the potential-flow design is the analytical method first outlined by Freidrichs, to which has been added some results which furnish a higher order of approximation to the nozzle contours. The implied mathematical convergence in the method is discussed and reference is made to continuous curvature streamlines. The accumulated computations of the Laboratory are tabulated for the convenience of those wishing to avail themselves of the family of contours successfully used. Experimental results from calibrations conducted within the uniform flow regime and at the nozzle boundaries are presented, and interpretations are given in terms of the design, fabrication, measurement accuracy, and model requirements.

*Naval Supersonic Laboratory, Massachusetts Institute of Technology, USA.


A semi-analytical method for the design of flexible-plate, potential-flow supersonic nozzle contours is presented. The method consists essentially of a fairly rigid control of the second and third derivatives of the nozzle displacement curve, both upstream and downstream from the inflection point. Using the method of characteristics as a tool, the technique of designing a completely flexible or combination of solid- and flexible-plate nozzle is illustrated. The presentation includes a table and graphs that have proved helpful in the design of supersonic nozzles at the Jet Propulsion Laboratory of the California Institute of Technology.

*Jet Propulsion Laboratory, Pasadena, California, USA.


Heat-transfer rates have been measured at the plane boundary of the flow through two-dimensional supersonic nozzles in the variable density, continuous-type wind tunnel at the Naval Supersonic Laboratory. A range of Mach number, Reynolds number, and pressure gradient was obtained by the use of fixed-block nozzles designed for exhaust flow at Mach numbers of 2.5, 3.0, and 3.5. Heat transfer was induced by cooling the outer surface of the channel wall to ice and to dry-ice temperatures which produced a ratio of surface-to-stagnation temperatures as low as 0.7. Despite the presence of pressure gradients, the turbulent recovery factor of 0.89 was found to be independent of Mach and Reynolds number. Local heat-transfer coefficients were in reasonable agreement with the flat-plate constant-surface-temperature theory of Van Driest after correcting for the varying conditions.
temperature potential history. Pressure-gradient influences were not apparent downstream of the nozzle throat but were distinguishable in the subsonic region. The results are in qualitative agreement with an analysis presented for the turbulent boundary layer with constant pressure gradient.

*Naval Supersonic Laboratory, Massachusetts Institute of Technology, Massachusetts, USA.


*Ordinance Aerophysics Laboratory, Daingerfield, Texas, USA.

AD-48625

An analysis is made of the effects of non-uniformity of flow on the pressure measurements on the surface of the model and also on the force and moment measurements and the following standards of flow uniformity are derived - variations in flow direction to be less than ±0.1° in the range M = 1.4 to 3; variation in Mach number to be less than ±0.003 at M = 1.4 increasing to ±0.01 at M = 3. A brief analysis is made of the errors in model manufacture and their effects on force and pressure measurements. Using the same standards as were used in deducing the requirements for flow uniformity quoted above it is concluded that present standards of model manufacture are satisfactory overall, though for accurate pressure plotting tests at low supersonic Mach numbers (M ≈ 1.4) a higher standard is desirable.

*National Aeronautical Establishment, Ottawa, Canada.


AD-56640

This report is a survey of the techniques of two-dimensional wind tunnel nozzle design. A procedure for the aerodynamic design of flexible nozzles capable of continuous Mach number variation is developed in detail. The special structural, mechanical, calibration and cost estimation problems involved in flexible nozzle construction are discussed.

*Sandberg-Serrell Corp., Pasadena, California, USA.

(Ames Research Center library no.)

During the design of the supersonic wind tunnel of the Gas Dynamics Facility at AEDC, a considerable effort has been expended in developing methods of prediction of precision, largely because the intended use of this tunnel places unusually high demands on the precision of test data. In this report, some of the concepts which have been utilized in this prediction are summarized. The influence of nozzle shape errors and condensation on test data is reviewed.

*ARO Inc., Arnold AFS, Tullahoma, Tennessee, USA.


AD-074144

Heat transfer rates have been measured at the plane boundary of the flow through two-dimensional supersonic nozzles in the variable density, continuous-type wind tunnel at the Naval Supersonic Laboratory. A range of Mach number, Reynolds number, and pressure.
gradient was obtained by the use of fixed block nozzles designed for exhaust flow at Mach numbers of 2.5, 3.0, and 3.5. Heat transfer was induced by cooling the outer surface of the channel wall to ice and to dry-ice temperatures which produced a ratio of surface-to-stagnation temperature as low as 0.7. Despite the presence of pressure gradients, the turbulent recovery factor of 0.89 was found to be independent of Mach and Reynolds numbers. Local heat-transfer coefficients were in reasonable agreement with flat-plate constant-surface-temperature theory of Van Driest after correcting for the varying temperature potential history. Reynolds-number bases (extending to \( 3.5 \times 10^7 \)) were free-stream conditions and the effective lengths of boundary layer growth. Pressure-gradient influences were no apparent downstream of the nozzle throat but were distinguishable in the subsonic region. The results are in qualitative agreement with an analysis presented for the turbulent boundary layer with constant pressure gradient.

*Naval Supersonic Laboratory, Massachusetts Institute of Technology, USA.


A Procedure is given for designing twodimensional nozzles in which the streamline coordinates are computed directly from tabulated flow parameters and appropriate equations. The method of characteristics is used to obtain the first part of the flow, which consists of a continuous expansion from a uniform sonic flow to a radial flow. The Foelsch equation's are then used for the transition from radial flow to the final uniform flow. Information is presented which enables the designer to select and compute rapidly the wall contour for any nozzle or series of nozzles for a wide range of length-to-height ratio, Mach number, and wall angle at the inflection point. In general, a nozzle is determined by specifying any two of these three parameters.

*Langley Aeronautical Laboratory, Langley Field, Virginia, USA.


Methods of obtaining the contour corrections necessary for the production of a satisfactory flow uniformity at the exit of a flexible plate supersonic nozzle are reviewed. Basic requirements of the methods are considered, and in particular it is found that contour corrections, small in comparison with the boundary-layer thickness, produce flow changes which cannot be adequately described by the simple theory of characteristics. A procedure involving use of experimentally determined jack influences was successfully applied to correction of nozzle profiles in a 12-in. supersonic wind tunnel. Considerations of further application of this method are discussed.

*ARO Inc, Arnold AFS, Tullahoma, Tennessee, USA.


Some difficulties encountered in the calibration of a supersonic wind-tunnel nozzle have recently been reported by Murphy (1953). We have found that the expansions in supersonic wind-tunnel nozzles are generally isentropic (in the absence of condensation) and that the most reliable method of calibration is to determine Mach number from the ratio of pitot pressure \( P_p \) to \( P_T \). A duplication of this work in a hypersonic nozzle seems desirable.

*Naval Supersonic Laboratory, Massachusetts Institute of Technology, USA.


A method of designing a symmetric Mach number nozzle with adjustable flexible walls has been developed. The continuous variation of the Mach number is controlled by one single
suitable choice of the linear thickness derivatives of the inlet and throat curvatures simultaneously affects the coordinates and the Mach number gradient must go to zero, the nozzle contour can be calculated. After the patching Mach line is curvature conditions established, the coordinates of the perfect-fluid profile, a set of partial differential equations is obtained. These equations are solved by expanding the dependent variables - velocity \( q \), flow direction \( \theta \), and curvilinear coordinates\((x, y)\) of a streamline - in powers of the stream function. This solution determines the nozzle contours for any constant value of the stream function as well as all flow characteristics in the subsonic region and the supersonic expansion region of the nozzle. From the general solution and the boundary conditions at the sonic line it follows that the curvature of the contour must be zero at the throat. After the patching Mach line is calculated numerically the recompression contour can be determined to complete the nozzle calculations. These equations show that the Mach number gradient must go to zero at the design Mach wave. This requirement implies continuous wall curvature at the corners of the recompression region. The variation of Mach number by relative translation of fixed contours of the asymmetric type is then discussed. An iterative method is derived for this purpose which should converge very rapidly as compared to the iteration required by the characteristics method of design.

*Division of Steam Engineering, The Royal Institute of Technology, Stockholm, Sweden.


An analytic method is presented for the design of two-dimensional asymmetric curved nozzles. Such nozzles may be used to continuously vary the Mach number by relative translation of one contour with respect to the other. The continuity equation and the condition of irrotationality for steady compressible flow are given in curvilinear coordinates with stream function and velocity potential as independent variables. Using Bernoulli's equation for isentropic gas and a suitable chosen analytic velocity distribution along the initial expansion contour, which is specified in terms of its curvature, a set of partial differential equations is obtained. These equations are solved by expanding the dependent variables - velocity \( q \), flow direction \( \theta \), and curvilinear coordinates\((x, y)\) of a streamline - in powers of the stream function. This solution determines the nozzle contours for any constant value of the stream function as well as all flow characteristics in the subsonic region and the supersonic expansion region of the nozzle. From the general solution and the boundary conditions at the sonic line it follows that the curvature of the contour must be zero at the throat. After the patching Mach line is calculated numerically the recompression contour can be determined to complete the nozzle calculations. These equations show that the Mach number gradient must go to zero at the design Mach wave. This requirement implies continuous wall curvature at the corners of the recompression region. The variation of Mach number by relative translation of fixed contours of the asymmetric type is then discussed. An iterative method is derived for this purpose which should converge very rapidly as compared to the iteration required by the characteristics method of design.

*Division of Steam Engineering, The Royal Institute of Technology, Stockholm, Sweden.


In order to cover continuously a rather wide range of Mach numbers, several supersonic wind tunnels have flexible-plate nozzles incorporated in their design. The elastic wall of a flexible plate supported at discrete points must have continuous curvature. It is, therefore, desirable that the curvature of the supersonic contour be continuous to enable the flexible plate to conform to the desired contour. A method is presented herein for designing two-dimensional supersonic nozzles having continuous curvature. Approximations are made for determining the length of the nozzle and for correcting the perfect-fluid profile for the effect of the boundary-layer growth. Computing terms are included for making the successive approximations that are necessary to establish the angles of the inflection and characteristic points that will satisfy the desired requirements of nozzle length, throat radius, and continuous third derivative at the inflection point. After these angles are established, the coordinates of the perfect-fluid profile, downstream of the inflection point, can be obtained by the characteristics method.

*ARO Inc, Arnold AFS, Tullahoma, Tennessee, USA.

152. *Carafoli, E.: High Speed Aerodynamics (Compressible Flow). Chapter 20 -

TL 573 C313
(Ames Research Center library no.)

This chapter describes the calculation of a two-dimensional effusor. Both graphical and analytic methods are used to determine the contour of the effusor.

*Institute of Applied Mechanics, Bucharest, Rumania.


Small, steady perturbations of general two-dimensional steady, shock-free, supersonic flows are studied and the perturbation fields of symmetrical nozzle flows are described in detail. The relation between errors in the shape of the supersonic part of the nozzle liners and the deviations from uniformity of the flow of the test section is given to an approximation sufficient for the treatment of a number of problems arising from experiment and in the design of nozzles.

*Department of Aeronautics, University of Sydney, Australia.


*Aeronautics Department, The John Hopkins University, Baltimore, Maryland, USA.


*Ordnance Aerophysics Laboratory, Daingerfield, Texas, USA.


An analytical method is presented for determining the contours of two-dimensional supersonic nozzles having continuous curvature. The continuity of curvature is necessary for flexible-plate type nozzles inasmuch as the aerodynamic contour must be such that it can be closely simulated by the elastic curve of the flexible plate. The method described is more accurate and less time consuming than the characteristics method of obtaining contours. The assumption is made that radial flow can be obtained at the inflection point through the use of a simple polynomial for the initial part of the contour. This radial flow is converted to parallel flow at the test section Mach number as described herein.

*ARO Inc., Arnold AFS, Tullahoma, Tennessee, USA.


The mechanism by which errors are propagated through a computation is analyzed, and the resultant effect of various types of errors determined. A procedure is outlined for planning computations to achieve specified accuracy with minimum labour.

*Department of Aeronautical Engineering, The University of Sydney, Australia.

Measurements of subsonic and supersonic Mach numbers in air are discussed from the point of view of calibration measurements of an empty wind tunnel, of measurements of local Mach numbers at points in the flow field around a model, and of simulating free flight Mach number in the presence of wind tunnel wall interference. Errors in deducing Mach number from particular measurements are discussed and certain measuring procedures recommended.

*Naval Supersonic Laboratory, Massachusetts Institute of Technology, USA.


AD-121412
N-51372

This report describes a new method for computing supersonic nozzle potential flow contours which is based on a combination of three existing theories. The contour is determined by specifying the nozzle exit dimension and by choosing a Mach number distribution function for the centerline of the entire nozzle. The new method can be used to design fixed block, semi-flexible, flexible nozzles by choosing a suitable centerline Mach number distribution. The new method is also adaptable to high-speed computing machines; therefore, the contours can be computed sufficiently accurately to obtain maximum utilization of present fabrication methods.

*US Naval Ordnance Laboratory, White Oak, Maryland, USA.


This paper reviews the systematic studies of drag, stability, control, and aerodynamic heating of supersonic bodies which had to be initiated before long range rockets like the A-4 could be successfully launched. Supersonic wind tunnel tests are discussed together with the facilities used. It is concluded that this pioneering work did indeed help the V-2 to fly, and it prepared the ground for the future investigations in which we are still involved.

*Aeroballistic Research Department, US Naval Ordnance Laboratory, White Oak, Maryland, USA.


QC 145 H35 (Ames Research Center library no.)

Two-dimensional laminar boundary-layer flow is considered for a perfect gas with Prandtl number unity and viscosity proportional to temperature. The physical situation represented by these equations is flow through a convergent-divergent channel with constant surface temperature and with sonic velocity at the throat. It is shown that flows with large Falkner-Skan parameters may be investigated analytically by means of series expansions provided that there is no heat transfer at the walls. If there is heat transfer, however, it becomes necessary to use a singular expansion procedure in two space variables.

*Consultant to California Institute of Technology, Pasadena, California, USA.


N-57799
(Ames Research Center library no.)

An iterative technique for rapid and accurate calibration of a flexible-plate supersonic-wind-tunnel nozzle is described. The technique depends mainly on an automatic pressure-ratio or difference-plotting apparatus and a knowledge of the jack influence curves. Calibration results are presented for a family of
continuous-third-derivative nozzles at 10 Mach numbers between 2.20 and 4.04 in the 12-in. wind tunnel at the Jet Propulsion Laboratory. The design of continuous-third-derivative supersonic-wind-tunnel nozzles was described in Report no. 20-74 by Riise (1954).

*Jet Propulsion Laboratory, Pasadena, California, USA.


The peculiarity of transition through the velocity of sound in a plane nozzle, i.e., in the case when the tangent to the sonic line coincides with the direction of the characteristics which pass through the axis of the channel, was pointed out by Khristianovich. Later Frankl, on the basis of hodograph transformation, investigated in detail the character of a plane stream in the vicinity of the sonic line. Applying a direct method Falkovich obtained the main term of the solution in the form of a third degree polynomial which considerably simplified all the calculations of the transition region of the nozzle. In the present work on the basis of this solution, some properties of flow with axial symmetry are investigated. The investigation is based on the method of Falkovich, since, in this case, it is not necessary to make use of the hodograph transformation.


This article describes an application of a computer to the aerodynamic design and automatic control of a large wind tunnel. A new mathematical approach was used to avoid singularities in the partial differential equations governing the flow. The work was started in 1953 on the ACE Pilot model at the N.P.L., Teddington, and completed on one of the DEUCE computers at the RAE Farnborough.

*Royal Aircraft Establishment, Farnborough, UK


A method for designing the wall contour of an exhaust nozzle to yield optimum thrust is established. The nozzle length, ambient pressure and flow conditions in the immediate vicinity of the throat appear as governing conditions under which the thrust on the nozzle is maximized. Isentropic flow is assumed and the variational integral of this maximizing problem is formulated by considering a suitable chosen control surface. The solution of the variational problem yields certain flow properties on the control surface, and the nozzle contour is constructed by the method of characteristics to give this flow. An example is carried out and typical nozzle contours given.

*Marquardt Aircraft Co., Van Nuys, California, USA.


The investigation reported in this paper constitutes a study of flow separation in nozzles within a large experimental universe. The results of these investigations show that, for flow through conical nozzles pressure expansion along the wall is almost one-dimensional and isentropic and the wall pressure depends on the area ratio only. For flow through curved nozzles, the expansion depends on the radius of curvature as well as the throat area. For the three types of nozzles tested (Axi-symmetric and slit-nozzles with curved and straight divergence), there is one separation pressure for each ambient discharge pressure, and is a function of nozzle divergence angle only. It has been observed that the jet separates asymmetrically at a low reservoir pressure and, on lowering the pressure still further, the separation becomes unstable, with the jet oscillating rapidly from one side of the nozzle wall to the other.
This report describes an application of a digital computer to the aerodynamic design and automatic control of a large wind tunnel with flexible walls. The computational problem was threefold: first, calculation of the wall shapes for a set of 'pivotal' operating speeds between $M = 1$ and $2.8$; second, computation of the necessary movements of the set of screw jacks which flex the walls; and finally, preparation of the set of digitally punched control tapes. A new mathematical approach was used for the first part in order to avoid singularities in the partial differential equations governing the flow in the convergent-divergent nozzle of the tunnel. The work was started in 1953 on the ACE Pilot Model at the National Physical Laboratory and completed on one of the Royal Aircraft Establishment DEUCE computers.

Royal Aircraft Establishment Bedford, UK.

Measurements in supersonic wind tunnels have become almost routine in the last decade and yet little is known about the free-stream disturbances, which are present, and about the means of minimizing them. The present note outlines the problems of minimizing the mean as well as the unsteady variations of velocity and temperature and gives guidance where the trends are clear. It is found that in many wind tunnels the free-stream fluctuations are likely to be dominated by aerodynamic sound in the sense of Lighthill and by shivering Mach waves and wall vibrations. Only conjectures can be offered on the means of minimizing the resulting sound intensity at the present time.

