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High Reynolds Number Research - 1980

Proceedings of a workshop held at
NASA Langley Research Center
Hampton, Virginia
December 9-11, 1980

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PREFACE

The 1980 Workshop on High Reynolds Number Research, representing an update of its 1976 counterpart documented in NASA CP-2009, was organized to provide a national forum for interchange of plans and ideas for future research in the high Reynolds number area. In addition, the workshop provided an opportunity for potential users to review the operational characteristics, design features, and initial calibration plans of the National Transonic Facility (NTF), which is now in the final phases of construction at the Langley Research Center. Applicable operational experience with the Langley 0.3-Meter Transonic Cryogenic Tunnel was also reviewed. Since the NTF is truly a national facility, participants in the workshop included technical experts representing a cross section of potential users from NASA, DOD, other governmental agencies, the aerospace industry, and the university community. A list of the attendees is included in this volume.

The basic purpose of the workshop was the examination of the fundamental aerodynamic questions for which high Reynolds number experimental capability is required. A directed effort was made to outline and prioritize potential experiments which would maximize the early research returns from the use of the National Transonic Facility. In addition, NTF calibration plans were reviewed and submitted for comment. The recommendations of the technical panels are recorded in this conference publication.

The 1980 workshop was expanded to the following seven technical panels for initial research planning:

- Fluid Dynamics
- High Lift
- Configuration Aerodynamics
- Aeroelasticity and Unsteady Aerodynamics
- Wind Tunnel/Flight Correlation
- Space Vehicles
- Theoretical Aerodynamics

The 1980 Workshop on High Reynolds Number Research was sponsored by the NASA Langley Research Center in association with the Joint Institute for Advancement of Flight Sciences, the George Washington University. The editors wish to express their appreciation to the chairmen and technical secretaries of the various panels for coordinating the panel discussions and for presentation and documentation of the panel recommendations. The active support of the potential users of NTF in attending the workshop and their recommendations made this workshop a success. Special recognition is due Raymond Siewert of the Department of Defense (ODDRE) and Clinton Brown of NASA Headquarters (OAST), who took time from their busy schedules to participate in the workshop proceedings.

L. Wayne McKinney
Donald D. Baals

CONTENTS

THE NATIONAL TRANSONIC FACILITY:

REVIEW AND STATUS REPORT

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INTRODUCTION

This paper will provide an update on the status of the National Transonic Facility (NTF) Project (refs. 1 to 4). It will also review some of the pertinent performance characteristics and features that the new wind tunnel will have. Finally, it will discuss the liquid nitrogen supply, current views of user cost, and the activation schedule.

STATUS OF PROJECT

Review of Pertinent Features

The NTF (fig. 1) is a closed circuit transonic pressure tunnel designed to provide tests at or near full-scale Reynolds numbers. High Reynolds numbers are achieved by using a very cold (cryogenic) test medium. In the case of the NTF, cryogenic temperatures are obtained by spraying liquid nitrogen, stored in the right cylindrical tank in the upper right of the picture, into the circuit and using the heat of vaporization to reduce or maintain temperature. Temperature increase is provided by the heat of compression from the drive fan. A venting system attached at the low speed end of the circuit is used to control pressure. The gas discharged from the vent is released through a 37-m vertical stack at the left of the picture. The tunnel circuit (fig. 2) is about 61 meters between centerlines in the long direction and about 14.6 meters between centerlines in the short direction. The aerodynamic design is compact and efficient. The short length is achieved by using a rapid diffuser which requires a uniform resistance across the discharge end to assure that it flows full. A cooling coil is used to produce the required resistance, thus affording the added capability to operate the tunnel at near atmospheric temperature without the need for liquid nitrogen. The cooling coil is followed by four antiturbulence screens and a 15-to-1 contraction ratio to assure low turbulence levels in the test flow. A slotted test section is used to produce the transonic test capability. The high speed and low speed diffusers are near optimum conical diffusers.

Inasmuch as the tunnel must operate at and maintain cold temperatures for long periods of time, the gas volume must be thermally insulated. The insulation system used in the NTF design (fig. 3) is a rigidized closed cell foam bonded to the inside surface of the tunnel shell. Using the insulation inside the shell places the shell mass outside the thermally controlled volume and avoids the energy cost to control the shell temperature. Although the insulation is bonded to the shell, there is an additional mechanical retaining system which prevents the insulation from getting into the flow if the bond fails and provides a liner that is the interface with the moving gas. The insulation and liner system is designed to accommodate the very large temperature differences between the internal gas and the tunnel shell.

There are a number of internal structures (fig. 4) that because of their function must feel the gas temperature. These structures are made of aluminum alloy and are designed to accommodate the temperature change rates required for foreseen testing needs. Because of their size, the large internal structures were placed in the tunnel shell in the sequence of fabrication. (See fig. 5 for example.)

Status of Construction

Currently, all of the large internal structural components are inside the shell. The only structure that remains to be delivered to the site is the test section which is scheduled for delivery in April 1981. The pressure shell which was designed, fabricated, and tested in accordance with the ASME pressure vessel code is complete. An aerial photograph of the site (fig. 6) taken November 1, 1980, shows the liquid nitrogen tank complete and the shell complete. The building addition which provides space for model preparation, control room, etc. is being erected, and the vent system foundations and stack components are in the foreground.

The NTF Project master schedule (fig. 7) indicates about 1 year remaining in the construction schedule. Most of the remaining work is associated with the installation and alignment of the internal structures, the installation of the insulation system, and the alignment of the electrical hookup and plumbing required to complete the systems. Note that two major reviews remain to be held. The first is the Integrated Systems Review (ISR), which is required before activation of discrete systems. The second review is the Operational Readiness Review, which is required before the facility is turned over to the user for operation as a testing tool. If the schedule is maintained, the NTF should be completed in 1982.

REVIEW OF PERFORMANCE

The basic characteristics and capability of the NTF were established by an ad hoc facilities panel at the request of the joint NASA/Air Force Astronautics and Aeronautics Coordinating Board (AACB). The characteristics and capabilities approved by the AACB are shown in figure 8. Note that the concept for achieving high Reynolds number, the size of the tunnel test section, the Mach number range, the Reynolds number capability, and the productivity were all stipulated. These requirements have been regarded as the minimum acceptable in the design of the tunnel.

The cryogenic approach to achieving high Reynolds number provides a degree of that temperature control which has not been available in large wind tunnels heretofore; a typical example of the operating envelope at a fixed Mach number is shown as figure 9. The envelope is for Mach 0.8 and is shown in terms of operating stagnation pressure versus Reynolds number produced for operating isotherms. The boundaries of the envelope defined by dotted lines are the result of operating constraints. On the left of the envelope, the constraint is inlet guide vane performance. Variable inlet guide vanes are used to control Mach number at power levels above 48×10^6 watts. Above 48×10^6 watts a synchronous motor is employed which rotates the compressor fan at a fixed RPM. The inlet guide vane design and movement determine the range of temperature over which the Mach number can be maintained. In the upper left corner a power limit defines the boundary. At the top of the envelope, the maximum operating pressure for the shell (9 bars) defines the boundary. At the right, liquefication in the test stream, defined by a local Mach number of 1.4, limits the temperature for operation. More recent experimental work indicates that this $M_L = 1.4$ limit is very conservative. The shaded portion at the bottom of the envelope shows the part that can be covered with air in the circuit if the drive system is the only constraint. In reality the cooling coil system will limit the operating pressure to near two atmospheres.

If the maximum Reynolds number at each Mach number is plotted, the overall performance of the NTF is obtained (fig. 10). The boundary on the left is the maximum working pressure for the shell (9 bars), the slanting boundary at the upper right is a power limit (93×10^6 watts to the fan), and the vertical boundary is a fan compression ratio limit. The vertical dashed line at $M = 1.0$ indicates the design goal of $RN_{\bar{c}} = 120 \times 10^6$ can be met.

The lines of constant dynamic pressure are shown on figure 10 to indicate how dynamic pressure will vary with Reynolds number and to show that at the higher Reynolds numbers, the NTF will have dynamic pressures larger than existing tunnels. This fact will have to be accommodated as required in the design of models and support systems. The ability of the NTF to perform full-scale Reynolds number testing is indicated in figure 11. In this figure, a comparison is made of the flight and NTF test envelopes for the Boeing 747 air transport. On the left, the flight envelope is presented in the conventional form of altitude versus velocity (Mach number). The boundaries of this envelope are formed by a thrust limit on the right and C_L max on the left. The upper end closure is defined by buffet and/or stability constraint. Note the design cruise point indicated by the open circle located at an altitude of about 9×10^3 meters, $M = 0.81$ and $RN = 54 \times 10^6$. The plot of $RN_{\bar{c}}$ versus Mach number on the right shows the test capability of the NTF with the airplane scaled to the proper model size for testing. The flight envelope transformed to this plot is represented by the dashed boundary identified as the 747 envelope. On this plot, the zero altitude flight is the straight side of the envelope on the left. The shaded area on both plots indicates the portion of the full-scale flight envelope that can be simulated in the NTF.

The cross-hatched envelope at the bottom of the figure on the right shows the Reynolds number capability of all existing wind tunnels. The improvement in Reynolds number capability with the NTF is obvious. It should be noted that the NTF will not be capable of covering the complete flight envelope of all airplanes. It appears, however, that it will cover the important region of cruise and high performance for most airplanes and will in all cases provide Reynolds numbers close to full scale such that Reynolds number extrapolation errors will be minimal.

PRODUCTIVITY/EFFICIENCY FEATURES OF THE NTF

Automated Data Acquisition and Controls

Productivity and efficiency were goals of the NTF design. To assure that the electronic equipment was not a source of large losses of operating time, the computer complex (fig. 12) was designed around four moderate size computers rather than one large one. These computers are paired: two to handle data acquisition and management and two to handle tunnel process controls and monitoring. All four computers are linked together to permit cross communication as required for efficient control and rapid response. Additionally, each pair of computers is tied together through a bus transfer switch. These switches permit one computer to take on the essential parts of the work load of both in the event one of the two goes out. This feature allows continued tunnel operation when a single computer is out. There is a set of peripheral equipment that can be shared among the computers to display, record, or plot information or data in a predetermined manner.

The basic data acquisition capability for the NTF is shown as table I. The system of channels of analog and digital data is modular and can be enlarged at a later date if desired. By use of the miniaturized electronic sensing pressure measurement system, two analog channels permit accurate recording of steady pressures from 1024 pressure ports.

TABLE I.- DATA ACQUISITION AND DISPLAY CAPABILITY FOR NTF

(a) Acquisition

Data characteristics	Type	Number of channels	
		Test section	Ass'y room
Steady state	Analog*	256	64
	Digital	32	16
Unsteady	Analog	14	14

*1024 pressure measurements.

(b) Display/Reduction

	Type	Form
On line	CRT	Tabular Graphic Hard copy
	Printer	Tabular
Off-line		Batch data analysis Plotting

Plenum Access System

As mentioned previously, the tunnel shell is insulated to contain cryogenic gas at pressure for long periods of time. An isolation system has been designed to allow personnel access to the plenum volume without having to discharge the cold gas from the entire tunnel circuit. This isolation system (fig. 13) requires that the contraction cone and high-speed diffuser be moved away from the plenum bulkheads and large domed heads (isolation valves) be placed in the bulkhead openings. With these isolation valves in the closed position, the plenum volume can be vented to reduce the plenum pressure to atmospheric. Dry air is circulated in the volume to bring the oxygen constituency and temperature to the required level for personnel entry. There are 2.7-m by 3.7-m rectangular doors on either side of the plenum to permit personnel entry for model change and for service to actuators and instrumentation inside the plenum volume.

To return the tunnel to operational status, the access doors are closed and the pressure is equalized across the bulkheads. The isolation valves are returned to their stored position and the contraction cone and high-speed diffuser are placed in their operating position. A clamping system is used to attach these components as well as the isolation valves to the bulkheads (ref. 5).

The warm-up and cool-down of the large structural elements inside the plenum are time consuming because of constraints on the rate of their temperature change to control thermal stress levels. As a consequence, plenum access is planned only when absolutely necessary.

Model Access

A model access system (ref. 5) is provided (figs. 14 and 15) to permit relatively rapid access to the model for configuration change during a test program. The system required the plenum access to be implemented to the point of reducing the plenum pressure to atmospheric. It does not require the plenum volume to be purged and warmed. With the plenum pressure at atmospheric and two rectangular access tubes in place on either side of the plenum, the 2.7-m by 3.7-m doors are opened and the test section side walls are lowered. The two access tubes are then inserted until they meet at the centerline capturing the model inside the rectangular volume. The tubes seal at the butt joint where they come together and around the sting. The volume inside the tubes is then purged with dry air to bring the oxygen and temperature to the required level and the model surface is warmed to avoid condensation when outside air is admitted. Once internal conditions meet requirements, the doors at the outer ends of the tubes can be opened to provide a through passage and a 2.1-m high by 3-m wide work space around the model. Upon completion of the model changes, the return to operation is accomplished by reversing the described process. The estimated time for model access is shown in figure 16. Because there was no change in the temperature of the large structures in the plenum during model access, little time is required other than for the movement of components. An average time of model configuration changes is anticipated to be about 2 hours.

USER INTERFACES

As a result of the facility being designated as a national facility and the anticipation that it will be used extensively by organizations other than NASA, a process for model delivery, checkout, installation, testing, and recrating has been developed and used in the facility design. The building addition (fig. 17) will provide three rooms on the first floor designated as user space which can be isolated to accommodate classified or proprietary models. The model will be delivered into one of these rooms uncrated and assembled on its support sting. The sting will be supported on a backstop (fig. 18) which will simulate the model support strut in the tunnel in that it will provide the ability to level and roll the model. Weight baskets will be provided to check the calibration of strain gage balances under combined loads with or without a cryogenic environment. The cryogenic environment is provided by a portable cryogenic chamber which will enclose the model. The outputs from the strain gage balance and all other on-board instrumentation can be patched to the data acquisition system located in the control room directly above the user space.

When the model is ready to be installed in the tunnel, it will be placed on a model handling cart (fig. 19), moved by elevator to the second floor, and installed in the tunnel. The model handling cart floats on air bearings and is designed to provide easy alignment of stings with the model support sting (fig. 20) and with the backstop in the user spaces (fig. 18). Three-dimensional models whose aerodynamic loads do not exceed those shown at the bottom of figure 20 can be accommodated. The model roll capability in the tunnel is $\pm 180^{\circ}$ at a rate of 10° per second.

A side wall mount (fig. 21) will also be provided as a support for half models. Models will be mounted on a splitter plate which will span the tunnel height (2.5 m). The angle-of-attack capability is $\pm 180^{\circ}$ for this system. The support also provides the capability of oscillation in pitch at frequencies from 1 to 100 Hz and amplitude from 0.1° to 1.0° for the angle-of-attack range of $\pm 10^{\circ}$. The load capability of this system is also shown in figure 21.

A jet simulation system will be provided (see fig. 22). This system is currently being designed to permit jet simulation at a pressure ratio of 8.1, $M = 0.9$, and maximum Reynolds number with temperature control from 100 K to 339 K and an angle-of-attack range from -2° to $+10^{\circ}$. Although it will be designed initially to support three-dimensional models, it can also be routed (with piping change) to provide jet simulation for half models tests.

LIQUID NITROGEN SUPPLY

Liquid nitrogen for the NTF operation is to be purchased through the Air Force San Antonio Air Logistic Center at Kelly Air Base in Texas. Because of the large quantities of LN_2 required, the Air Force has recently awarded a contract to Union Carbide Corp., which is in the process of building a new LN_2 plant in close proximity to the NTF site and will pipe LN_2 to an on-site storage facility. The characteristics of the LN_2 fill system are shown in figure 23. Note that the plant site will have its own bulk storage system which will act as a buffer for erratic operation of the tunnel. Additionally, the capability of off-loading over-the-road LN_2 tanker trucks is provided at the site against untimely plant shut down. The system assures an LN_2 supply with considerable flexibility.

USER COSTS

User costs have been a continuing concern throughout the design of the NTF. Now that we have completed the design and are developing operating sequences and procedures and establishing manpower and time requirements for the operation, we have tried to reevaluate the cost of the NTF operation and put the user cost in perspective.

There are two parts to the user cost for NTF testing. The first part is what is known as occupancy cost, which is fixed overhead and burden. This cost deals primarily with the expense of the personnel, management, and supplies necessary to operate and maintain the facility. The second part of the cost is called energy cost. This cost is the expense of the electric power and liquid nitrogen consumed in the user's test program. The occupancy cost has been estimated for the NTF and is shown in figure 24. The format is the same as that used for unitary plan wind tunnels and shows the total cost of civil service labor and the overhead rates that are applied. The other charges are estimates for annual consumption of materials, supplies, and maintenance based on experience with other wind tunnels. The contract support reflects the planned use of a support service contract for the operation of the ancillary systems. The total annual cost is estimated to be $3,800,000 (1983 money). Based on operation for 46 weeks out of the year, the weekly operating cost is $83,000 or about $8,330 per shift for a 5-day week (two shifts/day).

The user energy cost is solely program dependent. In the process of developing the NTF requirements, a number of different testing programs were developed and evaluated. These programs, which are believed to be typical for specific testing objectives, are shown in figure 25. Note that the list includes two different subsonic transports, a supersonic transport configuration, two fighter test programs, and a basic research program. Each of these programs was developed in considerable detail to assure a credible estimate of the time and costs for its implementation. To provide a typical program for discussion in this paper, the subsonic transport no. 2 was selected. The testing plan for this model (fig. 26) was developed without regard for cost and is based on experience with typical development programs in other wind tunnels. As can be noted, the effects of Reynolds number are explored extensively in the beginning of the program. Then the remainder of the testing is done at a selected Reynolds number which in this case was relatively high. Also note that the program is rather large, 142 data polars.

To evaluate the cost of the program, certain assumptions were made as shown in figure 27. A pitch pause mode was selected for data acquisition with 5 seconds required at each angle of attack. An angle-of-attack range to 20° was selected with 13 discrete angles for data measurement. The total testing time required for each data polar was 78 seconds. The time required for each model configuration change was estimated to be 2 hours, and the model change required two shifts or 16 hours. Based on these assumptions, the program testing cost is as shown in figure 28. The liquid nitrogen consumed was 2837 metric tons. The applied cost of $95 per metric ton is believed realistic for the next 5 years. The cost of liquid nitrogen was $266,678. The electric power costs were $7,753 for a total energy cost of $274,431.

The occupancy time was estimated to be about seven shifts for a total cost of $58,000 and the overall cost for the program was computed to be $332,531. For 142 data polars, the average total cost was $2,300 per polar.

As mentioned previously, this was a relatively large program and could probably be reduced considerably if costs were a concern. Additional cost reduction could be achieved by minimizing the Reynolds number for developmental work where possible and testing only the final configuration at high Reynolds number.

ACTIVATION SCHEDULE

The NTF activation schedule is shown in figure 29. As mentioned in the discussion of the project status, the integrated systems review is currently scheduled to be held at the end of 1981. That review will be followed by a period of checkout and systems integration, bringing the tunnel to full operational readiness in the latter part of 1982. At that point there will be a trained operating crew capable of one shift per day operation. Two shifts per day operation will be available by mid 1983.

A user's guide has been written and is in the editorial process. This guide should be published and issued by the end of the first quarter of 1981. It is planned to update this guide to incorporate experimental data and experience with the operating tunnel.

Tunnel calibration will start with the completion of the operational readiness review. The period that has been set aside to assure a good definitive calibration is 18 months and includes some testing of correlation models. The target date for the tunnel to be available to do work is in the first quarter of 1984.

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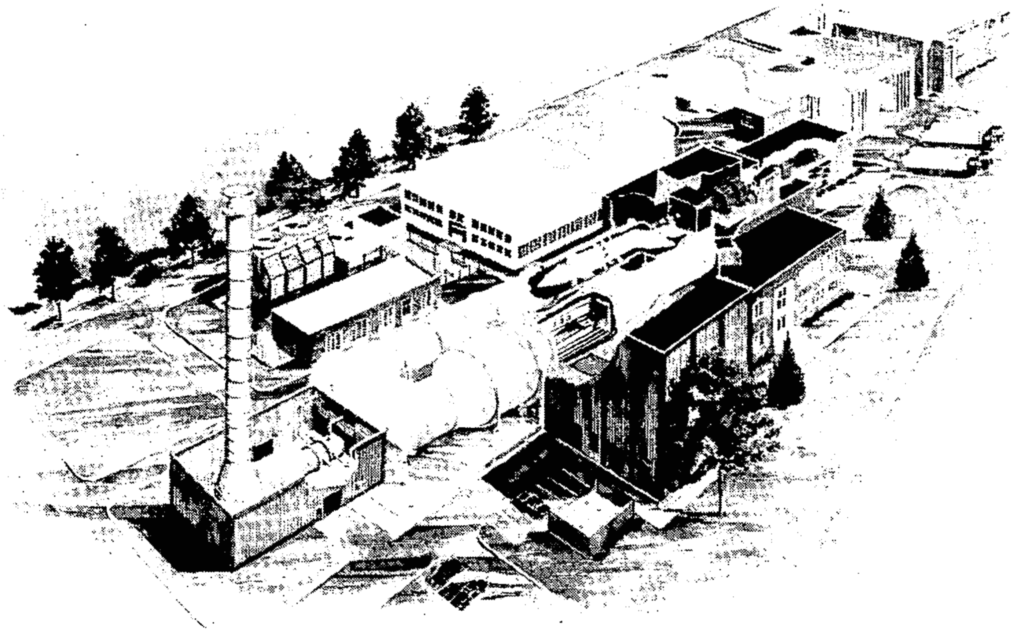

Figure 1.- Artist's perspective of National Transonic Facility site.

