NASA Contractor Report 4294

Supersonic Wind Tunnel Nozzles

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

Cooperative Agreement NCC2-604

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TO AID IN THE DEVELOPMENT OF QUIET WIND
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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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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 the 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:

Acquisition Number	Type of Material	Where Obtained
A??-????? (Example: A88-12345)	AIAA papers and worldwide published literature from conferences and periodicals available from AIAA.	Technical Information Service American Institute of Aeronautics and Astronautics (AIAA) 555 West 57th Street, 12th Floor New York, NY 10019 USA
N??-????? (Example: N77-12345)	Report literature with no distribution limitations.	National Technical Information Service (NTIS) 5285 Royal Port Road Springfield, VA 22161 USA
X??-????? (Example: X66-12345)	Report literature with distribution limitation of some type.	NASA Scientific and Technical Information Facility (STIF) PO Box 8757 B.W.I. Airport, MD 21240 USA
AD-?????? ATI-?????? (Example: AD-123456)	Report literature which has been declassified	Defense Technical Information Center (DTIC) Defense Logistics Agency Cameron Station Alexandria, VA 22304 USA
N-??????	Pre-1962 reports and papers.	NASA Libraries
(Example: TL 123 C66)	Books	Libraries

Please note that an "#" after and acquisition number (for example A66-12345#) indicates that the document is available in microfiche form for your convenience.

Bibliography

1. *Stanton, T.E.: **On the Flow of Gases at High Speeds.** Proceedings of the Royal Society - Series A, vol. 111, July 1926, pp. 306-339.

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.

2. *Stanton, T.E.; **High-Speed Wind Channel for Tests on Airfoils.** ARC R&M 1130, 1928.

*Engineering Department, National Physical Laboratory, UK.

3. *Briggs, L.J.; and Dryden, H.L.: **Aerodynamic Characteristics of Twenty-Four Airfoils at High Speeds.** NACA Report 319, 1929.

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

4. Prandtl, L.; and Busemann, A.: **Approximate Method of Graphically Solving Problems of Two-Dimensional Supersonic Flows.** Stodola Festschrift, 1929, pp. 499-509. In German.

5. *Stanton, T.E.: **The Variation of Velocity in the Neighbourhood of the Throat of a Constriction in a Wind Tunnel.** ARC Report N1388, 1931.

*Engineering Department, National Physical Laboratory, UK.

6. Busemann, A.: **Orifices, Nozzles and Diffusers.** In: Handbook of Experimental Physics, vol. 4 - Gas Dynamics - Part 3, 1931, pp. 383-407, 26 refs. In German. English translation is RTP Translation no. 2209 by the British Ministry of Aircraft Production.

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(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.

7. Karner, L.; and Ackeret, J.: **Aeronautical Education and Research at the Swiss Institute of Technology in Zurich.** NACA TM-616, 1931.

8. *Bailey, A.; and Wood, S.A.: **High Speed Induced Wind Tunnel.** British ARC R&M 1468, May 1932.

*National Physical Laboratory, Teddington, UK.

9. *Stack, J.: **The NACA High-Speed Wind Tunnel and Tests of Six Propeller Sections.** NACA Report 463, 1933.

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

10. Frankl, F.I.: **Supersonic Flows with Axial Symmetry.** Izvestia Art, Akademii RKKA, vol. 1, 1934. In Russian.

11. *Santon, L.: **Experimental Contribution to the Study of Supersonic Wind Tunnels.** Publication Scientifiques et Techniques du Ministere de l'Air Publication no. 68, 1935. In French. English translation is RTP Translation no. 1942 by the British Ministry of Aircraft Production, 60 pp, 15 refs.

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(Ames Research Center library no.)

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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.

12. Crocco, L.: **High Speed Wind Tunnels.** In: L'Aerotecnica, 1935. In Italian. English translation is USAAC, Material Command, AMC Translation no. 366, July 1943.

13. Idelchik, I.E.: **Investigation of the Nozzle of Wind Tunnels.** TsAGI Report no. 80, 1935. In Russian.

14. Ackeret, J.: **High Speed Wind Tunnels.** Presented at the 5th Convention of the Volta Congress, Italy, September 30 - October 6, 1935. In German. English translation is NACA TM-808, November 1936, 47 pp.

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.

15. Ferri, A.: **Investigations and Experiments in the Guidonia Supersonic Wind Tunnel.** NACA TM-901, 1939.

16. Kurzweg, H.H.: **Report on the Supersonic Wind Tunnels of the Army Research Station at Peenemunde.** Peenemunde Archives Report no. 66/11, June 1939, 52 pp. In German. English translation is DTIC ATI-1812.

17. *Kisenko, M.S.: **Comparative Results of Tests on Several Different Types of Nozzles.** TsAGI Report no. 478, 1940. In Russian. English translation is NACA TM-1066, June 1944, 44 pp, 4 refs.

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 assured 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 25^0 to 27^0. 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.

18. *Lehnert, R.: **Report on Three Component Measurements of a Model of A-4 in the Supersonic Wind Tunnel of the Army Research**

Station Peenemunde. Peenemunde Archives Report no. 66/33, October 1940. 37 pp. In German. English translation published by Office of Naval Intelligence, Technical Intelligence Center, 1947.

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(Ames Research Center library no.)

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 provides 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

19. Hermann, R.: **The Supersonic Wind Tunnel of the Heerswaffenamt and Its Application in External Ballistics.** Peenemunde Archives Report nos. 66/0 and 66/11, December 1940, 54 pp. In German. English translation is DTIC ATI-1801, June 1945.

The supersonic tunnel of the Heerswaffenamt was constructed to facilitate the development of spin- and fin-stabalized 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.

20. Busemann, A.: **Laval Nozzles for Uniform Supersonic Flow.** Zeltschrift VDI, vol. 84, no. 45, November 1940, pp. 857-862. In German. English translations are RTP Translation no. 2511, by the British Ministry of Aircraft Production, and NACA TT-152, June 1941 (Langley Research Center library no. 6410 907).

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 curly 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

21. *Heybey, W.H.: **The Prandtl-Busemann Method for the Graphical Evaluation of 2-D Super. Flow. Its Application to the Design of Wind Tunnel Nozzles and the Appropriate Corrections. Parts I, II, and III.** Peenemunde Archives Report no. 66/56, February 1941, 112 pp, 8 refs. In German. English translation by Cornell Aeronautical Laboratories is DTIC ATI-1855, August 1946. An English translation of Part III only (Peenemunde Archives Report 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.

22. *Berthold: **Research on the Design of a Continuous-Operation Wind Tunnel for Subsonic and Supersonic Velocities with Maximum Possible Efficiency**. Peenemunde Archives Report no. 66/41, May 1941, 28 pp. In German. English translation by Cornell Aeronautical Laboratory is included with original report in DTIC ATI-1841, April 1946.

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.

23. Seippel, C.: **Supersonic Wind Tunnels.** The Brown Boveri Review, vol. 30, no. 7/8, August 1943, pp. 178-183, 1 ref.

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.

24. Green, J.R.; and Southwell, R.V.: **Relaxation Methods Applied to Engineering Problems. Part IX. High-Speed Flow of Compressible Fluid Through a Two-Dimensional Nozzle.** Philosophical Transactions of the Royal Society of London - Series A. Mathematical and Physical Sciences, vol. 239, no. 808, April 1944, pp. 367-386, 8 refs.

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.

25. *Freidrichs, K.O.: **Theoretical Studies on the Flow Through Nozzles and Related Problems - Applied Mathematics Panel 82.1R.** New York University AMG-NYU report no. 43, April 1944.

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

26. *Shapiro, A.H.: **Nozzles for Supersonic Flow Without Shock Fronts.** Presented at the ASME Annual Meeting, November 29 - December 3, 1943. Journal of Applied Mechanics, vol. 11, no. 2, June 1944, pp. A93-A100, 7 refs.

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 fro 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.

27. *Puckett, A.E.: **Final report on the Model Supersonic Wind Tunnel Project.** NDRC-AO-A-269, July 1944, 120 pp, 17 refs.

5341 CIT/15
(Ames Research Center library no.)

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.

28. Fox, L.; and Southwell, R.V.: **On the Flow of Gas Through a Nozzle with Velocities Exceeding the Speed of Sound.** Proceedings of the Royal Society - Series A, vol. 183, no. 992, August 1944, pp. 38-54, 5 refs.

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.

29. *Freidrichs, K.O.: **On Supersonic Compressors and Nozzles. Applied Mathematics Panel report 82.2R.** New York University AMG-NYU Report no. 77, October 1944.

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

30. *Stack, J.: **Compressible Flows in Aeronautics.** The 8th Wright Brothers Lecture, December 1944. In: Journal of the Aeronautical Sciences, vol. 12, no. 2, April 1945, pp. 127-148, 34 refs.

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.

31. Lukasiewicz, J.: **Design and Calibration Tests of a 5.5 inch Square Supersonic Wind Tunnel.** ARC R&M 2745, 1945.

32. *Eber, G.R.: **The Supersonic Wind Tunnel of Wasserbau-Versuchsanstalt Kochelsee.** Peenemunde Archives Report no. 66/182, June 1945, 57 pp. In German.

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.

33. *Hermann, R.: **The Efficiency of the Supersonic Wind Tunnel Installation of Wasserbau-Versuchsanstalt Kochelsee.** Peenemunde Archives Report no. 66/183, June 1945, 18 pp. In German.

DTIC ATI-1963

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

34. *Atkin, A.O.L.: **Two-Dimensional Supersonic Channel Design - Part I.** ARC 9194, November 1945, 12 pp, 3 refs. Also: ARC R&M 2174, 1946.

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.

35. *Eaton, A.R.; and Bradfield, W.S.: **Report on Calibration Methods for Supersonic Wind Tunnel Nozzles M=1.25, 1.50, 1.75, 2.00, 2.25, and 2.50.** John Hopkins University APL-JHU/CM-144, November 1945, 33 pp.

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.

36. *Pinkel, I.I.; and Shames, H.: **Supersonic Nozzle Design Procedure.** Presented at NACA/AERL Advanced Topics in Supersonic Flow Seminar. Lecture no. 7, 1946, 7 pp.

5341.1 41
(Langley Research Center library no.)

This paper presents a solution to the problem of transforming a parallel one-dimensional flow of sonic velocity into a parallel one-dimensional flow of supersonic velocity. The method is applicable to two-dimensional asymmetric nozzle designs.

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

37. *Ferri, A.: **Application of the Method of Characteristics to Supersonic Rotational Flow.** NACA Report 841, 1946, 12 pp, 12 refs.