Royal Aircraft Establishment, UK.

A test program was conducted to calibrate the von Karman Gas Dynamics Facility's 40-in., continuous, supersonic wind tunnel (Tunnel A). Theoretical nozzle contours were adjusted, and local Mach numbers and flow angles were measured within a region of the test section. Results of the calibration at Mach numbers from 1.5 to 6 in half Mach number increments are presented. Included in the nozzle calibration program were tests of two typical models, AGARD Calibration Model B
and a 15-deg cone cylinder. Along the axis of the test section the total variation in Mach number was less than ±0.5 percent at all Mach numbers and slightly larger variations were measured off axis within a 12-in. diameter core. The average flow angles on the test section axis were less than 0.1 deg in both the horizontal and vertical planes. Significant changes in average Mach numbers were observed for decreases in free-stream Reynolds number at the high Mach numbers. The pressure distribution measurements on the cone-cylinder model were in reasonable agreement with theory, and the force characteristics of AGARD Calibration Model B agreed well with results from another wind tunnel.

*ARO Inc, Arnold AFS, Tullahoma, Tennessee, USA.


A number of contour nozzles for an ideal gas with $\gamma = 1.22$ are presented. The variation of gamma is considered. The design is based upon the assumption of a radial flow section in the initial divergent section which is turned to uniform axially directed flow at the exit. For each specified exit Mach number, about 95 per cent of the maximum expansion angles are used. The contours obtained were cut back in order to yield nozzles of various exit areas. The surface area of the divergent section, thrust coefficient and length of these cutback nozzles are compared with those properties of 15-deg half-angle conical nozzles with the same exit area. Simple heat flux calculations for some of the designs are also presented. These designs result in moderate savings in length, surface area and heat flux over a 15-deg nozzle with the same thrust coefficient. Greater savings are possible at the expense of reducing the thrust coefficient.

*Jet Propulsion Laboratory, Pasadena, California, USA.


Besides the nozzle thrust, a design engineer would like to know a nozzle's weight and cooling requirements to enable him to make the proper choice. For this purpose an approximation to the optimum thrust contour is enough, and a simple geometric method is described here.

*Rocketdyne Division, North American Aviation Inc., Canoga Park, California, USA.


This response discusses some of the results from the JPL 12 in. by 20 in. supersonic wind tunnels which add proof to the ideas presented in Morkovin's paper. Other results indicate that our understanding is not complete. Obviously it is difficult to reduce any fluctuations originating from the turbulent wall boundary layers; but the large amplitude of these fluctuations at the higher Mach numbers may have significant effects on supersonic wind tunnel model testing and should be investigated. The author's suggestions for reducing disturbance levels and their effects in wind tunnels should be considered as part of good supersonic wind tunnel design.

*Jet Propulsion Laboratory, Pasadena, California, USA.


TL 573 H52
(Ames Research Center library no.)

This book section reviews the design of all major components of supersonic wind tunnels including a study of supersonic nozzle design. Operational requirements, performance calculations and measurement techniques are discussed.


A graphical technique for selecting optimum nozzle contours from a family of truncated perfect nozzles is presented which permits simultaneous consideration of various types of optima. This procedure includes the effects of friction, as well as the thermodynamic properties of the reacting gas. Results are given of hot and cold flow tests, showing the effect of wall contour and wall cooling upon oblique shock separation.


A low-density supersonic nozzle was operated with carbon dioxide over a range of supply pressures from 10 microns to 600 microns and the transition from a boundary layer-inviscid core flow to a fully developed viscous flow was observed. A liquid-nitrogen cryogenic pump was used to pump the flow through this nozzle. Sizable reductions of viscous effects in the nozzle flow were observed when the cryogenic pump was extended to the wall of the nozzle.


The method developed by Maurice Tucker for calculating the growth of turbulent boundary layers in two-dimensional compressible flow has been adapted to the computation of a dimensionless boundary layer parameter. Equations and calculational procedures are presented for correcting two-dimensional supersonic nozzles for boundary layer growth using only one or two tables, depending on the geometry. The tables are supplied for the Mach number range from 1.5 to 8.0. Agreement of calculated data with measured data and effect of Mach number distribution are shown.

181. *Holt, M.: The Design of Plane and Axisymmetric Nozzles by the Method of Integral Relations. AFOSR-3140. In:
Applications to the nozzle problem of the method of integral relations, which is very effective in solving problems governed by nonlinear partial differential equations with the aid of electronic computers. The method is so presented that it can be applied with equal facility either to the design or flow problem. In the application of the method to the problem of hypersonic flow past a blunt body, the numerical integration scheme is started on the axis, where symmetry conditions must be satisfied, and proceeds towards the saddle points, the positions of which are not known in advance. An alternative scheme is proposed for the nozzle problem. In this case there is no need to satisfy symmetry conditions at the start of the integration.


By means of suitable expansions in inverse powers of R, the radius of curvature of the nozzle profile at the throat measured in throat half-heights, the velocity components in the throat region of a convergent-divergent nozzle can be calculated. The first three terms of the series solution have been obtained both for two-dimensional and for axially-symmetric nozzles. The numerical accuracy of the solution is confirmed by comparison with the known exact solution along the branchline.


Conical supersonic nozzles were analyzed by the method of characteristics to show the effects of wall angle, throat-to-cone fairing, area ratio, and the thermodynamic properties as characterized by the isentropic expansion coefficient. A detailed analysis indicated that the results differ from those obtained with the usual source flow assumptions and that both the vacuum thrust efficiency and the wall pressure distribution are functions of the aforementioned variables.


Test section flow characteristics are presented for the nominal Mach number range from 1.5 to 5 over the available Reynolds number range. Along the axis of the test section for all contours, the maximum variation in Mach Number is ±0.01, and the average flow angles in pitch and yaw are less than 0.2 deg. Somewhat larger variations in Mach number and flow angularity are found off-axis. At the higher Mach numbers, significant changes in the average test section Mach number with decreasing Reynolds number are accompanied by corresponding changes in the tunnel boundary layer.


A method of calculation is presented for a potential uniform flow of a perfect gas in the subsonic and transonic part of a plane symmetrical Laval nozzle with a given form of contour.

An analytical study is made of the viscous flow in slender channels. Similar solutions to the approximate equations of motion, valid for flow at moderate or high Reynolds numbers in slender channels, are found for incompressible two-dimensional and axisymmetric flows and for compressible flows through two-dimensional channels with adiabatic walls. A study of compressible flows in convergent-divergent channels yields results regarding the effect of viscosity on the location of the sonic line, on the pressure ratio at the geometric throat and on the discharge coefficient for such channels.

*Department of Mechanical Engineering, North Carolina State College, North Carolina, USA.


AD-400746
N63-14607

Analytical studies have been made concerning many of the problems associated with the expansion of extremely high temperature and pressure plasmas through the converging section of a hypersonic wind tunnel nozzle. The problems in this study relate to the establishment of heat fluxes to the nozzle wall, methods of removing the heat load or protecting the nozzle wall, determining the stress levels in the nozzle liner, analyzing various materials for strength and thermal properties. In all cases parametric studies have been made resulting in design criteria which can be used for specific conditions. The conditions are established from the performance envelope for a typical low density hypersonic wind tunnel. Throughout the report the performance envelope is plotted on stagnation pressure and temperature coordinates which correspond to free stream velocities and altitudes up to 25 kilofeet/sec and 350 kilofeet respectively. Ultimate use of the information will depend upon the reliability of the assumptions made which in many cases must be determined by new experiments. The appendices contain listings of FORTRAN programs for nozzle and rib stress calculations, and backside cooling calculations.

*Nuclear Engineering Department, University of Tennessee, Knoxville, Tennessee, USA.


Evaluation of axially symmetric gas flow in conventional conical nozzles by the method of characteristics has revealed the possibility of shock formation within the nozzle. Negative Mach lines, originating at or just downstream of the junction of the throat profile and the cone, intersect near the axis. Modified computer programs have been run to determine the nature of this phenomenon, and it is concluded that the effect is real. It is shown that the shock formation can be removed by changes in the wall contour near the junction.

*Imperial Metal Industries Ltd., Summerfield Research Station, Kidderminster, Worcestershire, UK.


*Mechanics of Fluids Department, University of Manchester, UK.


N66-13589
This handbook section presents the basic techniques and related information needed for the control of duct flow in high-speed propulsion devices and in supersonic and hypersonic wind tunnels. The general mechanical and thermodynamic theory of steady flow in a duct is simplified as much as possible and is concisely reviewed. Non-steady, one-dimensional flows are discussed and methods of solution for a great number of practical problems are outlined. Generalized techniques are presented for the inclusion of viscous and heat-transfer effects by means of which boundary-layer correction may be made to profiles designed for potential flow. The general solutions of duct flow are applied to specific problems and are amply supplemented by experimental data. Also presented are many practical geometric modifications and design techniques. An included Bibliography directs the reader to related research past and present.


N67-34665#

This paper presents a method of design of a supersonic nozzle which incorporates recent developments in compressible flow theory. Continuous curvature of the contour is ensured by defining a continuous gradient of Prandtl-Meyer angle along the nozzle axis. The flow in the throat was calculated from results given by Hall (1962). A matching technique was used to determine a triplet of values for the throat radius of curvature, the flow deflection at the inflection point and the Prandtl-Meyer angle at the point on the nozzle axis where the left running characteristic through the inflection point intersects the nozzle axis. The majority of the new work is to be found in the Appendices. Appendix B presents a method of determining the co-ordinates along a particular characteristic by inverting the results given by Hall (1962). Appendix C reviews the accuracy of the method of characteristics and, Appendix D suggests a method by which it may be possible to increase the accuracy without resorting to calculations made at very small step sizes.

*The Mechanics of Fluids Department, University of Manchester, UK.


The results of an experimental investigation of convective heat transfer from turbulent boundary layers accelerated under the influence of large pressure gradients in a cooled convergent-divergent nozzle are presented. The most significant unexpected trend in the results is the reduction in the heat-transfer coefficient, below that typical of a turbulent boundary layer, at stagnation pressures less than 75 psia. Heat transfer predictions with which the data were compared either incorporate a prediction of the boundary-layer characteristics or are related to pipe flow. At the higher stagnation pressures, predicted values from a modification of Bartz's turbulent-boundary-layer analysis are in fair agreement with the data. As a possible explanation of the low heat transfer at the lower stagnation pressures, a parameter is found which is a measure of the importance of flow acceleration in reducing the turbulent transport below that typical of a fully turbulent boundary layer.

*Jet Propulsion Laboratory, Pasadena, California, USA.


N64-31672#

An experimental study was made of the internal air-film cooling of a Mach 2.4, nonadiabatic, wall, axially symmetric nozzle. The main stream air was heated to supply temperatures from 672 to 1212°R at supply pressures from 115 to 465 psia. The film coolant air was injected through a single peripheral slot at an angle of 10° from the nozzle wall. The coolant-to-main stream mass flow ratios were varied up to 20 percent. Steady-state nozzle wall temperatures were measured in both the subsonic and supersonic flow regimes. The turbulent pipe flow equations of Dittus and Boelter was found to be applicable in predicting the heat transfer rates in the absence of film cooling. A modified
version of the semi-empirical equation of Hatch and Papell was found applicable in estimating the film cooled nozzle wall temperatures.

*US Naval Ordnance Laboratory, White Oak, Maryland, USA.


Chapter 1 gives an extensive overview of supersonic wind tunnel design. Both theoretical and practical aspects are covered in considerable depth. Chapter 10 considers the calibration of supersonic nozzles and the operation of supersonic tunnels. This book is to be considered essential reading for any supersonic wind tunnel designer.

*Sandia Corp., Albuquerque, New Mexico, USA.


X65-13236 (Available to US Government Agencies and US Government Contractors Only)

A program was conducted to determine the effects of favorable pressure gradients, mainstream turbulence, wall cooling, and turbulent boundary layer removal in the convergent section on the heat flux and boundary layer characteristics at the throat of a supersonic nozzle. The objective of this investigation was to determine whether a laminar boundary layer could be maintained in the throat region of such a nozzle for the purpose of reducing the throat heat flux. The effort under this program was divided into both theoretical and experimental investigations. The theoretical investigation consisted of analysis of the laminar and turbulent heat-transfer rates for a range of nozzle conditions. The experimental investigation consisted of the measurement of the heat transfer rates for both laminar and turbulent flow in the throat of a asymmetric nozzle installed in a continuous-flow wind tunnel. It was found that up to some lower limit on Reynolds number, both the boundary layer profiles and heat flux at the throat were laminar in the high pressure gradient nozzle both with and without suction removal of the turbulent boundary layer upstream of the convergent section. This indicates that natural reverse transition (laminarization) was occurring in the nozzle for these conditions. It was also found that high favorable (i.e. negative) pressure gradients and increased wall cooling tend to promote the occurrence of reverse transition. It was also found that flush suction slots with sharp downstream corners which are located upstream of the convergent section of the nozzle are not an effective means of initiating a laminar boundary layer in the convergent section of a nozzle.

*United Aircraft Research Laboratory, East Hartford, Connecticut, USA.


N66-18405
AD-464401

The effect of fairing a contraction, starting from a square cross section, into a two-dimensional supersonic nozzle is considered with particular emphasis upon the avoidance of pressure gradients along the contraction walls that would be unfavourable to the development of the boundary layer flow. A contraction wall profile, derived from two-dimensional irrotational incompressible flow, was found from model tests to give satisfactory results.

*Royal Aircraft Establishment, UK.

Measurements of mean values in time of pitot pressure and flow direction have been made in the boundary layer over the insulated side-wall of a specially-constructed supersonic nozzle. The external flow first accelerated and then decelerated, and the cross-wise pressure gradients were such that the boundary layer cross-flow was first in one direction and then in the opposite, as happens over wings with swept leading-edges. The external Mach number ranged between 1.6 and 2.0. Boundary layer traverses were made at intervals along external streamlines; from each traverse, profiles of Mach number and streamwise and cross-wise components of velocity were derived. The results were supplemented by measurements of skin friction using surface-tubes. A preliminary analysis reveals serious limitations in the assumed forms for the cross-flow profiles, but there is evidence that the streamwise component behaves as it would in an equivalent two-dimensional boundary layer.

*Royal Aircraft Establishment, Farnborough, UK.


N66-24647#

A computer program is described which is capable of computing the laminar boundary layer in an axisymmetric supersonic nozzle. The program also determines the nozzle contour which will maximize the thrust obtained under a specified set of operating conditions, taking into account the effects of the boundary layer. The boundary layer thickness is computed by using a correlation suggested by Howarth; and the optimum nozzle contour is found by the approximate method of Rao. The program was originally developed to determine the design and performance of electrothermal thrusters, but it has also been applied to supersonic nozzles for other applications such as chemical rockets and wind tunnel facilities.

*Space Power and Propulsion Section, General Electric Company, Evendale, Ohio, USA.


N66-87191

*Grumman Aircraft Engineering Corp., Bethpage, New York, USA.


To understand better equilibrium flows through supersonic nozzles, wall static pressures have been measured in nozzles with circular-arc throats having different ratios of throat radius of curvature to throat radius $r_c/r_{th}$, circular-arc or conical convergent sections, and conical divergent sections. These measurements were made with air at stagnation temperatures of 530° and 1500°R and over a stagnation pressure range from 45 to 250 psia. The flow through the transonic regime was found to depend essentially on local configuration, i.e., on the ratio $r_c/r_{th}$; two-dimensional isentropic flow predictions agreed with the data in this region for the nozzles with $r_c/r_{th} = 2.0$, but were inadequate for the nozzle with $r_c/r_{th} = 0.625$. By comparison, the simple one-dimensional isentropic flow prediction was as much as 45% high in the throat region for one nozzle; in the conical sections, deviations of a smaller magnitude were found. The effects of wall cooling and variation in the boundary-layer thickness at the nozzle inlet were investigated, as were differences in pressure readings with taps of various sizes. Some separation pressure data are presented to show the effect of wall cooling. Other flow features that indicate the extent of deviations from one-dimensional flow include flow coefficients, thrust ratios, and local mass fluxes. It is hoped that these comparisons between measurements and predictions will be useful in studying nozzle flows with the additional complexity of chemical reactions.

*Jet Propulsion Laboratory, Pasadena, California, USA.

A66-11015#

Consideration of the problem of steady-state irrotational flow of a perfect gas in plane and axisymmetric Laval nozzles. Solutions are obtained for both the direct problem of calculating the flow in nozzles of a certain form and for the inverse problem of constructing a nozzle wall according to a given velocity distribution on the axis of symmetry. The solutions are carried out numerically according to a variant of the method of integral relations.


An experimental investigation of the local heat-transfer coefficient in the throat region of a nozzle operating under conditions of cold wall, low-density, high-speed flow has been conducted. Results are presented for nitrogen gas for reservoir temperatures from 5200° to 6500°R and reservoir pressures from 0.9 to 1.5 atm. The experimental results have been compared with several analytic procedures for predicting the heat-transfer coefficient in laminar flow. The simple flat-plate equation is shown to underestimate the heat-transfer coefficient. The method of Cohen and Reshotko predicts coefficients that are in agreement with the experimental results. The incremental flat-plate method of Pasqua and Stevens and a modification of a solution of Beckwith and Cohen exhibit good results when referenced to the value calculated by the simple flat-plate equation at the nozzle exit. The effects of thermal radiation upstream of the nozzle throat are indicated.