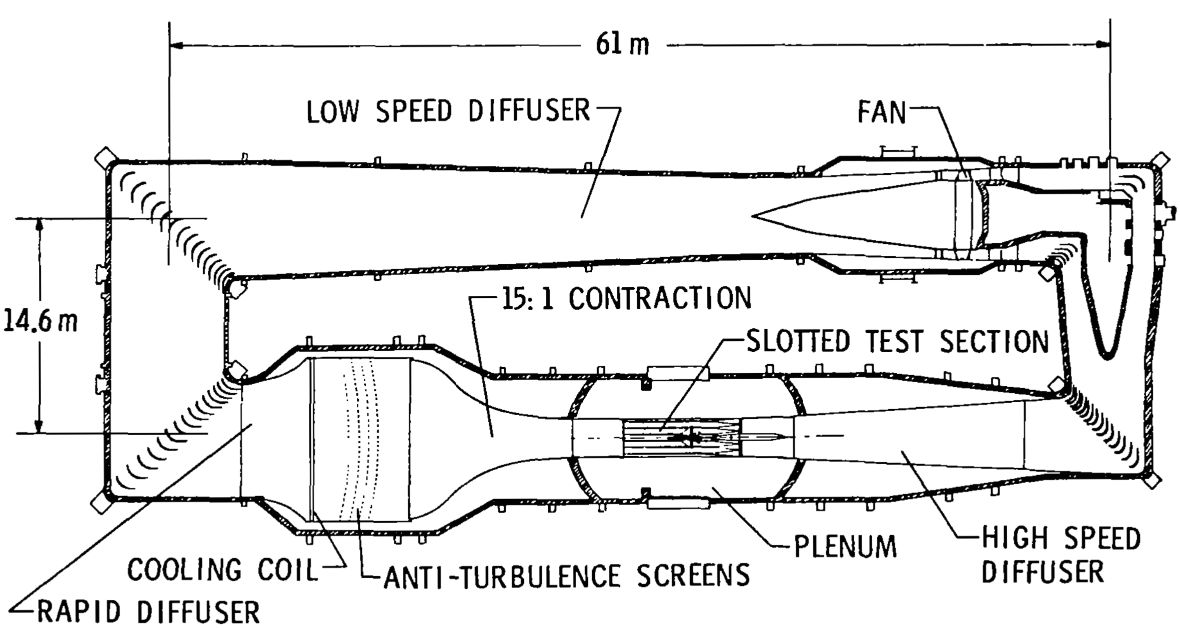

Figure 2.- Plan view of NTF tunnel circuit showing basic components and arrangements.

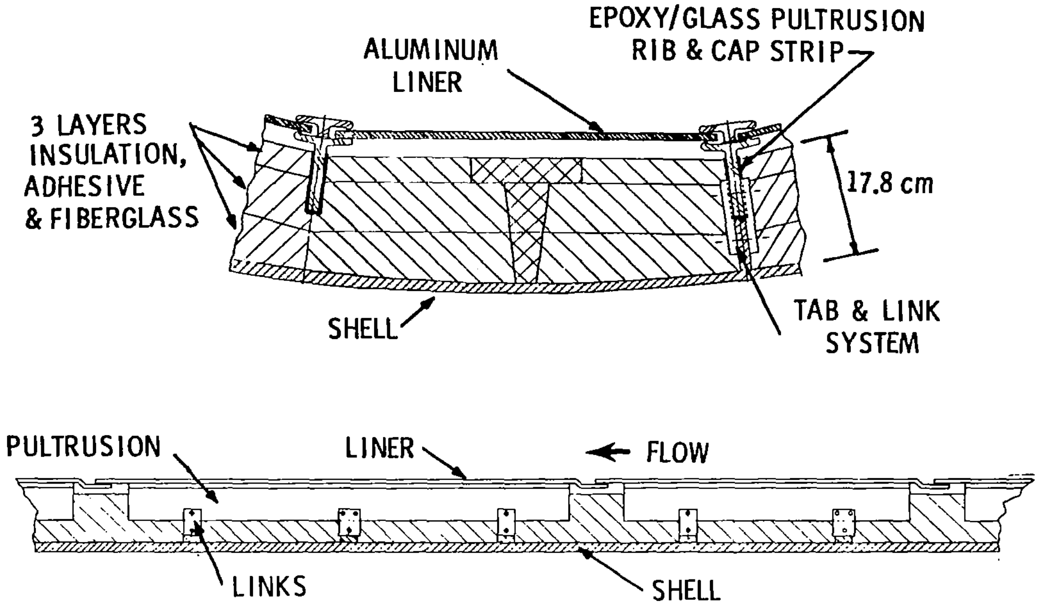

Figure 3.- Sketch of cross sectional and longitudinal views of insulation and liner system for NTF.

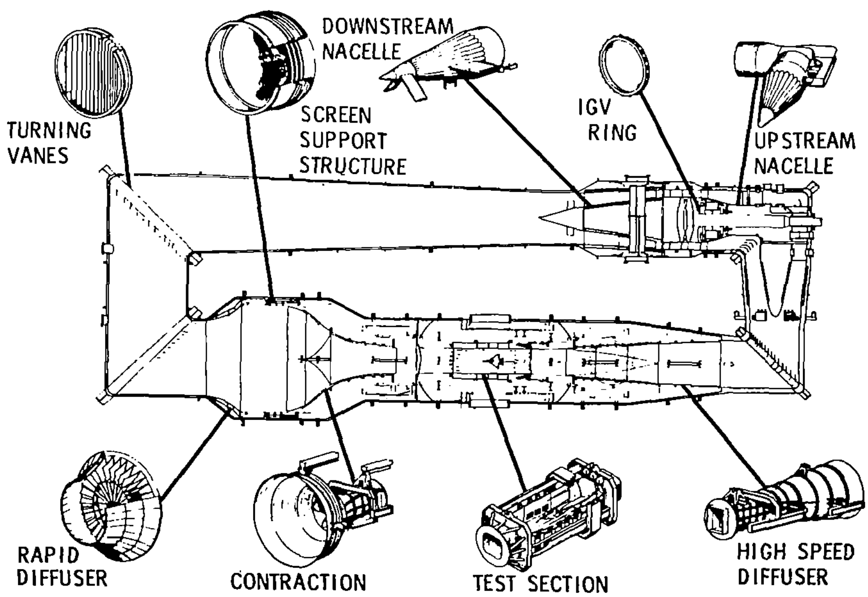

Figure 4.- Diagram of NTF circuit showing internal structural components.

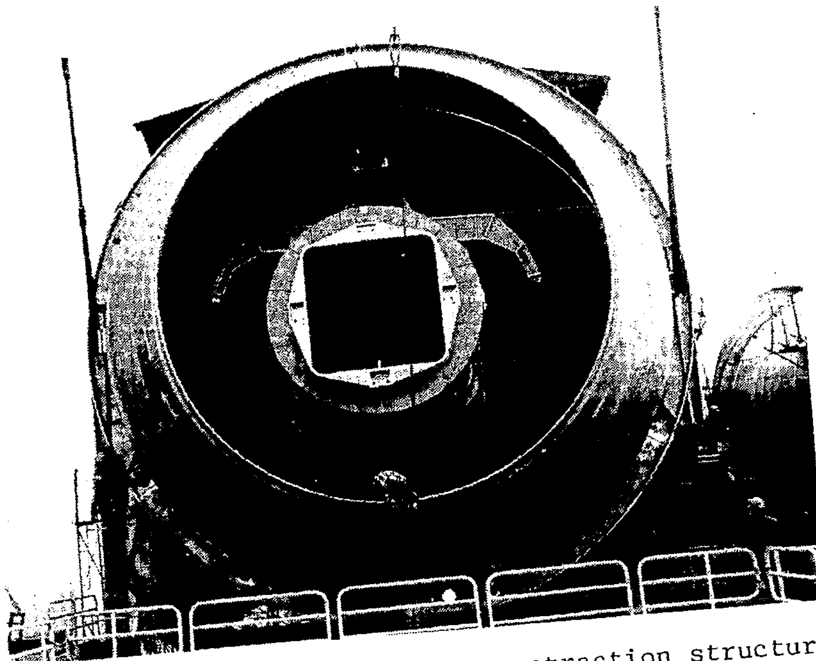

Figure 5.- Photograph showing contraction structure placed inside tunnel pressure shell.

Figure 6.- Aerial photograph of construction site as it was November 1, 1980.

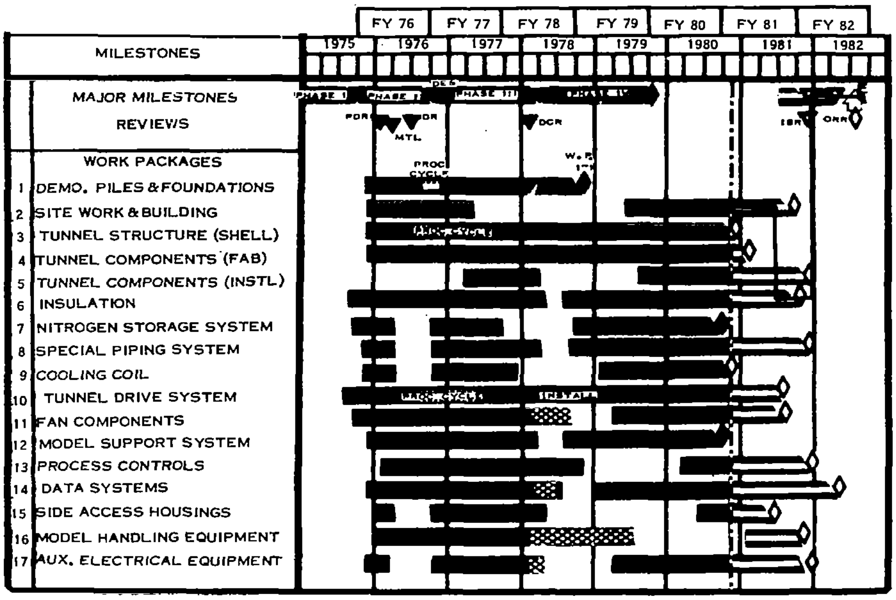

Figure 7.- Master schedule for NTF project as of November 24, 1980.

A SINGLE TRANSONIC TEST FACILITY IDENTIFIED AS THE NATIONAL TRANSONIC FACILITY (NTF)

CRYOGENIC CONCEPT

CHARACTERISTICS:

TEST SECTION SIZE	2.5 m SQUARE
DESIGN PRESSURE	9 BAR
DESIGN MACH NUMBER RANGE	0.2 - 1.2
STREAM FLUID	NITROGEN
BASIC DRIVE POWER	9×10^7 W
PRODUCTIVITY/EFFICIENCY	8000 POLARS/yr
REYNOLDS NUMBER	120×10^6 (M = 1.0)

LOCATED AT LANGLEY RESEARCH CENTER

Figure 8.- Requirements for design and performance of National Transonic Facility.

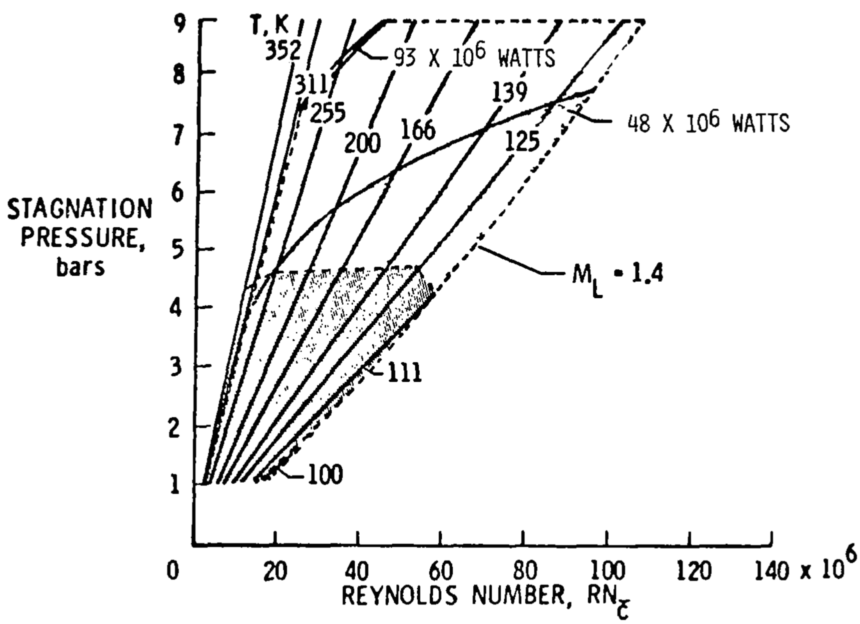

Figure 9.- Operating envelope for NTF.
$M = 0.8$; $\bar{c} = 0.25$ m.

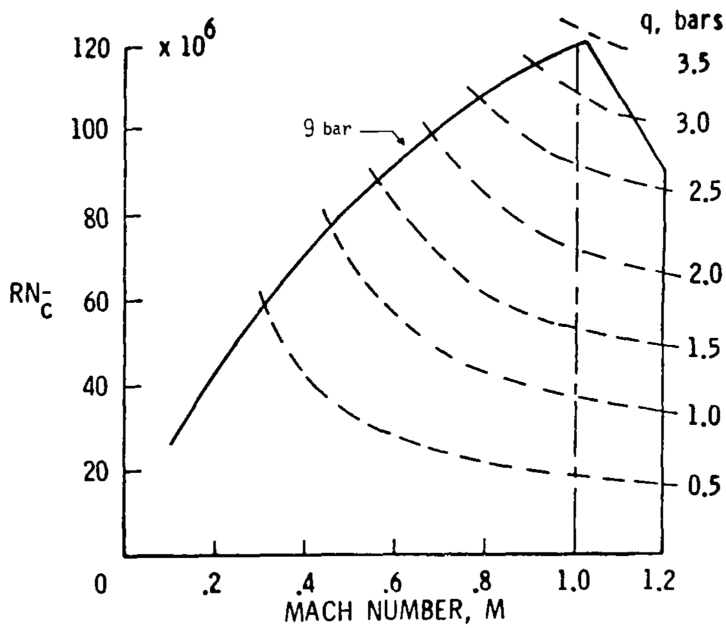

Figure 10.- Overall operating envelope for NTF.
$\bar{c} = 0.25$ m.

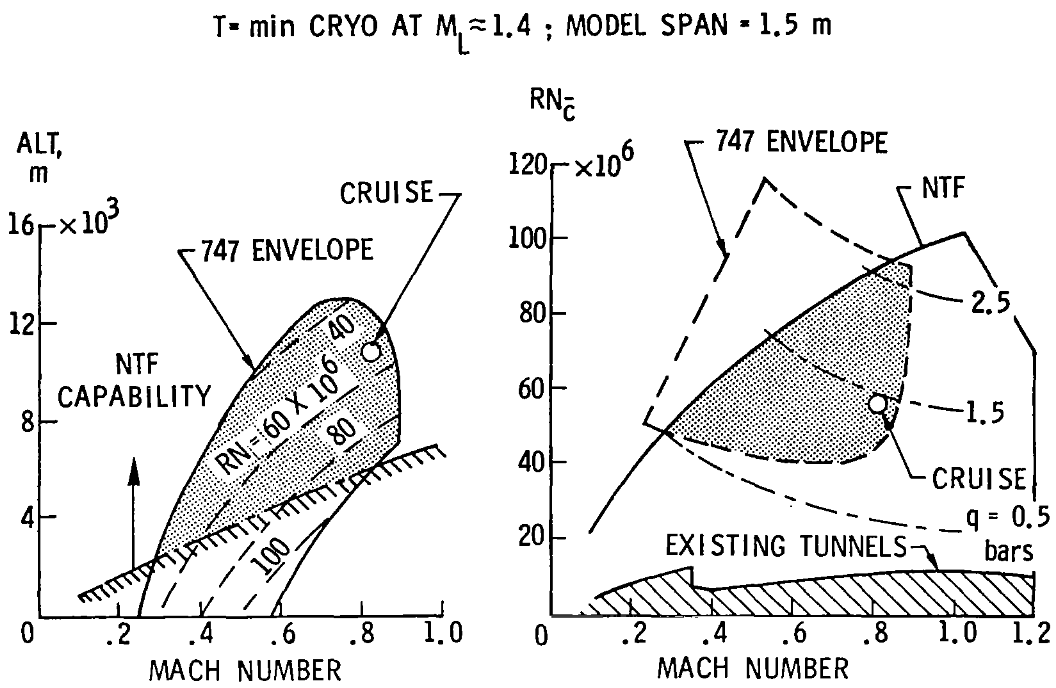

Figure 11.- Comparison of flight envelope of B-747 airplane with NTF test envelope. (Model span = 0.6 test section width.)

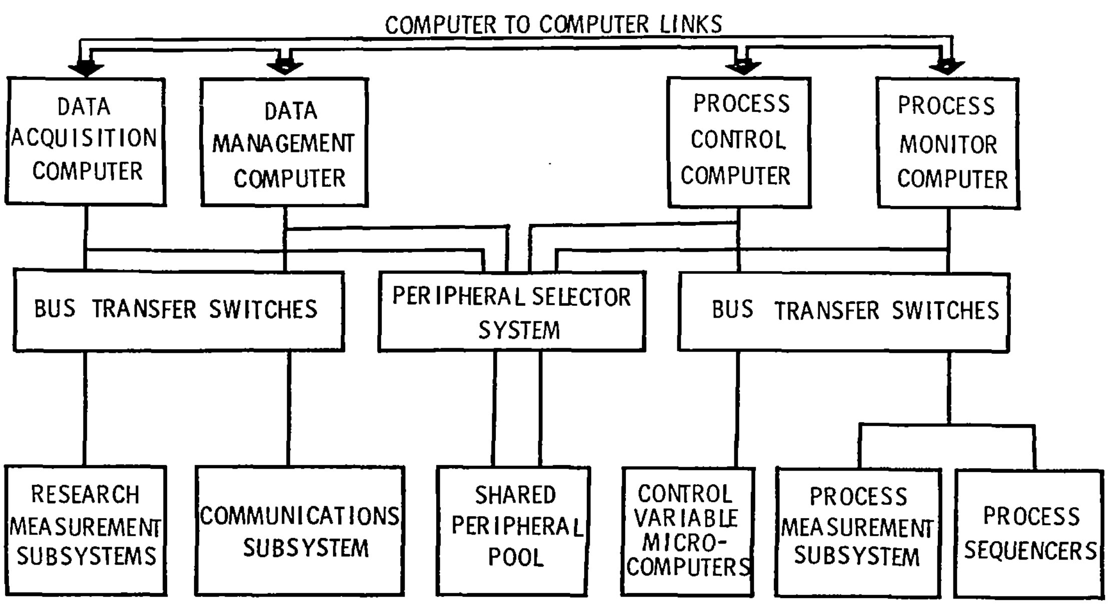

Figure 12.- Schematic diagram of linkage and switching arrangement for NTF computer complex.

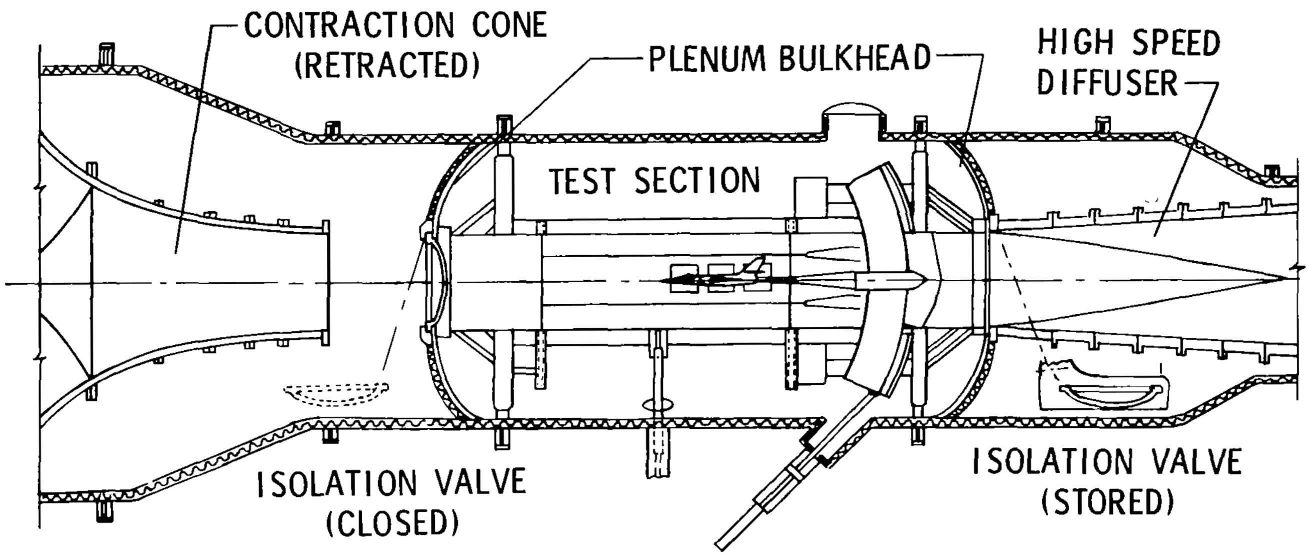

Figure 13.- Sketch of plenum isolation system for National Transonic Facility.

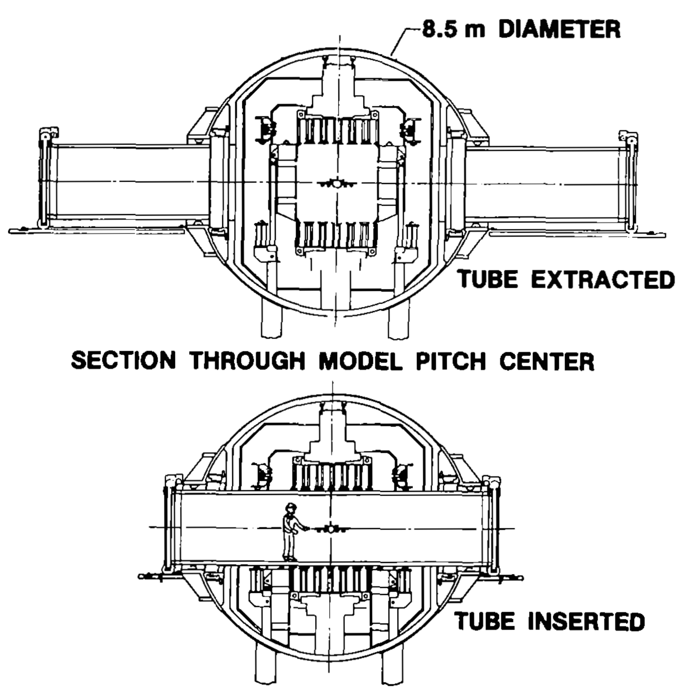

Figure 14.- Sketch showing model access tubes in extracted and inserted positions.