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.

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

38. Frossel, W.: **Experimental Investigation of the Compressible Flow At and Near a Cambered Wall - Part IV Nozzle Flow.** January 1946, 33 pp. In German. English translation is DTIC ATI-102029.

DTIC ATI-102029

39. *Fejer, A.A.: **Report on Discussion with German Engineers and Scientists - Supersonic Air Flow Research.** PMR-359-0-2, May 1946, 4 pp.

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.

40. *Foelsch, K.: **A New Method of Designing Two-Dimensional Laval Nozzles for a Parallel and Uniform Jet.** NA-46-235-2, May 1946, 50 pp, 4 refs.

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.

41. *Emmons, H.W.: **The Theoretical Flow of a Frictionless Adiabatic Perfect Gas Inside of a Two Dimensional Hyperbolic Nozzle.** NACA TN-1003, May 1946, 54 pp, 10 refs.

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.

43. *Smelt, R.: **A Critical Review of German Research on High-Speed Airflow.** The 696th Royal Aeronautical Society Lecture, October 1946. In: The Royal Aeronautical Society Journal, vol. 50, 1946, pp. 899-934.

This lecture examines the German work on the compressibility phenomena intensified by the need to develop high-speed fighters and jet propulsion. The appearance of the V2 in England was evidence of a considerable background of knowledge of supersonic aerodynamics. This review concentrates on the highlights of the German work, in particular those aspects which have given strong confirmation of our own ideas or have modified greatly our line of attack on the compressibility problem. German high-speed wind tunnels are described. In supersonic research, the German contribution to sweepback and low aspect ratio wings is discussed. It is concluded that German progress was hampered by a lack of cooperation between aircraft manufacturers and the research organizations. Consequently, the Germans have given the British research worker a whole series of suggestions for high-speed aircraft which are almost entirely untried or inadequately tried in flight.

*Gas Dynamics Department, Royal Aircraft Establishment, UK.

44. *Puckett, A.E.: **Supersonic Nozzle Design.** Journal of Applied Mechanics, vol. 13, no. 4, December 1946, pp. A265-A270, 8 refs.

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.

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

45. *Heybey, W.H.: **Nozzle Design and Correction for Two Dimensional Supersonic Flow.** NOL Memo 9132, 1947.

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

46. *Liepmann, H.W.; and Puckett, A.E.: **Introduction to Aerodynamics of a Compressible Fluid. Chapter 13 - The Method of Characteristics.** Published by John Wiley and Sons Inc., 1947, pp. 210-232.

TL 573 L5
(Ames Research Center library no.)

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.

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

47. Sauer, R.: **Method of Characteristics for Three-Dimensional Axially Symmetric Supersonic Flows.** English translation of unpublished August 1940 German paper. NACA TM-1133, January 1947, 36 pp, 6 refs.

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.

48. *Stram, L.: **Shock Free Supersonic Nozzle Design.** CAEC Memo Report no. 2004, February 1947, 12 pp, 6 refs.

5391 14
(Langley Research Center library no.)

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.

*Continental Aviation and Engineering Corp., Detroit, Michigan, USA.

49. *Shapiro, A.H.; and Edelman, G.M.: **Method of Characteristics for Two Dimensional Supersonic Flow - Graphical and Numerical Procedures.** MIT Meteor Report no. 7, February 1947.

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

50. *Cronvich, L.L.: **Numerical-Graphical Methods of Characteristics for Plane Potential Shock-Free Flow Problems.** Journal of the Aeronautical Sciences, vol. 14, no. 4, April 1947, pp. 237-242, 6 refs.

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.

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

51. *Shapiro, A.H.; and Edelman, G.M.: **Method of Characteristics for Two Dimensional Supersonic Flow - Graphical and Numerical Procedures.** Journal of Applied Mechanics, vol. 14, no. 2, June 1947, pp. A154-A162, 6 refs.

The method of characteristics for two-dimensional supersonic flow is reviewed and summarized. A characteristics chart is presented for use with the graphical procedure of Prandtl and Busemann. A new numerical procedure is described which eliminates graphical operations and which allows more accurate solutions. To facilitate the numerical method, a table of useful functions is included.

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

52. Sauer, R.: **General Characteristics of the Flow Through Nozzles at Near Critical Speeds.** English translation of German paper presented in 1944. NACA TM-1147, June 1947, 20 pp, 5 refs.

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.

53. *Nilson, E.N.: **The Method of Characteristics for Three-Dimensional Axially Symmetric Supersonic Flow.** UACRD M-2101-1, September 1947.

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

54. Royle, J.K.; Bowling, A.G.; and Lukasiewicz, J.: **Calibration of Two Dimensional and Conical Supersonic Multi-Nozzles.** RAE Aero 2221 SD 23, September 1947.

55. *Nilson, E.N.: **Design of an Inlet for a Two-Dimensional Supersonic Nozzle.** Meteor Report UAC-13, December 1947, 25 pp, 13 refs.

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.

56. *Allen, H.J.: **The Asymmetrical Adjustable Supersonic Nozzle for Wind-Tunnel Applications.** NACA RM-A8E17, 1948.

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

57. Hawthorne, W.R.; et al: **Analysis of Various Systems for Supersonic Wind Tunnels.** MIT Meteor R UAC-23, 1948.

58. Nilson, E.N.: **Analytical Correction of a Two-Dimensional Nozzle for Uniform Exhaust Flow.** MIT Meteor R UAC-15, 1948

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

59. *Peterson, J.W.: **Nozzle Design for the Supersonic Wind Tunnel.** University of Michigan Report EMP-7, 1948.

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

60. *Bogdonoff, S.M.: **The Princeton Pilot Variable Density Supersonic Wind Tunnel. Project Squid.** Princeton University TM PR-8, 1948.

*Princeton University, Princeton, New Jersey, USA.

61. *Courant, R.; and Freidrichs, K.O.: **Supersonic Flow and Shock Waves. Chapter 5 - Flow in Nozzles and Jets.** Published by Interscience Publishers, New York, 1948, pp. 377-396, 11 refs.

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.

62. *Pinkel, I.I.: **Equations for the Design of Two-Dimensional Supersonic Nozzles.** NACA Report 907, 1948, 24 pp, 2 refs.

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.

63. *Cronvich, L.L.: **A Numerical-Graphical Methods of Characteristics for Axially Symmetric Isentropic Flow.** Journal of the Aeronautical Sciences, vol. 15, no. 3, March 1948, pp. 155-162, 11 refs.

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.

64. Edelman, G.M.: **The Design, Development and Testing of Two-Dimensional Sharp Cornered Supersonic Nozzles.** MIT Meteor Report 22, May 1948.

65. *Meyer, R.E.: **The Method of Characteristics for Problems of Compressible Flow Involving Two Independent Variables. Part 1 - The General Theory.** Quarterly Journal of Mechanics and Applied Mathematics, vol. 1, no. 2, June 1948, pp. 196-219, 14 refs.

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.

66. *Crown, J.C.: **Supersonic Nozzle Design.** NACA TN-1651, June 1948, 34 pp, 10 refs.

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.

67. *Kane, E.D.; Chambre, P.L.; and Schaaf, S.A.: **Supersonic Nozzle Designs for Low Pressure Gas Flows.** HE-150-48, July 1948, 15 pp.

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.

68. *Puckett, A.E.: **Design and Operation of a 12-Inch Supersonic Wind Tunnel.** Presented at the Annual Summer Meeting of Institute of the Aeronautical Sciences, July 1948. Preprint #160 published by IAS, 30 pp.

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.

69. *Kurzweg, H.H.: **The Use of Grating Nozzles and Diffusers.** NOL Memo 9848, July 1948.

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

70. *Holt, M.: **The Behaviour of the Velocity Along a Straight Characteristic in Steady Irrotational Isentropic Flow with Axial Symmetry.** Quarterly Journal of Mechanics and Applied Mathematics, vol. 1, part 3, September 1948, pp. 358-364, 1 ref.

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 long 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.

71. *Tsien, H.S.: **Wind-Tunnel Testing Problems in Superaerodynamics.** Journal of the Aeronautical Sciences, vol. 15, no. 10, October 1948, pp. 573-580, 9 refs.

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.

72. *Evvard, J.C.; and Wyatt, D.D.: **Investigation of a Variable Mach Number Supersonic Tunnel with Nonintersecting Characteristics.** NACA RM-E8J13, November 1948, 28 pp, 3 refs.

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.

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

73. *Meyer, R.E.: **The Method of Characteristics for Problems of Compressible Flow Involving Two Independent Variables. Part 2 - Integration Along a Mach Line. The Radial Focusing Effect in Axially Symmetrical Flow.** Quarterly Journal of Mechanics and Applied Mathematics, vol. 1, part 4, December 1948, pp. 451-469, 7 refs.

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.

74. Boden, O.W.: **Supersonic Wind Tunnel.** NA Report no. 14, C/49, 1949.

75. Hulbert, J.K.; Sutton, M.A.; and Von Eschen, G.L: **An Experimental and Analytical Investigation of Critical Design Parameters for High Mach Number Open-Jet Supersonic Wind Tunnel.** Ohio State University P-344 R-9, 1949.

76. Drougge, G.: **A Method for Continuous Variation of the Mach Number in a Supersonic Wind Tunnel and Some Experimental Results Obtained at Low Supersonic Speeds.**, Technical Research Institute of Sweden TRIS Report 29, 1949.

77. Kane, E.D.; and Emerson, D.E.: **Construction and Operation of a Small Low Density Supersonic Wind Tunnel.** University of California Report HE-150-57, 1949.

78. *Monroe, L.L.: **Nozzle Contours for a Flexible-Wall Supersonic Nozzle - Theoretical and Experimental Investigation.** CSAS Thesis, 1949, 86 pp.

5341.1 48
(Ames Research Center library no.)

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

79. *Kurzweg, H.H.: **A Few Aspects of Future Supersonic Wind-Tunnel Design and Test Techniques.** NOL Report no. 1133, 1949, pp. 103-119.

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.

80. *Ferri, A.: **Elements of Aerodynamics of Supersonic Flows. Chapter 8 - Two Dimensional Effusers.** Published by The MacMillan Company, 1949, pp. 161-178, 74 refs.

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.

81. *Gompf, G.F.: **Supersonic Nozzle Design for Viscous Fluids.** GALCIT Thesis 10, 1949.

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

82. *Holder, D.W.; and North, R.J.: **The 9 x 3 inch National Physical Laboratory Induced Flow High Speed Wind Tunnel.** ARC 12387, 1949.