An inviscid transonic theory appears to be inadequate to describe the flow near the throat of a converging-diverging nozzle during the transition from the symmetrical Taylor (1930) type of flow to the subsonic-supersonic Meyer (1908) flow. A viscous transonic equation taking account of heat conduction and longitudinal viscosity has been developed previously (Cole 1949; Sichel 1963; Szaniawski 1963) An exact, nozzle-type of similarity solution of the viscous transonic equation, similar to the inviscid solution of Tomotika & Tamada (1950), has been found. This solution does provide a description of the gradual transition from the Taylor to the Meyer flow and shows the initial stages in the development of a shock wave downstream of the nozzle throat. The solution provides a viscous, shock-like transition from an inviscid, supersonic, accelerating flow to an inviscid, subsonic, decelerating flow.

*Department of Aerospace Engineering, University of Michigan, Ann Arbor, Michigan, USA.


An analysis was made of the transonic flow in axisymmetric nozzles having wall radii between one-quarter and three times the throat radius. Entrance angles were varied from 30° to 75°. The analysis is based on Freidrichs' equations, by which the flowfield is developed for a prescribed velocity distribution along the nozzle axis. Each variable is expressed as a series in terms of the stream function, and fourth order terms are retained to give an accurate solution for a small throat radius of curvature. The usual assumption of steady, isentropic, irrotational flow is made, and the specific heat ratio is constant. In the throat region, the solution has been found to compare favorably with available experimental data for both small
and large throat curvatures. With a large wall radius of curvature, the solution is comparable to those of Sauer, Oswatitsch, and Hall. With a small radius of curvature, the sonic line is found to differ appreciably from the parabolic sonic line of Sauer's solution. For the wall radii equal to or less than the throat radius, an inflection point occurs in the sonic line near the nozzle wall. The entrance angle influences the flow in the transonic region only when the wall radius is less than 1 1/2 times the throat radius.

*Douglas Aircraft Company, Santa Monica, California, USA.


AD-646748
N67-25477#

A Mach 3 to 7 blow-down wind tunnel facility for investigating boundary layer phenomena in the transitional and turbulent range is described. The facility, referred to as the Boundary Layer Channel, utilizes a flexible plate and a flat plate to form the two opposite walls of a two-dimensional supersonic nozzle. The plate is 8 feet long. The channel operates at supply temperatures up to 1000°F and supply pressures from 0.1 to 10 atmospheres. The Reynolds number per foot capability is from $3 \times 10^4$ to $2.4 \times 10^7$.

*US Naval Ordnance Laboratory, White Oak, Maryland, USA.


Method of characteristics predictions of supersonic flow through conical nozzles reveal shock formation where characteristics originating just downstream of the tangent between the conical divergent section and throat curvature section approach the nozzle axis. This note presents some measurements that confirm the predicted shock formation. To eliminate this shock formation, Darwell (1963) and Migdal (1965) suggest attaching the conical divergent section at the inflection point of a conventionally designed contour nozzle.

*Jet Propulsion Laboratory, Pasadena, California, USA.


An experimental investigation of gasdynamics and convective heat transfer in a cooled conical nozzle with a 45-deg convergent and 15-deg divergent half-angle and a small circular-arc throat has revealed a variety of flow phenomena that can occur and, consequently, influence the convective heat transfer significantly. The investigation with air spanned a range of stagnation pressures from 30 to 250 psia and stagnation temperatures from $1000^\circ$ to $2000^\circ$R. With the inlet boundary layer turbulent, a reduction in heat transfer below that typical of a turbulent boundary-layer occurred along the convergent section, so that at the throat at the lower Reynolds numbers apparently reverse transition to a laminar boundary layer had occurred. The largest reduction in heat transfer amounted to about 50%. Wall static pressure measurements indicate, in addition to large deviations from one-dimensional flow in the transonic region, adverse pressure gradients in the nozzle inlet curvature section and just downstream of the tangency between the circular-arc throat and conical divergent section. In the nozzle inlet region, the heat transfer is typical of that found in separated flow regions; whereas just downstream of the tangency, apparently retransition from a laminar or partially laminar boundary layer to a turbulent layer occurred for those tests where reverse or partial reverse transition had taken place along the convergent section. Heat-transfer predictions are in fair agreement with the data, providing that the boundary layer is either turbulent or essentially laminar.

*Jet Propulsion Laboratory, Pasadena, California, USA.

A study is made of source-type viscous compressible flow in a conical nozzle. For such a flow an exact solution to the Navier-Stokes equations together with the first-order velocity-slip and temperature-jump boundary condition is possible. It is found that solutions of this type are possible only for certain combinations of centerline Mach number, Reynolds number, wall angle and ratio of wall temperature to centerline temperature. Solutions of this type are obtained for Reynolds numbers from 5 to 500 and are used to estimate the magnitude of velocity slip and temperature jump in this Reynolds number range, and to assess the effect of slip on the wall skin friction, on the mass flow through the nozzle, and on the pressure difference between the nozzle centerline and the wall.

*North Carolina State University, Raleigh, North Carolina, USA.


The purpose of this note is to present the results obtained with relatively thin inlet velocity and temperature boundary layers and compare them to the thicker turbulent inlet layer results recently published. Heat-transfer coefficients were generally higher along a supersonic nozzle with relatively thin inlet velocity and temperature boundary layers than with thicker, turbulent inlet layers.

*Jet Propulsion Laboratory, Pasadena, California, USA.


N68-14747#


The description of the compressible gas flow in the throat region of unconventional rocket nozzles must be known in order to optimize the wall contours and to predict the performance. The existence of both subsonic and supersonic flow in close proximity does not permit the use of conventional analytical techniques to obtain a mathematical description of the flow. The analysis presented in this paper can treat annular flow passages that are either parallel to the nozzle axis or inclined to it at some arbitrary angle. The solution is not limited to nozzles with moderate wall radii of curvature; sufficient terms have been retained in the series solution to treat nearly sharp wall curvatures in the throat region. The usual assumption of steady, isentropic, irrotational flow with constant specific heats is made. The shape of the sonic line is not symmetrical about the geometric center of the annular throat as in the case of bell nozzles. The flow reaches sonic velocity further upstream of the geometric throat on the inner wall than on the outer wall curvatures being equal.

*NASA Lewis Research Center, Cleveland, Ohio, USA.

Heat transfer measurements were obtained in 30° and 60° half-angle of convergence nozzles at a nominal stagnation temperature of 970°R (539 K) and over a range of stagnation pressures of 2.03 to 20.4 atmospheres (2.03 x 10^6 to 20.67 x 10^6 N/m^2). These conditions provided nozzle throat Reynolds numbers based on diameter of about 6.0 x 10^5 to 5.0 x 10^6.

Boundary layer time-mean velocity and temperature measurements were obtained at one station in a water cooled pipe inlet and at a subsonic (Mach number <0.08) station in each nozzle at the stagnation pressures of 3.1 and 20.4 atmospheres (3.1 x 10^6 and 20.67 x 10^6 N/m^2). The heat transfer and boundary layer surveys suggested the occurrence of laminarization of an initially turbulent boundary layer.

Utilizing normalized coordinates, the equations governing inviscid, isentropic expansions of perfect gases through convergent-divergent nozzles can be solved as inverse power expansions in the normalized throat wall radius of curvature. The solution for the complete nozzle flowfield is obtained in the form of a perturbation about the one-dimensional flowfield. The corrections are found to be polynomial in the perpendicular coordinate. Both the first- and second-order corrections to the one-dimensional flowfield have been worked out. Solutions are obtained for both choked and unchoked nozzle flows as well as for multistream nozzle expansion such as characteristically occurs in a barrier cooled rocket engine. Comparison with Hall's transonic analysis reveals that the transonic solution is obtained from the full nozzle solution through a coordinate transformation and reordering of the complete solution.

Flow and heat-transfer measurements are presented for heated air flowing subsonically through a contraction section smoothly connecting a tube to a conical section. These measurements spanned a range of cooled wall-to-stagnation temperature ratios, T_w/T_t from 0.1/3 to 2/3, Reynolds numbers from 2 x 10^4 to 2 x 10^6, with data obtained for both laminar and turbulent boundary layers. Flow visualization results indicated that the flow separated just upstream of the curved contraction section and reattached near its end. As a consequence of the curved section and the associated separated flow effects, the heat transfer in the separation region under certain operating conditions was reduced below values measured upstream in the tube. Values of the heat-transfer coefficient were higher with thin boundary layers produced by a short, cooled upstream length, and the distribution of the heat-transfer coefficient in the redeveloped region depended on whether the boundary-layer was turbulent or laminar. Effects of wall cooling in terms of T_w/T_t that were found upstream in the tube were not observable in the separation and reattachment regions.
Wall pressure measurements in the transonic regime of supersonic nozzles indicate that there are no satisfactory theoretical predictions methods when wall curvature effects are large. In this note we examine the use of a nonconservative form of the basic equations in the analysis of axisymmetric, transonic nozzle flows and compare the results with experimental data. In addition, the technique of coordinate stretching in time-dependent calculations is introduced to account properly for subsonic boundary conditions.

Previous investigations of turbulent boundary-layer heat transfer from heated air flowing through cooled supersonic nozzles have lead to the observation of a reduction in turbulent transport of heat under certain operating conditions, and effect also apparent in rocket engine tests. In order to better understand the conditions for which laminarization occurs and the effect of laminarization on the friction coefficient, an investigation of the structure of the boundary layer was undertaken in an axisymmetric nozzle. Boundary layer measurements upstream and within the nozzle are presented and this data reveal a strong effect of flow acceleration on the structure of an originally turbulent boundary layer.

For most rocket nozzles of current interest, the normalized wall radius of curvature R is less than or equal to 1, and it is generally believed that expansion methods are inapplicable to the study of the transonic flow region in a convergent-divergent nozzle. It is shown, however, that this limitation is due to the coordinate system employed in the analysis rather than to a fundamental limitation of the expansion method itself.

This note presents some internal flow measurements in the transonic regime of a nozzle with a small ratio of throat radius of curvature to throat radius \( \frac{r_c}{r_{th}} = 0.625 \) and compares some recently developed prediction methods of other investigators with the data. The measurements revealed radial variations in the flow. At the physical throat, the Mach number was 0.8 at the axis and 1.4 near the wall. The three methods considered predicted the transonic flowfield reasonably well in nozzles which have a small throat radius of curvature \( \frac{r_c}{r_{th}}<1.0 \), such as that for the nozzle investigated.

This comment concerns discrepancies in the published analysis by Hopkins and Hill (AIAA Journal, May 1968). The "corrected" equations are presented.

Experimental values of adiabatic flow coefficient are presented for choked flow in a supersonic nozzle. The adiabatic flow coefficient is influenced by the nozzle contour, particularly in the throat region which establishes the inviscid core flow distribution, and by the viscous (boundary-layer) effects. In the experiments, which were conducted with argon, nitrogen, and helium, the boundary layers in the throat region of the nozzle are believed to have been laminar even at the higher throat Reynolds number (350,000), because of the high values of the acceleration factor K. The experimental trend of an increasing flow coefficient with Reynolds number is in agreement with a laminar boundary-layer prediction, although most of the data points at the intermediate and higher Reynolds numbers were somewhat lower than the predicted values of $C_D$.

*Jet Propulsion Laboratory, Pasadena, California, USA.


The structure of laminar boundary layers is investigated analytically over a large range of flow acceleration, surface cooling, and flow speeds. For flow of a perfect gas over an isothermal surface, the transformed boundary-layer equations with the similarity assumption were solved numerically for values of an acceleration parameter $\beta$ up to 20, a value not uncommon to transonic flow in supersonic nozzles, and for flat-faced bodies placed in supersonic flow, as well as for values of the surface-to-total-gas enthalpy $g_w$ ranging from 0 (severely cooled surface) to 1. The solutions were inverted and shown in the physical plane where the velocity and total enthalpy profiles depend upon the flow speed or compressibility parameter $S$ in addition to $\beta$ and $g_w$. Friction and heat transfer parameters, as well as thicknesses associated with the boundary layer, are also shown. In general, the effects of acceleration, cooling, and flow speed are significant, but the dependence of the heat-transfer parameter on $\beta$ and $g_w$ is weak when $g_w$ is small. Application of the similarity solutions to accelerated flow of real gases for which viscosity is not proportional to temperature and for which Prandtl number is not unity is discussed.

*Jet Propulsion Laboratory, Pasadena, California, USA.


Boundary layer and heat transfer measurements are presented for flow through a cooled conical nozzle with a convergent and divergent half-angle of 10 deg for a wall-to-total-gas temperature ratio of 0.5. A reduction in heat transfer below values typical of a turbulent boundary layer was found when values of the parameter K exceeded about 2 to $3 \times 10^{-6}$. The boundary layer measurements, when viewed in conjunction with the heat transfer measurements, reveal the complicated nature of the flow and thermal behavior and their interrelationship when laminarization occurs.

*Jet Propulsion Laboratory, Pasadena, California, USA.


N71-10444#

The transonic equations of motion for a converging diverging nozzle, including the effect of variable gamma, were solved in toroidal coordinates using a combination of an asymptotic small parameter expansion and a double coordinate expansion. The analysis was kept general so that high order solutions could be recursively calculated. It was found that the use of toroidal coordinates and different expansion parameters did not significantly
extend the range of normalized throat wall radii of curvature for which expansion solutions could be accurately calculated. An explanation of why expansion methods fail for small R is given. Calculations made, including the effect of variable gamma (for a homogeneous unstriated flow), indicate that its effect is negligible in the transonic region. A new technique for solving the subsonic portion of the nozzle flow is also described.

*Dynamic Science, Irvine, California, USA.


This note is concerned with the determination of the mass flow rate through choked nozzles with emphasis on comparatively small radius of curvature throats. In the flow regime investigated (throat Reynolds numbers larger than $10^5$) viscous (boundary layer) effects are not believed to be significant, so that the flow field can be regarded as essentially isentropic.

*Jet Propulsion Laboratory, Pasadena, California, USA.


The effect of large wall heating and flow acceleration on the structure of laminar boundary layers was analyzed over a range of flow speeds. The heat-transfer parameter was found to increase significantly with the amount of wall heating in flow acceleration regions provided that frictional effects are not important and that the laminar boundary layer is not on the verge of transition to a turbulent boundary layer before the flow is accelerated. Because of flow acceleration, the velocity profiles with wall heating do not have an inflection point in the region near the wall where the velocity increases and therefore would appear to be more stable to small disturbances.

*Jet Propulsion Laboratory, Pasadena, California, USA.


Performance predictions for attitude control jets of satellites and manned spacecraft suffer from the lack of well-substantiated theoretical and experimental data in the fully viscous nozzle-flow regime. This paper presents an experimental investigation of the internal and external flow for nozzle Reynolds numbers in the general range between $10^2$ and $10^3$ with nitrogen as the test gas. Electron-beam techniques are used for measuring gas density and rotational temperatures at selected points throughout the flow. Discharge coefficients are also measured. In addition, some effects of ambient pressure on the external flow structure are studied by flow visualization experiments. At the lower Reynolds numbers studied, experimentally determined temperatures indicate the existence of a supersonic bubble inside the nozzle expansion cone, with a subsequent shock-free viscous transition to subsonic flow. These results substantiate the theoretical prediction of this phenomenon, first made by Pae in an earlier phase of this program.

*Jet Propulsion Laboratory, Pasadena, California, USA.


N72-18306#
AD-731677

A cold flow-field study of a rapid expansion, Mach 4.73, two-dimensional multiple nozzle array has been carried out in the Hypersonic Research Tunnel No.4 at the Naval Ordnance Laboratory. The nozzles are of the type used in a gas dynamics laser where the quality of the downstream flow is important. Consequently, this report examines the uniformity of the flowfield in such a multiple nozzle array. Tests were conducted at supply conditions of 45 psia...
and 540°R. Pitot pressure, static pressure, and total pressure were measured along the axial, vertical, and horizontal centerlines within the nozzle and to 21 exit heights downstream. Shadowgraph pictures provided quantitative flow field observations. The shock structure consisted of sidewall shocks, nozzle trailing edge shocks, and nozzle throat shocks. Viscous phenomena consisted of the boundary layers on the sidewalls and contoured walls, as well as the turbulent wake caused by the trailing edges of the center nozzle blades. Mach number, static temperature, static density, and velocity profiles are presented. This flowfield information is useful to the prediction of the degradation of beam quality resulting from the non-homogeneity of a gas-dynamic-laser flowfield.

*US Naval Ordnance Laboratory, White Oak, Maryland, USA.


Exact numerical solutions have been obtained for highly accelerated self-similar laminar boundary layer flows with and without mass transfer. Values of the acceleration parameter $\beta$ in the range 0 to 20 were considered. Variable gas properties were realistically modeled by assuming $\rho \propto h^{-1}$, $\mu \propto h^{\omega}$, and $Pr = constant$. The results presented show the dependence of wall shear stress, heat transfer rate, and displacement thickness on the problem parameters which include $\beta$, Mach number, wall enthalpy ratio, mass transfer rate, $\omega$ and $Pr$. The inadequacy of solutions obtained under the simplifying assumptions of $Pr = 1.0$ and $\omega = 1.0$ is clearly displayed. The numerical solution procedure employed proved quite adequate for a class of problem which has presented serious difficulties to previous investigators.

*Aerodynamics Research Branch, Northrop Corp., Hawthorne, California, USA.


Wall static pressure measurements and performance parameters are presented for axisymmetric supersonic nozzles with relatively steep convergent sections and comparatively small radius-of-curvature throats. The nozzle walls were essentially adiabatic. These results are compared with those obtained in other nozzles tested previously to appraise the influence of contraction shape on performance. Both the flow coefficient and the thrust were less than the corresponding values for one-dimensional, isentropic, plane flow for both the axial and radial inflow nozzles considered, but the specific impulse, the most important performance parameter, was found to be relatively unchanged. The thrust decrement for the axial inflow nozzles was established primarily by the shape of the contraction section, and could be estimated reasonably well from a conical sink flow consideration. The radial inflow nozzle has a potential advantage from a cooling point of view if used in a rocket engine.