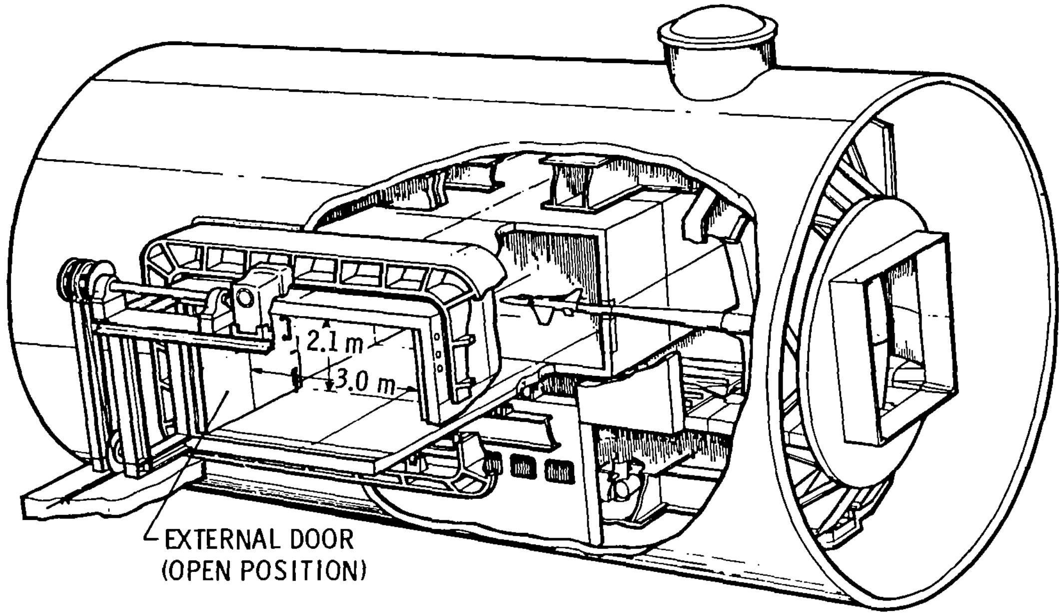

Figure 15.- Isometric sketch showing model access tubes inserted as required for servicing a test model.

FUNCTION	TIME
CONDITION PLENUM	18 min
INSERT TUBES	3 min
CONDITION TUBES/WARM MODEL	37 min
CHANGE/SERVICE MODEL	VARIABLE
PREPARE FOR TUBE EXTRACTION	5 min
RETRACT TUBES	3 min
RETURN TO OPERATING CONDITIONS	18 min
TOTAL	84 min + VARIABLE

Figure 16.- Actuating times for the different processes leading to model access.

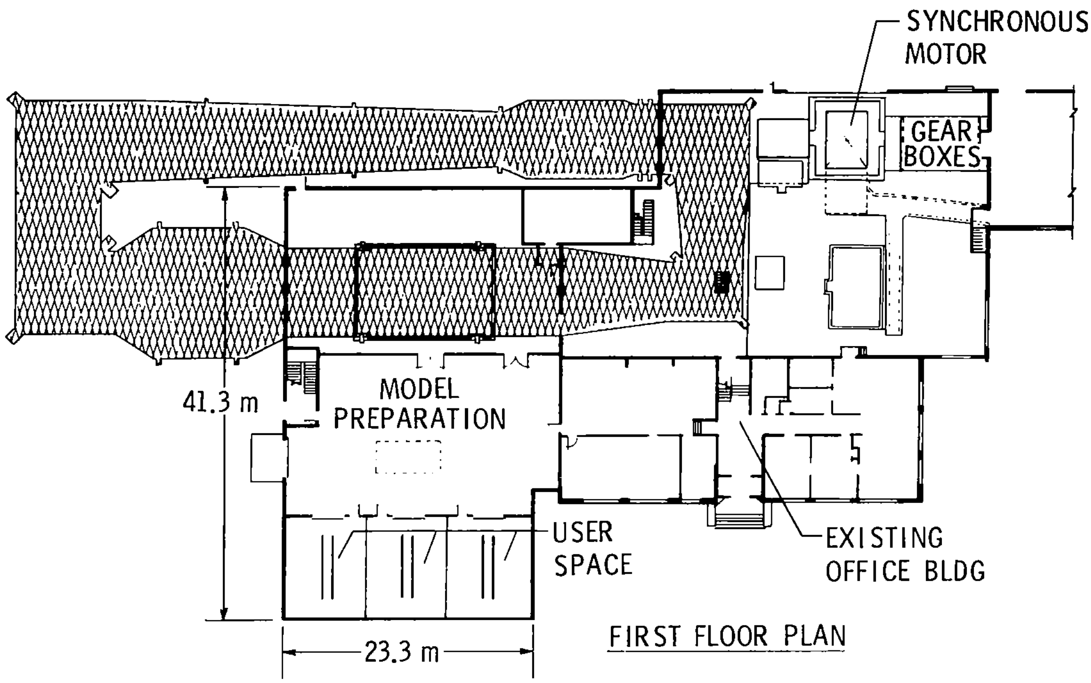

Figure 17.- Sketch showing first floor plan of building addition.

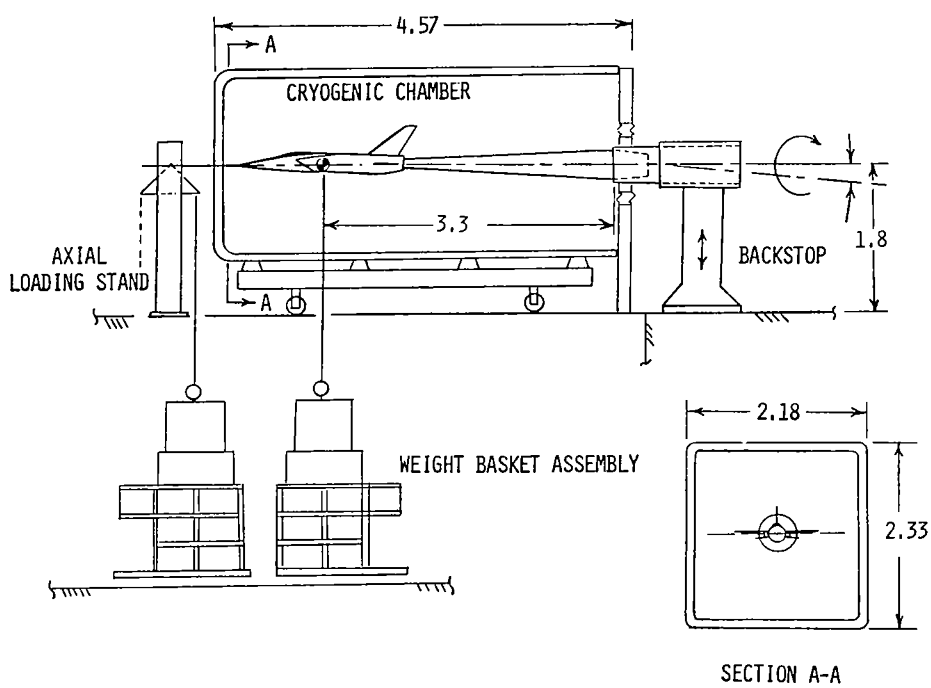

Figure 18.- Sketch showing arrangement for model calibration. All dimensions are in meters.

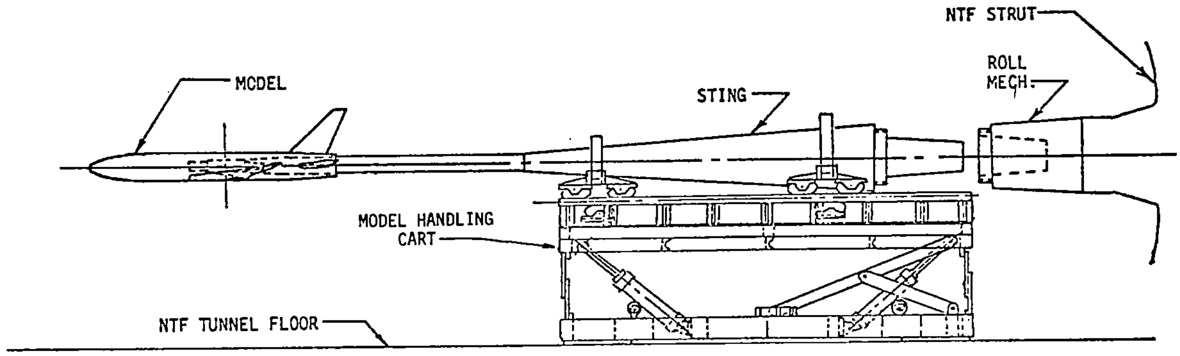

Figure 19.- Sketch of model handling cart used for transporting models within building.

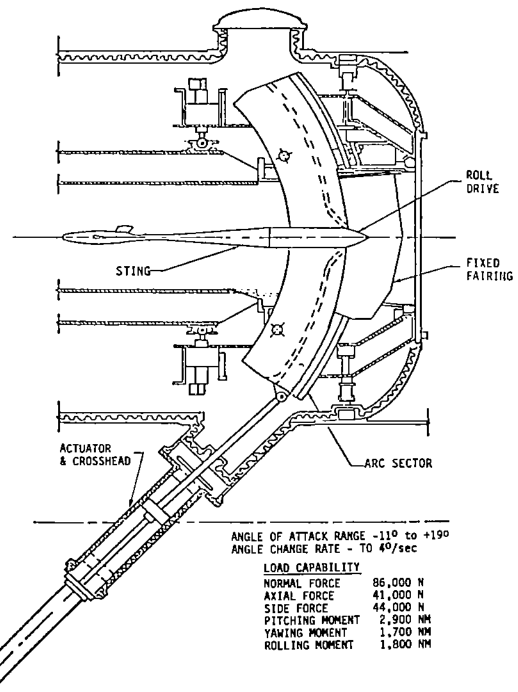

Figure 20.- Sketch of three-dimensional model support and pitch control system.

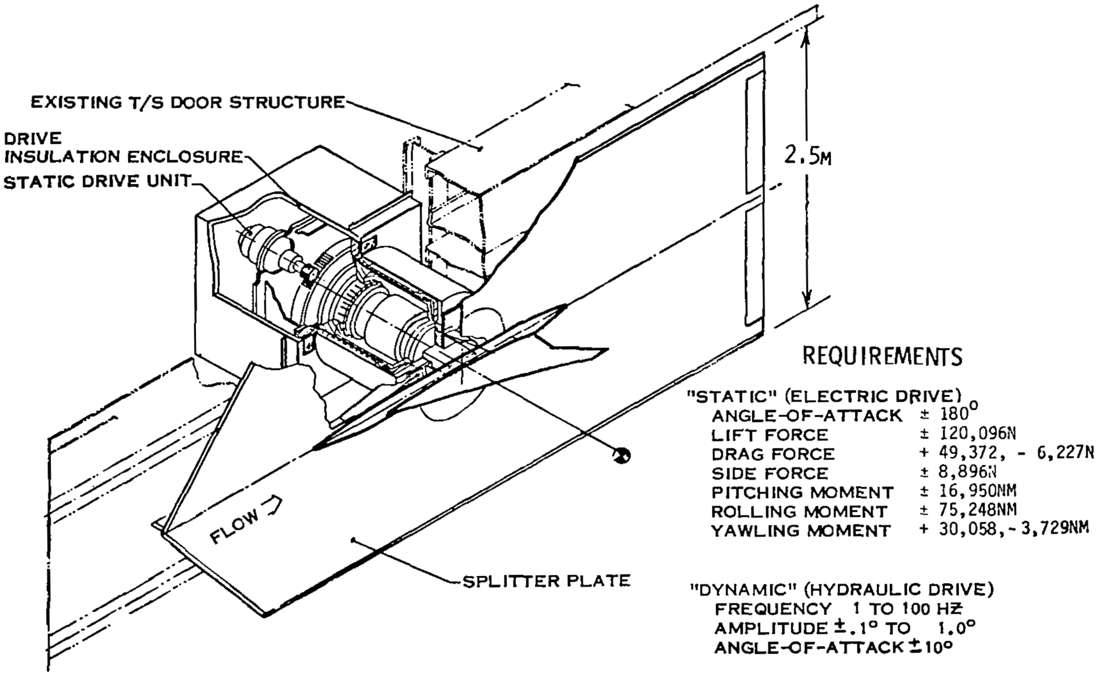

Figure 21.- Sketch of half model support system.

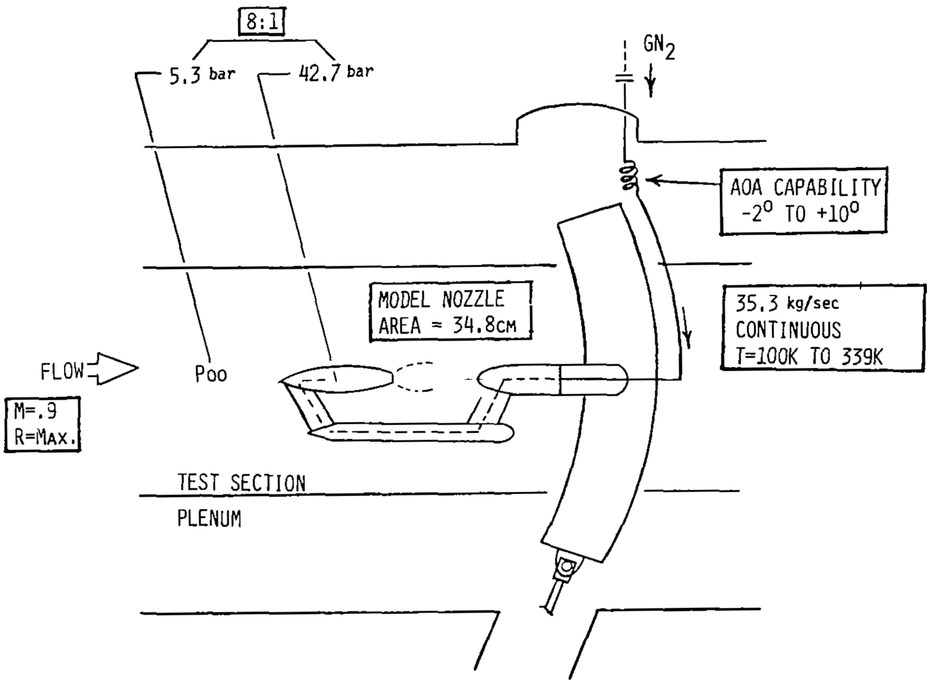

Figure 22.- Schematic of gas supply system for jet exhaust simulation.

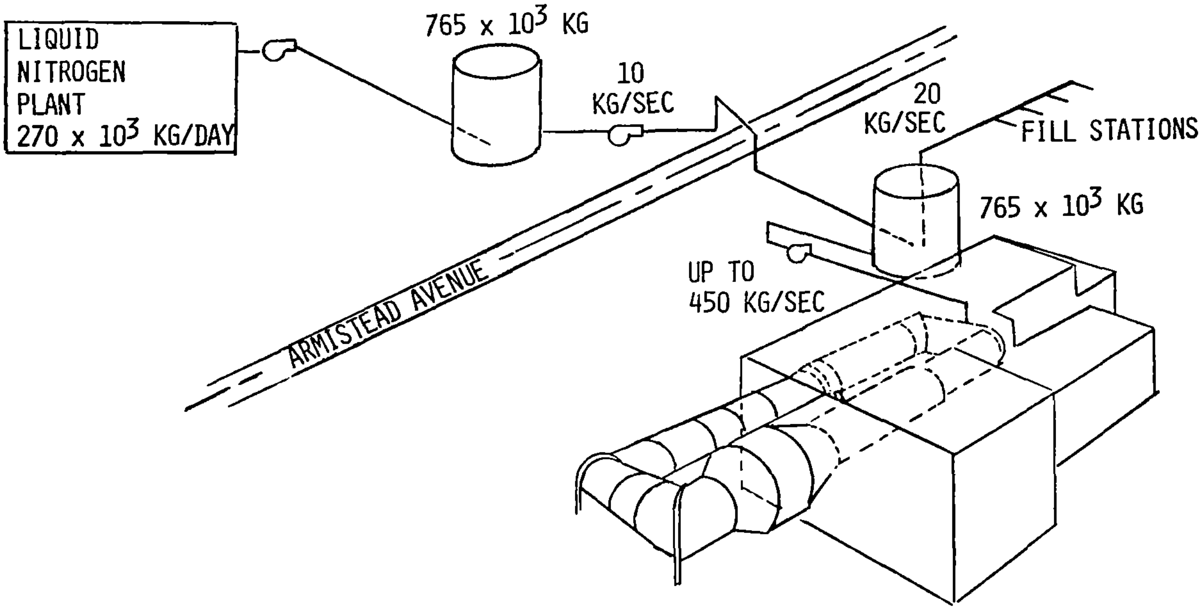

Figure 23.- Schematic of bulk LN_2 supply system for NTF.

LABOR:		
DIRECT	$ 686K/YR.	
INDIRECT	186	
TOTAL LABOR	$ 872K	
LARC SUPPORT OVERHEAD @ 96%	837	
HEADQUARTERS OVERHEAD @ 10%	171	
TOTAL LABOR AND OVERHEAD	$1,880K/YR.	
OTHER CHARGES:		
MATERIALS AND SUPPLIES	$ 105K	
MAINTENANCE	320	
CONTRACT SUPPORT	1,510	
TOTAL OTHER CHARGES	$1,935K/YR.	
TOTAL OCCUPANCY COST/YEAR		$3,815K
*WEEKLY OCCUPANCY RATE		83K

* o BASED ON 46 OPERATING WEEKS PER YEAR, 10 SHIFTS PER WEEK
  o COSTS ARE BASED ON 1983 DOLLARS

Figure 24.- Breakdown of projected occupancy cost for NTF showing annual and weekly rate.

	CONFIGURATION	M	RN x 10^{-6}
1	SUBSONIC TRANSPORT #1	0.5 - 0.85	20 - 80
2	SUBSONIC TRANSPORT #2	0.3 - 0.9	5 - 40
3	SCAR (LOW SPEED)	0.3	25 - 50
4	FIGHTER (MANEUVERING)	.5 - 1.2	20 - 40
5	FIGHTER (STALL-SPIN)	0.6 - 0.9	10 - 50
6	BASIC RN Q & M EFFECTS	0.5 - 0.9	15 - 60

Figure 25.- List of sample test programs developed for use in projecting operational procedures, LN_2 requirements, and electrical power requirements.

CONFIGURATION			M										RN x 10^{-6}						#POLARS
	δ_h	δ_a	.3	.5	.7	.75	.775	.8	.825	.85	.875	.9	5	75	10	20	30	40	
# 1 WFV Grit off	off	0			X			X			X		X	X	X	X	X	X	18
# 2 WFV	off	0			X			X			X		X	X	X	X	X	X	18
# 3 WFVH	0	0			X			X			X						X		3
# 4 WFVHN	0	0			X			X			X		X	X	X	X	X	X	18
# 5 WFVHN-inverted	0	0			X			X			X						X		3
# 6 WFVHN-Tuning	0	0			X			X			X						X		3
# 7 WFVHN-Tuning	0	0			X			X			X						X		3
# 8 WFVHN-Tuning	0	0			X			X			X						X		3
# 9 WFVHN-Tuning	0	0			X			X			X						X		3
#10 WFVHN-Tuning	0	0			X			X			X						X		3
#11 WFVHN-Tuning	0	0			X			X			X						X		3
#12 WFVHN-Tuning	0	0			X			X			X						X		3
#13 WFVHN-Tuning	0	0			X			X			X						X		3
#14 WFVHN	-5	0	X	X	X	X	X	X	X	X	X	X					X		10
#15 WFVHN	-10	0			X			X			X						X		3
#16 WFVHN	+5	0			X			X			X						X		3
#17 WFVHN B=+5^{0}	0	0	X	X	X	X	X	X	X	X	X	X					X		10
WFVHN B=-5^{0}	0	0	X	X	X	X	X	X	X	X	X	X					X		10
WFVHN B range	0	0	X	X	X	X	X	X	X	X	X	X					X		10
WFVHN B=0	0	0			X			X			X						X		3
#18 WFVHN	0	±5			X			X			X						X		3
#19 WFVHN	0	±10			X			X			X						X		3
#20 WFVHN	0	±15			X			X			X						X		3
																		TOTAL POLARS	142

W WING
F FUSELAGE
V V TAIL
H H TAIL
N NACELLES

- Unless noted all polars are angle-of-attack sweeps @ 0^{o} sideslip
- δ_h=horizontal tail deflection
- δ_a=aileron deflection
- Angle-of-attack range to 20^{o} @ 13 points
- Polar time - 78 seconds
- B=Sideslip <

Figure 26.- Testing plan for subsonic transport no. 2.

TIME PER DATA POINT	5 SEC.
ANGLE-OF-ATTACK RANGE	20° (13 POINTS)
TIME PER POLAR	78 SEC.
MODEL ADJUSTMENT	2 HOURS
MODEL CHANGE	16 HOURS

OPERATION IS ON A TWO SHIFT PER DAY BASIS WITH THE TUNNEL STATIC (COLD AND PRESSURIZED) ON THE THIRD SHIFT.

OPERATION IS FOR FIVE DAYS PER WEEK.