*National Physical Laboratory, Teddington, UK.

83. *Foelsch, K.: **The Analytical Design of an Axially Symmetric Laval Nozzle for a Parallel and Uniform Jet.** Journal of the Aeronautical Sciences, vol. 16, no. 3, March 1949, pp. 161-166 and 188, 6 refs.

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 1, the use of which shortens the calculations considerably.

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

84. Oswatitsch, K.; and Rothstein, W.: **Flow Pattern in a Converging-Diverging Nozzle.** English translation of unpublished 1942 paper. NACA TM-1215, March 1949, 42 pp, 4 refs.

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.

85. *Bershader, D.: **An Interferometric Study of Supersonic Channel Flow.** Review of Scientific Instruments, vol. 20, no. 4, April 1949, pp. 260-275, 10 refs.

The Zehnder-Mach interferometer has been applied to the study of supersonic flow of air expanding from a pressure tank into the atmosphere through two channels of rectangular cross-section. Both channels were about 1 cm in width. One was a divergent channel (half angle = 4.0^o) and the other a Laval nozzle constructed by the "characteristics" method for M=1.7. It was found possible to measure the density distribution with a rather small estimated probable error (2%). Knowledge of the stagnation temperature and pressure made it possible also to compute pressure, temperature, and velocity. The report takes up, in some detail, accuracy problems in supersonic nozzles. Results obtained for the potential flow agree, for the most part, within one percent with the well-known theory of isentropic channel flow.

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

86. *Schaaf, S.A.: **An Axially Symmetric Nozzle with Boundary Layer Correction.** UC HE-150-58, June 1949, 13 pp, 7 refs.

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

87. *Henderson, H.W.: **NOL 18 x 18cm Supersonic Wind Tunnel - Design Criteria, Description, Operating Technique.** NOL Memo 10379, August 1949, 42 pp, 1 ref.

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.

88. *Culbertson, P.E: **Calibration Report on the University of Michigan Supersonic Wind Tunnel.** University of Michigan UMM-36, November 1949, 39 pp, 10 refs.

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.

89. *Bonney, E.A.: **Engineering Supersonic Aerodynamics. Chapter 2 - One-Dimensional Flow - Nozzles and Diffusers.** Published by McGraw-Hill Book Company Inc., 1950, pp. 33-70, 4 refs.

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 to show the effect of back pressure on flows through a De Laval nozzle

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

90. Emerson, D.E.; and Schaaf, S.A.: **Performance of a Supersonic Nozzle in the Rarefied Gas Dynamics Regime.** University of California Report HE-150-72, 1950.

91. *Pope, A.: **Aerodynamics of Supersonic Flight. Chapter 6 - Supersonic Wind Tunnels.** Published by Pitman Publishing Corp., 1950, pp. 104-137, 3 refs.

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.

92. *Garby, L.C.; and Nelson, W.C.: **University of Michigan 8 x 13 inch Intermittent Flow Supersonic Wind Tunnel.** University of Michigan UMM-59, 1950.

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

93. Miller, L.T.: **The Supersonic Wind Tunnel.** John Hopkins University Report 44-11-02, 1950.

94. *Frost, R.C.: **Nozzle Design Considerations.** University of Michigan Report WTM-188, 1950.

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

95. *Harrop, R.; Bright, P.I.F.; and Caiger, M.T.: **The Design and Testing of Supersonic Nozzles.** ARC R&M 2712, May 1950, 40 pp, 10 refs.

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 it is thought will need further consideration.

*Royal Aircraft Establishment, UK.

96. *Johannesen, N.H; and Meyer, R.E.: **Axially Symmetrical Supersonic Flows Near the Center of an Expansion.** The Aeronautical Quarterly, vol. 2, August 1950, pp. 127-142, 7 refs.

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**

Physical Laboratory. The 800th Royal Aeronautical Society Lecture. In: The Journal of the Royal Aeronautical Society, vol. 54, September 1950, pp. 545-586, 63 refs.

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.

98. *Lindsey, W.F.; and Chew, W.L.: **The Development and Performance of Two Small Tunnels Capable of Intermittent Operation at Mach Numbers Between 0.4 and 4.0.** NACA TN-2189, September 1950, 90 pp, 27 refs.

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.

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

99. *Brinich, P.F.: **Boundary Layer Measurements in 3.84-by-10 inch Supersonic Channel.** NACA TN-2203, October 1950, 51 pp, 10 refs.

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.

100. *Meyer, R.E.; and Holt, M.: **The Correction of Wind Tunnel Nozzles for Two-Dimensional, Supersonic Flow.** The Aeronautical Quarterly, vol. 2, November 1950, pp. 195-208, 6 refs.

The paper is concerned with the two-dimensional, steady, irrotational, iscentropic 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.

101. *Cohen, C.B.; and Valerino, A.S.: **Investigation of Operating Pressure Ratio of a Supersonic Wind Tunnel Utilizing Distributed Boundary-Layer Suction in Test Section.** NACA RM E50H04, November 1950, 39 pp, 1 ref.

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.

102. *Hilton, W.F.: **High Speed Aerodynamics: Part III - Wind Tunnels.** Published by Longmans, Green and Co., London, 1951, pp. 359-477, 2 refs.

TL 570 H55
(Ames Research Center library no.)

Part III of this book is split into two sections. The first deals with high-speed wind tunnels including discussion of design, effects of humidity, and the physical principles underlying the operation of supersonic tunnels. 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

103. *McKenney, J.D.: **An Investigation of Flow Separation in a Two-Dimensional Transparent Nozzle.** JPL PR 20-129, 1951.

*Jet Propulsion Laboratory, Pasadena, California, USA.

104. *Bogdonoff, S.M.: **Design, Construction and Operation of the Princeton 4 x 8 inch Variable Density Supersonic Tunnel. Project Squid.** Princeton University PS TR-34, 1951.

*Princeton University, Princeton, New Jersey, USA.

105. *Dhawan, S.; and Roshko, A.: **A Flexible Nozzle for a Small Supersonic Wind Tunnel.** Journal of the Aeronautical Sciences, vol. 18, no. 4, April 1951, pp. 253-258, 1 ref.

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.

106. Cunsolo, D.: **Calculation of the Shape of a Two-Dimensional Supersonic Nozzle in Closed Form.** L'Aerotecnica, vol. 31, no. 4, August 1951. In Italian. English translation is NACA TM-1358, January 1953, 29 pp.

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."

107. *Armstrong, A.H.; and Smith, M.G.: **Two-Dimensional Supersonic Nozzle Design Part I: Theory.** Armament Research Establishment Report 5/51, October 1951, 36 pp.

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.

108. *Temple, E.B.: **The Effect of Small Variations in the Nozzle Contour on the Flow in the Test Section of a Supersonic Nozzle.** NSLM A&R 100-81, December 1951.

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

109. *Clippinger, R.F.: **Supersonic Axially Symmetric Nozzles.** BRL Report 794, December 1951.

*Ballistic Research Laboratory, Aberdeen Proving Ground, Maryland, USA.

110. Drougge, G.: **Some Measurements at Low Supersonic Speeds by a Method for Continuous Variation of the Mach Number.** Aeronautical Research Institute of Sweden ARIS Report 42, 1952.

111. Serruys, M.: **General Study of the Flow of Gas Through a Supersonic Nozzle (In the Steady or Unsteady State) with Possible Heat Addition.** Ministere de l'Air Publications Scientifiques et Techniques (PST) no. 272, 1952, 60 pp. In French.

N-21010
(Ames Research Center library no.)

112. *Evvard, J.C.; and Marcus, L.R.: **Achievement of Continuous Wall Curvature in Design of Two-Dimensional Symmetrical Supersonic Nozzles.** NACA TN-2616, January 1952, 8 pp, 5 refs.

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.

113. *Corrsin, S.: **Effect of Wind Tunnel Nozzle on Steady-Flow Nonuniformities.** Readers' Forum. Journal of the Aeronautical Sciences, vol. 19, no. 2, February 1952, pp. 135-136, 1 ref.

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^{-1}) 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 John Hopkins University, Baltimore, Maryland, USA.

114. *Evvard, J.C.; and Maslen, S.H.: **Three-Dimensional Supersonic Nozzles and Inlets of Arbitrary Exit Cross Section.** NACA TN-2688, April 1952, 12 pp, 6 refs.

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.

115. *Crown, J.C.: **Design of Nozzles Having Continuous Wall Curvature.** Readers' Forum. Journal of the Aeronautical Sciences, vol. 19, no. 5, May 1952, pp. 358-359, 2 refs.

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.

116. *Ruptash, J.: **Supersonic Wind Tunnels - Theory, Design and Performance.** UTIA Review 5, June 1952, 190 pp, 122 refs.

AD-001377
N-51372

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 been only 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.

117. *Haefeli, R.C.: **Use of Fences to Increase Uniformity of Boundary Layer on Side Walls of Supersonic Wind Tunnels.** NACA RM-E52E19, July 1952, 15 pp, 8 refs.

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.

118. *Frost, R.C.; and Liepman, H.P.: **Supersonic Nozzles with Continuous Wall Curvature.** Readers' Forum. Journal of the Aeronautical Sciences, vol. 19, no. 10, October 1952, pp. 716, 4 refs.

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.

*Consolidated Vultee Aircraft Corp., Fort Worth, Texas, USA.

119. *Reshotko, E.; and Haefeli, R.C.: **Investigation of Axially Symmetric and Two-Dimensional Multi-Nozzles for Producing Supersonic Streams.** NACA RM-E52H28, October 1952, 35 pp, 4 refs.

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.

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

120. LaBerge, J.G.: **Design, Calibration and Correction of a 1.6 Mach Number Nozzle.** NAE LR-41, November 1952.

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

122. Fallis, W.B.; Johnson, G.W.; et al: **Design and Calibration of the Institute of Aerophysics 16 x 16 inch Supersonic Wind Tunnel.** UTIA Report 15, 1953

123. *Shapiro, A.H.: **The Dynamics and Thermodynamics of Compressible Fluid Flow. Volume 1. Chapter 15 - Method of Characteristics for Two-Dimensional Supersonic Flow.** Published by Ronald Press Company, 1953, pp. 462-524, 10 refs.

QA 913 S497
(Ames Research Center library no.)

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.

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

124. **Intermittent Supersonic Wind Tunnels: Notes on British Experience.** RAE TM Aero 345, 1953.

125. Cunsolo, D.: **Calculation of the Shape of a Two-Dimensional Supersonic Nozzle in Closed Form.** English translation of 1951 Italian paper. NACA TM-1358, January 1953, 29 pp.