*Jet Propulsion Laboratory, Pasadena, California, USA.


The present studies include a quantitative experimental and theoretical assessment of the role of wind-tunnel disturbances in the boundary-layer transition process at hypersonic speeds. The various approaches and recent results for the development of a low-noise-level tunnel are presented. A statistical parametric study of transition data with a large computer is shown for cones in free flight, ballistic ranges, and wind tunnels at essentially zero angle of attack. New transition results for slender cones at small angle of attack are also given, as are studies of transition at high angle of attack, which are compared with various correlation attempts. Included are results which indicate that hypersonic transition in the
outer part of the boundary layer precedes the manifestation of transition at the wall ("precursor" transition).

*NASA Langley Research Center, Hampton, Virginia, USA.


Flow and thermal regimes found in relatively low Reynolds-number flows of high-temperature gases in cooled convergent-divergent nozzles used in propulsion systems and in research facilities are investigated by a combined experimental and numerical approach. The experiments were conducted with argon at temperatures up to 14,200 deg R, and the throat Reynolds number ranged from 2200 to 2800. The numerical calculations involved the laminar-flow equations in differential form. Taken together, the experiments and the numerical calculations provide information on the pressure, heat-flux, and shear-stress distributions along internal flows with heat transfer, and on the velocity and enthalpy distributions across the flow as well as along the flow. The influence of heat conduction and of viscous shear extended to the centerline all along the nozzle.

*Jet Propulsion Laboratory, Pasadena, California, USA.


Despite intensive research into laminarization over the past twenty years, there has resulted hitherto no prediction method suitable for ab into engineering design applications. Following a critical survey of currently proposed prediction criteria, this paper presents experimental evidence for a new prediction scheme applicable for the first time to compressible adiabatic flows. It is shown that this scheme is readily applicable for engineering design purposes. Extension to nonadiabatic flows is discussed.

*Northern Research and Engineering Corp., Cambridge, Massachusetts, USA.


The computation of ideal non-viscous gas flow in a two-dimensional supersonic nozzle is presented. The transonic flow was computed using an approximate method and the supersonic flow was computed using the method of characteristics. The program written for an IBM 360-40 computer is described and numerical examples, including the evolution of Mach number in the nozzle plane of symmetry, at the nozzle wall, and along a line parallel to the plane of symmetry of the nozzle are given.

*Institut Franco-Allemand de Recherches, Saint Louis, France.


One of the principal design requirements for a "quiet" supersonic or hypersonic wind tunnel is to maintain laminar boundary layers on the nozzle walls and thereby reduce disturbance levels in the test flow. The purpose of this report is to review the conditions and apparent reasons for laminar boundary layers which have been observed during previous investigations on the walls of several nozzles for exit Mach numbers from 2 to 20. Based on these results, an analysis and an assessment of nozzle design requirements for laminar boundary layers including low Reynolds numbers, high acceleration, suction slots, wall temperature control, wall roughness, and area suction are presented.

A procedure for designing ducts for subsonic and transonic speeds is described. Examples discussed are a wind-tunnel contraction cone, a supersonic nozzle, and a diffuser. A listing of the computer program is included.


The equations of motion governing steady, inviscid flow are of a mixed type, that is, hyperbolic in the supersonic region and elliptic in the subsonic region. These mathematical difficulties may be removed by using the so called time-dependent method, where the governing equations become hyperbolic everywhere. The steady-state solution may be obtained as the asymptotic solution for large time. This technique has been used to compute converging-diverging nozzle flows by Prozan (as reported by Saunders and Cuffel et al.), Migdal et al., Wehofer and Moger, Laval, and Serra. This technique has also been used to compute converging nozzle flows by Wehofer and Moger and Brown and Ozcan. While the results of the preceding calculations are for the most part good, the computational times are rather large. In addition, although the computer program of Serra included a centerbody and those of Wehofer and Brown included the exhaust jet, none of the preceding codes is able to calculate both, that is, plug nozzles. Therefore, the object of this research was to develop a production type computer program capable of solving converging, converging-diverging, and plug two-dimensional nozzle flows in computational times of 1 minute or less on a CDC 6600 computer.


Performance data are presented for the tunnel, which has a model support system, auxiliary systems, instrumentation, control room equipment, and automatic recording and computing equipment are also described. Information is presented on criteria for designing models and on shop facilities available to prospective users.


By means of comparisons between theoretical predictions and experimental data, the accuracy of a boundary procedure to predict the simultaneous solution of the boundary layer partial differential equations present investigation show the ability of the procedure to accurately predict properties of the boundary layers subjected to large streamwise accelerations. The procedure is used to conduct a parametric study of the effect of free stream turbulence, heat transfer, Reynolds number, acceleration, and Mach number on boundary layers in supersonic nozzles to assist in the design of a quiet tunnel. Results of the investigation show that, even in the presence of moderate free-stream turbulence levels, the boundary layer in the approach section of the quiet tunnel nozzle relaminarizes and becomes thin enough to be removed by a small slot.

Performance data are presented for this tunnel, which has a Mach number range from 2.0 to 3.5. Also described are the tunnel circuit, model support systems, auxiliary systems, instrumentation, control room equipment, and automatic recording and computing equipment. Information is presented on criteria for designing models and on shop facilities available to users.

*NASA Lewis Research Center, Cleveland, Ohio, USA.


The feasibility of quiet, suction laminarized, high Reynolds number (Re) supersonic wind tunnel nozzles was studied. According to nozzle wall boundary layer development and stability studies, relatively weak area suction can prevent amplified nozzle wall TS (Tollmien-Schlichting) boundary layer oscillations. Stronger suction is needed in and shortly upstream of the supersonic concave curvature nozzle area to avoid transition due to amplified TG (Taylor-Goertler) vortices. To control TG instability, moderately rapid and slow expansion nozzles require smaller total suction rates than rapid expansion nozzles, at the cost of larger nozzle length Re and increased TS disturbances. The total suction mass flow ratios for the laminarization of high Re supersonic air nozzles increase from 0.005 at Mach 3 (test section) to 0.0105 at Mach 9. Nozzle wall cooling decreases TS oscillations; TG instability in the concave curvature region, though, may be worse. Due to smaller nozzle length Re and Goertler parameters, Mach 9 helium nozzles require half as much suction for their laminarization as Mach 9 air nozzles of the same $U^*D^*/\nu^*$ (test section). Boundary layer crossflow instability on the side walls of two-dimensional high Re supersonic nozzles due to streamline curvature requires strong local suction to avoid transition. Nozzle wall surface roughness is critical in the throat area, especially at high Mach numbers, but not in the downstream nozzle region. Allowable surface roughness in the throat area of a Mach 9 helium nozzle is five times larger than for a comparable Mach 9 air nozzle. Test section mean flow irregularities can be minimized with suction through longitudinal or highly swept slots (swept behind local Mach cone) as well as finely perforated surfaces (hole spacing ≤ subsonic nozzle wall boundary layer thickness). Longitudinal slot suction is optimized when the suction-induced crossflow velocity increases linearly with surface distance from the slot “attachment line” toward the slot (through suitable slot geometry). Suction in supersonic blowdown tunnels may be operated by one or several individual vacuum spheres.

*Boeing Commercial Airplane Co., Seattle, Washington, USA.


High noise levels in conventional supersonic and hypersonic wind tunnels modify or dominate transition on test models. Transition research and predictions for flight conditions then require at least a partial simulation in wind tunnels of the much lower disturbance levels usually present in flight. The minimum operational limit for Reynolds numbers is a “quiet” wind tunnel will depend on unit Reynolds number and are established by transition Reynolds number correlations of free-flight data. High facility noise levels also dominate fluctuating pressure loads under fully turbulent boundary layers. Reliable measurements of these loads are required for panel vibration and flutter research. By analogy with recent data on the effects of stream turbulence and noise on low-speed turbulent shear layers, the basic structure of supersonic turbulent shear layers may be modified by high facility noise levels. Required research in these areas determines the maximum design Reynolds numbers for a quiet tunnel. Experimental data for current techniques to control and reduce noise levels in supersonic and hypersonic wind tunnels by laminarization of nozzle wall boundary layers and by noise radiation shields are presented. These results
and possible effects of Taylor-Goertler vortices observed in a nozzle wall boundary layer are used to predict the limits of quiet performance for a proposed 20 in. quiet tunnel.

*NASA Langley Research Center, Hampton, Virginia, USA.


This paper deals with determination of a class of minimum-length axisymmetric Laval nozzles, which are compared to the simple conical type, emphasizing the extra length needed to ensure uniform flow conditions. Submission of the paper has been prompted by the publication of Reddall; in fact, most of the following results have been obtained (and applied) over ten years ago, including the derivation of the hodograph equation for axisymmetric source flow, but were left dormant in a journal of limited circulation.

*Institute of Fluid Mechanics and Aerospace Technology, Bucharest, Rumania.


N77-11012#

This program, minimizing operator interventions, provides the necessary codes for the manufacturing of the nozzle using the following data: ratio of heat mass of gas, Mach number required at the nozzle exit, ratio of throat curvature radius to height or to throat radius, and height or diameter of exit section. The program and numerical methods used are described in this paper.

*Institut Franco-Allemand de Recherches, Saint Louis, France.


This paper reviews methods capable of solving the problems of two-dimensional (planar or axisymmetric) isentropic (inviscid and shock-free) flow of a perfect gas. The methods cited are divided into two groups: indirect methods and direct methods. Indirect methods (or inverse methods) involve the specification of a velocity (or pressure) distribution along some reference line in the flowfield (usually the centerline). These methods compute the boundary geometry and are sometimes called design methods. In direct methods, the boundary geometry (nozzle wall) is specified and the resulting solution then represents a complete description of the nozzle flowfield. It is the authors' hope that the classification and discussion of relative merits of the methods cited in this paper will bring to the attention of the nozzle designer the vast number of analytical methods available to him and will enable him to select the method that possess the appropriate combination of accuracy, speed, and computational flexibility.

*Virginia Polytechnic Institute and State University, Blacksburg, Virginia, USA.


The calculation of viscous nozzle flows can be accomplished by either solving the inviscid-core and viscous-boundary-layer equations separately or by solving the viscous equations for the entire flowfield. In the inviscid-core, boundary-layer approach, the assumption is made that the boundary layer is thin when compared to the nozzle diameter. However, for Reynolds numbers on the order of $10^3$ based on the throat diameter, this assumption is questionable. On the other hand, while the viscous equation approach is physically desirable, the computations tend to be rather lengthy. Therefore, the object of this research was to modify an efficient inviscid code to solve the viscous equations. This new code, called VNAP (Viscous Nozzle Analysis Program), is then used to calculate the flow in a chemical laser nozzle. This numerical solution is compared with both the inviscid-core, boundary-layer solution and experimental data presented by Hyde.

This note extends an approximate prediction method involving the integral form of the momentum equation to deduce the flow qualities of interest when compressible effects become important and heat transfer may occur. The approximate method is applicable to a two-dimensional laminar boundary layer on an impermeable surface of negligible curvature for a specified integral momentum equation. The proposed approximate method involves the use of similar solutions in conjunction with the integral momentum equation for isentropic freestream flow of a perfect gas by assuming that viscosity is proportional to stagnation condition and Prandtl number is unity. The study pertains to larger values of acceleration (acceleration parameter to 20 rather than 2) than previously considered to account for rapidly accelerating flows such as in a supersonic nozzle. A discussion of some applications confirms the advantages of the method in establishing better confidence in the predictions.

*Jet Propulsion Laboratory, Pasadena, California, USA.


In a wind tunnel with flexible liners the movement of a single supporting jack causes ripples along the plate, and consequently complicated changes in the flow pattern. A method is proposed for containing these ripples by moving the adjacent jacks. A system called the 1-4-1 system is derived and test data is shown to verify the smoothing and shortening effect it has on the resulting flow disturbances.

*Royal Aircraft Establishment, UK.


A computational procedure is described for rotational transonic flows. The effects of rotationality were introduced by a rotation function, and the resulting "potential-like" equations were solved by type-dependent relaxation. The method has been used to predict the flowfield in propulsion-type nozzles using both arbitrarily specified and experimentally measured entrance conditions. In these runs, good agreement with analysis and available experimental results was obtained, and a significant influence due to the rotationality of the flow was observed.

*Mechanical Engineering Department, Virginia Polytechnic Institute and State University, Blacksburg, Virginia, USA.


N78-15058# State-of-the art instrumentation and procedures for calibrating transonic (0.6 < M < 1.4) and supersonic (M >= 3.5) wind tunnels are reviewed and evaluated. Major emphasis is given to transonic tunnels. Background information was obtained via a literature search, personal contacts, and a questionnaire which was sent to 106 domestic and foreign facilities. Completed questionnaires were received for 88 tunnels and included government, industry and university-owned facilities. Continuous, blowdown and intermittent tunnels are considered. The required measurements of pressure, temperature, flow angularity, noise and humidity are discussed, and the effects of measurement uncertainties are summarized. Included is a comprehensive review of instrumentation currently used to calibrate empty-tunnel flow conditions. The recent results of relevant research are noted and recommendations for achieving improved data accuracy are made where appropriate. It is
concluded, for general testing purposes, that satisfactory calibration measurements can be achieved in both transonic and supersonic tunnels. The goal of calibrating transonic tunnels to within 0.001 in centerline Mach number appears to be feasible with existing instrumentation, provided correct calibration procedures are carefully followed. A comparable accuracy can be achieved off-centerline with carefully designed, conventional probes, except near Mach 1. In the range 0.95 < M < 1.05, the laser doppler velocimeter appears to offer the most promise for improved calibration accuracy off-centerline. With regard to procedures, tunnel operators are cautioned to: (1) verify by measurements that expansions from a settling chamber to a test section are indeed isentropic, and (2) obtain calibrations over the entire range of Reynolds number and humidity levels. Also, it is suggested that calibration data should include off-centerline measurements of Mach number and flow angularity. Finally, three problem areas for transonic tunnels are identified and discussed, viz. (1) the lack of standard criteria for flow uniformity and unsteadiness, (2) the undesirable noise generated by ventilated walls, and (3) wall interference.

*Vought Corporation, Dallas, Texas, USA.


A78-32371#

A new technique is presented for obtaining a low-noise test environment in a high-speed wind tunnel. This technique utilizes the fact that the primary noise source for high supersonic/hypersonic wind tunnels is radiated noise from the turbulent, tunnel-wall boundary layer. Because of the high directionality of sound in supersonic flows this test section noise originates far upstream on the walls of the nozzle at the "acoustic origin." It is shown that tailoring the nozzle contour to reduce the acoustic origin Mach number significantly reduces the noise level in the upstream half of the nozzle test rhombus. Experimental noise measurements are presented from a conventional, Mach 5 nozzle and are compared with measurements from a rapid-expansion, Mach 5 nozzle.

*NASA Langley Research Center, Hampton, Virginia, USA.


A78-43573#

A procedure is presented for the design of maximum thrust nozzle contours by direct optimization methods. The nozzle contour is modeled as a second-degree polynomial having a fixed initial expansion contour. The coefficients of the polynomial are varied by direct optimization methods to determine the maximum thrust contour. Three direct optimization methods are considered: multi-dimensional line search, the method of steepest decent, and Newton's method. Results are presented to illustrate the behavior of the three direct optimization methods, and to demonstrate that second-degree polynomial contours yield thrusts substantially identical to the thrusts developed by nozzle contours determined by calculus of variations methods.

*Mechanical Engineering Department, Purdue University, West Lafayette, Indiana, USA.


This paper describes the process of selecting a suitable scheme for the numerical solution of the full Navier-Stokes and energy equations in a Laval nozzle with different initial conditions and boundary conditions. A comparison is made between the computed flowfields and corresponding fully viscous channel flow solutions. The present work shows that nozzle flows with larger Reynolds numbers can also be handled with Crocco's scheme. In addition,
fully viscous channel flow solutions, when available, are an ideal choice as an initial field.

*DLR, West Germany.


The major problem in the design of a low-noise supersonic wind tunnel is to reduce the noise radiated from the turbulent, nozzle-wall boundary layer. This broadband noise is generally the dominant disturbance mode in supersonic flow; hence, their sources are located far upstream of the test section at so-called "acoustic origins" near the nozzle wall. It is shown that tailoring the nozzle contour to reduce the Mach number and boundary-layer thickness at the acoustic origins significantly reduces the noise levels in the upstream part of the test rhombus.

*NASA Langley Research Center, Hampton, Virginia, USA.


Twice during the spring of 1978, the two steel-plate "flex-walls" that form the variable-geometry nozzle of the 11- x 11-foot transonic wind tunnel at Ames Research Center experienced a severe dynamic instability. Both walls fluttered in the fundamental beam-bending mode and experienced stresses approaching the yield strength of the material. Both flutter incidents occurred at Mach numbers of about 1.15. The tunnel, operational for 24 years, had no history of such an instability. The cause of these flutter incidents, the steps taken to prevent a recurrence, and the requalification of the facility are described.

*NASA Ames Research Center, Moffett Field, California, USA.


*Department of Mechanical Engineering, Montana State University, Bozeman, Montana, USA.


A numerical method for computing potential flow through either a planar or axisymmetric nozzle is described. Some results obtained from a computer program based on this method are presented and compared with experimental data.

*Aircraft Research Association Ltd., Bedford, UK.


N81-20085#

Recently developed methods indicate that the total turbulence levels, which include both the acoustic and vorticity modes, can be reduced to 1 percent or less in the settling chamber of a typical blowdown tunnel. Based on recent data and the present analysis of two different blowdown facilities at Langley, methods to achieve these low levels of acoustic and vorticity distributions are recommended. Included are pertinent design details of the damping screens and honeycomb and also the recommended minimum pressure drop across the porous components which will provide the required two orders of magnitude attenuation of the acoustic noise levels.

*NASA Langley Research Center, Hampton, Virginia, USA.