Figure 27.- Assumptions used in estimating time required for installing and testing model of subsonic transport no. 2 in NTF.

LIQUID NITROGEN	= 2837 METRIC TONS @ $94/TON	=	$266,678
ELECTRICITY	= 177 000 KWH @ $.0438/KWH	=	7,753
			$274,431
OCCUPANCY TIME ESTIMATED TO BE ON THE ORDER OF 7 SHIFTS - 7 x $8,300		=	$ 58,100
			$332,531

Figure 28.- Breakdown of projected testing program cost for subsonic transport no. 2 tested in NTF.

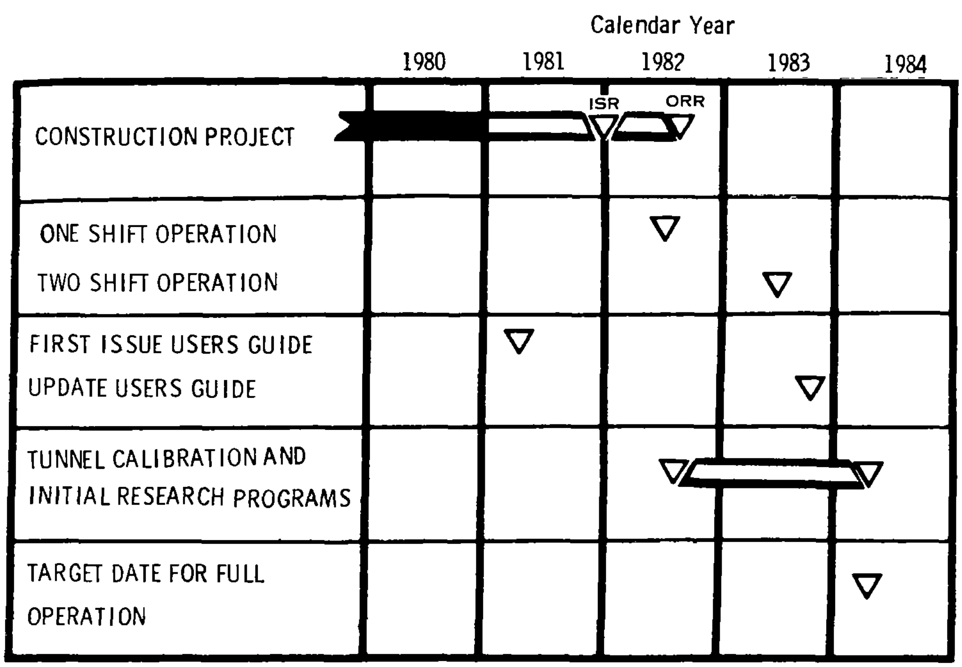

Figure 29.- Current schedule for startup and activation of NTF.

THE NTF AS A NATIONAL FACILITY

Clinton E. Brown
NASA Headquarters

The history of NTF development has been well documented at the first High Reynolds Number Research Workshop held at Langley in 1976. Progress made following the funding of NTF was summarized in the proceedings of last year's Conference on Cryogenic Technology. With the past being well taken care of, I would like to comment briefly on the term "national facility" and to discuss the future evolution of the NTF.

The word "national" has a special connotation indicating the concurrence of NASA, DOD, industry, and academia in its value to the nation as well as signifying its servitude to those four interested groups. The usage of the NTF is of course intended to serve the interest of the nation first and the individual interests of the user groups second; thus, in times of intense military development, greater use may be required for immediate aircraft or missile development whereas otherwise the major usage will be for research and development designed to maintain the national preeminence in aeronautics. The "national" label also infers and accentuates the idea of openness to the aeronautical community even though all of NASA's facilities hopefully are available for the exploitation of new ideas and concepts.

As custodian of this unique and advanced aerodynamic test facility, the Langley Research Center is charged with the task of maintaining the excellence of the test data as well as operating the facility with maximum productivity to increase the utility as well as to minimize the test costs. In regard to costs for testing, present projections indicate a number around $6,000 per hour and it follows that careful planning and carrying out of test programs is a requirement. NASA is aware that costs for aircraft development in wind tunnels have shown a steady rise over the years even in the face of the rapidly developing science of computational aerodynamics. The trend is illustrated in figure 1 as a history of wind tunnel hours utilized in the development of American aircraft. At present, efforts are under way at all the NASA centers to increase productivity of the wind tunnels and help bend the "hours-required" curve over. These comments are made to accentuate the need for high productivity in the NTF if its attributes are to be fully utilized.

The high capital cost of the NTF makes it unlikely that another facility of its Reynolds number range will be built in this country. It is probable, therefore, that efforts to achieve an ultimate in flight simulation must be centered around the NTF. The evolution of the NTF toward complete flight simulation is illustrated schematically in figure 2. The future developments needed include (a) the elimination of wall interference uncertainties, (b) the elimination or minimization of sting interference, and (c) the achievement of free-air quality low-turbulence airflow. Research in the wall interference area - smart walls - is progressing at several centers in the USA and in Europe. Sting interference elimination with attendant costly and time consuming tare testing may hopefully be achieved by means of magnetic suspension techniques. This research is currently being pursued at the Langley Research Center. The achievement of "quiet" flow in the test section of the NTF is a task of as yet unknown complexity. The influence of wind tunnel flow turbulence on boundary layer transition, on flow separation, and even on the friction drag at high Reynolds number is an area of continuing research effort in the aerodynamic community. The NTF should provide substantial help in reaching conclusions in this difficult research area.

I mention the continuing developmental requirements for the NTF because I believe it is necessary for planning purposes to have these well understood at this point in time. Even though it is not yet operational the NTF will require additional substantial funding in the not too distant future, and because of long lead times required we should not be timid in asking for funds to proceed. The NTF as presently configured is a valuable and necessary resource but it is not the ultimate in aerodynamic testing nor the ultimate in productivity. The NTF will become an essential partner in a three-element aircraft developmental process involving additionally the rapidly maturing field of computational fluid dynamics and the final standard of comparison, flight testing.

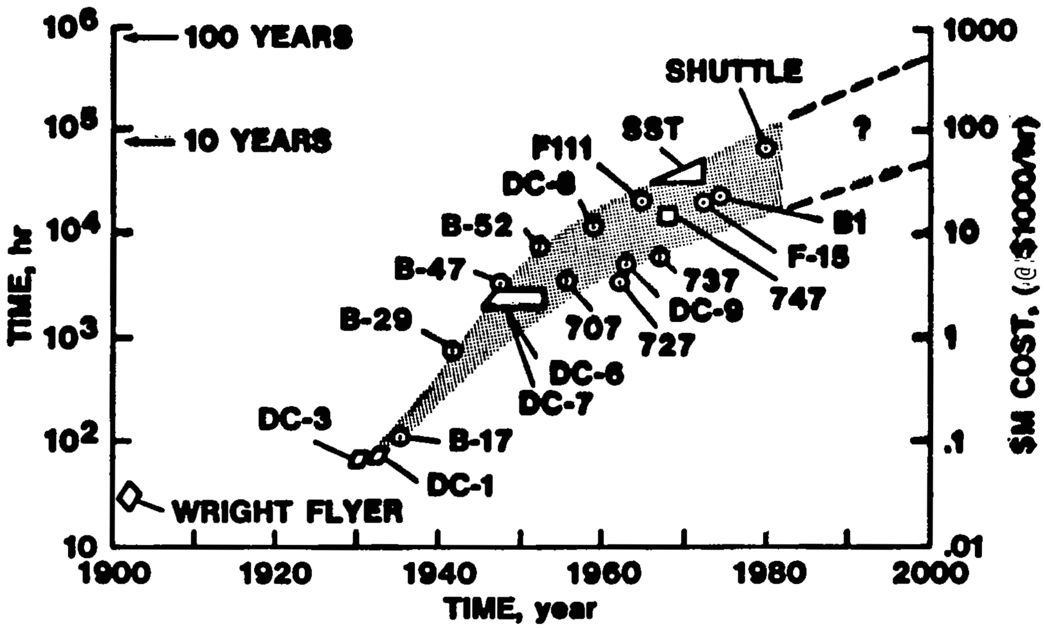

Figure 1.- Total wind tunnel test hours for development of various aircraft.

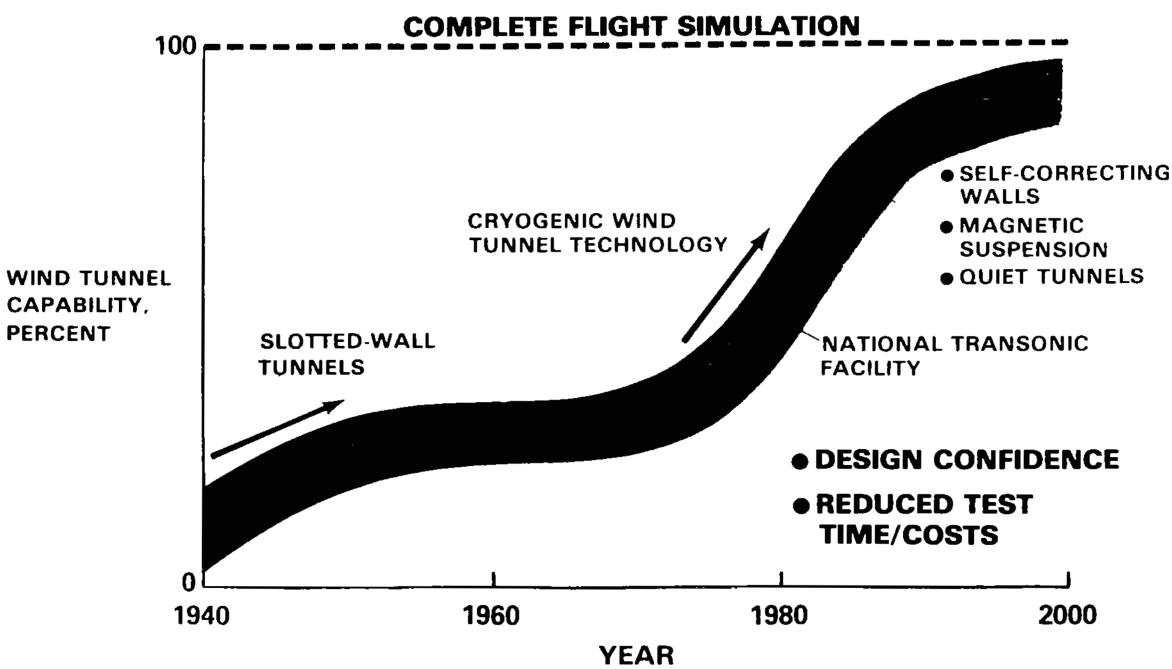

Figure 2.- Opportunities in wind tunnel testing.

NTF AND THE DEPARTMENT OF DEFENSE

Raymond F. Siewert
Staff Specialist for Aeronautics
Office of The Undersecretary of Defense for Research and Engineering

In discussing the relationship of the National Transonic Facility (NTF) to the Department of Defense (DOD), I would like to observe that the NTF has been described specifically as a national facility. This concept of national aeronautical facilities was first formalized in the Unitary Plan Wind Tunnel Act of 1949. I only mention this to show the close relationship between NASA and the DOD in developing aeronautical research facilities. The Unitary Plan Wind Tunnel Act not only established the concept of a plan for the construction of transonic and supersonic wind tunnels at the Ames, Langley, and Lewis Research Centers of NASA, but also established the Air Force Arnold Engineering Center at Tullahoma, Tennessee, as well as providing for the Navy transonic wind tunnel at the David Taylor Naval Ship Research and Development Center. However, I would like to observe that in reality, the concept of national facilities goes back to the founding of NASA, then NACA, here at Langley, where the development of new and unique aeronautical research facilities has been a way of life for over sixty years. These facilities have always been developed in close cooperation with the military services. I believe we can all agree that they have played a major role in the development of first rate military aircraft for those sixty plus years.

The development of the NTF has been a joint venture since its beginning some twelve to thirteen years ago. The need for high Reynolds number test capability was first identified by a joint NASA/DOD study group that was formed in 1968 under the auspices of the Aeronautics and Astronautics Coordinating Board (AACB).

The AACB is the uppermost level for coordination between NASA and DOD. The NASA co-chairman of the AACB is the Deputy Administrator, and the DOD co-chairman is the Under Secretary of Defense for Research and Engineering. The AACB in concept, but not by specific name, was established by the Space Act of 1958. There are several panels which function under the AACB. One of these is the Aeronautics Panel which has among its assigned responsibilities the promotion and coordination of requirements for those special ground facilities needed to support the aeronautical programs of both agencies. The Aeronautics Panel convened two separate study groups to determine the size and capabilities of what we now call the NTF. These studies covered a period of almost eight years, so you can see that the NTF did not just happen.

The preceding discussion should certainly establish DOD's interest in the NTF. This interest is due to a large extent to the fact that many of the problems which led to recognition of the need for the NTF capabilities were first encountered in military aircraft developments. The development of the C-141 aircraft and the problem of determining the location of the wing shock based on small scale tests is certainly well known. The F-102 and F-106 are but two of several tactical aircraft in which the after-body drag was higher than predicted by small scale tests, resulting in less than desired transonic acceleration performance.

Avoiding these problems is not enough. This can be accomplished through conservative design practices. The underlying physics of the problem must be understood so that the next generation of military aircraft can be designed closer to the margin. This will be necessary because flight efficiency will be more important than ever for military aircraft. Improved cruise performance will be required for the whole range of military missions. Some of the factors that make cruise efficiency important include:

- Rapid deployment airlift capability
- Self-deployment capability for tactical aircraft
- Reduced tanker dependence for strategic aircraft

Decreasing fuel consumption has always been a primary goal in aircraft design. Obviously, any vehicle that depends on the expenditure of energy to exist in its environment must be fuel efficient to be practical. However, the continuing escalating cost of fuel mandates that we strive to develop aircraft that are as fuel efficient as possible.

The usefulness of the NTF will not stop with developing improved transonic cruise efficiency. Other technical areas of high interest to DOD include:

- Developing a better understanding of high angle of attack aerodynamics, particularly in separated flow

- Obtaining a keener insight into unsteady aerodynamics and flutter mechanisms, particularly on aircraft with external stores

- Fully understanding the importance of Reynolds number on maximum lift coefficient, which has become more important in light of the developing interest in short take-off and landing capability on the part of both the Air Force and the Navy

There will still be other important uses for the NTF. An obvious example will be the calibration of existing transonic wind tunnels. Still, there will be uses we have not thought of yet. As with all new experimental facilities, as we begin to use the NTF we will undoubtedly discover new and previously unconsidered ways to utilize this capability. The history of NACA and NASA is replete with examples of this type of activity. Employing the lunar landing gantry here at Langley to do research on the characteristics of general aviation aircraft crashes under controlled conditions is a prime example.

In conclusion, I would like to observe that the NTF is nearing completion. The range of capabilities of the NTF is still difficult for some of us to grasp. The purpose of this workshop is to identify and prioritize those initial experiments which will make the NTF productive from the very beginning. I wish you great success in this important undertaking.

NTF MANAGEMENT CONSIDERATIONS

Robert E. Bower
Director for Aeronautics
NASA Langley Research Center

INTRODUCTION

The National Transonic Facility (NTF) is expected to come on-line in mid-1982. It was designed from the start to be a national testing facility satisfying the research and development needs of NASA, DOD, industry, and universities. The NTF offers for the first time simulation of full-scale Reynolds numbers in the critical flight regions of most current and planned aerospace vehicles. Despite this unique capability, however, the degree of interest by users will depend greatly upon tunnel productivity, data quality, and cost. These three considerations have driven the design of the NTF from its inception.

A commitment to serve both the researcher and the vehicle designer extends over into the operational management of the NTF. A great deal of thought has gone into these management considerations, both within NASA and the DOD, as well as input from non-government users. Significant in this input was a recent study by the National Research Council's Aeronautics and Space Engineering Board (ref. 1). A special ad hoc committee, headed by Professor Hans Liepmann, was formed with representatives from industry and the university community to review and critique NASA's management plans. The recommendations of this group were very helpful and most have been incorporated in NASA's plans. It is the purpose of this paper to discuss the current status of NASA's plans for the management of the NTF with emphasis on organization, operations, user interfaces, and user charges.

NTF MANAGEMENT

The NTF will be managed by the NASA Langley Research Center as shown in figure 1. Responsibility for its operation will reside in the Transonic Aerodynamics Division (TAD) which reports to the Aeronautics Directorate. The NTF will be headed by an assistant division chief of TAD. An operations branch will be responsible for facility operation and documentation. Its staff will be a combination of civil servants and support service contractors and will be sized to support a two-shift operation with a productivity of 8000 data polars per year. The skill mix of this dedicated branch will be determined by NASA's commitment to support both research and production/development testing. Emphasis will be on fast response and rapid data turn-around. Support service contractors will only be used to support facility operational functions. Overall management, technical guidance, actual facility operation, and interfaces with users will be provided by civil servants.

An aerodynamics branch will be responsible for independent research, facility improvement, user consultation, and test engineers to support user programs. It is believed that maximum effectiveness will be obtained if engineers performing all of these functions are closely associated in the same branch during the early years of operation. User consultation is particularly important to insure efficient use of the NTF in concert with other facilities.

An oversight committee reporting to the Langley Research Center Director will review the previous year's operation to insure that the NTF is best serving the interests of all users. At the present time, it is planned that the existing Aeronautics Panel of the DOD/NASA Aeronautics and Astronautics Coordinating Board will serve this function. It is also expected that the current Aerodynamics Subcommittee of NASA's Aeronautics Advisory Committee will serve in an advisory role to Langley to suggest research programs, facility improvements, and facility use.

USER CONSIDERATIONS

After initial calibration and documentation of the NTF, the expected use of the facility is shown in figure 2. Four types of programs involving different combinations of users are envisioned. The first is support of the regular NASA Research and Technology programs funded by NASA and managed by the NASA centers. The users are industry and universities in addition to the NASA centers.

The second type of program involves DOD testing in support of their vehicle development and advanced technology programs. This type of testing will be paid for by DOD on an actual use basis. It is expected that both industry and DOD may assign personnel to the NTF during test periods or for extended periods of time as appropriate.

The third type of testing would support industry proprietary programs. The funding and approval procedures for such tests would follow the current practice used in the NASA Unitary Plan Wind Tunnels as outlined in reference 2.

The fourth use of the NTF is somewhat unique. In an effort to encourage maximum use by the scientific community, provision will be made under NASA funding for unsolicited research proposals from universities and industry. These programs would be expected to encourage resident participation by researchers from the scientific community.

Included in figure 2 are preliminary guidelines for allocating test time among the four types of programs. This distribution, of course, is subject to change if operating experience and/or demand indicate a change to be desirable. The main reason for indicating such a distribution at this time is to demonstrate that balance will be sought among program types and users, e.g., research and facility documentation and improvement will not be sacrificed to excessive developmental testing. Also, the small university type user will be protected.

REIMBURSEMENT POLICY

It is clear that the reimbursement policy for the NTF will influence both the effectiveness of its operation and its attractiveness to the user. A final policy is still being discussed but the current proposed plan is as shown in figure 3. Under this plan, charges for DOD development testing and industry proprietary tests will be on an actual use basis, i.e., occupancy charges plus energy charges (both electrical and liquid nitrogen). It is expected that NASA joint programs with government and non-government participants will not be charged any fees. This is in keeping with present NASA policy.

Although user costs have not yet been formally established for the NTF, a rough order of magnitude estimate in 1983 dollars is shown in figure 3. Occupancy charges including a fixed overhead and burden are estimated to be $1000 per hour. Energy costs, of course, are a function of the test program and the operating Reynolds number. Energy costs have been developed for six different types of programs, including typical research type testing, fighter development, and transport developments, utilizing various aspects of the NTF capability. The results for one of these test programs, a subsonic transport, are summarized in figure 3. This program involved testing predominantly at $R_N = 30 \times 10^6$ and Mach numbers 0.7 to 0.875, and was chosen because it is believed to be the most typical of NTF high Reynolds number testing. Liquid nitrogen costs ran about $2000 per polar and electricity $50 per polar. This would result in a total cost of about $2300 per polar or $6000 per hour including occupancy charges. This test cost is competitive with other large facility costs in the United States and Europe and is considered reasonable in view of the value of the high Reynolds number data obtained.

CONCLUDING REMARKS

In conclusion, the following points should be emphasized:

1. The NTF provides scientists and engineers with a new and unique capability to evaluate their theories and perfect their designs at flight Reynolds numbers.

2. The user appeal of the NTF will depend greatly upon tunnel productivity, data quality, user cost, and management responsiveness.

3. NASA is committed to respond to these needs with a superb facility, a strong and competent management team, and minimal user charges.

4. Participation by the scientific community is mandatory at this stage of facility planning.

This workshop on high Reynolds number research provides the opportunity to suggest cooperative programs during the early years of operation of the NTF. All potential users of the NTF are strongly encouraged to start their planning now.

REFERENCES

1. "National Transonic Facility: A Review of the Operational Plan." A report of the ad hoc National Transonic Facility Committee, Aeronautics and Space Engineering Board, National Academy of Sciences, March 1980.

2. "Development Work for Industry in NASA Wind Tunnels." NMI 1300.1, October 4, 1978.