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."

126. LaBerge, J.G.: **Design and Calibration of Supersonic Wind Tunnel Nozzles.** In: Proceedings of Symposium on High Speed Aerodynamics, NAE Ottawa, January 1953, 35 pp.

127. *Orlin, W.J.; Heins, A.; and Douglas, B.: **A New Type Symmetrical Two-Dimensional Variable Mach Number Nozzle.** ARO Inc., RJA Branch, NP Report 2, January 1953.

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

128. Syvertson, C.A.; and Savin, R.C.: **The Design of Variable Mach Number Asymmetric Supersonic Nozzles by Two Procedures Employing Inclined and Curved Sonic Lines.** NACA TN-2922, March 1953.

129. *Heybey, W.H.: **The Prandtl-Busemann Method for the Graphical Evaluation of 2-D Super. Flow. Its Application to the Design of Wind Tunnel Nozzles and the Appropriate Corrections. Part III Principles and Accomplishments of the Correction of Wind Tunnel Nozzles.** English translation of Peenemunde Archives Report no. 66/36g, February 1941. North American Report AL-1715, May 1953, 63 pp.

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 windtunnel 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.

*Wasserbau-Versuchsanstalt, Kochel, Germany.

130. *Riise, H.N.; and Serrell, P.V.H.: **Flexible Nozzles for Supersonic Wind Tunnels.** Presented at AGARD Wind Tunnel and Model Testing Panel Meeting, May 1953, 26 pp.

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.

*Jet Propulsion Laboratory, Pasadena, California, USA.

131. *Orlin, W.J.: **The Semi-Flexible Supersonic Nozzle.** Presented at AGARD Wind Tunnel and Model Testing Panel Meeting, May 1953, 12 pp.

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.

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

132. *Riise, H.N.: **Functions of Prandtl-Meyer Angle for Supersonic Nozzle Design.** JPL Publication no. 26, December 1953, 47 pp, 1 ref.

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^{o} in Prandtl-Meyer angle through the range $\psi=0$ to $\psi=102.34$.

*Jet Propulsion Laboratory, Pasadena, California, USA.

133. *MacDermott, W.N.: **Correction of Flexible Plate Supersonic Nozzle Contours by Influence Methods.** AEDC-TN-53-8, December 1953, 21 pp, 11 refs.

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.

134. *Shapiro, A.H.: **The Dynamics and Thermodynamics of Compressible Fluid Flow. Volume 2. Section 21.4 - Flow in Throat of Converging-Diverging Nozzle and Section 21.5 - relaxation Method.** Published by Ronald Press Company, 1954, pp. 826-840, 7 refs.

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.

135. *LaBerge, J.G.; and Conlin, L.T.: **A Technique of Manufacture of Supersonic Nozzles.** In: Proceedings of 2nd Canadian Symposium on Aerodynamics. Paper no. 5, February 1954, pp 264-279, 1 ref.

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.

136. *Ferri, A.; and Bogdonoff, S.M.: **Design and Operation of Intermittent Supersonic Wind Tunnels.** AGARDograph-1, May 1954, 113 pp, 72 refs.

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.

137. *Baron, J.R.: **Analytic Design of a Family of Supersonic Nozzles by the Freidrichs' Method, Including Computation Tables and a Summary of Calibration Data.** WADC TR-54-279, June 1954, 432 pp, 48 refs.

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.

138. *Riise, H.N.: **Flexible-Plate Nozzle Design for Two-Dimensional Supersonic Wind Tunnels.** JPL Report no. 20-74, June 1954. 46 pp, 5 refs.

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.

139. *Laufer, J.: **Factors Affecting Transition Reynolds Numbers on Models in Supersonic Wind Tunnels.** Readers' Forum. Journal of the Aeronautical Sciences, vol. 21, no. 7, July 1954, pp. 497-498, 5 refs.

The great scatter in transition Reynolds numbers measured in different supersonic wind tunnels is believed to be caused by disturbances of various degrees present in the tunnels. In general, these disturbances found in supersonic wind tunnel nozzle flows may be of three different types: (a) vorticity fluctuations (or transverse waves); (b) sound, or longitudinal waves - including weak compression waves emanating from the imperfect nozzle walls; and (c) temperature fluctuations. The JPL research into this problem will attempt to successively eliminate the above mentioned three contributions to the perturbation field in the supersonic tunnel. Data is presented from the JPL 18 in. by 20 in. supersonic tunnel on a 5^0 included angle cone. The first set of experiments indicate that in low Mach number flows ($M<2.5$) the turbulence level of the settling chamber had a strong effect on the boundary layer transition Reynolds number. However, no such effect could be detected for higher Mach number flows.

*Jet Propulsion Laboratory, Pasadena, California, USA.

140. *Baron, J.R.; and Durgin, F.H.: **An Experimental Investigation of Heat Transfer to the Cooled Sidewall of M = 2.5, 3.0, and 3.5 Supersonic Nozzles.** NSLM Wind Tunnel Report 65, July 1954, 82 pp, 27 refs.

AD-74144

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 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.

141. *Westkaemper, J: **The Design of a Mach Number 2.75 Nozzle and Variable Throat Diffuser for the OAL Supersonic Wind Tunnel.** OAL Memo 80, September 1954.

*Ordnance Aerophysics Laboratory, Daingerfield, Texas, USA.

142. *Morris, D.E.; and Winter, K.G.: **Requirements for Uniformity of Flow in Supersonic Wind Tunnels.** RAE TN no. Aero.2342, September 1954, 9 pp.

N-48101
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^{0} 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 ($M \approx 1.4$) a higher standard is desirable.

*National Aeronautical Establishment, Ottawa, Canada.

143. *Kenney, J.T.; and Webb, L.M.: **A Summary of the Techniques of Variable Mach Number Supersonic Wind Tunnel Nozzle Design.** AGARDograph-3, October 1954, 142 pp, 47 refs.

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.

144. *Smelt, R.: **Factors Influencing Precision in Supersonic Wind Tunnels.** AEDC-TN-54-25, November 1954, 21 pp, 2 refs.

N-33527
(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.

145. *Baron, J.R.; and Durgin, F.H.: **An Experimental Investigation of Heat Transfer at the Boundaries of Supersonic Nozzles.** WADC-TR-54-541, December 1954, 81 pp, 27 refs.

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 3.5×10^7) 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.

146. *Beckwith, I.E.; and Moore, J.A.: **An Accurate and Rapid Method for the Design of Supersonic Nozzles.** NACA TN-3322, February 1955. 57 pp, 10 refs.

A Procedure is given for designing two-dimensional 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.

147. *MacDermott, W.N.: **The Correction of Flexible Plate Supersonic Nozzle Contours by Influence Methods.** Journal of the Aeronautical Sciences, vol. 22, no. 5, May 1955, pp. 289-296, 10 refs.

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.

148. *Hill, J.A.F.: **On the Calibration of Supersonic Wind-Tunnels.** Readers' Forum. Journal of the Aeronautical Sciences, vol. 22, no. 6, June 1955, pp. 441-443, 2 refs.

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.

149. *Rosen, J.: **The Design and Calibration of a Variable Mach Number Nozzle.** Journal of the Aeronautical Sciences, vol. 22, no. 7, July 1955, pp. 484-490.

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 screw mechanism, which in a defined manner simultaneously affects the coordinates and derivatives of the inlet and throat curvatures up to the inflection point. The flexible part of the curvature between the inflection point and the test section consists of a spring, with linear thickness distribution, rigidly clamped at the level of the test section (150 x 150 mm^2 = 5.9 in. x 5.9 in.). Due to small curvatures and a suitable choice of the linear thickness distribution of the flexible part of the wall, a good Mach number distribution (±0.015 or less) has been realized within the whole Mach number range from 1.5 to 2.5.

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

150. *Liepman, H.P.: **An Analytic Method for the Design of Two-Dimensional Asymmetric Nozzles.** Journal of the Aeronautical Sciences, vol. 22, no. 10, October 1955, pp. 701-709, 17 refs.

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 θ, 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.

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

151. *Sivells, J.C.: **Design of Two-Dimensional Continuous-Curvature Supersonic Nozzles.** Journal of the Aeronautical Sciences, vol. 22, no. 10, October 1955, pp. 685-692 and 732, 6 refs.

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 curve of a flexible plate supported at discrete points must have continuous curvature. It is, therefore, desirable that the curvature of the aerodynamic contour also 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 -**

Application of Characteristics to the Design of Plane Effusors, Diffusers and Jets. English version published by Editura Tehnica, 1956, pp. 425-444.

TL 573 C313
(Ames Research Center library no.)

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

*Institute of Applied Mechanics, Bucharest, Rumania.

153. *Meyer, R.E.: **Perturbations of Supersonic Nozzle Flows.** The Aeronautical Quarterly, February 1956, pp. 71-84, 8 refs.

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.

154. *Morkovin, M.V.: **Critical Review no 1223 of MacDermott paper in Journal of the Aeronautical Sciences, vol. 22, no. 5, May 1955.** In: Mechanics of Fluids Section of Applied Mechanics Review, vol. 9, no. 4, April 1956, pp. 172.

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

155. *Kennedy, E.C.: **Design of an Axially Symmetric Contraction-Expansion Nozzle For Exit Mach Number 2.7203.** OAL Memo 89, July 1956.

*Ordnance Aerophysics Laboratory, Daingerfield, Texas, USA.

156. *Sivells, J.C.: **Analytic Determination of Two-Dimensional Supersonic Nozzle Contours having Continuous Curvature.** AEDC-TR-56-11, July 1956, 29 pp, 5 refs.

AD-88606

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.

157. Leavy, L.E.: **A Supersonic Wind Tunnel for Mach Numbers up to 3.5.** AGARD Report no. 70, August 1956.

158. *Hall, M.G.: **The Accuracy of the Method of Characteristics for Plane Supersonic Flow.** The Quarterly Journal of Mechanics and Applied Mathematics, vol. 9, part 3, September 1956, pp. 320-333, 7 refs.

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.

159. *Hill, J.A.F.; Baron, J.R.; et al: **Mach Number Measurements in High Speed Wind Tunnels.** AGARDograph-22, October 1956, 125 pp, 81 refs.

TL 500 N6 no.22
(Ames Research Center library no.)

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.

160. *Squires, R.K.; Roberts, R.C.; and Fisher, E.R.: **A Method for Designing Supersonic Nozzles using the Centerline Mach Number Distribution.** Aeroballistic Report 285, NAVORD-3995, October 1956, 31 pp, 12 refs.