A boundary-layer transition study was carried out in the throat region of the DeLaval nozzle of a supersonic wind tunnel. The study was motivated equally by the need to find thresholds for laminar boundary-layer flow in the tunnel walls when roughness is present and by the desire to simulate transition on roughened blunt bodies in supersonic and hypersonic flow. Detailed inviscid and viscous flow measurements were done from the low subsonic to the supersonic regions of the nozzle throat. The roughness was caused by attaching distributed roughness overlays on the nozzle surface. Transition, detected by hot-film anemometers, was found to move upstream as the flow Reynolds number and/or the roughness height increased. Cast in the coordinates of some of the empirical blunt-body transition correlations currently in use, the present transition data agree with the available blunt-body data when the nondimensional roughness exceeds unity and support the concept of a constant roughness Reynolds number for transition in that regime. At the lower roughness heights, the results show that the transition Reynolds number departs from the aforementioned correlations and approaches a limit insensitive to roughness but characteristic of the experimental facility.

*Department of Mechanical Engineering, Texas A&M University, College Station, Texas, USA.


The purpose of this Note is to present the series solution for axisymmetric nozzles which satisfies the governing differential equations in cylindrical coordinates. The expansion variable employed is \( \epsilon = (R_c + \eta)^{-1} \), where the value of the parameter \( \eta \) is arbitrary. The results of this solution are compared to two data sets for which the nozzle wall radius of curvature is relatively small.

*Mechanical Engineering Department, Montana State University, Bozeman, Montana, USA.


Observations of boundary-layer transition in a supersonic wind-tunnel nozzle throat were made in the region where the local Mach number increased from 0.4 to 2. The wall temperature ranged from adiabatic to 0.66 times the stagnation temperature, and its surface configuration varied from random sand-grain to waffle-type and wire-screen roughness. The roughness height was changed from a fraction to a multiple of the local momentum thickness. It was found that wall cooling accelerates transition markedly, and that for a given average peak-to-valley roughness height, the random sand-type roughness is the most efficient trigger of transition, followed by the waffle-type roughness and then by the wire-screen roughness.
roughness. Weave orientation had a weak effect on transition, but on the other hand the transition Reynolds number decreased drastically when the weave wavelength increased. Transition tripping by lowering the wall temperature would be even more pronounced were it not for local re-laminarization phenomena observed at these lower temperatures. Such isolated re-laminarization or restabilization regions were seen only when the roughness heights were relatively small. A substantial portion of these data cannot be accounted for by existing blunt-body boundary-layer transition correlations.

*Mechanical Engineering Department, Montana State University, Bozeman, Montana, USA.


An inviscid analytical study has been conducted to determine the upstream flow perturbations caused by placing choke bumps in a wind tunnel. A computer program based on the stream-tube curvature method was used to calculate the resulting flow fields for a nominal free-stream Mach number range of 0.6 to 0.9. The choke bump geometry was also varied to investigate the effect of bump shape on the disturbance produced. Results from the study indicate that a region of significant variation from the free-stream conditions exists upstream of the throat of the tunnel. The greatest upstream disturbance distance generally occurred along the tunnel center line and extended from a minimum of about 0.25 tunnel height ahead of the throat to a maximum of about 2.0 tunnel heights for the cases considered. The extent of the disturbance region was, as a rule, dependent on Mach number and the geometry of the choke bump. In general, the upstream disturbance distance decreased for increasing nominal free-stream Mach number and for decreasing length-to-height ratio of the bump. A polynomial-curve choke bump usually produced less of a disturbance than did a circular-arc bump, and going to an axisymmetric configuration (modeling choke bumps on all the tunnel walls) generally resulted in a lower upstream disturbance distance than with the corresponding two-dimensional case. Finally, for some of the circular-arc configurations, certain flow parameters could be estimated by using the throat geometry.

*NASA Langley Research Center, Hampton, Virginia, USA.


A83-11285

The goal in 1937 was to build an aerodynamic-ballistic research institute to provide for all aerodynamic, stability and heat transfer research needs. The construction of the supersonic tunnel, with a 40 cm x 40 cm test-section, is described, which was operational at Peenemunde in 1939 and moved to Kochel in 1943. Seven wind tunnel Laval nozzles, ranging up to Mach number 5, were in use. Their theory, design and construction is outlined and their wave patterns are illustrated. Detailed measurements of delta wings such as the 'Glider A9' or the 'Wasserfall' are presented. The influence of exhaust jet on drag and stability is discussed.


A82-17813

Analytical solutions of two-dimensional and axisymmetric transonic potential flow with shock in slender hyperbolic nozzles are obtained by using asymptotic expansions. A measure of the slenderness of the nozzle is defined by a parameter delta, with only the case of fully developed transonic shock with strength of order delta considered; however, alternative expansions based on either the velocity potential of the stream function formulation, are included. In either case, the entropy increase across the shock is of the
order delta to the 3rd power, while the vorticity for the flow behind the shock is of order delta to the 4th power. The present theory is, in principle, more accurate than transonic discretization schemes which solve the governing equations with jump conditions. In a two-dimensional example, the comparison with a state-of-the-art finite-difference computation shows excellent agreement in the predicted shock shape.

*Cornell University, Ithaca, New York, USA.


A small scale, Mach 3.5 low disturbance tunnel for boundary-layer transition research has been designed, built, and tested. Free stream noise levels and transition Reynolds numbers on a 5° half-angle cone have been measured over a unit Reynolds number range from 10 to 60 million per meter. A two-dimensional nozzle used in the facility incorporates unusual design features of a rapid expansion contour, boundary-layer bleed slots upstream of the throat, and an exit width to height ratio of 1.7 to provide noise levels in the upstream regions of the test rhombus that are substantially lower than in conventional nozzles. The normalized rms levels or fluctuating static pressures obtained from hot-wire measurements vary from extremely low values of less than 0.03 percent (essentially with in the instrument noise range) up to about 0.8 percent depending on the unit Reynolds number, the axial location in the test rhombus, and the bleed slot flow. As is well known, the higher noise levels are caused by eddy Mach wave radiation from the turbulent boundary layers on the nozzle walls. When the valve which controls the boundary-layer bleed flow is open, the wall boundary layers over upstream regions of the nozzle are laminar at the lower unit Reynolds numbers and the absence of high frequency radiated noise then results in cone transition Reynolds numbers that are in the range of free-flight data. As the unit Reynolds numbers are increased, the nozzle wall boundary layers become transitional and turbulent. The noise then increases to peak levels of about 0.5 percent with significant energy up to 150 KHz. The cone transition Reynolds number then decrease considerably to values that are in the range of those measured in conventional wind tunnels.

*NASA Langley Research Center, Hampton, Virginia, USA.


The method of characteristics is commonly used to design a supersonic nozzle. This method is widely applied to large nozzles where the boundary-layer displacement thickness is small compared to the nonviscous flow. As the nozzle becomes larger and the Mach number higher, one should use more characteristics lines in order to achieve an accurate supersonic profile. Such a profile is necessary to obtain a uniform flowfield at the exit of the nozzle in the supersonic test chamber. A simple method that considerably increases the accuracy of the solution for a given number of characteristics lines is described in this paper.

*Armament Development Authority, Haifa, Israel.


N83-34229# This paper describes experiments with a 10 x 6 in. Mach 3.5 two-dimensional nozzle to demonstrate a reduction in free stream disturbance levels over conventional nozzle designs and to observe the subsequent effects upon transition on a slender cone. Three notable features of the nozzle design are: first, the use of a rapid-expansion contour; second, the selection of a wide separation of the two planar sidewalls in order that noise generated by the sidewalls reaches the tunnel centerline downstream of the model location; and third, the provision for boundary layer bleed slots shortly upstream of the nozzle throat for removal of the turbulent boundary layer on the
contraction walls. For unit Reynolds numbers from $2.5 \times 10^5$ up to $15 \times 10^5$ per inch, the observed root-mean-square values of the fluctuating static pressures were found to vary from extremely low values of less than 0.03 percent up to 0.8 percent of the mean static pressure. When the boundary-layer bleed valve was open, the upstream regions of the nozzle wall boundary layers were laminar, which resulted in the cone transition Reynolds number approaching those reported for free flight.

*NASA Langley Research Center, Hampton, Virginia, USA.


TL 357 E54
(Ames Research Center library no.)

Linear compressibility stability theory is used to analyze the recently available flight transition data ($M_\infty=1.2, 1.35, 1.6, 1.92$) and “quiet” tunnel data ($M_\infty=3.5$) for a 5 degree half angle cone. The results show that, for these, low supersonic Mach numbers, the exponent $N$ in the $e^N$ type transition correlation ranges between 9 and 11. These values of $N$ are within the range of values previously determined for subsonic flows (both flight and ground, two- and three-dimensional mean flows). Computations show that the transition in both the flight and quiet tunnel experiment was caused by oblique first mode disturbances. The paper contains detailed results of the calculations including disturbance wave number, wave angle, and group velocity.

*High Technology Corp., Hampton, Virginia, USA.


Recent results from a Mach 3.5 pilot quiet tunnel at the NASA Langley Research Center have shown that very low stream noise levels can be achieved only when the nozzle wall boundary layers are laminar. For the key part of the present study an $e^N$ method transition criteria was applied to several nozzles with a range of expansion rates and wall curvatures. The design Mach number, exit dimensions, and unit Reynolds numbers were the same as for the pilot nozzle. The effect of nozzle design parameters on the extent of laminar wall boundary-layer flow and the corresponding lengths of the quiet test regions depend on the rate of change of the concave curvature and on the wall angle at the inflection point. By manipulation of these parameters, the calculations indicate that it maybe possible to increase significantly the size of the quiet test regions compared with those observed in the pilot nozzle. Some limited experimental results from the pilot nozzle on the influence of nozzle finish and flow contaminants on maintaining and increasing the extent of laminar wall boundary layers are also presented.

*NASA Langley Research Center, Hampton, Virginia, USA.


TA 357 L35
(Ames Research Center library no.)

To advance boundary-layer stability and transition research and to ultimately provide reliable predictions of transition for supersonic flight vehicles, a wind tunnel is required with much lower stream disturbance levels than in conventional supersonic tunnels. Recent results from a Mach 3.5 pilot quiet tunnel at the NASA Langley Research Center have shown that very low stream noise levels can be achieved only when the nozzle wall boundary layers are laminar. Transition Reynolds numbers measured on a slender cone were then in the same range as flight data. The linear amplification of both Tollmein-Schlichting (TS) and Goertler type instabilities was calculated.
for the wall boundary layers in this Mach 3.5 pilot nozzle. By using an eN method, it was determined that amplified Goertler vortices were involved in the transition process. The resulting transition criteria were then applied to several nozzles with different expansion rates and wall curvatures. By manipulation of these parameters, the calculations indicate that it may be possible to increase significantly the size of the quiet test regions compared with those observed in the pilot nozzle.

*NASA Langley Research Center, Hampton, Virginia, USA.


N85-18991

This catalog updates and supplements previous surveys conducted on aeronautical facilities. NASA undertook this survey primarily to form a database from which to assess its own capabilities and that of the United States in aeronautical research and development, particularly in relation to that of the western world. No information on Russian wind tunnels is presented. Thirty-nine supersonic wind tunnels are identified in this catalogue.

*Office of Aeronautics and Space Technology, NASA HQ, Washington, DC, USA.


To advance boundary-layer stability and transition research at supersonic speeds, a wind tunnel with very low stream disturbance levels is required. Recent work at NASA Langley has shown that the required low noise levels can be obtained only when the boundary layers are laminar over the upstream regions of the nozzle walls where the radiated noise originates. The eN method based on data from four test nozzles was used to predict transition in four study nozzles. The results indicate that even larger quiet test length Reynolds numbers are possible than observed in the test nozzles may be obtained in longer nozzles designed with the wall inflection point located far downstream.

*High Technology Corp., Hampton, Virginia, USA.


The S3 blowdown tunnel of the ONERA Modane test center has been in operation since 1960. Removable nozzles or test sections are used: transonic inserts, 2D insert, supersonic nozzles at M = 2 - 3.4 - 4.5. A supersonic nozzle with controllable Mach number in the range 1.65 to 3.8 has just been made, mainly for missile and launcher tests. The design of this nozzle is described along with its operating range and main characteristics. This nozzle consists of two symmetrical rigid blocks rotated by an original single actuator system (a single actuator for each half nozzle) which changes the throat by rotating the rigid blocks and adapting the divergent part, made of a flexible plate.

*ONERA, Chatillon-Cedex, France.


A86-24755#

A flexible wall nozzle design approach is presented that couples optimization concepts and current design methodologies. The optimization process provides for a tuned nozzle design which meets specific performance and operating environment requirements. A case study involving a generic transonic flexible nozzle wall design is utilized to focus on the important steps in the nozzle design optimization process. The integrated nozzle design is based on the definition of the optimum plate thickness, number of actuators, actuator distribution and contour shape covering the Mach number and Reynolds number range of the nozzle. Use of this technique provides the designer with the
capability to totally integrate geometric, operation and performance requirements into the nozzle design, yielding design simplicity and controllability and minimizing construction costs.

*Sverdrup Technology Inc., Tullahoma, Tennessee, USA.


Recent results from a research program at NASA Langley to develop a quiet supersonic tunnel have shown that limited upstream regions of the nozzle-wall boundary layer in a small Mach 3.5 pilot tunnel can be maintained laminar at high unit Reynolds numbers. The high level acoustic disturbances are then eliminated and transition Reynolds numbers measured on cones approach those for atmospheric flight. Critical design requirements for a large facility (which will be a blow-down tunnel) are presented in this paper. These requirements are based on studies completed or underway in the pilot quiet tunnels at Mach numbers of 3, 3.5, and 5. In particular, the design requirements for high quality supply air filters, noise attenuation devices in the settling chamber, and nozzle coordinate accuracy and surface finish are illustrated and quantified with data from the pilot tunnels.

*S NASA Langley Research Center, Hampton, Virginia, USA.


The renewed emphasis on fundamental aerodynamic research in high-speed flows at the NASA Langley Research Center has resulted in the acquisition of a new wind tunnel. The new 20-in. Supersonic Wind Tunnel (SWT), formerly located at the Jet Propulsion Laboratory (JPL), is in the process of being assembled following extensive modifications. Several unique features have been incorporated into the new tunnel. Some of these features are a “quiet” flow control valve, thick sintered screens with very large pressure drops, special fairings in the contraction section, an “instantaneous” relief hatch in the test section, and a switching exhaust system to permit flow into a vacuum sphere during startup and shutdown and to atmosphere during data acquisition. This paper will describe the modifications, unique features, and expected operating map of the new tunnel.

*N NASA Langley Research Center, Hampton, Virginia, USA.


An optimum supersonic facility is defined by identifying (a) major shortfalls in current supersonic facility capability, and (b) advanced technology capable of addressing these deficiencies. Major problem areas include (l) large amplitude stream disturbances caused by acoustic radiation from nozzle wall boundary layers, of particular importance to the boundary layer transition problem on models, (2) flow field and model shape distortion caused by sting supports and their installation, (3) lack of sufficient three-dimensional flow visualization and flow diagnostic capability to identify “Reynolds number effects,” (4) lack of aerodynamic/propulsion facilities to investigate the necessary range of vehicle attitudes and transient propulsion behavior, and (5) inordinately large energy usage, caused primarily by inefficient diffusers. Remedies proffered for these shortfalls include (a) development of a quiet supersonic wind tunnel with laminar wall boundary layers, (b) further development/usage of magnetic balances, (c) scanning “stimulated Raman spectroscopy” to measure time-average, three-dimensional flow field details, the data base to serve as both diagnostics and, through computer graphics, visualization, (d) an “effectively flexible” wall free-jet wind tunnel capable of transient attitude and speed changes, and (e) improved diffusers. The optimum wind tunnel, which is
both technically and economically justifiable (especially in view of the emerging CFD capability), is not a single facility and not even a new one. The major emphasis should be placed upon retrofit of existing tunnels.

*NASA Langley Research Center, Hampton, Virginia, USA.


The classical problem of transonic flow through a hyperbolic nozzle with or without a shock wave, has been revisited by applying recently developed numerical methods. Both planar and axisymmetrical cases have been considered. The full potential equation is solved in conservation form using the finite volume method of Jameson and Caughey. To treat the mixed nature of the equation, either a first- or a second-order numerical viscosity in the direction of the flow is added explicitly in conservation form. A multigrid, alternating direction implicit (ADI) method is used to solve the difference equations and the results are compared with analytical and numerical results of earlier researchers. The scheme converges rapidly, and results are in good agreement with those of earlier workers, including analytical results for flows containing shock waves.

*Cornell University, Ithaca, New York, USA.


This section gives a very brief overview of supersonic and hypersonic aerodynamics. Comments about nozzle design are included.


The paper describes the results of nozzle model tests used in steady and unsteady gas turbine engine operating states at low available expansion degrees. It is concluded that the shape of the subsonic part of the nozzle and the different operation intensity variation rates have a significant influence on the flow pattern, pulsations, vibrations, and the noise level.

The experimental and theoretical program at NASA Langley to develop a low-disturbance wind tunnel is described. Theoretical analysis of the experimental test results by means of a linear stability theory has shown that amplification of Goertler vortices on the concave walls of nozzles at Mach numbers from 3 to 5 are the cause of transition. The downstream boundaries of the quiet test region are determined by Mach lines extended downstream from the transition locations on the nozzle walls. The theory was then used to design advanced nozzles for Mach numbers 3.5 and 6 that will provide much longer quiet test regions. Transition on the nozzle walls is extremely sensitive to nozzle wall roughness. Allowable roughness on the two advanced nozzles and the new proposed large-scale facility were then calculated from a roughness Reynolds number criterion based on the test results.

*NASA Langley Research Center, Hampton, Virginia, USA.