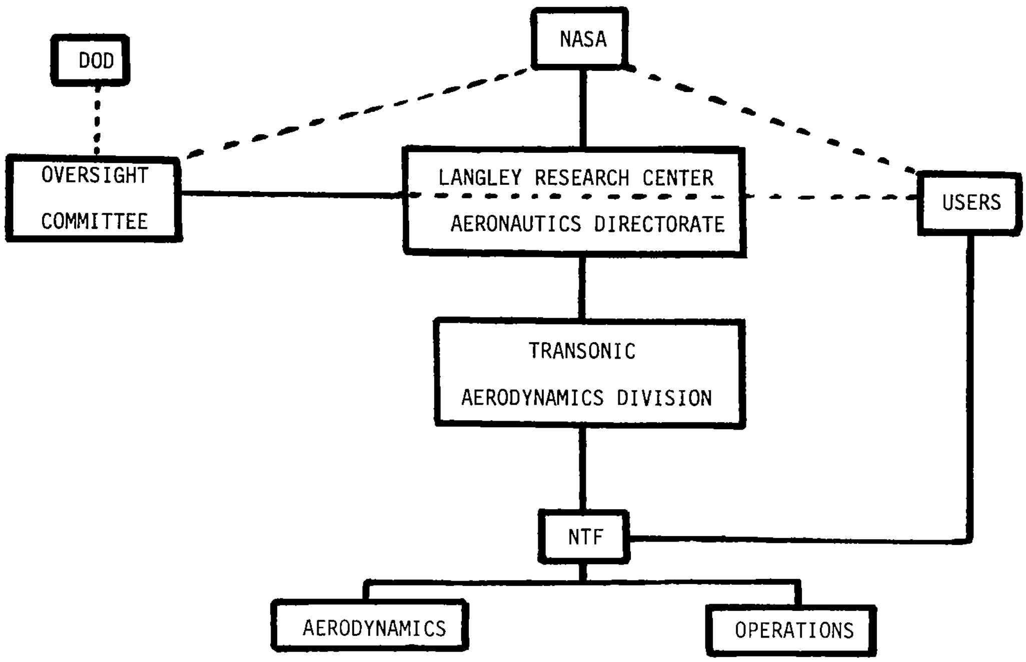

Figure 1.- NTF management organization.

USER	PROGRAM TYPE	FUNDING	APPROVAL PROCEDURE	TUNNEL UTILIZATION
NASA CENTERS UNIVERSITIES INDUSTRY	RESEARCH	NASA	REGULAR NASA CYCLE	40 PERCENT
DOD INDUSTRY	DEVELOPMENT	DOD	REGULAR DOD CYCLE	40 PERCENT
INDUSTRY	PROPRIETARY	INDUSTRY	PROPOSAL TO LANGLEY	15 PERCENT
UNIVERSITIES INDUSTRY	UNSOLICITED	NASA	PROPOSAL TO NASA	5 PERCENT

Figure 2.- Expected NTF utilization.

o DOD DEVELOPMENT AND INDUSTRY PROPRIETARY

- Occupancy charge (fixed overhead and burden)

- Energy cost (actual use)

o GOVERNMENT AND NON-GOVERNMENT JOINT PROGRAMS WITH NASA

- *No user charges (NASA funded)*

o ROUGH ORDER OF MAGNITUDE USER CHARGES (1983 dollars)

- Occupancy charge: $1000 per hour

- Energy costs ("Typical" Subsonic Transport Test Program):

  o Liquid Nitrogen - $2000 per polar; $4900 per hour

  o Electrical - $50 per polar; $100 per hour

- Total Charges: $2300 per polar; $6000 per hour

Figure 3.- Proposed NTF reimbursement policy.

PATHFINDER MODEL PROGRAM FOR THE NATIONAL TRANSONIC FACILITY

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NASA Langley Research Center

SUMMARY

An overview of the Pathfinder Models Program is presented in this paper. The Pathfinder program is a major research and development activity that is underway in support of the National Transonic Facility Activation Plan. The program scope, models design approach, and Pathfinder model configurations are presented along with a discussion of major supportive program activities. In addition, the anticipated design criteria for NTF models are presented.

INTRODUCTION

The advent of high Reynolds number (R_e) high dynamic pressure testing in a cryogenic environment presents unique and difficult challenges for the researcher and model/sting systems design engineers and analysts. The Pathfinder models program was initiated as a part of the R & D effort to develop the technology for design and fabrication of cryogenic models and sting systems for the National Transonic Facility. The purpose of this paper is to present an overview of this activity. This activity was reported on in detail at the Cryogenic Technology Conference held at the Langley Research Center in November 1979 and is documented in Session V - Model/Sting Technology in reference 1.

PROGRAM SCOPE

In support of the activation plan for the National Transonic Facility (NTF) a major R & D effort is underway to develop the technology needed for the design and fabrication of models and stings for the NTF. This technology will be integrated into the design and fabrication of two developmental models and stings which are referred to as the Pathfinder models. The first model, Pathfinder I, is a force and pressure model representative of a transport configuration. The second model, Pathfinder II, is a generalized fighter model. Both of these models will be discussed subsequently. A major end product of the Pathfinder models program will be a Users Criteria Document that will be developed around the Pathfinder models experience, with publication planned approximately 1 year before the NTF becomes operational. In addition to the in-house activities, an out-of-house (contractual) design study for fighter models is planned with an invitation for proposals expected by early summer of 1981. The Pathfinder Models Program objectives and tasks are summarized in figure 1.

DESIGN APPROACH

The major design difficulties for NTF cryo models are summarized in figure 2. The fact that no precedents exist for cryo models, coupled with the high Reynolds number (R_e) test environment, results in major challenges for both the researcher and engineer. The Langley Research Center 0.3-meter Transonic Cryogenic Tunnel (TCT) has provided valuable thermal environment experience; however, models in the NTF will be more complex, supported on long flexible stings, and loaded much higher as compared to the more rigid, simply supported models in the 0.3-m TCT. The potential for brittle fracture coupled with very high working stresses disqualifies most available structural alloys. Aeroelastic effects take on added significance, and stiffness requirements become a major factor in the design.

The Pathfinder model design philosophy is based primarily on three considerations. First, it was obvious that present model design practices for conventional tunnels could not be imposed, but should be considered and applied wherever and whenever possible. Second, it is necessary to strive to build flexibility into design practices in order to fully utilize the NTF high R_e capability. Finally, the philosophy is based on safe design practices that can be established through the application of state-of-the-art design, analysis, and testing techniques.

The approach adopted for the Pathfinder Models Program is referred to as design-by-analysis and is depicted in the flow chart of figure 3. In view of the material strength limitations, coupled with the high R_e test environment, many models for NTF will be designed to small safety factors. In order to insure that model systems are designed adequately and for safe operation, various detailed analyses and tests must be performed as the design progresses. The designer and analyst must work very closely in this regard since each element of the analysis process can affect the design, as illustrated in the flow chart. The importance of more in-depth analysis and testing of NTF models, particularly for critical application, cannot be overemphasized (see ref. 2).

PATHFINDER I

Description

The Pathfinder I model (fig. 4) is an advanced transport configuration. The geometric characteristics, which are typical of this class of airplane, are given in figure 4. The primary design point is for a 1 g cruise; however, to insure capability of testing to off-design conditions, the model strength design point is 1.8 g at 0.8 Mach number and 10 700 meters altitude.

The principal considerations that led to the selection of the Pathfinder I configuration were: (1) investigate problems associated with design and fabrication of a transport configuration for full scale R_e conditions and (2) provide a basic model for research studies. The first consideration focuses on problems associated with design to high loads, hence high stresses and increased aeroelastic effects. Also for the high-aspect-ratio wings, methods of pressure orifice and tube installation would require developmental work and the problems associated with material selection would be exercised. A basic model selection results in a clean wing for study of R_e number and aeroelastic effects at both design and off-design conditions. Also, the model is designed so that nacelles could be added at a later date, and the aft fuselage is designed for variable sting configurations. The foregoing considerations are summarized in figure 5. In essence, a configuration that would sufficiently address the design and fabrication problems as well as serve as a basic research model for high R_e testing was the main goal in the selection.

The Pathfinder I dimensions are given in figure 6 and the various model components are illustrated in figure 7. At the present time, the nose section, aft fuselage, and the horizontal and vertical tails are in fabrication. The wing design is complete and machining is expected to begin no later than mid-summer of 1981.

Stress Considerations and Test Envelope

The Pathfinder I model is discussed extensively in reference 3, however it is of interest to examine the wing stress state and the test envelope for the model as influenced by the design criteria and material selection. A planform view of the wing semispan is given in figure 8. The tube passage areas are indicated by the dashed lines. There are six orifice tube rows per wing as indicated on the figure. The model will have a total of approximately 200 orifices located on the upper surface of the left wing and the lower surface of the right wing. The wing cross section is a tongue and groove design which was dictated by strength and stiffness requirements (see ref. 3). A plot of stress versus wing semispan station is provided in figure 9. The stress values are given for two different dynamic pressure and test temperature conditions. Note that the peak stress is 698×10^6 N/m^2 and occurs at the orifice row near the wing break. By comparison the yield strength for the selected Nitronic 40 material (see ref. 4) is 1396×10^6 N/m^2 at 88.9 K. Thus the wing is designed to a working stress of 2/3 yield corresponding to a design safety factor of 1.5 on yield with stress concentration. Applying these criteria to a test temperature of 339 K would limit the dynamic pressure (q) to 0.054×10^6 N/m^2 at a working stress of 279×10^6 N/m^2 since the material yield at this temperature is 419×10^6 N/m^2. As a result of the variation in material strength with temperature a test envelope for the Pathfinder I exists as a function of yield stress (1.5 times the allowable stress), temperature, and dynamic pressure. This envelope is illustrated in figure 10. Such an envelope will be, in general, unique for each model.

Sting Configuration

The present support system for Pathfinder I is illustrated in figure 11. A general purpose stub sting will be used for the Pathfinder models and will have the strength capability for testing much higher loaded models. The stub sting engages the roll mechanism and will accommodate various model sting geometries. The Pathfinder I sting will be a double taper design having a diameter of 7.62 cm at the model base with an initial 1° half angle taper. The Pathfinder I sting geometry is dictated by aeroelastic design criteria and has a safety factor of 2 or greater against divergence and flutter at the maximum "q" test condition. The divergence problem was found to be quite sensitive to the sting stiffness over the portion just aft of the model base, as one might expect. In special cases a one piece sting may be required, but for the foreseeable future the utilization of a strut sting should meet most test requirements and be much more cost effective.

Material Selection

One of the more important activities in the Pathfinder Models Program is the material studies necessary for evaluating materials for NTF models. The factors and criteria used in selecting the material for Pathfinder I are reported in reference 4. Also a study made by the National Bureau of Standards (NBS) under contract to LaRC is documented in reference 5. Reference 5 is perhaps the most recent and up-to-date source that can be used for evaluation and selection of materials. The report was derived from an extensive literature survey and contains a long list of reference publications. Also, at the present time, other materials are being evaluated at LaRC for potential use in model/sting systems.

Fabrication

A major area of activity in the Pathfinder I program is that of fabrication and process technology development. Because of the toughness requirements needed for cryogenic materials, virtually all model materials (metals) will be difficult to machine. The anticipated requirement for smoother surfaces, closer tolerances, and high quality orifice holes for NTF models (see ref. 6) poses fabrication processes and methodology concerns. For example, the lack of machining experience with the Nitronic 40 material resulted in a program of reheat treatment and machining studies (tool bits, oil flow rates, etc.) to assure stability during the machining process. In addition, the methodology had to be established for pressure orifice tube installation in the Pathfinder I wing. In order to evaluate the wing design and gain experience with the pressure tube installation, two 2-D wings were fabricated per the Pathfinder I design and will be tested in the 0.3-m TCT.

Test Activities

Other related work includes wing load/displacement tests that were performed on a wing similar to the Pathfinder I. The wing load/displacement test results verified the math model that was used to establish the wing jig shape (see fig. 3). Other load and/or verification testing is planned, such as a loads test which is to be performed on the horizontal and vertical tail to verify the adequacy of the attachment design. The principal elements of the fabrication and test activities are summarized in figure 12.

Design Criteria for NTF Models

The work described up to this point contributed to the determination of the design criteria that were used for the Pathfinder I model. The design criteria are summarized in figure 13. It became apparent very early in the design process that LaRC conventional safety factors of three on yield or four on ultimate (whichever is greater) could not be used for the given design conditions. The use of conventional safety factors would have been highly desirable but disqualified all commercially available alloys. The criteria in figure 13 relax LaRC standard practice safety factors on strength and fatigue, introduce a new requirement in terms of fracture toughness, and maintain current practice on aeroelastic safety factors.

PATHFINDER II

Description

Much broader considerations were used for selecting the fighter model configuration (Pathfinder II). Two test configurations are illustrated in figures 14 and 15. In figure 15 the same wing is shown in the swept forward condition with a canard added, to illustrate the model versatility. This configuration was selected to assess fabrication and design problems and material requirements for highly loaded thin wings at cryo conditions. Also, the structural design limitations will determine the maximum R_e design envelopes and design safety factors for this class of configuration. In addition, the generalized research model will be configured to accommodate two balances to separate the effects of forebody, canard and/or strake from wing effects. Also, as indicated in the figures, the configuration will have changeable components for different planform, canard and/or strake configurations. The foregoing considerations are summarized in figure 16.

Fighter Design Study

The principal technical objectives of the planned contractual fighter design study are given in figure 17. This study will focus on the design of models of production type (actual) configurations. Also, it will be a mechanism whereby industry can participate in the cryogenic models design process and will supplement the LaRC in-house effort.

ANTICIPATED DESIGN CRITERIA FOR NTF MODELS/STINGS

Of particular interest to NTF users are the design criteria that will be applied to NTF models and support systems. As indicated in figure 18, it is anticipated that Langley standard practice (LHB 8850.1) will be adhered to wherever possible. Also, new criteria will be established to allow some relaxation of present LaRC practices subject to the conditions set forth by design-by-analysis and test requirements. A listing of some of the envisioned conditions for designing to relaxed criteria is given in figure 19. It should be apparent that more in-depth design, analysis, and testing will be required for NTF models. Also, all phases of quality control in model construction take on added significance. In addition, for highly critical applications, appropriate model instrumentation for use in on-line loads monitoring will likely be required. The ability to detect the onset of problems and to unload the model quickly will be of paramount importance during operation.

CONCLUDING REMARKS

An overview of the Pathfinder Models Program is presented. The narrative is intended to cover in a very broad manner the major activities ongoing at LaRC to address the problems associated with the design and fabrication of models for the NTF. Much work remains to be done in the coming months and is expected to be completed in a timely fashion to allow for the successful activation and utilization of the high Reynolds number test capability provided by the National Transonic Facility. It is anticipated that the Pathfinder I model will be ready by the spring of 1982 with Pathfinder II scheduled for completion by mid 1982. The fighter design study is also expected to be completed by mid 1982. The Users Criteria Document is scheduled to be completed by mid 1981, approximately 1 year before NTF is scheduled to become operational.

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5. Tobler, R. L.: Materials for Cryogenic Wind Tunnel Testing. National Bureau of Standards Report NBSIR 79-1624, May 1980.

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- OBJECTIVES
  - DEVELOP TECHNOLOGY FOR DESIGN AND FABRICATION OF MODELS/STINGS FOR NTF
  - INTEGRATE TECHNOLOGY INTO DESIGN AND FABRICATION OF TWO DEVELOPMENTAL MODELS AND STING SUPPORTS
- TASKS
  - IN-HOUSE
    - PATHFINDER I MODEL (TRANSPORT)
    - PATHFINDER II MODEL (FIGHTER)
    - SUPPORT STINGS
    - USERS CRITERIA DOCUMENT
  - OUT-OF-HOUSE (CONTRACTUAL)
    - DESIGN STUDY FOR FIGHTER MODELS - IFP, SUMMER 1981

Figure 1.- Pathfinder Models Program.

- NO PRECEDENTS
- NTF HIGH R_e TEST ENVIRONMENT INTRODUCES
  - THERMAL COMPLICATIONS
  - BRITTLE FRACTURE FAILURE MODES
  - HIGHER DYNAMIC PRESSURES
    - HIGH WORKING STRESS
      - USE OF CONVENTIONAL SAFETY FACTORS WOULD DISQUALIFY MOST AVAILABLE STRUCTURAL ALLOYS
    - AEROELASTIC EFFECTS
      - STIFFNESS BECOMES A MAJOR FACTOR

Figure 2.- Design difficulties for NTF cryo models.

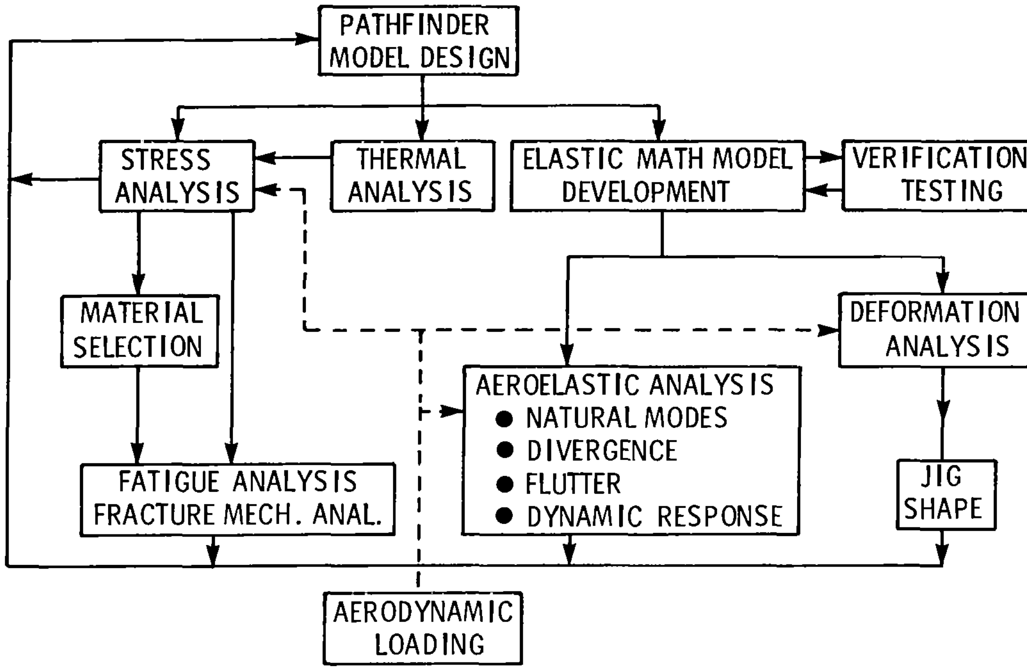

Figure 3.- Design-by-analysis flow diagram.

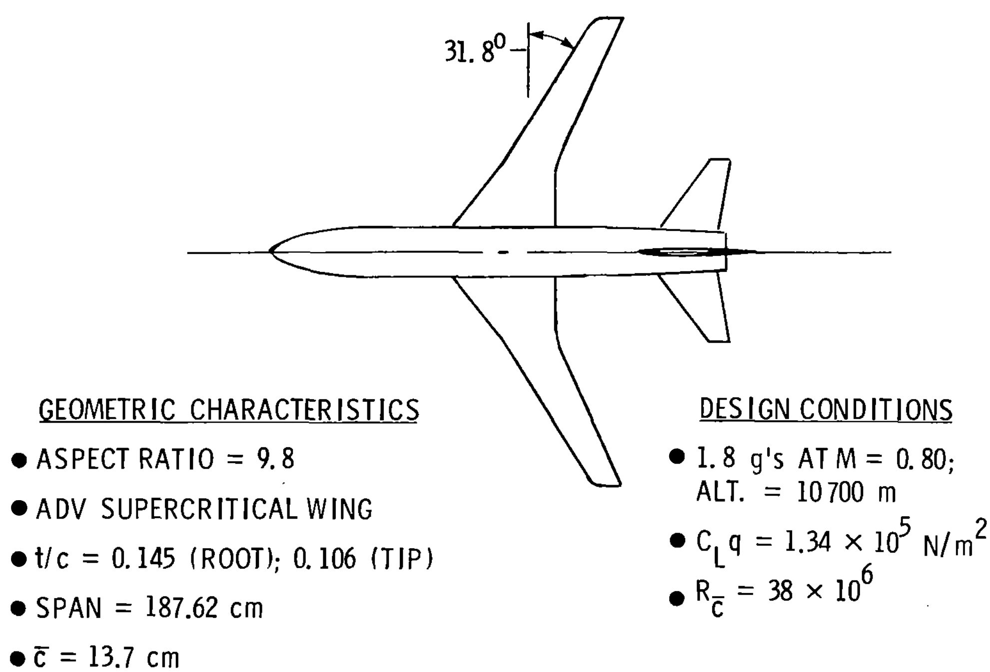

Figure 4.- NTF Pathfinder I: Advanced subsonic transport configuration.

- INVESTIGATE PROBLEMS ASSOCIATED WITH DESIGN AND FABRICATION OF A TRANSPORT CONFIGURATION FOR FULL SCALE REYNOLDS NUMBER CONDITIONS
    - HIGH LOADS (HIGH STRESSES)
    - PRESSURE ORIFICE/TUBE INSTALLATION
    - MATERIAL SELECTION
- PROVIDE A BASIC MODEL FOR RESEARCH STUDIES
    - CLEAN WING FOR STUDY OF REYNOLDS NUMBER AND AEROELASTIC EFFECTS AT BOTH DESIGN AND OFF-DESIGN CONDITIONS
    - NACELLES TO BE ADDED AT A LATER DATE
    - AFT FUSELAGE DESIGNED FOR VARIABLE STING CONFIGURATION

Figure 5.- Key considerations in Pathfinder I configuration selection.

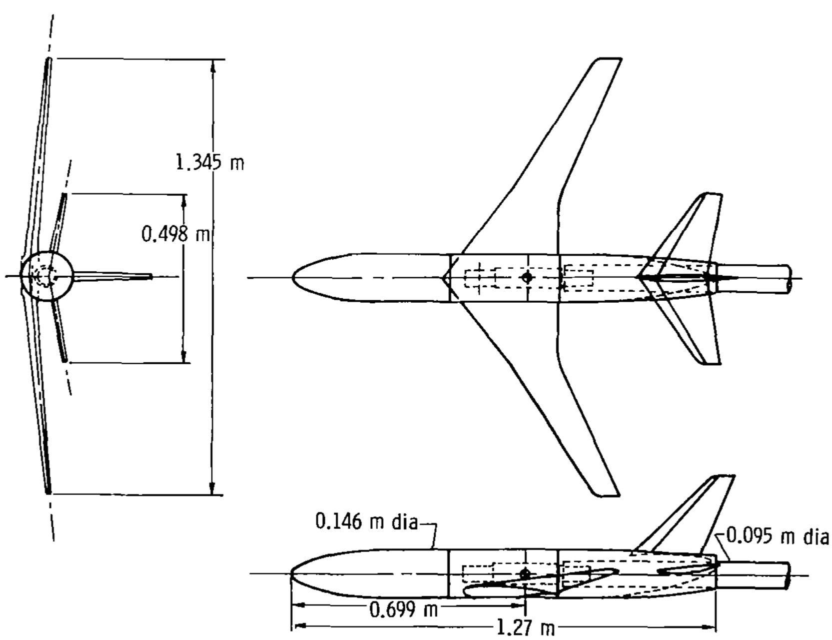

Figure 6.- Pathfinder I model.