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.

161. *Kurzweg, H.H.: **The Aerodynamic Development of the V-2.** In: History of German Guided Missiles. AGARDograph-20, 1957, pp. 50-69, 15 refs.

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.

162. *Cole, D.: **The Laminar Boundary Layer Near a Sonic Throat.** In: Proceedings of the Heat Transfer and Fluid Mechanics Institute, June 1957, pp. 119-137, 7 refs.

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.

163. *Goranson, G.G.; Barnett, R.M.; et al: **A Method for the Calibration of Flexible-Plate Supersonic Wind Tunnels and Calibration Results for the 12-inch Wind Tunnel at the Jet Propulsion Laboratory.** JPL Report 20-110, June 1957, 78 pp, 2 refs.

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.

164. Ryzhov, O.S.: **On Flows in the Region of the Transition Surface.** English translation of Prik. Matematika i Mexahika (Journal of Applied Mathematics and Mechanics) published by Pergaman Press, vol. 22, no. 4, 1958, pp. 608-621, 5 refs.

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.

165. *Hollingdale, S.; and Barritt, M.M.: **An Application of a Computer to Wind Tunnel Design.** The Computer Journal, vol. 1, April 1958, pp. 42-48 (Part 1), pp. 64-68 (Part 2), 10 refs.

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

166. *Rao, G.V.R.: **Exhaust Nozzle Contour for Optimum Thrust.** Journal of the American Rocket Society (Jet Propulsion), vol. 28, no. 6, June 1958, pp. 377-382, 6 refs.

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.

167. *Frazer, R.P.; Eisenklam, P.; and Wilkie, D.: **Investigation of Supersonic Flow Separation in Nozzles.** Journal of Mechanical Engineering Science, vol. 1, no. 3, 1959, pp. 267-279, 13 refs.

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 iscentropic 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.

*Department of Chemical Engineering, Imperial College of Science and Technology, London, UK.

168. *Barritt, M.M.: **Calculation of Flexible Wall Shapes and Preparation of Control Tapes for the Bedford 8-ft x 8-ft Wind Tunnel.** ARC R&M 3187, February 1959, 33 pp, 13 refs.

N73-25041

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.

169. *Morkovin, M.V.: **On Supersonic Wind Tunnels with Low Free-Stream Turbulence.** Journal of Applied Mechanics, vol. 26, no. 9, September 1959, pp. 319-323, 24 refs.

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.

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

170. *Lord, W.T.: **A Theoretical Study of Annular Supersonic Nozzles.** ARC R&M 3227, October 1959, 28 pp, 37 refs.

N73-25035

This paper is concerned with the design of annular supersonic nozzles to produce uniform flow in supersonic wind tunnels which are axi-symmetrical and which have an internal coaxial circular cylinder throughout. Symmetrical two-dimensional and conventional axi-symmetrical nozzles are special cases of annular design. Proposals are made for design criteria sufficient to ensure that the flow inside a nozzle is free from limit lines and shock waves; the criteria for (symmetrical) two-dimensional and (conventional) axi-symmetrical nozzles are new. The two outstanding procedures for designing two-dimensional and axi-symmetrical nozzles are generalized to apply to annular nozzles. One of the design procedures is mainly analytical and the other is mainly numerical; the analytical expressions in both procedures are made much more complicated by the presence of the internal cylinder but the numerical process (the method of characteristics) is not. The design criteria and the mainly numerical design procedure are successfully applied to the design of a particular annular nozzle.

*Royal Aircraft Establishment, UK.

171. *Schueler, C.J.; and Strike, W.T.: **Calibration of a 40-Inch Continuous Flow Tunnel at Mach Numbers 1.5 to 6.** AEDC-TN-59-136, November 1959, 41 pp, 9 refs.

A test program was conducted to calibrate the von Karman Gas Dynamics Facility's 40- by 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.

172. *Landsbaum, E.M.: **Contour Nozzles.** American Rocket Society Journal, vol. 30, no. 3, March 1960, pp. 244-250, 20 refs.

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.

173. *Rao, G.V.R.: **Approximation of Optimum Thrust Nozzle Contour.** Journal of the American Rocket Society (Jet Propulsion), vol. 30, no. 6, June 1960, pp. 561, 3 refs.

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.

174. *Vrebalovich, T.: **On Supersonic Wind Tunnels with Low Free-Stream Turbulence. (Discussion of the Morkovin report of same title with an Author's Closure.)** Discussion. Journal of Applied Mechanics, vol. 27, no. 6, June 1960, pp. 362-364, 6 refs.

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.

175. *Goddard, F.E.: **Supersonic Tunnels.** In: High Speed Problems of Aircraft and Experimental Methods, Volume VIII of High Speed Aerodynamics and Jet Propulsion, Published by Princeton University Press, 1961, pp. 491-532, 21 refs.

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.

*Jet Propulsion Laboratory, Pasadena, California, USA.

176. *Irani, P.A.; and Satyanarayana, B.: **Aerodynamic Design of Two-Dimensional Supersonic Nozzle Contours with Boundary Layer Corrections.** NAL TN-AE-7-61, February 1961, 62 pp.

N64-80779

*National Aeronautical Laboratory, Bangalore, India.

177. *Pope, A.: **Wind Tunnel Calibration Techniques. Section 5 - Calibration of Supersonic Wind Tunnels.** AGARDograph-54, April 1961, pp. 27-42, 104 refs.

N63-80505
TL 500 N6 no.54
(Ames Research Center library no.)

This paper discusses air-flow measuring instruments, their use, and the manner of working up and presenting the data obtained when calibrating low-speed, nearsonic, transonic, supersonic, and hypersonic wind tunnels.

*Sandia Corp., New Mexico, USA.

178. *Ahlberg, J.H.; Hamilton, S.; et al: **Truncated Perfect Nozzles in Optimum Nozzle Design.** American Rocket Society Journal, vol. 31, no. 5, May 1961, pp. 614-620, 16 refs.

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.

*United Aircraft Corp., East Hartford, Connecticut, USA.

179. *MacDermott, W.N.: **Preliminary Experimental Results of the Reduction of Viscous Effects in a Low-Density Supersonic Nozzle by Wall Cryopumping.** AEDC-TN-61-71, October 1961, 46 pp, 8 refs.

AD-264656
N66-87627

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.

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

180. *Maxwell, H.; and Jacocks, J.L.: **Nondimensional Calculation of Turbulent Boundary-Layer Development in Two-Dimensional Nozzles of Supersonic Wind Tunnels.** AEDC-TN-61-153, January 1962, 25 pp, 3 refs.

N62-11832

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.

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

181. *Holt, M.: **The Design of Plane and Axisymmetric Nozzles by the Method of Integral Relations.** AFOSR-3140. In:

Proceedings of IUATM Symposium Transsonicum, September 1962, pp. 310-324, 12 refs.

A65-12340

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.

*University of California, Berkeley, California, USA.

182. *Hall, I.M.: **Transonic Flow in Two-Dimensional and Axially-Symmetric Nozzles.** The Quarterly Journal of Mechanics and Applied Mathematics, vol. 15, part 4, November 1962, pp. 487-508, 10 refs.

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.

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

183. *Migdal, D.; and Landis, F.: **Characteristics of Conical Supersonic Nozzles.** American Rocket Society Journal, vol. 32, no. 12, December 1962, pp. 1898-1901, 8 refs.

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 iscentropic 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.

*Pratt and Whitney Aircraft, East Hartford, Connecticut. USA.

184. *Anderson, A.: **Flow Characteristics of a 12-inch Intermittent Supersonic Tunnel.** AEDC-TDR-63-203, 1963, 44 pp, 5 refs.

AD-418578
N63-22438

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.

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

185. Alikhashkin, Y.; Favorskii, A.P.; and Chushkin, P.I.: **On the Calculation of the Flow in a Plane Laval Nozzle.** Zhurnal Vychisletel'noy Matematika i atematicheskoi Fiziki (Journal of Computer Mathematics and Physics), vol. 3, 1963, pp 1130-1134, 4 refs. In Russian. English translation is NASA TT F-10494, December 1966, 9 pp. N67-13834#.

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.

186. *Williams, J.C.: **Viscous Compressible and Incompressible Flow in Slender Channels.** AIAA Journal, vol. 1, no. 1, January 1963, pp. 185-195, 13 refs.

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.

187. *Pasqua, P.F.; Stevens, P.N.; et al: **Analytical Studies of Nozzle Throat Cooling.** AEDC-TDR-63-58, April 1963, 163 pp, 20 refs.

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.

188. *Darwell, H.M.; and Badham, H.: **Shock Formation in Conical Nozzles.** Technical Note. AIAA Journal, vol. 1, no. 8, August 1963, pp. 1932-1934, 4 refs.

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.

189. Carriere, P.: **Calculation of Supersonic Nozzle Shapes using the Method of Characteristics.** Ecole Nationale Superior Aero ENSA Report, 1964. In French.

190. *Hall, I.M.; and Sutton, E.P.: **Transonic Flow in Ducts and Nozzles, a Survey.** In: Symposium Transsonicum published by Springer Verlag, 1964, pp. 325-344.

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

191. **Handbook of Supersonic Aerodynamics: Section 17 - Ducts, Nozzles and Diffusers.** Produced by The John Hopkins University. NAVWEPS-1488, January 1964, 450 pp, 413 refs.

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.

192. *McCabe, A.: **Design of Supersonic Nozzles.** ARC R&M 3440, March 1964, 36 pp, 25 refs.

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.

193. *Back, L.H.; Massier, P.F.; and Gier, H.L.: **Convective Heat Transfer in Convergent-Divergent Nozzles.** International Journal of Heat and Mass Transfer, vol. 7, no. 5, May 1964, pp. 549-568, 24 refs.

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.

194. *Lieu, B.H.: **Air Film Cooling of a Supersonic Nozzle.** NOLTR-64-65, August 1964, 35 pp, 15 refs.

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.

195. *Pope, A.; and Goin, K.L.: **High-Speed Wind Tunnel Testing. Chapter 1 - High-Speed Wind Tunnel Theory and Chapter 10 - Calibration and Use of Supersonic Tunnels** Published by John Wiley and Sons Inc., 1965, pp 1-65 and pp 349-401, 33 refs.

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.

196. *O'Brien, R.L.: **Laminarization of Nozzle Wall Boundary Layers as a Means of Reducing Heat Flux.** Report AFRPL-TR-65-40, February 1965, 156 pp, 31 refs.

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.