A89-29244#

To investigate the mechanism of flow breakdown in the S2 wind tunnel at the ONERA Modane-Avrieux center, a measurement system was designed, which makes it possible to visualize in real time the variations of the Mach number distributions along the walls of the first throat, the test section, the second throat, and the diffuser. The measurement system consists of a set of pressure multi-transducers producing the instantaneous Mach number distributions. Several cases are presented which clearly show the effectiveness of the device, and the monitoring and surveillance possibilities.

*ONERA, Modane-Avrieux, France.


A numerical one-dimensional calculation of the transonic Laval nozzle flow was performed. A computer program was developed and tested on a steady example. Flow separation was taken into account using a simplified model. In calculations with separation, it is shown that separated flow behind a shock wave, with a periodic pressure excitation at a certain frequency allows excitation and eigenfrequency to be observed, providing a possible explanation of the experimentally observed irregular shock oscillations.

*Max Planck Institute, Stuttgart, West Germany.


A89-47410#

This paper presents a method for ascertaining two-dimensional characteristics which can be applied to the calculation and design of a rocket supersonic nozzle. Some useful conclusions are drawn from the analysis of computational results. The results brought about by the method are so good that the method can be suggested as a reference for nozzle design and heat transfer analysis of rocket nozzles.


The problem of, and a particular solution for, the simulation of free-flight supersonic boundary-layer transition in ground facilities are discussed. Fundamental problems addressed include: (a) technology requirements for blowdown low-disturbance wind-tunnels from upstream piping through to the diffuser with emphasis on maintenance of laminar nozzle wall flows, (b) characteristics and operational aspects of the NASA-LaRC Pilot Supersonic Quiet Tunnel, and (c) comparisons of high-
speed quiet facility transition studies with results from free flight, stability theory, and conventional ground facilities. Maintenance of laminar nozzle wall flow is shown to be a necessary condition for accurate simulation of high-speed flight transition behavior in ground facilities.

*NASA Langley Research Center, Hampton, Virginia, USA.


N89-16860#

A procedure is described which allows the design of relatively short adjustable nozzles without resorting to the assumption of radial flow upstream of the nozzle inflection point. This design procedure is used to generate a number of fully flexible nozzle designs for a design Mach number of 1.4. Flexible nozzle parameters such as pressure load, number of actuators and nozzle length are varied in order that the influence of these parameters on test section flow quality may be evaluated. The influence of nozzle actuator setting accuracy is also estimated. The parametric study shows that high flow quality can be achieved for the shortest nozzle considered. The examination of jack setting errors shows that for well designed nozzles, jack setting error will be a significant source of test section flow nonuniformities.

*DSMA International Inc., Toronto, Canada.


A second-order accurate method-of-characteristics algorithm is used to determine the flow field and wall contour for a supersonic, axisymmetric, minimum length nozzle with a straight sonic line. Results are presented for this nozzle and compared with three other minimum length nozzle configurations. It is shown that the one investigated actually possesses the shortest length as well as the smallest initial wall turn angle at the throat. It also has an inflection point on the wall contour, in contrast to the other configurations.

*School of Aerospace and Mechanical Engineering, The University of Oklahoma, Norman, Oklahoma, USA.


A90-16799#

The transonic potential flows in the throat region of two-dimensional and axisymmetric nozzles were calculated using a biquadratic equation for the velocity potential in case of \( R/h = 3, 4 \) and \( 5 \), where \( R \) is the radius of the throat wall and \( h \) is the half-width of throat. In addition, the effects of the radius of curvature of the wall were calculated; the flow was also visualized using a schlieren technique.


Inviscid, stationary 2-D flows of moist air with nonequilibrium phase transition are calculated numerically using the Euler equation coupled with the classical nucleation theory. In Laval nozzles with supercritical heat addition the apparent normal shocks with local subsonic regions in the surrounding supersonic flow field are computed. Homogeneous condensation in local supersonic regions around profiles is calculated for the first time with detailed analysis of the 2-D structure for continuous change of state as well as for stationary double shock systems. For nozzles and profiles, all numerical results are well verified by experiments.

*Institute fur Stromungslehre und St-maschinen Universitat (TH) Karlsruhe, West Germany.

This note describes the transition process as a 6-stage affair which is initiated by the ambient disturbance environment. The importance of stream turbulence, particulates, electrostatics, and roughness are discussed. The overall disturbance levels in flight are compared with those disturbances found in wind tunnels.

*NASA Langley Research Center, Hampton, Virginia, USA.


Boundary-layer transition data on a cone and flat plate obtained in the Mach 3.5 Pilot Low-Disturbance Tunnel at NASA Langley are presented. Measured flat plate transition Reynolds numbers in this tunnel are an order of magnitude higher than previous results obtained in conventional noisy supersonic wind tunnels. Transition predictions based on compressible linear stability theory and the $e^N$ method with $N=10$ are in excellent agreement with the measured locations of transition onset for both the cone and flat plate under low-noise conditions in this tunnel. This investigation has resolved the discrepancies between the results of linear stability theory and the data from conventional supersonic wind tunnels regarding the ratio of cone to flat plate transition Reynolds numbers.

*University of Oklahoma, Norman, Oklahoma, USA.


The method of characteristics is used to generate supersonic wall contours for two-dimensional, straight sonic line (SSL) and curved sonic line (CSL) minimum length nozzles for exit Mach numbers of 2, 4, and 6. These contours are combined with subsonic inlets to determine the influence of the inlet geometry on the sonic-line shape and location and on the supersonic flow field. A modified version of the code VNP2 is used to compute the inviscid and laminar flow fields for Reynolds numbers of 1170, 11700, and 23400. Results indicate that the inlet geometry directly determines the sonic-line shape and location. Supersonic flow field phenomena, including boundary-layer separation and oblique shock waves, are observed to be a direct result of the inlet geometry. The sonic-line assumptions made for SSL prove to be superior to those of the CSL.

*High Technology Corp., Hampton, Virginia, USA.


A computer program was developed to study the characteristics of an actively cooled panel designed for autoheating flux on the order of 200 Btu/sq ft/sec. At this high heat flux, the panel experiences very high thermal stress that must be considered in addition to bending stress due to internal pressure. This forces the structural and thermal analyses to depend on each other. The methodology used in the computer program was developed to generate trends of how design parameters affect the panel structure and thermal performance. The analytical results can then be used to identify a design path leading to a panel design that is optimized with respect to the vehicle as a whole. Discussions of interactions between structural and thermal design parameters are provided.

*Space Systems Division, General Dynamics Corp., San Diego, California, USA.
To advance boundary-layer stability and transition research and to ultimately provide reliable predictions of transition for supersonic flight vehicles, a wind tunnel is required with very low stream disturbance levels comparable to free flight conditions. Experimental and theoretical research to develop a low-disturbance supersonic wind tunnel has achieved a breakthrough. A new concept for nozzle design is presented which promises a large increase in the length of the quiet test core. The Advanced Mach 3.5 Axisymmetric Quiet Nozzle is the first prototype built to prove the new design concept. Experimental results from this new nozzle on the extent of laminar wall boundary layers are compared with data from other nozzles and with theoretical predictions based on linear stability theory. The Reynolds numbers based on the measured length of the quiet test core for this new nozzle are in excellent agreement with the theoretical predictions. The effect of surface finish on the nozzle performance is also discussed.

*High Technology Corp., Hampton, Virginia, USA.
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# Supersonic Wind Tunnel Nozzles—
# A Selected, Annotated Bibliography To Aid in the Development of Quiet Wind Tunnel Technology

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## Abstract
This bibliography, with abstracts, consists of 298 citations arranged in chronological order. We selected the citations to be helpful to persons engaged in the design and development of quiet (low disturbance) nozzles for modern supersonic wind tunnels. We include author, subject, and corporate source indexes to assist with the location of specific information.

## Key Words (Suggested by Author(s))
- Supersonic nozzles
- Supersonic quiet flows
- Supersonic wind tunnels
- Nozzle design

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APPENDIX B
Status of Adaptive Wall Technology for Minimization of Wind Tunnel Boundary Interferences

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Stockholm, Sweden
10-14 September 1990
STATUS OF ADAPTIVE WALL TECHNOLOGY FOR MINIMIZATION OF WIND TUNNEL BOUNDARY INTERFERENCES

Stephen W. D. Wolf
MCAT Institute
Moffett Field, California, USA

Abstract

This paper reviews the status of adaptive wall technology to improve wind tunnel simulations. This technology relies on making the test section boundaries adjustable, with a tunnel/computer system to control the boundary shapes. This paper briefly considers the significant benefits of adaptive wall testing techniques. A brief historical overview covers the disjointed development of these testing techniques from 1938 to present. Currently operational Adaptive Wall Test Sections (AWTSs) are detailed. This review shows a preference for the simplest AWTS design with 2 solid flexible walls. A review of research experience with AWTSs shows the many advances in recent times. We find that quick wall adjustment procedures are available. Requirements for operating AWTSs on a production basis are discussed. Adaptive wall technology is mature enough for general use in 2-D testing, even in cryogenic wind tunnels. In 3-D testing, this technology is not so advanced because of low priority development and misconceptions.

Symbols

c - Chord

Cn - Normal force coefficient

h - Test section height

U∞ - Free stream velocity

Δw - Increment of local upwash velocity

Δu - Increment of local streamwise velocity

1. Introduction

The means to improve and gain more efficiency from our flight vehicles relies on better and better simulations of the "real" flow in our wind tunnel experiments. It is for this reason that improvements to wind tunnel data remain the subject of considerable research effort. Unfortunately, today's wind tunnel data still suffers from significant wall interference effects, particularly at transonic speeds. This is despite considerable efforts to remove this simulation problem over the last 44 years. 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 available). Ventilated test sections are used to relieve transonic blockage and prevent choking (introducing other complex boundary interferences). Post-test corrections, of varying sophistication, are applied to the model data in an effort to remove wall interferences. Usually, all three techniques are used together in transonic testing. Alas, these techniques still fail to achieve the high levels of accuracy we must now demand from wind tunnel simulations. In addition, these old techniques have led to expensive compromises for test section/model size.

A solution to this dilemma has existed, in a conceptual form, for about 52 years. It involves using testing techniques which minimize wall interferences at the very source of these disturbances. These techniques adapt the test section boundaries to streamline shapes so the test section walls become nearly invisible to the model. We know this concept as the Principle of Wall Streamlining which was first used in 1938 as a means of relieving transonic blockage.1

This paper briefly reviews the development of adaptive wall testing techniques as a background to the current status. We detail operational transonic AWTSs which are currently used for both conventional and turbomachinery research. This paper reviews 2- and 3-D research to illustrate the state of the art in adaptive wall testing techniques. Finally, we consider the operational aspects of AWTSs, since the practicabilities of adaptive walls play a critical factor in the use of this technology. In conclusion, an assessment of the accumulated adaptive wall experience is presented and possible directions for future developments are indicated.

2. Adaptive Wall Benefits

Although the basic advantages of adaptive wall testing techniques have been reported many times, a brief overview is appropriate. Adaptive walls offer several important advantages other than the major benefit of minimizing wall interferences. With wall interferences minimized, we are free to increase the size of the model for a given test section. We can double the test Reynolds number and have a larger model to work with. Alternatively, we can shrink the test section and reduce the tunnel size and operating costs. Interestingly, the task of magnetically suspending models (to remove 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 complex boundaries with ventilated walls. This smoothness minimizes disturbances to the tunnel free stream significantly improving flow quality. (An advantage that is becoming more important in transonic boundary layer transition research.) In addition, smooth walls reduce the tunnel drive power required for a given test condition, with the model and test section size fixed. The elimination of the plenum volume, when a closed AWTS is used for transonic testing, reduces settling times and minimizes flow resonance, which is particularly important for blowdown tunnels.

Adaptive walls can provide the aerodynamicist with real-time "corrected" data, even in the transonic regime. This fact presents another significant advantage to the wind tunnel user. Since, the final results are know real-time, test programmes can be much more efficient. Use of adaptive walls should significantly reduce the number of data points and tunnel entries necessary to achieve the test objectives.

It should be noted that the simulation of free-air conditions is one of 6 flow field simulations2 that adaptive wall technology can produce. It is possible to use multiple simulations with the same model and AWTS. This can and has been a useful advantage for CFD validation work and tunnel versatility.

3. 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). The National Physical Laboratory (NPL), UK, built the first adaptive wall test section in 1938, under the direction of Dr. H. J. Gough.1 Their pioneering research proved that streamlining the flexible walls of an AWTS was the first viable technique for achieving high speed (transonic) flows in a wind tunnel. They opted for minimum mechanical complexity in their AWTS and used only two flexible walls. The absence of computers made wall streamlining a slow and labour intensive
process. Sir G. I. Taylor developed the first wall adjustment procedure. NPL successfully used flexible walled AWTS up until the early 1950s, generating a vast amount of 2- and 3-D transonic data.

The arrival of ventilated test sections at NACA Langley in 1946, provided a "simpler" approach to high speed testing. The adjustments to the test section boundaries are passive in a ventilated test section and active in an AWTS. The apparent simplicity of ventilated test sections led to the political obsolescence of NPL's AWTS and the benefits of adaptive wall technology became forgotten.

After about 20 years, interest in AWTSs was rekindled. Around 1972, several researchers, in Europe and the USA, independently rediscovered the concept of adaptive wall testing techniques. These researchers sought better free air simulations in transonic wind tunnels. The adaptive wall approach offered them an elegant way to simplify the wall interference problem. Adaptive wall adjustment procedures need only consider the flow at the test section boundaries (in the farfield), the complex flow field around the model need not be considered. Therefore, the adaptive wall concept allows us to simplify the "correction codes" at the expense of increasing the complexity of the test section hardware.

This renewed interest, helped greatly by the availability of computers, has spawned the various adaptive wall research groups now found around the world. We have seen a variety of AWTS designs for testing 2- and 3-D models. Some unusual designs have been built including a rubber tube AWTS and a pilot multi-wall AWTS for automobile research. AWTSs are now available for commercial use at NASA Langley (USA), ONERA/CERT (France), and TsAGI (USSR). A complete list of currently operational AWTSs is shown on Table 1 below.

### Table 1 - Adaptive Wall Test Sections Currently in Use

<table>
<thead>
<tr>
<th>Organization</th>
<th>Tunnel</th>
<th>X-Section (h x w) m</th>
<th>Length, m</th>
<th>Approx. Max. Mach No.</th>
<th>Approx. Max. R. (millions)</th>
<th>Walls</th>
<th>Adaptation Control</th>
<th>Remarks</th>
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<tr>
<td>ONERA/ CERT</td>
<td>T2</td>
<td>0.37 x 0.39 Rectangular</td>
<td>1.3</td>
<td>&gt;1.0</td>
<td>2.0</td>
<td>2 Flexible 2 Solid</td>
<td>16 Jacks/Wall</td>
<td>Issue 2</td>
</tr>
<tr>
<td>NPI UVJ</td>
<td>Low Speed</td>
<td>0.256 x 0.238 Rectangular</td>
<td>1.0</td>
<td>&gt;1.5</td>
<td>2.0</td>
<td>2 Flexible 2 Solid</td>
<td>19 Jacks/Wall</td>
<td>Issues 2, 5, 9</td>
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<tr>
<td>Arizona University</td>
<td>HLAT</td>
<td>0.51 Square</td>
<td>0.914</td>
<td>0.2</td>
<td>2 Arrays of Venetian Blinds 2 Solid</td>
<td>16 Panels of Vanes and a Variable Angle Nozzle</td>
<td>Issue 3</td>
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<tr>
<td>Genova University</td>
<td>Low Def.</td>
<td>0.2 x 0.05 Rectangular</td>
<td>1.58</td>
<td>&gt;1.8</td>
<td>2 Flexible 2 Solid</td>
<td>13 Jacks-Ceiling 26 Jacks-Floor</td>
<td>Issue 7</td>
<td></td>
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<tr>
<td>Genova University</td>
<td>High Def.</td>
<td>0.2 x 0.05 Rectangular</td>
<td>1.6</td>
<td>&gt;1.18</td>
<td>2 Flexible 2 Solid</td>
<td>7 Jacks/Wall</td>
<td>Issue 7</td>
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<td>NASA Ames</td>
<td>ASWTS1</td>
<td>0.61 x 0.41 Rectangular</td>
<td>2.79</td>
<td>&gt;0.8</td>
<td>30</td>
<td>2 Flexible 2 Solid</td>
<td>11 Jacks/Wall</td>
<td>Issue 10</td>
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<tr>
<td>NASA Ames</td>
<td>ASWTS2</td>
<td>0.61 x 0.41 Rectangular</td>
<td>2.79</td>
<td>&gt;0.8</td>
<td>30</td>
<td>2 Flexible 2 Solid</td>
<td>-</td>
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<tr>
<td>NASA Langley</td>
<td>0.3-m TCT</td>
<td>0.33 Square</td>
<td>1.417</td>
<td>&gt;1.3</td>
<td>120</td>
<td>2 Flexible 2 Solid</td>
<td>18 Jacks/Wall</td>
<td>Issues 1-5, 7, 8</td>
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<td>RPI, Troy, NY</td>
<td>3 x 15</td>
<td>0.39 x 0.07 Rectangular</td>
<td>0.6</td>
<td>0.8</td>
<td>4 Solid</td>
<td>Multiple Top Wall Inserts</td>
<td>-</td>
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<tr>
<td>Sverdrup Technology</td>
<td>AWAT</td>
<td>0.365 x 0.61 Rectangular</td>
<td>2.438</td>
<td>0.2</td>
<td>3 Flexible 2 Solid</td>
<td>102 Dows-Ceiling 15 Dows-Sidewall</td>
<td>Issue 4</td>
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<td>TsAGI</td>
<td>T-128</td>
<td>2.75 Square</td>
<td>8.0</td>
<td>1.7</td>
<td>4 Porous</td>
<td>32 Control Panels per Wall</td>
<td>Issue 11</td>
<td></td>
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<tr>
<td>Umberto Nobili</td>
<td>FWWT</td>
<td>0.2 Square</td>
<td>1.0</td>
<td>0.6</td>
<td>3.5</td>
<td>2 Flexible 2 Solid</td>
<td>18 Jacks/Wall</td>
<td>-</td>
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</tbody>
</table>

Note: The Remarks column refers to Adaptive Wall Newsletter Issues (published roughly quarterly by NASA researchers) which contain related articles.
4. Transonic AWTSs Currently Operational

(In alphabetical order by organization)

4.1 Aerodynamic Institute, RWTH Aachen, West 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 40 cm (15.75 inches) square and 1.414 m (4.64 feet) long. The top and bottom walls are flexible and mounted between two parallel sidewalls. The flexible walls are made from 1.3 mm (0.051 inch) thick spring steel. Each wall is supported by 24 motorized jacks (See Figure 1).