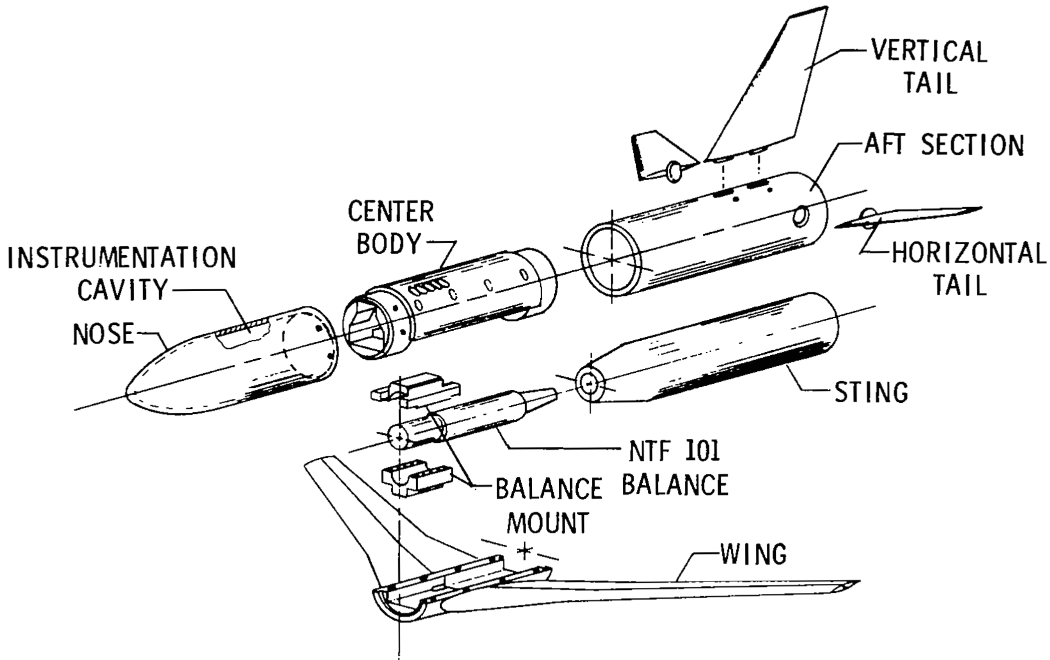

Figure 7.- Pathfinder I model components.

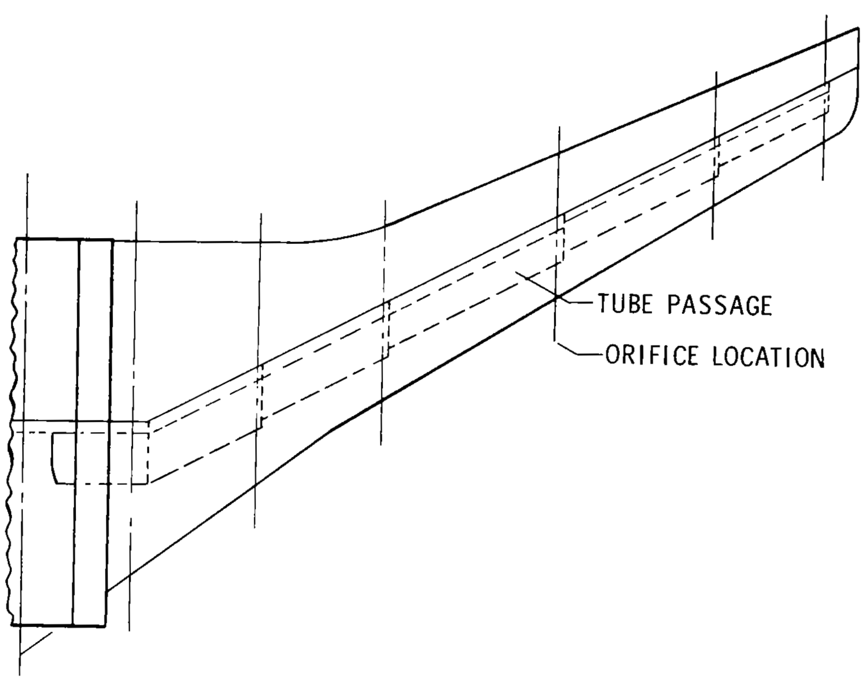

Figure 8.- Pathfinder I wing.

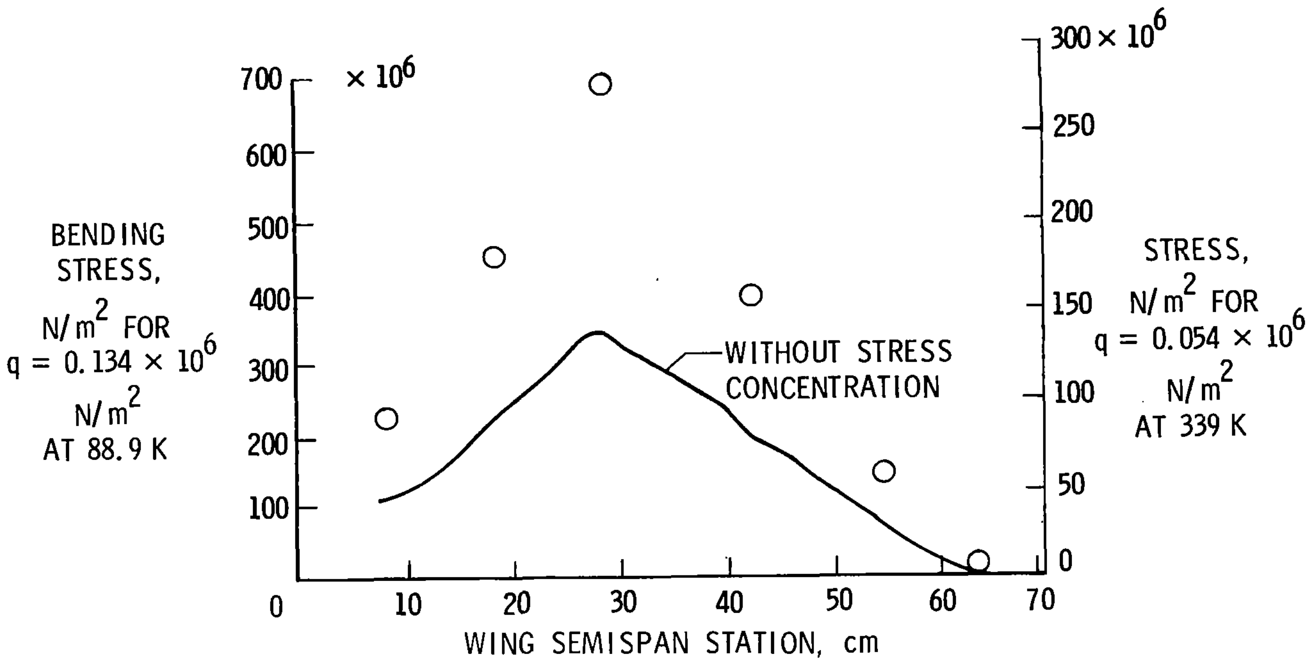

Figure 9.- Pathfinder I model wing stresses; $C_L = 1$.

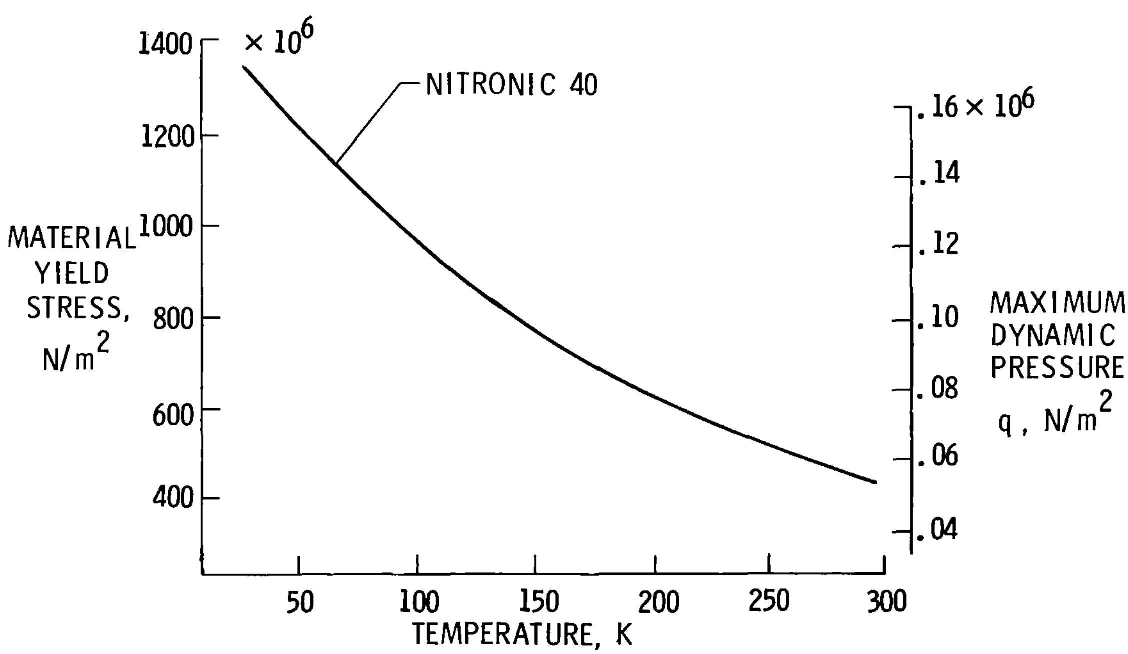

Figure 10.- Pathfinder I model test envelope for $C_L = 1$.

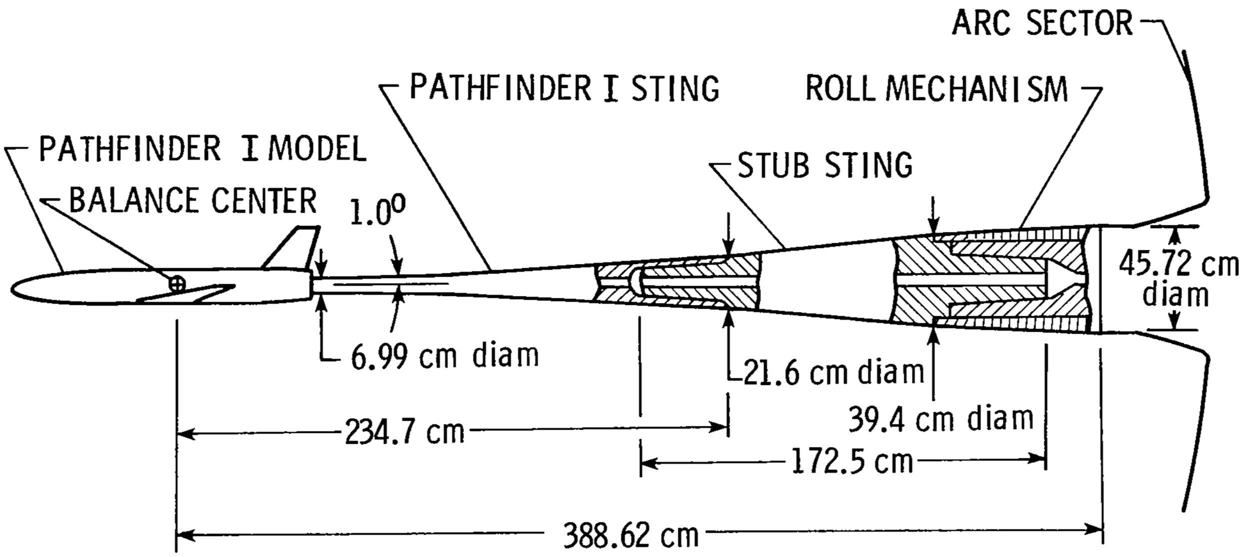

Figure 11.- Pathfinder I sting configuration.

- FABRICATION
  - MACHINING SPECS FOR SELECTED MATERIAL
  - METHODOLOGY FOR INSTALLING PRESSURE ORIFICE TUBES IN WING
    - BRAZE ALLOY STUDIES
    - PROCESS CONTROL
  - (2) 2-D WINGS FABRICATED PER PATHFINDER DESIGN FOR 0.3-M TUNNEL TESTS
  - WING MANUFACTURING METHODS
- TEST
  - WING LOAD/DISPLACEMENT - MATH MODEL VERIFICATION
  - HORIZONTAL AND VERTICAL TAIL LOADS TEST - ATTACHMENT DESIGN

Figure 12.- Pathfinder I program fabrication and test activities.

- STRENGTH SAFETY FACTOR - GREATER OF 1.5 ON YIELD OR 2 ON ULTIMATE AT TEST TEMPERATURE
- FATIGUE - SAFE-LIFE DESIGN PER MODIFIED GOODMAN DIAGRAM WITH APPLIED SAFETY FACTOR OF TWO (2)
- FRACTURE - MATERIAL CHARPY V-NOTCH IMPACT STRENGTH $\geq$ 33.89 N·m AT TEST TEMPERATURE
- AEROELASTIC - SAFETY FACTOR OF TWO (2) AGAINST DIVERGENCE AND FLUTTER (AT TEST DYNAMIC PRESSURE)

Figure 13.- Selected design criteria for Pathfinder I.

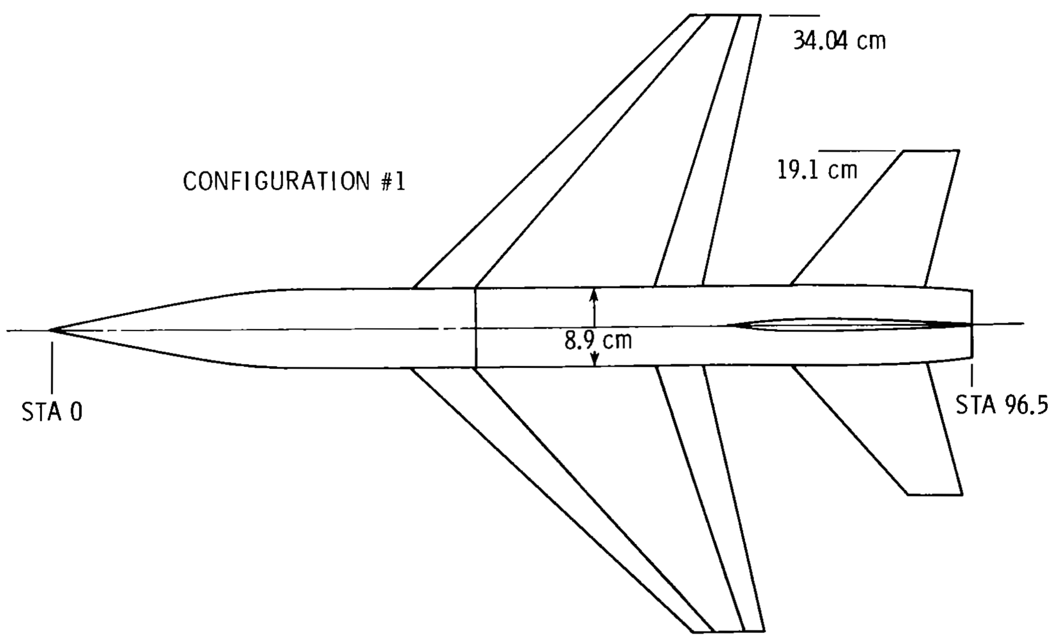

Figure 14.- NTF Pathfinder II model general research fighter configuration.

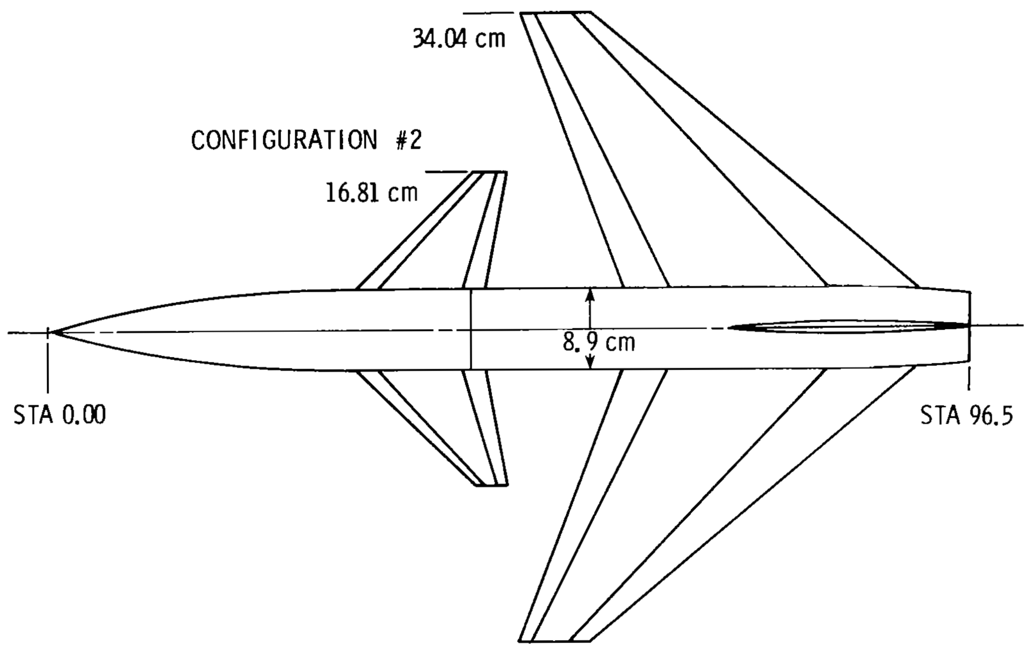

Figure 15.- NTF Pathfinder II model general research fighter configuration.

- ASSESS FABRICATION AND DESIGN PROBLEMS FOR HIGHLY LOADED THIN WINGS AT CRYO CONDITIONS
- ASSESS SUITABLE MATERIALS FOR THIS APPLICATION
- DETERMINE MAXIMUM R_e DESIGN ENVELOPES AND DESIGN SAFETY FACTORS FOR THIS CLASS OF CONFIGURATION
- PROVIDE GENERALIZED RESEARCH MODEL
    - 2 BALANCES - SEPARATE EFFECTS OF FOREBODY, CANARD AND/OR STRAKE FROM WING EFFECTS
    - CHANGEABLE COMPONENTS

Figure 16.- Key considerations in Pathfinder II configuration selection.

- DETERMINE MAXIMUM DESIGN TEST ENVELOPES FOR ACTUAL CONFIGURATIONS
  - SINGLE AND TWIN ENGINE
- ASSESS AERODYNAMIC VS. MODEL COMPROMISES FOR FLOW THROUGH NACELLES IN FUSELAGE
- ASSESS TRADE-OFFS FOR VARIOUS STING ARRANGEMENTS

Figure 17.- Fighter design study objectives.

- ADHERE TO LaRC STANDARD PRACTICE LHB 8850.1 WHENEVER AND WHEREVER POSSIBLE
- NEW CRITERIA WILL BE ESTABLISHED TO ALLOW SOME RELAXATION OF PRESENT LaRC STANDARD PRACTICES*

* SUBJECT TO CONDITIONS SET FORTH BY DESIGN-BY-ANALYSIS AND TEST REQUIREMENTS

Figure 18.- Anticipated design criteria for NTF models/stings.

- DESIGN BY IN-DEPTH ANALYSIS APPROACH
  - STRESS
  - FATIGUE
  - FRACTURE
  - AEROELASTICITY
- MATERIALS CERTIFICATION
  - MECHANICAL PROPERTIES
  - CHARPY OR FRACTURE TOUGHNESS
  - SCREEN FOR CRITICAL FLAW SIZE
- STRUCTURAL TESTING
  - MATH MODEL VERIFICATION
  - PROOF LOAD
    - STRENGTH
    - STIFFNESS
  - VIBRATION
    - NATURAL MODES AND FREQUENCIES
- QUALITY CONTROL
  - NON-DESTRUCTIVE EXAMINATION
  - PROCESSES
  - DOCUMENTATION
  - ETC.
- INSTRUMENT FOR ON-LINE LOADS MONITORING

Figure 19.- Proposed conditions for designing to relaxed criteria.

MODEL EXPERIENCE IN THE LANGLEY 0.3-m TRANSONIC CRYOGENIC TUNNEL

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SUMMARY

The development of wind tunnels that can be operated at cryogenic temperatures has placed several new demands on our ability to build and instrument wind-tunnel models. This paper presents a brief summary of the model building, development, and testing experience gained during 8 years of operation of the 0.3-m Transonic Cryogenic Tunnel at the Langley Research Center.

INTRODUCTION

The world's first transonic cryogenic tunnel was placed in operation at the NASA Langley Research Center in 1973. Since this tunnel, now known as the Langley 0.3-m TCT, was the first of its kind, a considerable portion of its first 8 years of operation has been spent in developing the testing techniques and technology peculiar to cryogenic wind tunnels rather than to the aerodynamic testing of models (refs. 1 to 5). Nevertheless, a large number of models of various types have been tested and a valuable body of model design and construction experience has been gained. The purpose of this paper is to provide a brief summary of the model building, development, and testing experience gained during 8 years of operation of the 0.3-m TCT. This summary will be presented in a chronological manner, being divided roughly into four portions: models tested in the 0.3-m TCT's original octagonal test section, models tested in the present two-dimensional test section, models tested as a part of tunnel calibration and the development of advanced technology airfoils, and the development of a new way to construct two-dimensional airfoil models. The model story is prefaced with a brief section on design requirements imposed on the models by high Reynolds number testing at cryogenic temperatures.