197. *Gibbings, J.C.: **A Note on the Combination of a Contraction with a Supersonic Nozzle for a Wind Tunnel.** RAE TR no. 65066, March 1965, 19 pp, 4 refs.

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.

198. *Hall, M.G.: **Experimental Measurements in a Three-Dimensional Turbulent Boundary Layer in Supersonic Flow.** In: Recent Developments in Boundary Layer Research, AGARDograph-97, May 1965, pp. 829-853, 11 refs.

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.

199. *Jeffries, N.P.; and Cumbers, L.L.: **Computer Program for Calculation of Laminar Boundary Layer in a Supersonic Nozzle.** GE-R65SD3007, June 1965, 48 pp, 5 refs.

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.

200. *Migdal, D.: **Study on Pressure Distributions in Conical Nozzles.** Grumman Aircraft Engineering Corp. paper, ADR-01-10-65.1, September 1965, 25 pp.

N66-87191

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

201. *Back, L.H.; Massier, P.F.; and Gier, H.L.: **Comparison of Measured and Predicted Flows Through Conical Supersonic Nozzles with Emphasis on the Transonic Regime.** AIAA Journal, vol. 3, no. 9, September 1965, pp. 1606-1614, 18 refs.

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^o and 1500^oR 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.

202. Favorskii, A.P.: **Design of Laval Nozzles.** Zhurnal Vychisletel'noy Matematika i Matematicheskoi Fiziki (Journal of Computer Mathematics and Physics), October 1965, pp. 955-959, 4 refs. In Russian.

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.

203. Demetriades, A.; and Von Seggern, L.: **Design, Performance and Operation of the Mach 3 Wind Tunnel.** Aeronautics Division of Philco Corp., Newport Beach, CA, PC Report U-3390, November 1965.

204. *Carden, W.H.: **Local Heat-Transfer Coefficients in a Nozzle with High Speed Laminar Flow.** Also AEDC-TDR-64-61, April 1964. AIAA Journal, vol. 3, no. 12, December 1965, pp. 2183-2188, 17 refs.

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^{o} to 6500^{o}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.

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

205. *Sichel, M.: **The Effect of Longitudinal Viscosity on the Flow at a Nozzle Throat.** Journal of Fluid Mechanics, vol. 25, part 4, August 1966, pp. 769-786, 28 refs.

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.

206. *Hopkins, D.F.; and Hill, D.E.: **Effect of Small Radius of Curvature on Transonic Flow in Axisymmetric Nozzles.** AIAA Journal, vol. 4, no. 8, August 1966, pp. 1337-1343, 8 refs.

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^{o} to 75^{o}. 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 time the throat radius.

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

207. *Lee, R.E.; Yanta, W.J.; et al: **The NOL Boundary Layer Channel.** NOLTR 66-185, November 1966, 39 pp, 13 refs.

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^{o}F and supply pressures from 0.1 to 10 atmospheres. The Reynolds number per foot capability is from 3×10^{4} to 2.4×10^{7}.

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

208. *Back, L.H.; and Cuffel, R.F.: **Detection of Oblique Shocks in a Conical Nozzle with a Circular-Arc Throat.** Technical Note. AIAA Journal, vol. 4, no. 12, December 1966, pp. 2219-2221, 9 refs.

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.

209. *Back, L.H.; Massier, P.F.; and Cuffel, R.F.: **Flow Phenomena and Convective Heat Transfer in a Conical Supersonic Nozzle.** Journal of Spacecraft and Rockets, vol. 4, no. 8, August 1967, pp. 1040-1047, 30 refs.

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^{o} to 2000^{o}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.

210. *Williams, J.C.: **Conical Nozzle Flow with Velocity Slip and Temperature Jump.** AIAA Journal, vol. 5, no. 12, December 1967, pp. 2128-2134, 7 refs.

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.

211. *Back, L.H.; Massier, P.F.; and Cuffel, R.F.: **Effect of Inlet Boundary-Layer Thickness and Structure on Heat Transfer in a Supersonic Nozzle.** Engineering Notes. AIAA Journal of Spacecraft and Rockets, vol. 5, no. 1, January 1968, pp. 121-123, 3 refs.

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.

212. *Vanco, M.R.; and Goldman, L.J.: **Computer Program for Design of Two-Dimensional Supersonic Nozzle with Sharp Edged Throat.** NASA TM-X-1502, January 1968, 19 pp.

N68-14747#

The FORTRAN IV computer program for the design of a two-dimensional supersonic nozzle with sharp-edged throat is presented along with the equations used. The nozzle, which has uniform parallel flow at the exit, is designed on the basis of two-dimensional isentropic flow of a perfect gas. The program requires as input the exit Mach number and the specific-heat ratio. The output yields the nozzle contour for the supersonic portion. Input and output for a sample case are included.

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

213. *Hopkins, D.F.; and Hill, D.E.: **Transonic Flow in Unconventional Nozzles.** AIAA Journal, vol. 6, no. 5, May 1968, pp. 838-842, 4 refs.

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.

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

214. *Boldman, D.R.; Schmidt, J.F.; and Gallagher, A.K.: **Laminarization of a Turbulent Boundary Layer as Observed from Heat Transfer and Boundary Layer**

Measurements in Conical Nozzles. NASA TN-D-4788, September 1968, 34 pp.

N68-33642#

Heat transfer measurements were obtained in 30^0 and 60^0 half-angle of convergence nozzles at a nominal stagnation temperature of 970^0R (539 K) and over a range of stagnation pressures of 2.0 to 20.4 atmospheres (2.03 x 10^5 to 20.67 x 10^5 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 (3.14 x 10^5 and 20.67 x 10^5 N/m^2). The heat transfer and boundary layer surveys suggested the occurrence of laminarization of an initially turbulent boundary layer.

*NASA Lewis R.C., Cleveland, Ohio, USA.

215. *Kliegel, J.R.; and Quan, V.: **Convergent-Divergent Nozzle Flows.** AIAA Journal, vol. 6, no. 9, September 1968, pp. 1728-1734, 9 refs.

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.

*Propulsion Analysis Department, TRW Systems, Redondo Beach, California, USA.

216. *Nelms, L.T.: **Design of Minimum Length Supersonic Nozzle Contours and Associated Sub-Contours.** AEDC-TR-68-212, November 1968, 96 pp.

AD-843755
X69-11596 (Available to US Government Agency and US Government Contractors Only)

A method for designing the minimum length, supersonic, inviscid nozzle contour based on a uniform starting line will be presented. A family of subcontours, all of which have the same exit flow, can also be calculated. The method applies to both axisymmetric and two dimensional nozzles and utilizes either perfect or real gas. A computer program utilizing this method has been developed and is also given.

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

217. *Back, L.H.; Massier, P.F.; and Cuffel, R.F.: **Flow and Heat Transfer Measurements in Subsonic Air Flow Through a Contraction Section.** International Journal of Heat and Mass Transfer, vol. 12, no. 1, January 1969, pp. 1-13.

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_{to} from 1/3 to 2/3, Reynolds numbers from 2 x 10^4 to 2 x 10^5, 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_{to} that were found upstream in the tube were not observable in the separation and reattachment regions.

*Jet Propulsion Laboratory, Pasadena, California, USA.

218. *Migdal, D.; Klein, K.; and Moretti, G.: **Time-Dependent Calculations for Transonic Nozzle Flow.** Technical Note. AIAA Journal, vol. 7, no. 2, February 1969, pp. 372-374, 11 refs.

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.

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

219. *Back, L.H.; Cuffel, R.F.; and Massier, P.F.: **Laminarization of a Turbulent Boundary Layer in Nozzle Flow.** Technical Note. AIAA Journal, vol. 7, no. 4, April 1969, pp. 730-733, 12 refs.

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.

*Jet Propulsion Laboratory, Pasadena, California, USA.

220. *Kliegel, J.R.; and Levine, J.N.: **Transonic Flow in Small Throat Radius of Curvature Nozzles.** AIAA Journal, vol. 7, no. 7, July 1969, pp. 1375-1378, 7 refs.

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.

*Dynamic Science, Monrovia, California, USA.

221. *Cuffel, R.F.; Back, L.H.; and Massier, P.F.: **Transonic Flowfield in a Supersonic Nozzle with Small Throat Radius of Curvature.** Technical Note. AIAA Journal, vol. 7, no. 7, July 1969, pp. 1364-1366, 15 refs.

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 (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 ($r_c/r_{th}<1.0$), such as that for the nozzle investigated.

*Jet Propulsion Laboratory, Pasadena, California, USA.

222. *Morden, D.B.; and Farquhar, B.W.: **Comment on Transonic Flow in Unconventional Nozzles.** Technical Comment. AIAA Journal, vol. 7, no. 8, August 1969, pp. 1661-1662, 1 ref.

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

*The Boeing Company, Seattle, Washington, USA.

223. *Massier, P.F.; Back, L.H.; et al: **Viscous Effects on the Flow Coefficient for a Supersonic Nozzle.** Technical Note. AIAA Journal, vol. 8, no. 3, March 1970, pp. 605-607, 7 refs.

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.

224. *Back, L.H.: **Acceleration and Cooling Effects in Laminar Boundary Layers - Subsonic, Transonic, and Supersonic Speeds.** AIAA Journal, vol. 8, no. 4, April 1970, pp. 794-802.

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 β 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 β 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 β 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.

225. *Back, L.H.; Cuffel, R.F.; and Massier, P.F.: **Laminarization of a Turbulent Boundary Layer in Nozzle Flow - Boundary Layer and Heat Transfer Measurements with Wall Cool.** ASME Journal of Heat Transfer, vol. 92, no. 3, August 1970, pp. 333-344, 33 refs.

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 x 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.

226. *Levine, J.N.; and Coats, D.E.; **Transonic Flow in a Converging-Diverging Nozzle.** NASA CR-111104, September 1970, 125 pp.

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.

227. *Back, L.H.; and Cuffel, R.F.: **Flow Coefficients for Supersonic Nozzles with Comparatively Small Radius of Curvature Throats.** Journal of Spacecraft and Rockets, vol. 8, no. 2, February 1971, pp. 196-198, 20 refs.

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 10^6) 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.

228. *Back, L.H.: **Laminar Boundary Layers with Large Wall Heating and Flow Acceleration.** Technical Note. AIAA Journal, vol. 9, no. 5, May 1971, pp. 966-969, 13 refs.

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.

229. *Rothe, D.E.: **Electron-Beam Studies of Viscous Flow in Supersonic Nozzle.** AIAA Journal, vol. 9, no. 5, May 1971, pp. 804-811, 19 refs.