Fig. 1 - The exposed TST adaptive wall test section.

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. 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. Wall adaptation calculations and automatic wall adjustments are made between tunnel runs.

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 ±0.1 mm (±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 0.64 m (25.2 inches) wide, 0.52 m (20.47 inches) high and 1.75 m (5.74 feet) 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 is split into 11 equal length segments. The porosity of each segment can be independently varied between 0% and 11%, using some manual adjustments.

Researchers have made preliminary 2-D tests at Mach numbers up to 0.8 at zero lift conditions. The wall adaptation procedure is experimental at this stage. Boundary measurements were made at two control surfaces/lines near one of the porous walls, probably using Calspan pipes.

4.3 DLR - Institute of Experimental Fluid Mechanics, Goettingen, West Germany

During 1987/8, researchers at DLR modified the 2-D supersonic nozzle of the DLR High Speed Wind Tunnel (HKG) into an AWTS. The top and bottom nozzle walls are made of highly flexible 4 mm (0.157 inch) thick steel plates. The shape of each wall is set by 17 pairs of equally spaced hydraulic jacks (See Figure 2).

The AWTS consists of an initial contraction followed by a 2.2 m (7.22 foot) straight section. This straight portion, in which the model is mounted, is nominally 0.67 m (2.2 feet) high and 0.725 m (2.38 feet) wide. Each wall of the test section is equipped with 3 rows of pressure taps for boundary measurements. The wall adjustment procedure of Wedemeyer/Lamarche is used to minimize interferences along the tunnel centerline.

This AWTS is used for evaluation of 2-D wall adaptation in 3-D testing. Researchers have tested sting mounted 3-D models, both lifting and non-lifting, up to about Mach 0.8.

4.4 Genoa University, Italy

The Department of Energy Engineering at the University of Genoa operates two adaptive wall cascade tunnels. Both tunnels have a cross-section of 0.2 m (7.87 inches) high and 5 cm (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 walls. The flexible walls are 1.58 m (5.18 feet) long and each is shaped by 36 manual jacks (see Figure 3). Wall streamlining is performed upstream and downstream of the cascade.

Fig. 3 - A side view of the Genoa University LDBCT.

The HDBCT has a similar configuration except the AWTS is 1.6 m (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. 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.
Both AWTS need only approximate wall adaptation procedures 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 has 2 AWTS for use in their intermittent 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 AWTS have a rectangular cross-section which is 0.61 m (24 inches) high and 0.41 m (16 inches) wide. The AWTSs are 2.79 m (9.15 feet) long.  

AWTS (#1) has 7 manually adjusted jacks supporting each flexible wall, while AWTS (#2) has 11 jacks powered by stepper motors (See Figure 4). 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 walls. The wall jacks on AWTS #2 are fast moving because of the short duration tunnel runs. (Wall movement is at about 5 mm (0.2 inch) per second.)  

Fig. 4 - A view of the NASA Ames AWTS (#2) showing the many sidewall apertures.  

The flexible walls are made of 17-4 PH stainless steel plates and are 2.53 m (8.32 feet) long. In AWTS (#1), the flexible walls are 15.9 mm (0.625 inch) thick at the ends tapering to 2.39 mm (0.094 inch) in the middle. In AWTS (#2), the flexible walls taper down to 2.39 mm (.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 suitable wall adaptation based on simple influence coefficients.  

AWTS #1 has been used for 2-D and 3-D CFD code validation. No wall adjustment procedure is used. The flexible walls are simply set to predetermined shapes depending on the investigation underway. Studies of LDA wake measurements behind 2-D aerofoils have also been carried out. Preliminary 3-D tests with a sidewall mounted half model were performed with straight walls. The AWTS (#2) has yet to be installed in HRC-2.  

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, 3.17 mm (0.125 inch) thick at the ends and thin down to 1.57 mm (0.062 inch) thick in the middle.  

The cross-section of the AWTS is 0.33 m (13 inches) square and the AWTS is 1.417 m (55.8 inches) long. The flexible walls are 1.417 m (55.8 inches) long and are shaped by 18 motorized jacks per wall. The downstream ends of the flexible walls are attached, by sliding joints, to a 2-D variable diffuser (formed by flexible wall extensions) between the AWTS and the rigid tunnel circuit. The shape of the variable diffuser is controlled by 6 motorized jacks. The wall jacks are designed with 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). 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 up to stagnation pressures of 90 psia (6 bars). The jack motors and position sensors are located outside the pressure shell in a near ambient environment (see Figure 5). Side wall boundary layer control is available by fitting porous plates in the sidewalls, upstream of the model position. Boundary layer suction has been successfully used in 2-D testing with normal wall adaptation. We take 2-D wake measurements using a traversing pivot/static probe mounted in one of 3 positions downstream of the aerofoil location.  

Fig. 5 - NASA Langley 0.3-m TCT adaptive wall test section: view above shows AWTS with the left side of the pressure vessel for 2-D testing with normal wall adaptation. We take 2-D wake measurements using a traversing pivot/static probe mounted in one of 3 positions downstream of the aerofoil location.  

We have used the wall adjustment procedure of Judd et al. for 2-D testing. The 2-D test section includes normal force coefficients up to 1.54 and Mach numbers up to 0.82 with a model blockage of 12%. Boundary measurements are static pressures measured along the centerline of the flexible walls at the jack locations. Wall streamlining takes on average less than 2 minutes and is paced by slow wall movements. A generalized and documented non-expert system 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 hour period.  

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 procedure of Rebstock to minimize interferences along a pre-set 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. Downstream flexible wall curvature is automatically minimized by rotation of the tunnel centerline. For low supersonic tests, the adapted wall shapes are based on wave theory and form a 2-D supersonic nozzle ahead of the model.
The flexible walls are set to a nominal accuracy of ±0.127 mm (±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 PC computer) to better than 0.002 during each wall adaptation process (streamlining).

4.7 ONERA/CERT, Toulouse, France

The AWTS fitted in the intermittent ONERA/CERT T2 transonic cryogenic tunnel became operational in 1978. This AWTS became the first cryogenic AWTS in 1981, when the T2 tunnel was modified to operate cryogenically for 1 to 2 minutes at a time. This French AWTS is 0.37 m (14.57 inches) high, 0.39 m (15.35 inches) wide and 1.32 m (51.97 inches) long. The AWTS has 2 flexible walls and 2 parallel solid sidewalls. The flexible walls are made of 1.5 mm (0.059 inch) thick Invar steel plate. The shape of each flexible wall is controlled by 16 hydraulic jacks attached to wall ribs (See Figure 6). These ribs are electron beam welded to the outside of the flexible walls. The hydraulic jacks move the flexible walls very rapidly at about 6 mm (0.24 inch) per second. The wall jacks have enough power to damage the flexible walls. During a cryogenic run, the flexible walls rapidly reach the low test temperatures, while the jack mechanisms remain at normal ambient temperatures. Sidewall BLC is available for 2-D testing by placing porous plates around the aerofoil(sidewall) junctions. BLC suction is routinely used with wall adaptation. In 2-D tests, a pitot/static rake, mounted on a sting support downstream of the wing, is used for wake measurements.

Fig. 6 - ONERA/CERT T2 AWTS with a C5 model installed for 3-D riblet tests.

A wall adjustment procedure developed by Chevallier et al is used for 2-D testing. This procedure is tunnel dependent and has no documented test envelope for non-expert use. Computer controlled wall streamlining in about 10 seconds is possible. However, 2 short tunnel runs are normally required per test point for 2-D tests at about Mach 0.8. Boundary measurements are static pressures measured equidistant along the centerline of the flexible walls.

For 3-D testing, the Wedemeyer/Lamarche wall adjustment procedure is used. Both lifting and non-lifting models have been tested up to Mach 0.97.14 The T2 AWTS is the closest we have come to a production-type 3-D AWTS. Researchers have carried out several production-type studies of riblets with 3-D models (See Figure 6). Boundary measurements are static pressures measured along 3 rows on each flexible wall and a single row on one sidewall.

The shape of the flexible walls can be measured to 0.05 mm (0.002 inch). The wall curvature is checked before any wall movement is initiated. Mach number is control by a downstream sonic throat which acts as a fairing between the AWTS and the fixed diffuser. In general, the Mach number is not held constant during each wall adaptation process.

4.8 ONERA, Chalais-Meudon, France

The ONERA SSCh wind tunnel was fitted with an AWTS about 1984, primarily to investigate shock wave cancellation with adaptable but solid test section boundaries. The AWTS is 22 cm (8.66 inches) high, 18 cm (7.09 inches) wide and 30 cm (11.8 inches) long. The impervious and adjustable floor and ceiling are mounted between solid parallel sidewalls (See Figure 7). The floor and ceiling are made up of 151 transverse sliding plates. Each of the 302 plates is 18 cm (7.09 inches) wide and 1.5 mm (0.059 inch) thick. The plates are manually adjusted to match specially machined profiles for each test condition. Upstream of the AWTS is a fixed supersonic nozzle which produces a Mach 1.2 stream at the test section entrance.

Both 2-D and 3-D models have been tested where strong shock waves reach the floor and ceiling. The model's streamwise position is adjusted to where the bow shock impinges on the floor and ceiling at the junction between the fixed nozzle and the AWTS. Suitable wall curvature is then used to cancel the shock wave reflection or deflect the reflection harmlessly downstream of the model. Boundary measurements are made along streamwise lines using a five hole probe. No special problems were reported during tests at low supersonic Mach numbers.16

4.9 Rensselaer Polytechnic Institute, New York, USA

Since the mid-1980s, the Rensselaer Polytechnic Institute has operated two AWTSs for rotorcraft research in particular the study of 2-D aerofoils with passive boundary layer control. The RPI 3 x 8 transonic wind tunnel is fitted with a rectangular AWTS, 20.3 cm (8 inches) high, 7.6 cm (3 inches) wide, and 0.6 m (23.62 inches) long. The top wall is flexible and supported by 6 jacks. The other three walls are solid. The 2-D aerofoil is mounted in the bottom wall with a boundary layer removal slot ahead of the leading edge. A relatively large aerofoil with a 10.16 cm (4 inch) chord has been tested in this AWTS at Mach numbers up to 0.86.

The RPI 3 x 15 transonic tunnel has a similar AWTS arrangement except the test section height is increased to 38 cm (15 inches). Also the top wall is not flexible and different wall shapes are set in the AWTS by using interchangeable wooden wall inserts. Tests of 14% thick aerofoils at Mach numbers up to 0.9 are reported.18

Researchers use a simple wall adjustment procedure in these AWTSs. One-dimensional wall influence coefficients are used to remove the blockage effects associated with testing a large aerofoil in these small test sections. Boundary measurements are static pressures measured along the test section walls.
The Transonic Self-Streamlining Tunnel (TSWT) at the University of Southampton is one of the first fully automated AWTSs. Built in 1976/7, TSWT has a 15 cm (6 inch) square test section which is 1.12 m (3.67 feet) long. The floor and ceiling are flexible and made from woven man-made fibre (Terylene). The flexible walls are 5 mm (0.2 inch) thick at the ends tapering to 2.5 mm (0.1 inch) thick in the middle. Each flexible wall is supported by 19 motorized jacks (See Figure 8). A sliding joint attaches the downstream ends of the flexible walls to a 2-D variable diffuser (which is 2 plates, each controlled by a single motorized jack). The wall jacks are designed with insufficient stepper motor power to permanently damage the flexible walls. The 2 sidewalls are solid and parallel.

The wall adjustment procedure of Judd et al. for 2-D testing was developed in TSWT, and is used routinely for all 2-D tests where the flow at the flexible walls is up to just sonic. Wall streamlining is generally achieved in less than 2 minutes. If the walls become sonic, a Transonic Small Perturbation code is included in the Judd procedure and 2-D testing has been successfully carried out up to Mach 0.96. For low supersonic 2-D testing at up to Mach 1.2, a wall adjustment procedure based on wave theory is used to generate a simple 2-D supersonic nozzle in the AWTS, upstream of the model. Since 1978, researchers have used TSWT to build up a substantial database on 2-D testing in AWTS with blockage ratios up to 12% and test section height to model chord ratios down to unity.

In addition, TSWT has been used for 3-D tests with sidewall and sting mounted models with blockage ratios up to 4%. A wall adjustment procedure developed by Goodyer et al. is used for 3-D test up to about Mach 0.9. 3-D tests have been performed at transonic speeds up to Mach 1.2 using wall adjustment procedures still under development. Boundary measurements are static pressures measured along 5 rows on each flexible wall and a single row on one sidewall.

The wall shapes are measured by potentiometers at each wall jack to an accuracy of ±0.127 mm (±0.005 inch). Free stream Mach number is controlled by automatic throttling of the inducing air pressure. Mach number variation up to 0.002 is typical during a test at Mach 0.8. Calibration of TSWT with an empty test section reveals a standard deviation in Mach number variation of about 0.003 at Mach 0.8.

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This 3-D AWTS uses a wall adjustment procedure developed by Rebstock et al. Wall adaptation is possible in 2 iterations at Mach numbers up to 0.95. Model blockage ratios up to 1.3% have been successfully tested, both with lifting and non-lifting sting mounted models. Boundary measurements are static pressures measured along the centerline of each flexible wall. Low supersonic tests of non-lifting bodies indicate that bow shock reflections from the flexible walls can be deflected away from the model.

The Experimental Techniques Branch at TsAGI currently operates the largest AWTS anywhere. The new Russian T-128 tunnel is fitted with a 2.75 m (9 foot) square AWTS which is 8 m (26.25 feet) long. All four walls are perforated. Each wall is made up of 32 segments. The porosity of each segment can be varied between 0 and 18%. Each segment is made up of 2 porous plates (one on top of the other). These 2 plates are moved relative to one another (manually) to achieve a desired porosity over the segment.

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Apparent wall adjustment procedures have been developed by Neyland for 2- and 3-D testing at transonic speeds. Boundary measurements are 5 static pressures measured on each of the 128 quadrilateral wall segments, then an average pressure is found for each segment. The T-128 has 5 interchangeable test sections which are probably configured for either 2- or 3-D testing. The maximum blockage ratios for 2-D testing are 6% and for 3-D testing about 3%. These high blockage ratios are beyond the capabilities of other reported variable porosity AWTS. Unfortunately, no data has been published to substantiate these claims. Nevertheless, the T-128 tunnel is supposed to have been used for production-type testing. Automation of the AWTS is planned in the near future.

5. An Overview of AWTS Designs

In 2-D testing, only two walls need to be adaptable and a simple AWTS is sufficient. The complexity of controlling a 3-D boundary has led to a variety of AWTS designs. Moreover, some approximation in the shape of the test section boundaries is inevitable. The magnitude of this approximation has been the subject of much research. The best number of adaptive walls for a 3-D AWTS is still unknown and must ultimately be a compromise. From
practical considerations, this design compromise is between size/correctability of residual wall interferences (after streamlining), hardware complexity, model accessibility, and the existence of rapid wall adjustment procedures.

There are strong theoretical\(^7\) and experimental\(^1\) indications that the simpler the AWTS design the better the testing technique (see sub-section 6.2). A simple design reduces both the complexity of calculating the residual wall interferences and the complexity of the tunnel hardware, and gives better model access as a bonus. A major factor in the design of new AWTSs will undoubtedly be the trade-off between the complexity of the boundary adjustments and the quality/cost of the residual wall interference corrections. Researchers have made preliminary 3-D tests in 2-D flexible walled AWTSs at NASA Langley\(^{12}\), University of Southampton\(^{14}\), ONERA/CERT\(^\text{1}\), TU-Berlin\(^8\), DLR Goettingen\(^8\), and China.\(^{23}\)

Published data clearly shows that flexible walled AWTSs provide testing capabilities superior to that of variable porosity AWTS designs. We can summarize the effectiveness of flexible walls thus:

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 assessment.

d) Flexible walls improve flow quality providing reduced tunnel interferences and reduced tunnel disturbances which lower operating costs.

e) No plenum is required around the test section.

Interestingly, of the 16 transonic AWTSs now operational worldwide, only 2 AWTSs do not have flexible walls (see Table 1).

The optimum 2-D AWTS has two flexible walls supported by jacks closely grouped in the vicinity of the model. A good example is the AWTS in the 0.3-m TCT shown on Figure 10. The flexible walls (made of thin metal) are anchored at the upstream ends and the downstream ends are attached by a sliding joint to a variable 2-D diffuser. The AWTS requires a square cross-section for optimum 2-D testing (i.e. maximizing Reynolds number capability). For 3-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.

1) Shortening of time attributed to wall streamlining.
2) Detailed examination of AWTS operating envelopes and measurement tolerances for 2- and 3-D testing.
3) Study of different applications.

However, we now know that the variable porosity AWTS is much less effective than the flexible walled AWTS. So, we will only consider the flexible wall research from here on.