MODEL REQUIREMENTS FOR THE 0.3-m TCT

One method of gaining insight into any special requirements placed on models for the 0.3-m TCT or any similar cryogenic tunnel is to compare the operating conditions of the 0.3-m TCT with those of a more conventional ambient temperature transonic tunnel. This is accomplished in figure 1, which shows typical operating conditions of the Langley 8-Foot Transonic Pressure Tunnel and the 0.3-m TCT. The most obvious difference is the very low value of temperature which can be achieved in the 0.3-m TCT. This low temperature level immediately restricts the materials available for model construction since some metals tend to become brittle at these temperatures while others will no longer meet minimum fracture toughness requirements. In addition, the very wide range of temperatures available in the 0.3-m TCT makes it essential that the thermal expansion properties of the materials used in a model be known and taken into account in the model design. The higher pressures available in the 0.3-m TCT, even at a maximum free-stream Mach number somewhat less than that of the 8-ft TPT, cause higher loads as reflected by the increase in dynamic pressure. This increased load again restricts the choice of materials available for model construction. Also, voids in the model necessary for instrumentation become more difficult to tolerate. Finally, the increased Reynolds number, which is the salient feature of cryogenic tunnels, requires improved surface finish and smaller pressure orifices due to the thinner boundary layers associated with the increased Reynolds number. In addition, testing at Reynolds numbers high enough to duplicate flight conditions has, as one of its purposes, the improvement of flight efficiency - typically dealing with the last one-half of one percent - and realization of these small increments in performance gain requires ever-increasing fidelity of contour and accuracy of measurement of wind-tunnel models.

The two existing test sections that can be used with the 0.3-m TCT and the general layout of the tunnel are shown in figure 2. The three-dimensional (3-D) octagonal test section is shown installed in the tunnel circuit. The two-dimensional (2-D) test section is shown separately. The tunnel is presently being used with the 2-D test section insert, but was initially used with the 3-D test section insert. Details of the 0.3-m TCT operation may be found in reference 1. Model testing experience in the 0.3-m TCT will be discussed first in terms of models tested in the 3-D test section and then in terms of models tested in the 2-D test section.

MODEL TESTING EXPERIENCE IN THE 0.3-m TCT

Models Tested in the 3-D Test Section

Figures 3 through 7 are photographs which summarize the model testing experience in the 3-D test section of the 0.3-m TCT. Each of these models will be briefly discussed.

Validation airfoil.- Shown in figure 3 is the first model to be built for the 0.3-m TCT, a modified NACA 0012-64 airfoil having a 13.72 cm chord. The model spanned the octagonal test section and was fastened to the walls in such a way that incidence could be varied. This model is of some historical significance in that pressure tests made with it provided the first experimental confirmation in compressible flow that gaseous nitrogen at cryogenic temperatures behaves in a manner very close to that of a perfect gas and that Reynolds number obtained by reducing temperature is fully equivalent to Reynolds number obtained by increasing pressure. Because of the comparative nature of the tests, surface finish and orifice size requirements were of no particular concern for this model. The pressure tubes were simply soldered into grooves cut into the model and the surface finished and the orifices drilled just as would be done for a model to be tested in an ambient temperature low Reynolds number tunnel.

Two problems developed with this model. The more serious was the breaking of the pressure tubes where they entered the ends of the model. This problem was caused by the tubes becoming brittle as they were soldered to the model. The second problem developed after a trailing edge orifice was added to the airfoil by using an electric discharge milling process that involved immersing the model in an oil bath. The oil seeped into minute voids around the solder and reappeared as not-so-small drops on the model surface which froze in place as the tunnel and model were cooled to cryogenic temperatures. Because of their proximity to the pressure orifices, the frozen drops of oil greatly affected the pressure data. It took a thorough cleaning in solvent using an ultrasonic cleaner to eliminate the problem.

Boattail models.- Two very successful series of models were built in the shops at Langley and tested in the 0.3-m TCT by Reubush, references 6 and 7, to determine the drag on isolated boattails and on boattails in the presence of a simple wing.

Isolated boattail models: There was a total of six models used in the isolated boattail tests: four short models (20.32 cm from the nose to the start of the boattail) having different boattail geometries and two long models (40.64 cm) which duplicated the boattail geometries of two of the short models. These models, shown in figure 4, were all sting-mounted with the diameter of the sting being equal to the model base diameter. Thus, in this case, the sting was used to advantage to simulate the geometry of a jet exhaust plume for a nozzle operating at its design point. This series of isolated boattail models was constructed of cast aluminum with stainless-steel pressure tubes cast as an integral part of the model. The tubes were placed in a sand mold in the proper position, the aluminum poured, and the model machined to the desired contour. Each of the models has 30 pressure orifices in three rows of 10 orifices each. Although it would have been desirable to have all of the orifices in one row, the fact that the models are only 2.54 cm in diameter combined with the relatively large number of orifices precluded this possibility.

Wing-body interference models: The models used in the wing-body tests, shown in figure 5, duplicated the forebody and boattail geometry of two of the short isolated boattail models. Construction of these models differed slightly from that of the isolated boattail models in that each of the models was cast around both the pressure tubes and a stainless-steel sting. By using this method, it was possible to fit each of these models with 50 pressure orifices in five rows of 10 orifices each. Provision was made for the mounting of a 10.16-cm span 60° delta wing at 0° incidence on the top of each of the models in one of three positions.

Delta-wing models.- A set of four simple delta-wing models was made to be used in evaluation tests of a three-component internal strain-gage balance (ref. 3). Each of the models has a 75° leading edge sweep. Two of the models were machined from 17-4 PH stainless steel which was heat-treated to conditions H 1150-M. With this heat treatment, 17-4 PH can be used at temperatures as low as 77 K. One of the steel models has a sharp leading edge and one has a rounded leading edge. Not knowing exactly what to expect with respect to heat transfer through the metal models, we built a duplicate set of models out of glass fiber reinforced plastic.

Although some tests were made using one of the plastic models, the final balance evaluation was made using the sharp leading edge steel model shown in figure 6. Due to the nature of the glass fiber reinforced plastic, it was impossible to make the surfaces of the plastic models as smooth as would be desired. Except for this single problem, these simple delta-wing models have proven to be satisfactory in every respect.

Space shuttle orbiter model.- The model of the space shuttle orbiter shown in figure 7 was constructed and tested to measure the support-interference-free base pressure drag and to determine Reynolds number effects on base drag to aid in extrapolating the values of base drag measured in the wind tunnel to full-scale flight conditions. Selected test results are presented in reference 8. The support-interference-free condition was obtained by supporting the model of the orbiter on extended wing tips attached to the tunnel walls in such a manner as to allow variation in incidence. The model itself was constructed of 9 percent nickel steel and accommodated 15 pressure orifices, 3 in the base of the vertical tail and the remaining 12 distributed over the base of the fuselage and rocket nozzles.

Models Tested in the 2-D Test Section

The 2-D test section is shown in figure 8 with the top wall of the test section removed. This top wall also forms part of the slotted test section and is identical to the slotted floor which can be seen in the photograph. The model shown in the photograph is a 2-D airfoil model. Also identified in the photograph are the AOA (angle-of-attack) transducer and the wake survey (momentum) rake. The airfoil testing program will be discussed in a later section, whereas in this section various other models that have been tested in the 2-D test section will be discussed.

Buffet models.- Two semispan buffet models are shown mounted in the 2-D test section in the photographs presented in figure 9. The view of the delta-wing model is shown looking down on the model with the slotted floor of the 2-D test section in the background. The buffet model of the NPL 9510 airfoil is seen mounted inverted on a removable section of the test section sidewall. Both models were mounted on one of the test section turntables to allow variation in model attitude. These models were constructed from 7075 T-6 aluminum alloy and the strain gages used to measure root bending were attached with an adhesive (Micromeasurement's M-bond GA-2). Further details of the design and testing of these models and selected buffet data may be found in reference 9.

Delta-wing model.- The simple 3-D delta-wing model shown in figure 10 was machined in one piece from A-286 stainless steel and was used in Langley's ongoing cryogenic strain-gage balance development program. As would be expected with such a simple model, it performed satisfactorily. Further details and results of the balance performance may be found in references 9 and 10.

NTF cooling coil.- A photograph of a model of the NTF cooling coil is shown in figure 11. The primary purpose of the test was to determine the pressure loss characteristics of various coil configurations over the NTF performance envelope. Due to the wide Reynolds number range of the 0.3-m TCT, the performance envelope of the NTF was duplicated. Selected data may be found in reference 8.

2-D airfoil models.- A photograph of an airfoil mounted in the 0.3-m TCT model module is shown in figure 12. The model module contains the turntables on which the 2-D airfoil models are mounted. There are two of these modules, allowing bench set-up of one model while testing another. The airfoil shown in this figure has the pressure tubing potted into grooves in the model with solder. Although this is a successful technique in conventional tunnels, it is suspect in cryogenic tunnels due to differential shrinkage of the A-286 stainless-steel model and the solder and disbonding and cracking at the solder-stainless-steel joint. Further comments on 2-D model construction problems may be found in reference 9.

General research airfoil: A general research airfoil, which also has the pressure tubes potted in place with solder, is shown in figure 13. This is one of a series of general research airfoils and in this test was being used to develop a technique for the measurement of surface adiabatic wall temperature rather than to make a precise determination of the aerodynamic characteristics of the model. The thermocouples for this test were potted into the surface with a 66 percent aluminum-filled epoxy which, incidentally, appears to be a very successful technique.

NPL 9510 airfoil: This airfoil model, shown in figure 14, was one of the first constructed for the 0.3-m TCT using a welded-cover-plate technique. To install pressure orifices in a model by this method, it is necessary to cut relatively deep channels in the model to allow room for the steel tube plumbing bundle to be routed to the various parts of the model. The ends of the steel tubes are soldered to the model and to the cover plate at the desired orifice locations and then the cover plate is electron-beam-welded to the model. For this model, a spanwise row of pressure orifices was fitted on the top surface in addition to the usual chordwise row of orifices on the top and bottom surfaces of the model. A total of 50 pressure orifices was fitted to this particular model.

NACA 0012 airfoil: This airfoil has been tested in many tunnels and is thus thought of as a "correlation" airfoil. For this reason extensive coverage of the model surface with pressure orifices was desired. There was a particular need for spanwise rows of orifices to assist in the evaluation of any possible tunnel sidewall effects. To this end, the model was fitted with an electron-beam-welded cover plate that covered most of the upper surface of the airfoil. In order to avoid oil-canning of the model surface during contour machining, the cover plate had to be supported by interior posts. This model had a total of 64 pressure orifices used in one bottom surface chordwise row and top surface chordwise and spanwise rows. This airfoil model is pictured in figure 15. Note the bundle of steel tube plumbing sticking out from each end of the model. The metal tangs on either side of the plumbing chase are used to attach the model to the turntables in the tunnel sidewalls.

Boeing airfoil: An airfoil designed and built by the Boeing Commercial Airplane Company is shown mounted in the test section in figure 16. The dots which can be seen near the leading edge of the model are thin pieces of aluminum foil glued to the surface and were used for fixing the location of transition. This airfoil also utilized the welded-cover-plate design, except for the pressure orifices on the top surface in the very thin region of the model at the trailing edge. There, orifices were installed by potting tubes in the bottom surface and then drilling the pressure orifices from the top. In figure 16, which is a view of the bottom surface of the airfoil, the potted channels are clearly visible.

Summary of 2-D airfoil models: Characteristics of the five 2-D airfoils tested to date are shown in table 1. The 12 percent supercritical wing airfoil model shown as the last entry on this table is the most recent airfoil model to be tested and has not yet been discussed; it is different from the welded cover plate models previously described in that its cover plate is adhesively bonded in place rather than welded. Table 1 also summarizes the type of construction and the material used for each of the models.

TABLE 1.- MODELS TESTED IN THE 2-D AIRFOIL PROGRAM

Designation	Material	Tubing installation	Comment
General research	304	Potted in solder	Thermocouples potted in epoxy - 66 percent Al filler
NACA 0012	A-286	Soldered tubes	Welded cover plates
NPL 9510	HP 9-4-20	Soldered tubes	Welded cover plates
Boeing	A-286	Soldered tubes	Welded cover plates
12 percent SCW	15-5 PH	Soldered tubes	Adhesively bonded cover plate

In this table most of the models are seen to be of the cover plate type with the adhesively bonded type possibly the most advanced. Thus far, five successively constructed and tested models have been discussed with little mention of the problems encountered during this program. It is appropriate to begin the discussion of problems with the photograph of the model and cover plate in figure 17. The complexity of the machining required for the plumbing voids and cover plate is evident and leads to a basic problem for this type of model; 1000 to 1500 manhours are required for construction which leads to a very high cost for these relatively small models, and much of this cost is incurred before the steel plumbing can be installed and checked so that leaking or plugged pressure tubes can result in a very nearly completed model being rejected as unacceptable. An analysis of the several unsuccessful attempts to build airfoil models has revealed an apparently unrelated set of causes. However, a common factor in many of the failures was the bundle of steel tubing within the model. Thus, a program was initiated to find an alternate method of getting the pressure information out of the model. This program is described in some detail in reference 9. Only the most promising technique and recent developments will be described herein.

NEW AIRFOIL CONSTRUCTION METHOD

The bundle of steel tubing has been identified as a major factor in the high cost of 2-D airfoil models; also, the model construction difficulties associated with the pressure plumbing have been directly responsible for a number of unacceptable models. A new airfoil construction method is being developed to eliminate the steel tubing and to circumvent the high construction cost and the high failure rate. The new method will be explained by following selected steps taken during the construction of an airfoil sample, beginning with the photograph in figure 18. Shown in the photograph are two blocks of steel laid out as the halves of an open book; the usual pressure tubing for a midspan row of orifices has been replaced by channels cut into the facing pages of the book. When the book is closed, the steel blocks will be joined, one block becoming the top half of the airfoil, the other block becoming the bottom half.

At the orifice row end of the channels (center of the blocks) a relatively large connector hole is drilled so that the bottom of the hole lies near the model surface. Next, the smaller orifice hole is drilled from the surface side at the desired orifice location and at an appropriate angle so that the pressure orifice will be normal to the local surface after the airfoil has been contour machined. The other end of the channel (towards the tunnel wall) ends in a relatively large connector hole, which is normal to an intersecting hole drilled into what will become the support end of the model, to form the pressure channel exit. Those channels that are to connect leading and trailing edge orifices are simply carried to the edge of the blocks, or in the book analogy, to the edge of the pages, and in this case the channels themselves serve as orifices. Note that the set of channels on one block will serve the upper surface of the model and the other set of channels the lower surface of the model.

For the next step, the smooth half of each block (where there are no channels) is first coated with a 0.008-mm thick layer of copper; the blocks are then placed face to face, as if the book were shut, and diffusion-brazed together in a vacuum oven. Tubing is brazed into the exit holes at the same time the blocks are joined, producing a relatively simple blank from which an airfoil shape can be machined with all the necessary plumbing existing internal to the block and ready to be checked for leaking or plugged tubes. If the plumbing is not satisfactory, the block can be discarded at this stage of model construction at a small fraction of the cost of the conventional method, in which most of the model building expense would have been incurred before the plumbing could have been installed and checked. After checking for plugged or leaking tubes, the block can then be examined ultrasonically to check for possible flaws in the bond between the two halves. A computer-generated view of an ultrasonic scan of a typical block is shown in figure 19. The internal channels can plainly be seen as well as the indicated flaw, or void, in the bond.

Figure 20 is a metallurgical-style photograph of the diffusion joint at a 400x magnification. The horizontal seam across the middle of the photograph is the bond joint, and what is left of the thin copper coating. Metallurgical examination shows that much of the copper has diffused away; also, examination of figure 20 will show that there has been direct grain migration in forming the joint, with the joint in many places taking on the characteristics of the parent metal. Figure 21 is a photograph of the completed airfoil sample with the top surface machined to its final finish. Figure 22 is an enlarged photograph showing two of the "V"-shaped channels which were extended to the edge of the blocks and which now serve as trailing edge orifices. The distance from the bond line to the top surface represents one-half of the trailing edge thickness. This serves to illustrate an additional advantage of this new construction method; trailing edge orifices and orifices on the surface near a thin trailing edge are now routinely attainable where they were frequently difficult and often impossible with previous model construction techniques.

Present Development of the Airfoil Construction Method

Although the construction of the sample airfoil just described (fig. 21) is viewed as a very successful demonstration of the new technique, more work is necessary to fully develop its potential. Steps in the development program are briefly described in the sections that follow.

Extension of diffusion brazing experience.- The sample model was constructed from 17-4 PH stainless steel, which is a material widely used for constructing models for conventional tunnels. However, this material has marginal fracture toughness parameters at the lowest operating temperatures of the 0.3-m TCT and was chosen for the sample model only because of the familiarity of the Langley personnel with its diffusion-brazing and machining characteristics. A more suitable material for use in building models for testing in the 0.3-m TCT is 15-5 PH stainless steel, and, since this model construction method has direct applicability to the NTF, it is desirable to gain experience with materials expected to be compatible with the 3-D model requirements of the NTF such as Nitronic-40 and 347 stainless steels. In line with the effort to develop diffusion-brazing techniques for the new materials, parallel programs are underway to understand the metallurgical process involved with each material as well as to measure tensile strength, shear strength, and fracture toughness properties of the bonds in the new materials.

Highly cambered airfoils.- The application of the new model construction technique to symmetrical airfoils is obvious, but application to highly cambered airfoils has not yet been attempted. Especially needed are pressure orifices in the curved trailing edge region of many supercritical airfoils. Since the models tend to be very thin in this region, it is desirable to have the bond line between the two model halves curved in approximately the same fashion as the airfoil camber line in order to have a means of transmitting information back from this region. An effort is currently underway to machine and bond curved plates in order to develop and demonstrate the ability to use this model construction technique on highly cambered airfoils.

3-D airfoils.- In applying this new model construction technique to 3-D wings, such as will be required for models to be tested in the NTF, it may be desirable to use three or more blocks of metal in order to have more than one bond line. This would not only reduce possible crowding of the pressure channels, but also would allow a separate family of channels to be provided for each chordwise row of orifices along the span of the wing. An initial 3-block sample (two bond lines) has been made and samples from it tested. Bonding problems with the initial sample are being resolved and work is proceeding on the development of multiblock 3-D airfoils.

Construction of a 0.3-m TCT airfoil.- An airfoil model for the 0.3-m TCT is currently under construction using this new technique. The airfoil is to be a 12-percent thick supercritical symmetrical airfoil designed to serve the same functions as the NACA 0012. The airfoil is to have a total of 94 pressure taps, approximately 50 percent more than previous 0.3-m TCT airfoil models. Included in the 94 orifices will be 8 trailing edge orifices, orifices at 96, 98, and 99 percent chord, and four stagnation line orifices at the tunnel wall-model stagnation line intersection within the tunnel wall boundary layer to assist in evaluating tunnel sidewall boundary-layer treatment.

CONCLUSIONS

Some of the experiences at the NASA Langley Research Center relative to the design, construction, and use of models for the Langley 0.3-m Transonic Cryogenic Tunnel during the last 8 years have been reviewed in this paper. The main conclusions to be drawn from our experience are:

1. A number of models having widely diverse research purposes have been tested, reflecting the testing versatility inherent in a cryogenic pressure tunnel.

2. A wide range of materials has been successfully used for building models for use in cryogenic tunnels.

3. A common source of trouble with models used in cryogenic tunnels is the practice of combining materials having different rates of thermal expansion.

4. A simple method for building airfoil models has been developed which offers many advantages over methods previously used. By brazing together two or more grooved blocks of metal it is possible to greatly increase the number of pressure orifices while reducing the volume within the model required for the pressure plumbing.

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	8-ft TPT	0.3-m TCT
T_t, K	320	78 - 340
p_t, atm	0.25 - 2	1 - 6.1
M_∞	0.2 - 1.2	0.1 - 0.9
q_∞, atm	0.007 - 0.54	0.007 - 2.0
R/m, max	14×10^6	390×10^6

Figure 1.- Range of operating conditions for the Langley 8-foot TPT and the 0.3-m TCT.

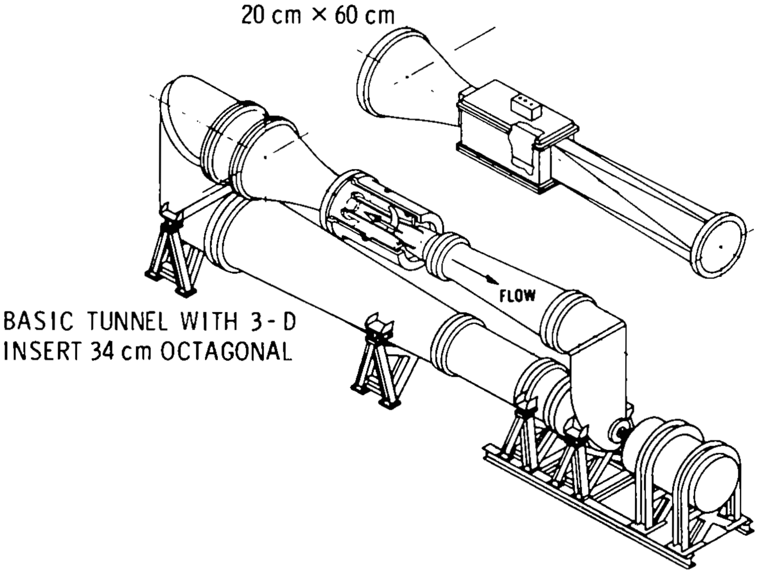


Figure 2.- Langley 0.3-m TCT.