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.

*Cornell Aeronautical Laboratory Inc., Buffalo, New York, USA.

230. *Wagner, J.L.: **A Cold Flow Field Experimental Study Associated with a 2-D Multiple Nozzle.** NOLTR 71-78, July 1971, 89 pp, 13 refs.

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^{0}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.

231. *Wortman, A.; and Mills, A.F.: **Highly Accelerated Compressible Laminar Boundary Layer Flows with Mass Transfer.** ASME Journal of Heat Transfer, vol. 93, no. 3, August 1971, pp. 281-289, 13 refs.

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 β 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 β, Mach number, wall enthalpy ratio, mass transfer rate, ω and Pr. The inadequacy of solutions obtained under the simplifying assumptions of Pr = 1.0 and ω = 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.

232. *Back, L.H.; Cuffel, R.F.; and Massier, P.F.: **Influence of Contraction Section Shape and Inlet Flow Direction on Supersonic Nozzle Flow Performance.** Journal of Spacecraft and Rockets, vol. 9, no. 6, June 1972, pp. 420-427, 16 refs.

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.

233. *Beckwith, I.E.; and Bertram, M.H.: A **Survey of NASA Langley Studies on High-Speed Transition and the Quiet Tunnel.** NASA TM-X-2566, July 1972, 67 pp, 41 refs.

N72-26239

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.

234. *Back, L.H.; and Massier, P.F.: **Viscous Non-Adiabatic Laminar Flow Through a Supersonic Nozzle: Experimental Results and Numerical Calculations.** ASME Journal of Heat Transfer, vol. 94, no. 4, November 1972, pp. 437-445, 29 refs.

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.

235. *Nash-Webber, J.; and Oates, G.C.: **An Engineering Approach to the Design of Laminarizing Nozzle Flows.** Journal of Basic Engineering, December 1972, pp. 897-904, 18 refs.

Despite intensive research into laminarization over the past twenty years, there has resulted hitherto no prediction method suitable for ab intio 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.

236. *Joly, M.: **CARACATH Program for the Computation of Two-Dimensional Nozzle Flow Using the Method of Characteristics.** Research Institute of St Louis ISL-12/73, May 1973, 49 pp, 5 refs. In French.

N74-17981#

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.

237. *Beckwith, I.E.; Harvey, W.D.; et al: **Control of Supersonic Wind-Tunnel Noise by Laminarization of Nozzle-Wall Boundary Layers.** NASA TM-X-2879, December 1973, 57 pp, 64 refs.

N74-11119

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.

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

238. *Barger, R.L.: **Streamline Curvature Design Procedure for Subsonic and Transonic Ducts.** NASA TN-D-7368, December 1973, 16 pp, 11 refs.

N74-11818

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.

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

239. *Cline, M.C.: **Computation of Steady Nozzle Flow by a Time-Dependent Method.** AIAA Journal, vol. 12, no. 4, April 1974, pp. 419-420, 11 refs. Also: NASA TM-X-71918.

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.

*University of California, Los Alamos Scientific Laboratory, Los Alamos, New Mexico, USA.

240. *Swallow, R.J.; and Aiello, R.A.: **NASA Lewis 8 by 6 Foot Supersonic Wind Tunnel.** NASA TM-X-71542, May 1974, 24 pp, 4 refs.

N74-25789#

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.

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

241. Kreskovsky, J.P.; Shamroth, S.J.; and McDonald, H.: **Parametric Study of Relaminarization of Turbulent Boundary Layers on Nozzle Walls.** NASA CR-2370, June 1974.

N74-28758

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.

242. *Aiello, R.A.: **NASA Lewis 10 by 10 Foot Supersonic Wind Tunnel.** NASA TM-X-71625, November 1974, 47 pp.

N75-14780#

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.

243. *Pfenninger, W.; and Syberg, J.: **Reduction of Acoustic Disturbances in the Test Section of Supersonic Wind Tunnels by Laminarizing their Nozzle and Test Section Wall Boundary Layers by Means of Suction.** NASA CR-2436, November 1974, 305 pp, 80 refs.

N75-12977#

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^*/ν^* (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.

244. *Beckwith, I.E.: **Development of a High Reynolds Number Quiet Tunnel for Transition Research.** AIAA Journal, vol. 13, no. 3, March 1975, pp. 300-306, 33 refs.

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.

245. *Dumitrescu, L.Z.: **Minimum Length Axisymmetric Laval Nozzle.** AIAA Journal, vol. 13, no. 4, April 1975, pp. 520-521, 5 refs.

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.

246. *Jaeggy, B.C.: **A FORTRAN Program for the Computation of Two-Dimensional or Axisymmetric Nozzles.** Anglo-German Research Institute of St Louis ISL-R-110/75, April 1975, 58 pp, 6 refs. In French.

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.

247. *Brown, E.F.; and Hamilton, G.L.: **Survey of Exhaust-Nozzle Flow Analysis.** AIAA Paper 75-60. AIAA Journal of Aircraft, vol. 13, no. 1, January 1976, pp. 4-11, 82 refs.

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.

248. *Cline, M.C.: **Computation of Two-Dimensional, Viscous Nozzle Flows.** AIAA Journal, vol. 14, no. 3, March 1976, pp. 295-296, 4 refs.

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.

*Theoretical Division, University of California, Los Alamos, New Mexico, USA.

249. *Back, L.H.; and Cuffel, R.F.: **Compressible Laminar Boundary Layers with Large Acceleration and Cooling.** Technical Note. AIAA Journal, vol. 14, no. 7, July 1976, pp. 968-971, 13 refs.

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 pertain 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.

250. *McIntosh, A.C.; and Pike, J.: **The 1-4-1 System of Jack Movements for the Flexible Liners of Supersonic Wind Tunnels.** RAE TM Aero.1684, August 1976, 20 pp, 4 refs.

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.

251. *Brown, E.F.; Brecht, T.J.; and Walsh, K.E.: **A Relaxation Solution of Transonic Nozzle Flows Including Rotational Effects.** AIAA Journal of Aircraft, vol. 14, no. 10, October 1977, pp. 944-951, 12 refs.

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.

252. *Reed, T.D.; Pope, T.C.; and Cooksey, J.M.: **Calibration of Transonic and Supersonic Wind Tunnels.** NASA CR-2920, November 1977, 287 pp, 123 refs.

N78-15058#

State-of-the art instrumentation and procedures for calibrating transonic ($0.6 < M < 1.4$) and supersonic ($M \leq 3.5$) 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.

253. *Anders, J.B.; Stainback, P.C.; and Beckwith, I.E.: **A New Technique for Reducing Test Section Noise in Supersonic Wind Tunnels.** AIAA Paper 78-817. In: Proceedings of AIAA 10th Aerodynamic Testing Conference, April 1978, pp. 354-364, 25 refs.

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.

254. *Hoffman, J.D.; and Allman, J.G.: **Design of Maximum Thrust Nozzle Contours by Direct Optimization Methods.** AIAA Paper 78-1048. Presented at AIAA/SAE 14th Joint Propulsion Conference, July 1978, 8 pp, 17 refs.

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: multidimensional 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.

255. *Mitra, N.K.; and Fiebig, M.: **Supersonic Nozzle Flowfields: A Comparison of Fully Viscous and Navier-Stokes Solutions.** In: Theoretical and Experimental Fluid Mechanics, Chapter 3 - Nozzle Flow, published by Springer-Verlag, 1979, pp. 157-165, 8 refs.

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.

256. *Anders, J.B.; Stainback, P.C.; and Beckwith, I.E.: **New Technique for Reducing Test Section Noise in Supersonic Wind Tunnels.** AIAA Journal, vol. 18, no. 1, January 1980, pp. 5-6, 11 refs.

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 wind tunnels at Mach numbers greater than about 2.5. These acoustic disturbances are propagated along Mach lines 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.

257. *Erickson, L.L.; Kassner, D.L.; et al: **Single-Mode Panel Flutter of a Wind Tunnel Flex-Wall.** AIAA Journal of Aircraft, vol. 17, no. 7, July 1980, pp. 521-527, 11 refs.

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.

258. *Drummond, D.; Rogers, B.; and Demetriades, A.: **Design and Operating Characteristics of the Supersonic Wind Tunnel.** Montana State University SWT-TR-81-01, January 1981.

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

259. *Baker, T.J.: **Numerical Computation of Transonic Potential Flow Through Nozzles.** The Aeronautical Quarterly, vol. 32, part 1, February 1981, pp. 31-42, 14 refs.

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.

260. *Beckwith, I.E.: **Comments on Settling Chamber Design for Quiet Blowdown Wind Tunnels.** NASA TM-81948, March 1981, 39 pp, 22 refs.

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.

261. *Demetriades, A.: **Roughness Effect on Boundary-Layer Transition in a Nozzle Throat.** AIAA Paper 81-4057. AIAA Journal, vol. 19, no. 3, March 1981, pp. 282-289, 16 refs.

A81-4057

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.

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

262. *Dutton, J.C.; and Addy, A.L.: **Transonic Flow in the Throat Region of Axisymmetric Nozzles.** AIAA Paper 81-4136. Technical Note. AIAA Journal, vol. 19, no. 6, June 1981, pp. 801-804, 6 refs.

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 η is arbitrary. The results of this solution are compared to two data sets for which the nozzle wall radius of curvature is relatively small.

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

263. *Allman, J.G.; and Hoffman, J.D.: **Design of Maximum Thrust Nozzle Contours by Direct Optimization Methods.** AIAA Paper 78-1048R. AIAA Journal, vol. 19, no. 6, June 1981, pp. 750-751, 4 refs.

A procedure for the design of maximum thrust nozzle contours by direct optimization methods is presented. The nozzle contour is 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: multidimensional line search, the method of steepest descent, and Newton's method. Results are presented to illustrate the behavior of the direct optimization methods, and to demonstrate that second-degree polynomial contours yield thrusts comparable to the thrusts developed by nozzle contours determined by calculus of variations methods.

*Purdue University, West Lafayette, Indiana, USA.

264. *Demetriades, A.: **Boundary Layer Transition in a Supersonic Nozzle Throat in the Presence of Cooling and Surface Roughness.** Montana State University MSU/SWT TR-82-01, August 1981, 181 pp, 18 refs.

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. 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 a counted for by existing blunt-body boundary-layer transition correlations.

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

265. *Campbell, R.L.: **Computer Analysis of Flow Perturbations Generated by Placement of Choke Bumps in a Wind Tunnel.** NASA TP-1892, August 1981, 42 pp, 7 refs.