Since 1938, researchers have made significant reductions to the time associated with wall streamlining. A major factor in this progress has been the development of rapid wall adjustment procedures for flexible walled AWTSs. (The term rapid refers to minimization of the number of iterations necessary in any wall adaptation procedure.) Early empirical type methods (requiring 8 iterations) have given way to analytical methods (requiring 1 or 2 iterations) as computer support has improved. These analytical methods now use both linear and non-linear theory. Nevertheless, simple empirical methods are still appropriate where the use of large models is not important, as found in some of the AWTSs (particularly the cascade AWTSs) described in this paper.

For 2-D free air simulations, the linear method of Judd, Goodyer, and Wolf? (University of Southampton, UK) is now well established for reasons of speed, accuracy, simplicity (Non-experts can easily use the method on any mini–computer), and adaptability for general use with any flexible walled AWTS. A non-linear version is also available for use in 2-D testing where the flow at the walls is sonic.\(^{18}\) For free air simulations in 3-D testing, researchers use the linear methods of Wedemeyer/Lamarche? (DLR Stuttgart (NASA), and Goodyer et al\(^{119}\) (Southampton). However, all these 3-D methods are still under development. Supersonic 2- and 3-D testing is possible using the method of characteristics (wave theory) to predict the wall shapes necessary to generate supersonic flow.\(^{21}\)

Another time-saving feature of modern AWTSs is the automation of the wall streamlining. Researchers have shown that computer controlled movement of the adaptive walls and automatic acquisition of wall data dramatically reduce the time attributed to wall streamlining from a week to seconds. In addition, researchers have found that fast wall streamlining requires a good practical definition of when the walls are streamlined. We call this definition the streamlining criterion (the point at which we stop wall adaptation). The criterion is directly related to the accuracy of the tunnel/wall measurements (discussed later). For 2-D free air simulations, the best approach appears to be a quantitative approach which is to set, as the streamlining criterion, an acceptable maxima for the residual wall interferences. This approach is used at the University of Southampton and NASA Langley. At present there are only qualitative streamlining criterions in 3-D testing, whereby the walls are streamlined when the model data is unaffected by subsequent iterations of the wall adjustment procedure. On-line residual wall interference codes are available but require development for 3-D testing techniques in AWTSs.\(^{19-24}\)

Researchers have probed the limits of 2-D adaptive wall testing techniques. These limits are related to aerodynamic, theoretical basis and mechanical aspects of the wind tunnel tests. The use of sidewall BLC is only a factor in altering the wall curvature requirements. In 2-D testing, the operating envelope of an AWTS can be assessed from the test section geometry, the wall adjustment procedure and the instrumentation. These are the same factors defined in the design phase of a new AWTS. Researchers have provided many design guidelines to eliminate wall hardware problems, so far encountered, from future AWTS designs. With good design, only theoretical assumptions should restrict the operating envelope for 2-D testing. In 3-D testing, the
situation is far from clear, as no AWTS operating envelopes are well defined. Research has been spread thinly over many AWTS designs and numerous model configurations. The result is that the favoured AWTS design for 3-D testing has gradually become the simplest design (as described earlier).

Researchers have examined the effects of measurement accuracy on AWTS operation, particularly for flexible wall designs. With flexible walls, we can only measure the position of each wall at a finite number of points. The measurement accuracy at each of these points is of the order ±0.127 mm (±0.005 inch) in current AWTSs. The relative position of these measurement points, along each wall, can be optimized for 2-D flexible walled AWTS designs (as shown on Figure 10). Operationally, flexible walled AWTSs have proved tolerant to wall jacks being disconnected due to hardware failures. Interestingly, because the wall position accuracy requirements are proportional to (l/h), the measurement accuracy requirements reduce significantly for a large AWTS. This should be an encouraging factor for potential large AWTS operators. A factor that is already proven in large supersonic nozzle systems operational to-day.

We have found the flexible wall adaptation procedures to be tolerant to uncertainties in the wall pressures. This important feature is due to the smearing effect of the wall boundary layers. However, at high Reynolds numbers (when the wall boundary layers are thin) or with near sonic flow at the adaptive walls, this tolerance to measurement uncertainty is reduced. The uncertainties in the wall pressures can be caused by wall imperfections or fluctuations in the tunnel test conditions. Again, large AWTS should provide more tolerance to these uncertainties. However, we do know that if the model perturbations at the adaptive walls are small (as found in 3-D testing), the accuracy of the wall pressures needs to be better than when the model perturbations are large (as found in 2-D testing).

Furthermore, the allowance necessary for the boundary layer growth on the test section flexible walls is dependent on the accuracy of the wall pressures. In theory, each test condition should require a different boundary layer allowance (i.e. a change in test section cross-sectional area). In practice, researchers have shown that a series of say 4 Aerodynamically Straight wall contours are sufficient to provide uniform Mach number distributions, through an empty AWTS, for Mach numbers up to 0.9. In addition, we do not need to make an allowance for the wall boundary layer thinning due to the presence of the model itself, until the flow on the flexible walls is sonic. Most AWTS operators monitor this boundary layer thinning real-time. Researchers have demonstrated that the adaptive wall testing techniques are tolerant to simple boundary layer allowances. In the 0.3-m TCT, for example, we use approximate Aerodynamically Straight contours which are simply linear divergence contours. This situation is a result of unacceptable wall waviness in the experimentally determined wall contours. The quality of TCT data does not show any problems due to this approximate wall boundary layer allowance.

In reviewing research goals for adaptive walls, there are still many applications yet to be studied. The classical transonic free-air and cascade simulations have received attention in this paper. There is basic research going on with high lift 3-D tests at low speeds at the University of Arizona (see Table 1); swept wing studies and minimum test section height studies in a low speed 2-D AWTS at the University of Southampton (See Table 1); and research at Sverdrup, Tennessee, USA is directed towards use of AWTSs in automotive testing. The 6 simulations possible with AWTSs were first studied experimentally by Goodyer back in 1974/6. However, the adaptive wall research effort has concentrated on free-air and cascade simulations. Although, we now find closed tunnel simulations are proving to be very useful for CFD code validation.

6.1 2-D Testing Experience in AWTSs

Validation data shows that real-time 2-D data from AWTSs is essentially free of top and bottom wall interferences. We have found no problems with testing an aerofoil through stall (no wall shape induced model hysteresis present). Data repeatability from day to day is excellent but, as with any wind tunnel measurements, calibration procedures affect long term repeatability.

We have observed that the model wake in an AWTS shows minimal spanwise variation. We can speculate that the use of large models (relative to the test section size) intrinsically minimizes secondary flows at the aerofoil-sidewall junction. This observation may explain why sidewall BLC does not significantly affect wing performance in a relatively small AWTS. There are strong indications that the flow in an AWTS can be an excellent simulation of a 2-D flow field. If we ever need to use sidewall BLC in an AWTS, then researchers have found that no special testing procedures are necessary.

Researchers have found many limitations to the various 2-D adaptive wall testing techniques, none of which are fundamental. These limitations are associated with wall movement (hardware), model size (theoretical assumptions) and Mach number (theoretical sophistication). Researchers have made 2-D tests close to Mach 1.0, and some limited tests at Mach 1.2. In the supersonic tests, researchers used local wall curvature to remove shock reflections on the model. However, the usefulness of 2-D testing in the supersonic regime is probably only academic, providing experience leading to production-type supersonic 3-D testing.

6.2 3-D Testing Experience in AWTSs

Limited 3-D validation tests support the claim that wall interferences are minimized in AWTSs. However, the wall interferences present before any wall streamlining tend to be already small. So the effectiveness of AWTSs to minimize severe wall interferences in 3-D testing has not been studied.

This situation is due to the low blockage of the 3-D models so far tested in AWTSs. We can increase the model disturbances in the test section by using larger models or testing only at high speeds. Unfortunately, the roughly square cross-section of current AWTSs restricts the size of non-axisymmetric lifting models. Researchers have found that they must use low aspect ratio models to increase the model blockage above the normally accepted value of 0.5 percent. (This is because the model span is limited to about 70 percent of the test section width by wind tunnel users.) Consequently, there is a need for new generation of 3-D AWTSs with a rectangular cross-sections, where the width is greater than the height. We still do not know the maximum model blockage we can successfully test in a 3-D AWTS.

Numerous 3-D AWTS designs have been studied (as discussed earlier). In fact, researchers have spent considerable time and effort to develop a wide range of complex 3-D AWTS designs, when it now appears the simpler 2-D design may well be adequate. (In hindsight, this effort appears unnecessary but the contribution to overall knowledge is nevertheless important.) An example of the promise of simple AWTSs in 3-D testing is shown on Figure 11. Data from residual interference codes are presented as contour plots of blockage and upwash wall interferences on a simple cropped delta wing, mounted on a sidewall of the 2-D Southampton TSWT. Notice on Figure 11a how the blockage interference patterns, with straight walls, are normal to the
flow and 2-D in nature. We can see 2-D wall streamlining significantly reduces the blockage interference. On Figure 11b, the upwash interference pattern with the walls straight still exhibits some two-dimensionality and again 2-D wall streamlining significantly reduces the upwash.

**University of Southampton TSWT 3-D Data**

Cropped Delta Wing - Mach 0.7; Alpha = 8°

Span/Width = 57%; Nominal Blockage = 2%

<table>
<thead>
<tr>
<th>Walls Straight</th>
<th>Walls Streamlined</th>
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<tbody>
<tr>
<td><img src="image1.png" alt="Image" /></td>
<td><img src="image2.png" alt="Image" /></td>
</tr>
</tbody>
</table>

**Fig. 11a - Effect of 2-D wall streamlining on blockage interference in a 3-D test.**

<table>
<thead>
<tr>
<th>Walls Straight</th>
<th>Walls Streamlined</th>
</tr>
</thead>
<tbody>
<tr>
<td><img src="image3.png" alt="Image" /></td>
<td><img src="image4.png" alt="Image" /></td>
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**Fig. 11b - Effect of 2-D wall streamlining on upwash interference in a 3-D test.**

We have not found any fundamental limits to Mach number when using AWTSs in 3-D testing. Preliminary tests at low supersonic speeds show we can bend the AWTS's flexible walls to eliminate oblique shock reflections onto the model, as found in 2-D testing. The smearing of the shock/wall interaction does much to ease the curvature requirements on the flexible walls. However, in supersonic testing, there is no clear indication of the quality of the model data after wall streamlining, nor is there yet a proven wall adjustment procedure.

The wall adjustment procedures for 3-D testing have taken advantage of the fast and large capacity minicomputers available for real-time 3-D flow computations. Research continues to identify the amount and type of wall interferences that can be successfully "corrected" by 3-D adaptive wall testing techniques. The different wall adjustment procedures minimize the 3-D wall interferences differently. For example, the Rebstock method minimizes interferences along a pre-set streamwise line anywhere in the test section. In addition, the Rebstock method minimizes wall curvature by introducing a uniform angle of attack error throughout the test section. We do not know where best to minimize the wall interferences for different model configurations nor do we know where the concept of a uniform angle of attack error will break down.

The type of wall pressure measurements necessary to adequately assess the residual wall interferences is also an unknown. The exploitation of real-time residual interference assessment codes is now critical to progress in 3-D adaptive wall testing techniques. This has come about because we now realize that 3-D wall interferences cannot be eliminated with even the most sophisticated AWTS.

Hardware limitations currently restrict AWTS test envelopes (in particular model lift) for reported 3-D tests. These hardware limitations arose from inappropriate AWTS design criteria and the use of AWTSs originally designed for only 2-D testing. Unfortunately, these limitations have hampered 3-D adaptive wall research. This situation would appear to be one of the outcomes of low priority funding.

**7. Production Requirements**

The production requirements for an adaptive wall testing technique is the same as for any modern testing technique. Firstly, the technique must be easy to use. Consequently, we need to make the complexities of the AWTS invisible to the tunnel operators (similar to operating large flexible supersonic nozzles). Secondly, the technique must not require excessive tunnel time. So we require the AWTS wall movements to be quick. Thirdly, the technique must have a known test envelope for successful use. Therefore, we must ensure the testing technique is well researched, so that we know the limitations and restrictions and can avoid them during normal operations. Fourthly, the technique must, of course, be financially viable.

How can the adaptive wall testing technique meet the production requirements shown above? First, lets consider the complexity of an adaptive wall testing technique. We must design the associated test section hardware so the wall shapes can be continually changed. We also need an interaction between the AWTS and a computer system to set the wall to streamline shapes. If we make the AWTS of simple design, then access to the model is unaffected. Furthermore, if we make the wall adjustments automatic via a user-friendly computer system, the operator need only issue Go/Stop commands (See Figure 12). Consequently, the complexity of the testing technique is invisible to the operator. The tunnel operator's contact with the AWTS becomes simply to setup the model and acquire test data.

![Fig. 12 - A schematic diagram of the interaction between AWTS, operator, and computer system for automatic control of an AWTS.](image5.png)

Second, lets consider the time factor. Adjusting walls in the test section takes time. How much time depends on the AWTS hardware (jack type) and the wall adjustment...
procedure. We can design the wall jacks to be very responsive. The wall adjustment procedure can find the streamline shapes in one or two iterations. The result is that wall streamlining can be quick. The French have already demonstrated wall streamlining in 10 seconds for 2-D testing. Computer advances will make this possible for 3-D testing in the future. Another time factor is the elimination of post-test corrections and lengthy test programmes, because real-time AWTS data is the final data. We show the importance of this fact on Figure 13. In this example, we compare real-time transonic 2-D lift data from a deep slotted walled test section with equivalent real-time data from a shallow flexible walled AWTS, at the same test conditions. The differences are alarming. With the final data known during the tunnel run, AWTSs can and should save overall tunnel run time.

**NASA Langley 0.3-m TCT Aerofoil Data**

Mach 0.765; Transition Fixed

![Graph](image)

**Fig. 13 - The importance of corrected real-time transonic 2-D aerofoil data from a shallow flexible walled AWTS.**

Third, let's consider the test envelopes for AWTSs. Researchers have defined the test envelope for various 2-D adaptive wall testing techniques (described earlier). So we can direct non-expert users away from these known limitations. Alas, in 3-D testing, we are still learning what limitations exist.

Fourth, consider the cost factor. The simple AWTS design can be incorporated in existing test sections by the replacement of only two walls. Also, the plenum, which surrounds ventilated transonic test sections, can provide adequate volume, within the pressure vessel, for the jack mechanisms. These factors will reduce the overall hardware costs. In addition, the AWTS control system requires the same computer/tunnel interface found with other tunnel features such as a motorized sting or speed controls. The era of cheap data acquisition systems based around powerful PC type computers means that the AWTS control system should be relatively inexpensive. In addition, an AWTS control system can be integrated with other tunnel systems, which do not need to operate at the same time as wall streamlining. Other favourable cost factors are the reduction of tunnel operating costs possible by using a smaller AWTS (as much as 75% smaller than the original) and having smooth walls.

Interestingly of the three wind tunnels with AWTSs which come closest to being production-type tunnels, non-experts can only use one. The Langley 0.3-m TCT has the only User Friendly AWTS control system that allows non-expert 2-D testing within defined test envelopes.

### 8. The Future of AWTSs?

The status of adaptive wall technology is ongoing and positive. The vast 2-D testing experience will continue to be very important to the development of 3-D adaptive wall testing techniques. Six research groups around the world are carefully pursuing the development of 3-D adaptive wall testing techniques. Work to find the best techniques to achieve specific test objectives at transonic speeds will also demonstrate all the AWTS advantages in 3-D testing. I speculate that only after this action will misconceptions, in the wind tunnel community, be dispelled leaving the way clear for adaptive wall technology to be properly utilized.

At the time of writing, a 30 cm (11.8 inches) square high speed tunnel in the Northwestern Polytechnical University, Xian, China is being fitted with a flexible walled AWTS. Also, a flexible walled AWTS is being built at DLR Goettingen, West Germany for transonic cascade testing with as few as a single blade installed. Another transonic cascade tunnel with a flexible walled test section is planned at the University of Genoa, Italy. This AWTS will have a 3 blade cascade. There is also a strong possibility of unreported adaptive wall activity in Russia where transonic boundary layer transition is receiving much attention.

This news shows there is still interest in improving our testing techniques. If production testing is the ultimate goal, then we have finished developing 2-D adaptive wall testing techniques for free-air simulations. However, work must continue to dispel the inevitable misconceptions about AWTS complexity. In 3-D testing, we still have test envelopes to define and testing techniques to optimize.

We can summarize the current status as the development of a "new" technology to a point where this technology could be made very useful to the aerodynamicist (both theoretician and experimentalist) given the right priority. I am certain that if adaptive wall research had been given similar priority to the development of transonic "correction codes", we would have a production 3-D adaptive wall testing technique available right now. Today, most, if not all, wind tunnel designers make allowances in their designs for that AWTS which will be fitted into their new wind tunnel someday. This situation demonstrates again that wind tunnel users agree there is a need for better testing techniques.

Now that the expectations of CFD have become more realistic, 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 (wind tunnel and computer) to improve our understanding of aerodynamics in the future. Perfection can only be achieved by making full use of all advanced technologies available to us.

### 9. Conclusions

1. Adaptive wall testing techniques, particularly those which utilize flexible walls, offer major advantages over conventional techniques in transonic testing.

2. We can significantly improve data quality by using adaptive wall technology available to us now.

3. Computer advances have removed any impractical aspects of adaptive wall technology.

4. Non-expert use of AWTSs for routine 2-D testing has been demonstrated.

5. We can now design an AWTS so there are no hardware restrictions to the operating envelope.
6. In 2-D testing, adaptive wall testing techniques are well proven and are already in use for production-type transonic testing in cryogenic wind tunnels.

7. Adaptive wall technology offers significant potential in 3-D testing which has yet to be fully demonstrated.

8. General acceptance of adaptive wall technology now relies on the development of testing techniques for general 3-D transonic testing.

References