N81-30088

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.

266. Hermann, R.: **The Supersonic Wind Tunnel Installations at Peenemunde and Kochel and Their Contributions to the Aerodynamics of Rockets.** In: Proceedings of the 32nd International Astrological Conference, September 1981, pp. 435-446.

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.

267. *Lin, C.Q.; and Shen, S.F.: **Analysis of Transonic Flow with Shocks in Slender Hyperbolic Nozzles.** AIAA Paper 82-0160. Presented at AIAA 20th Aerospace Sciences Meeting, January 1982, 9 pp, 4 refs.

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.

268. *Beckwith, I.E.; Creel, T.R.; et al: **Free Stream Noise and Transition Measurements in a Mach 3.5 Pilot Quiet Tunnel.** AIAA Paper 83-0042. Presented at AIAA 21st Aerospace Sciences Meeting, January 1983, 24 pp, 30 refs.

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 5^0 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.

269. *Nachshon, Y.: **Simple and Accurate Calculation of Supersonic Nozzle Contour.** Technical Note. AIAA Journal, vol. 21, no. 5, May 1983, pp. 783-784, 4 refs.

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 flowfieid 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.

270. *Beckwith, I.E.; Creel, T.R.; et al: **Free-Stream Noise and Transition Measurements on a Cone in a Mach 3.5 Pilot Low Disturbance Tunnel.** NASA TP-2180, September 1983, 68 pp, 34 refs.

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 x 10^5 up to 15 x 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.

271. *Malik, M.R.: **Instability and Transition in Supersonic Boundary Layers.** Presented at the Energy Sources Technology Conference, February 1984. In: Laminar Turbulent Boundary Layers published by ASME, pp. 139-147, 23 refs.

TL 357 E54
(Ames Research Center library no.)

Linear compressibility stability theory is used to analyze the recently available flight transition data (M_∞=1.2, 1.35, 1.6, 1.92) and "quiet" tunnel data (M_∞=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.

272. *Beckwith, I.E.; Malik, M.R.; and Chen, F.-J.: **Nozzle Optimization Study for Quiet Supersonic Wind Tunnels.** AIAA Paper 84-1628, Presented at AIAA 17th Fluid Dynamics, Plasma Dynamics, and Lasers Conference, June 1984, 23 pp, 25 refs.

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.

273. *Beckwith, I.E.; Malik, M.R.; et al: **Effects of Nozzle Design Parameters on the Extent of Quiet Test Flow at Mach 3.5.** Presented at IUTAM Symposium on Laminar-Turbulent Transition, July 1984 In: Laminar-Turbulent Transition published by Springer-Verlag, 1985, pp. 589-600, 17 refs.

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 e^N 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.

274. *Penaranda, F.E.; and Freda, M.S.: **Supersonic Wind Tunnels.** In: Aeronautical Facilities Catalogue - Volume 1, Wind Tunnels, NASA RP-1132, January 1985, pp. 187-236.

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.

275. *Chen, F.-J.; Malik, M.R.; and Beckwith, I.E.: **Instabilities and Transition in the Wall Boundary Layers of Low-Disturbance Supersonic Nozzles.** AIAA Paper 85-1573, June 1985, 14 pp, 29 refs.

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 e^N 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.

276. *Bazin, M.: **The New Controllable Mach Supersonic Nozzle for the S3MA Wind Tunnel.** Presented at the 64th STA Meeting, October 1985, 16 pp.

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.

277. *Varner, M.O.: **Application of Optimization Principles to Flexible Wall Nozzle Design - A Case Study.** AIAA Paper 86-0775, March 1986, 15 pp, 5 refs.

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.

278. *Beckwith, I.E.; Chen, F.-J.; and Creel, T.R.: **Design Requirements for the NASA Langley Supersonic Low-Disturbance Wind Tunnel.** AIAA Paper 86-0763. Presented at AIAA 14th Aerodynamic Testing Conference, March 1986, 15 pp, 22 refs.

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.

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

279. *Dillon, J.L.; Trimpi, R.L.; and Schultz, A.E.: **The NASA-Langley 20-inch Supersonic Wind Tunnel.** AIAA Paper 86-0765. Presented at the AIAA 14th Aerodynamic Testing Conference, March 1986, 10 pp, 8 refs.

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.

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

280. *Bushnell, D.M.; and Trimpi, R.L.: **Supersonic Wind Tunnel Optimization.** AIAA Paper 86-0773. In: Proceedings of the AIAA 14th Aerodynamic Testing Conference, June 1986, pp. 282-300, 90 refs.

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 (1) 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 or 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.

281. *Park, M.; and Caughey, D.A.: **Transonic Potential Flow in Hyperbolic Nozzles.** AIAA Journal, vol. 24, no. 6, June 1986, pp. 1037-1039, 7 refs.

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.

282. **Marks' Standard Handbook for Mechanical Engineers. Supersonic and Hypersonic Aerodynamics.** Ninth Edition, published by McGraw-Hill Book Co., 1987, pp. 1181-1192, 9 refs.

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

283. *Beckwith, I.E.; Chen, F.-J.; and Malik, M.R.: **Design and Fabrication Requirements for Low Noise Supersonic/Hypersonic Wind Tunnels.** In: Proceedings of Research in Natural Laminar Flow and Laminar-Flow Research Symposium, NASA CP-2487, March 1987, pp. 947-964, 13 refs.

This presentation covers such topics as a proposed supersonic low-disturbance tunnel, quiet test core lengths, effects of surface damage on transition asymmetry in a supersonic nozzle, allowable surface roughness, and the influence of Goertler vortices on nozzle design. The conclusions drawn are that large width two-dimensional rapid expansion nozzles guarantee wide quiet test cores that are well suited for testing models at large angle of attack and for swept wings. Also test results indicate that the surface finish of the pilot nozzle is critical. Experimental data and calculations for Mach 3.0, 3.5 and 5.0 nozzles give N-factors from 6 to 10 for transition caused by Goertler vortices. The use of $N \simeq 9.0$ for the Goertler instability predicts quiet cores in the new Mach 3.5 and Mach 6.0 axisymmetric-long pilot nozzles that are 3 to 4 times longer than observed in the test nozzles to date. The new nozzles utilize a region of radial flow which moves the inflection point far downstream and delays the onset and amplification of Goertler vortices.

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

284. Tsybizov, Y.I.; and Zagvozdkin, V.V: **Flows in Laval Nozzles of Gas Turbine Engines in Steady and Unsteady Operation.** Aviatsion. Tekhnika, vol. 31, no. 3, 1988, pp. 85-87, 3 refs. In Russian. English translation in Soviet Aeronautics published by Allerton Press, pp. 97-100.

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.

285. *Beckwith, I.E.; Chen, F.-J.; and Malik, M.R.: **Design and Fabrication Requirements for Low-Noise Supersonic/Hypersonic Wind Tunnels.** AIAA Paper 88-0143. Presented at AIAA 26th Aerospace Sciences Meeting, January 1988, 11 pp, 22 refs.

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.

286. *Charpin, F.; and Hugouvieux, P.: **Start and Unstart in S2 Supersonic Wind Tunnel in ONERA Modane-Avrieux Center.** Presented at 69th STA Meeting, Los Angeles, May 1988. ONERA TP 1988/9, 31 pp, 1 ref.

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.

287. *Ehrenfried, K.: **Dynamics of Laval Nozzle Flow Under Different Boundary Conditions.** Thesis from Max Planck Institute, May 1988, 74 pp. In German.

N89-22083

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.

288. Guoyao, F.: **Prediction of Two-Dimensional Supersonic Nozzle Flow Field.** Journal of Propulsion Technology, no. 3, June 1988, pp. 6-11, 2 refs. In Chinese.

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.

289. *Beckwith, I.E.; and Bushnell, D.M.: **Wind-Tunnel Simulation of High-Speed Flight Transition.** Presented at the USSR Academy of Sciences International Seminar on Problems of Simulations in Wind Tunnels, July 1988. In: Proceedings, vol. 1, pp. 33-46, 29 refs.

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.

290. *Van Every, D.; Raimondo, S.; and Elfstrom, G.M.: **Impact of Nozzle Design Constraint on Test Section Flow Quality.** In: AGARD Aerodynamic Data Accuracy and Quality: Requirements and Capabilities in Wind Tunnel Testing, FDP Meeting, July 1988, 11 pp, 11 refs.

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.

291. *Argrow, B.M.; and Emanuel, G.: **Comparison of Minimum Length Nozzles.** Journal of Fluids Engineering, vol. 110, no. 3, September 1988, pp. 283-288, 11 refs.

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.

292. Yamanaka, S.: **Transonic Flow in Throat Region of Supersonic Nozzles.** Japanese Society for Aeronautical and Space Sciences Journal, vol. 37, no. 428, 1989, pp. 451-455, 8 refs. In Japanese.

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.

293. *Schnerr, G.; and Dohrmann, U.: **Theoretical and Experimental Investigation of 2-D Diabatic Transonic and Supersonic Flow Fields.** In: Proceedings of IUTAM Symposium Transonicum 3 published by Springer, 1989, 10 pp, 18 refs.

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.

294. *Bushnell, D.M.: **Notes on Initial Disturbance Fields for the Transition Problem.** NASP TM-1051, March 1989, 27 pp, 115 refs.

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.

295. *Chen, F.-J.; Malik, M.R.; and Beckwith, I.E.: **Boundary-Layer Transition on a Cone and Flat Plate at Mach 3.5.** AIAA Journal, vol. 27, no. 6, June 1989, pp. 687-693, 33 refs.

A89-36904#

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.

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

296. *Argrow, B.M.; and Emanuel, G.: **A Computational Analysis of the Transonic Flow Field of Two-Dimensional Minimum Length Nozzles.** AIAA Paper 89-1822. Presented at the AIAA 20th Fluid Dynamics, Plasma Dynamics and Lasers Conference, June 1989, 21 pp, 44 refs.

A89-42052#

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 VNAP2 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.

*University of Oklahoma, Norman, Oklahoma, USA.

297. *Lee, C.Y.; McCool, R.C.; and Chitwood, N.M.: **Actively Cooled Panel Characteristics Study.** AIAA Paper 89-1726. Presented at AIAA 24th Thermophysics Conference, June 1989, 12 pp, 2 refs.

A89-43241#

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.

298. *Chen, F.-J.; Malik, M.R.; and Beckwith, I.E.: **Advanced Mach 3.5 Axisymmetric Quiet Nozzle.** High Technology Report no. HTC-8903, August 1989, 29 pp, 23 refs.

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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