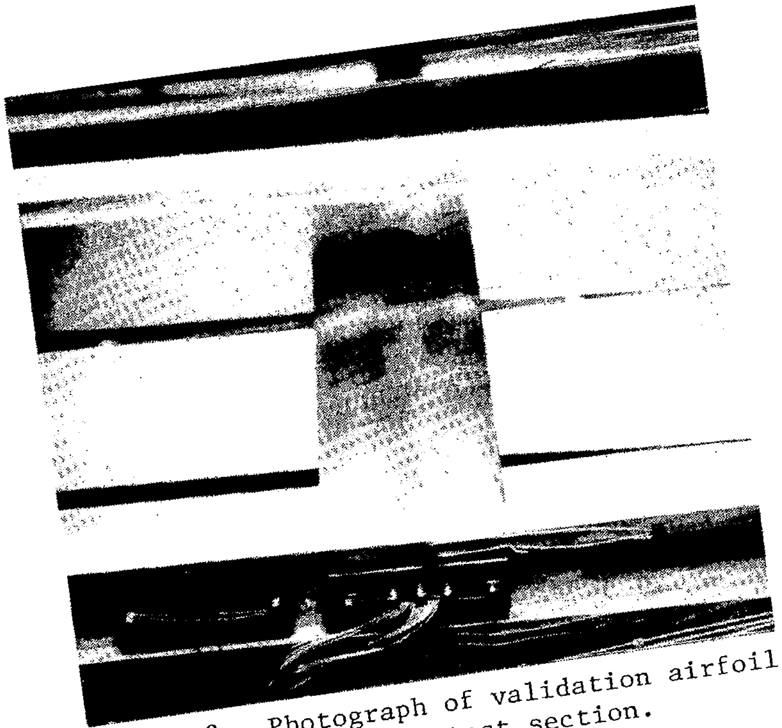


Figure 3.- Photograph of validation airfoil mounted in 3-D test section.

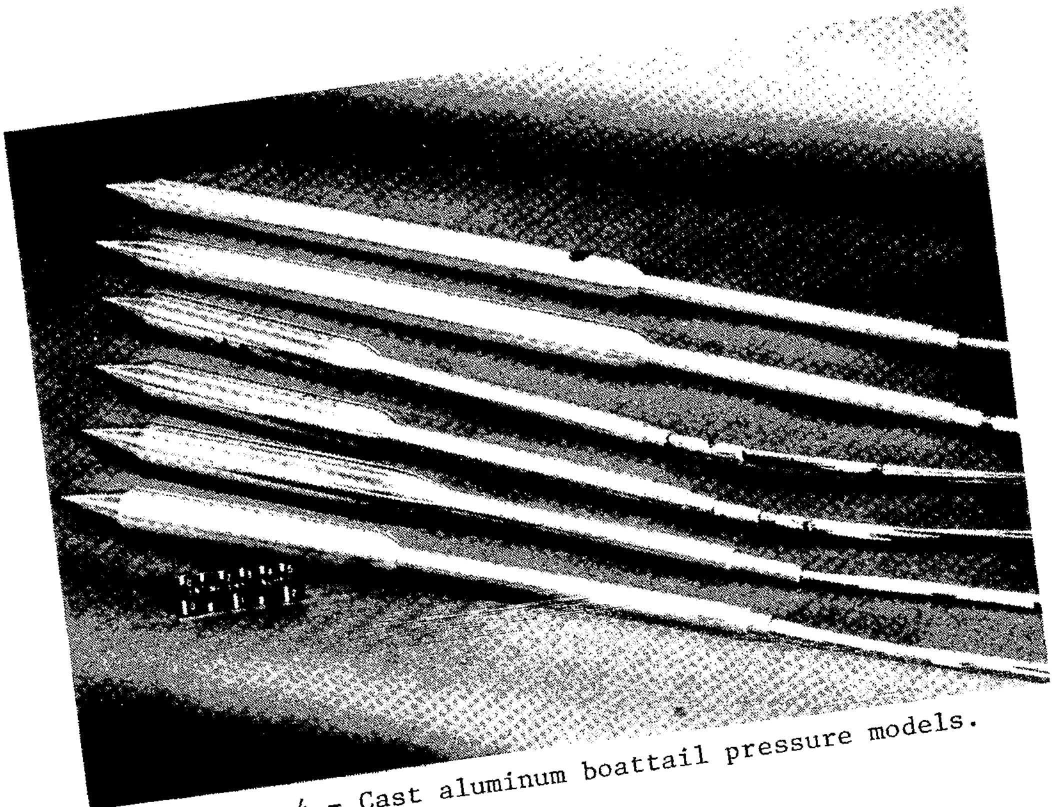


Figure 4.- Cast aluminum boattail pressure models.

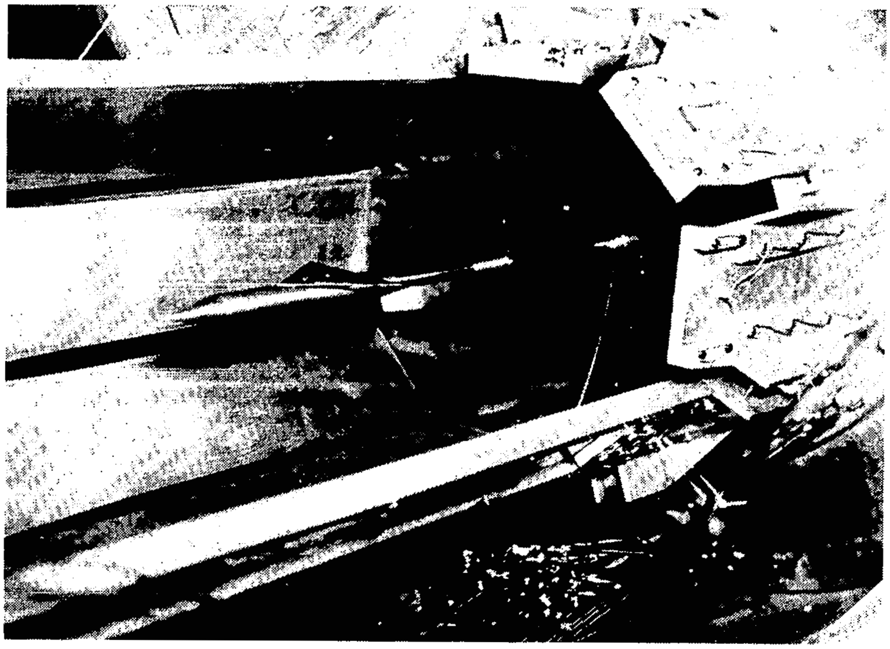


Figure 5.- Wing-body interference model mounted in 3-D test section.

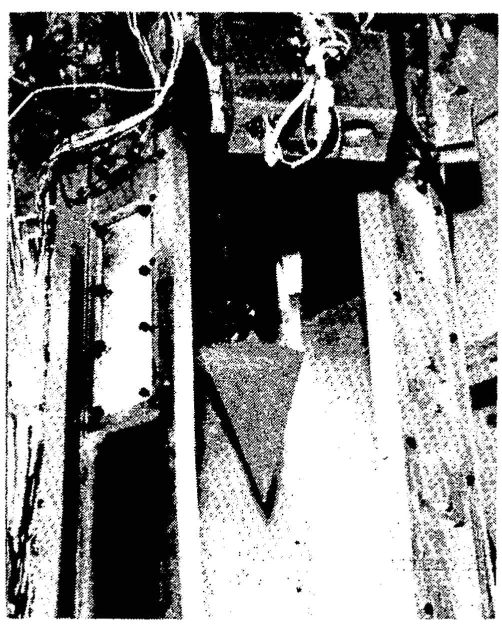


Figure 6.- Machined steel model used in strain-gage balance development.

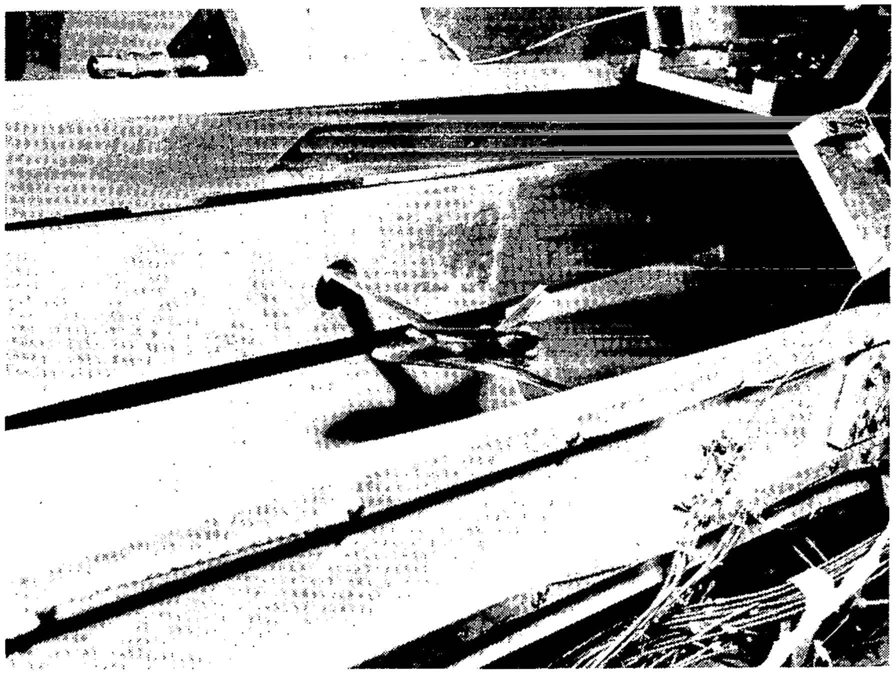


Figure 7.- Space Shuttle Orbiter model mounted in 3-D test section.

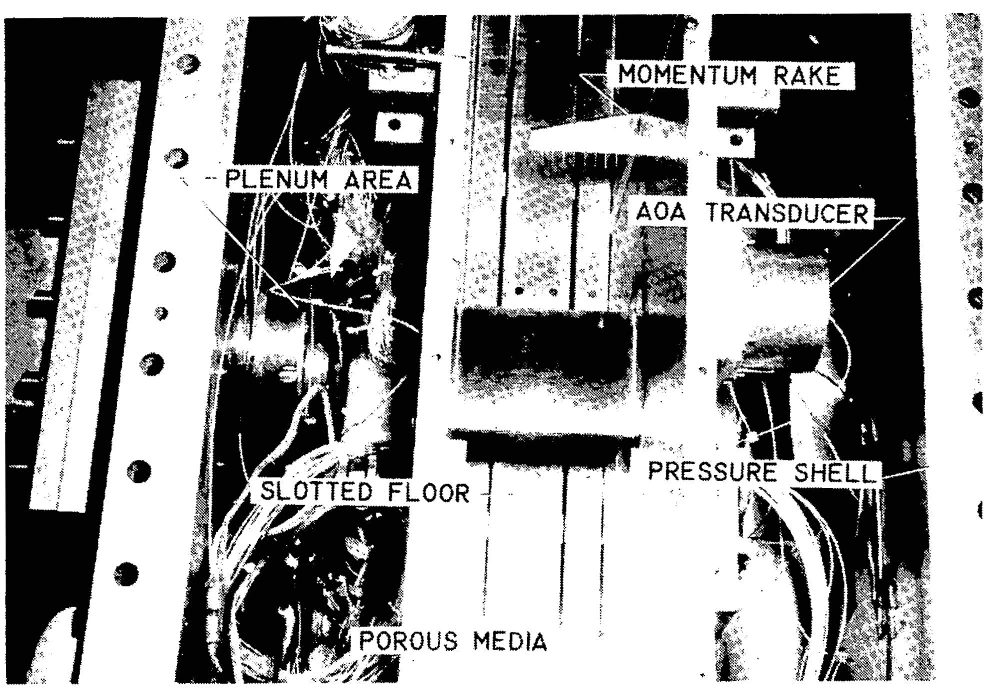
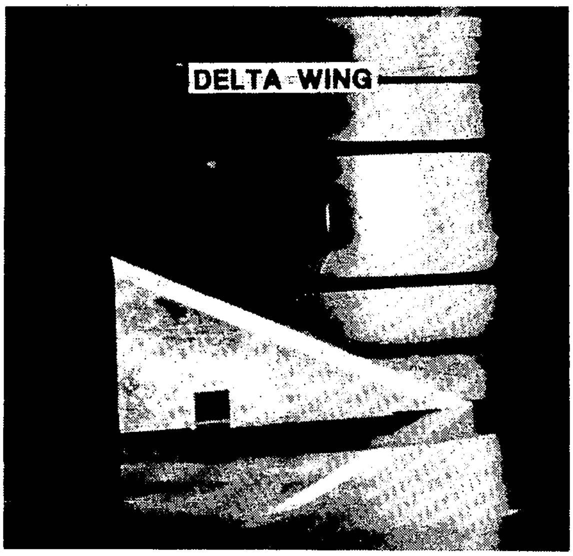


Figure 8.- 0.3-m TCT two-dimensional (2-D) test section.



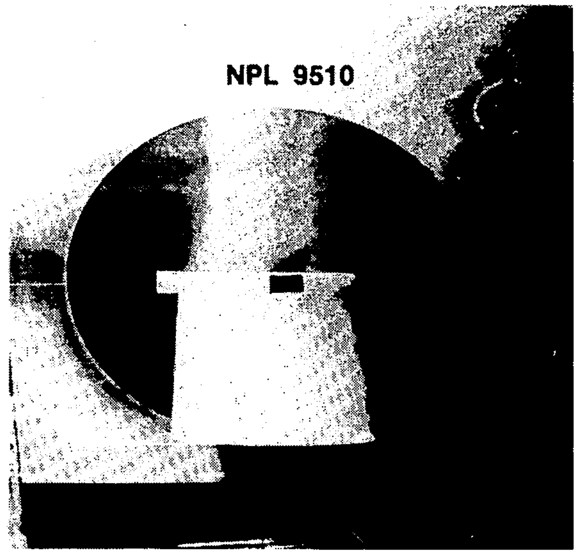


Figure 9.- Machined aluminum buffet models.

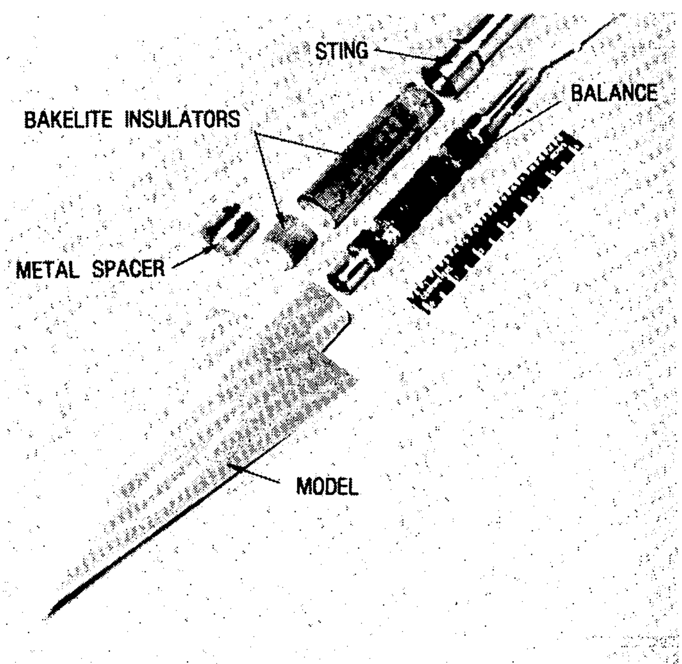


Figure 10.- Machined steel model used in tests of advanced strain-gage balance.

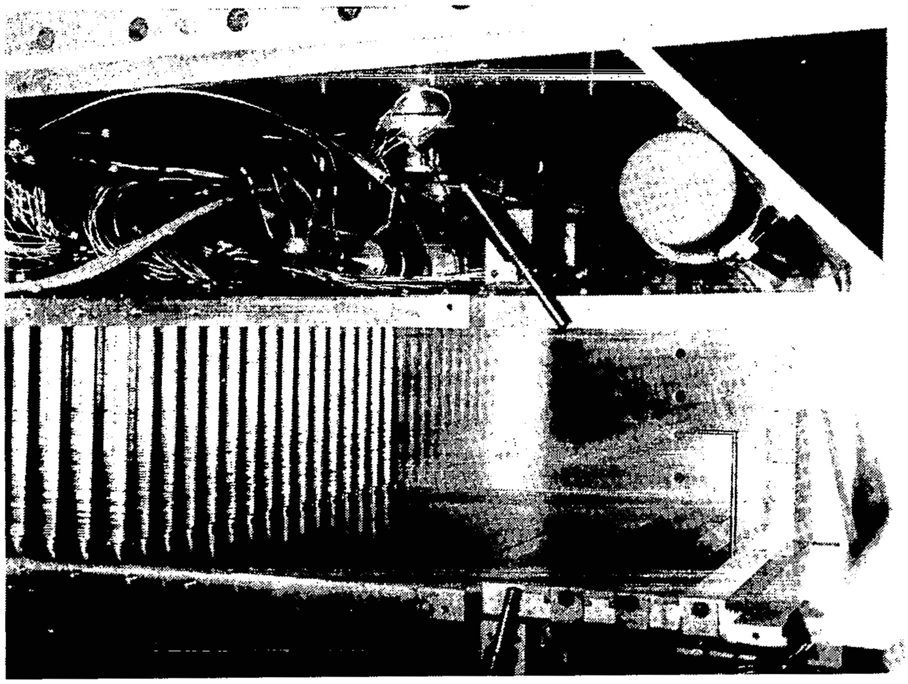


Figure 11.- Model of cooling coils for NTF tested in 0.3-m TCT.

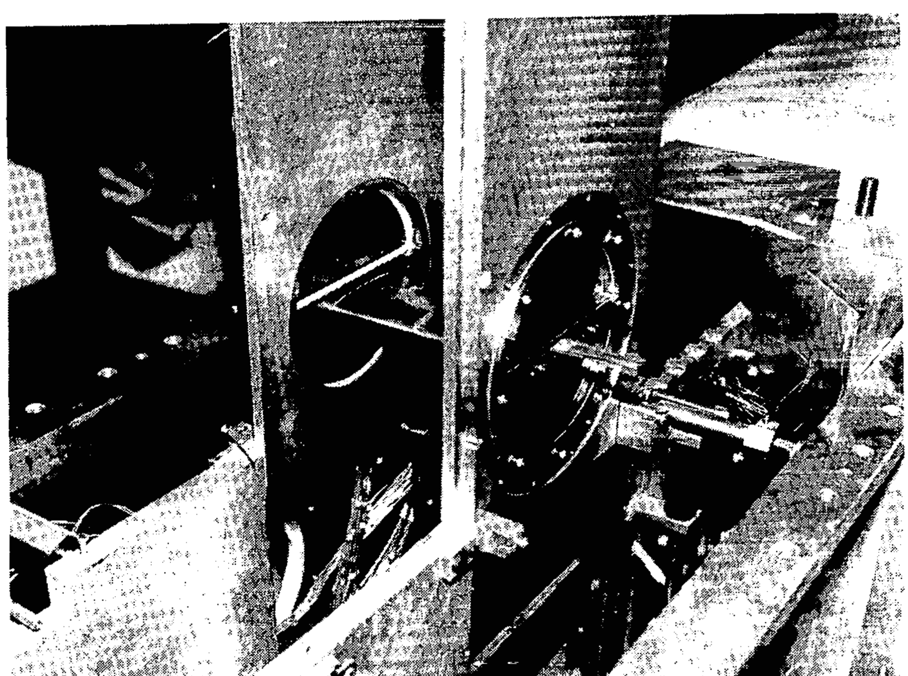


Figure 12.- Model module showing method of mounting airfoil models.

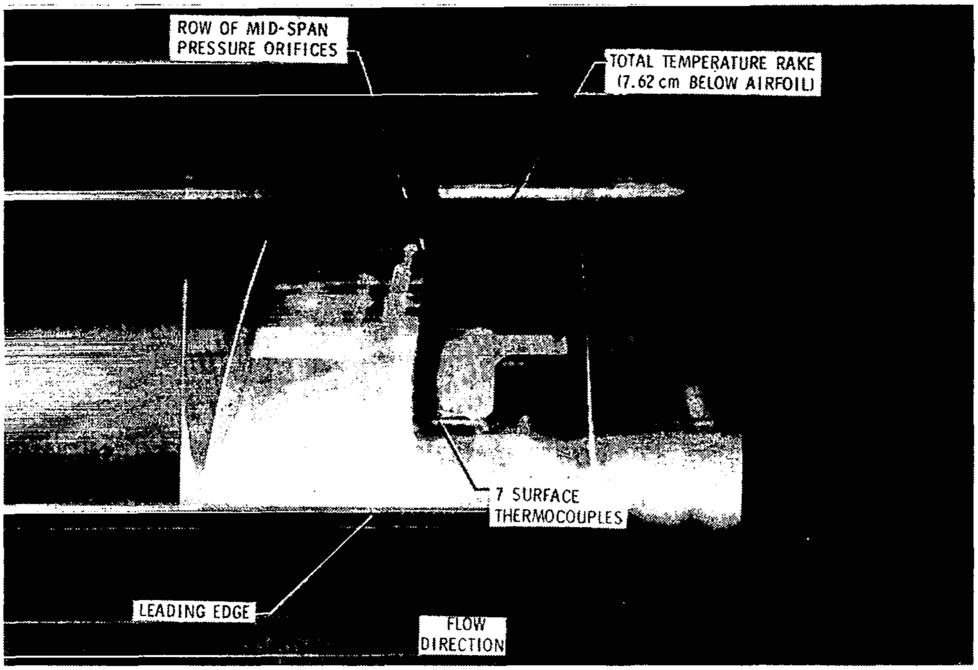


Figure 13.- General research airfoil model used in adiabatic wall temperature tests.

Figure 14.- NPL 9510 airfoil model as tested in 0.3-m TCT.

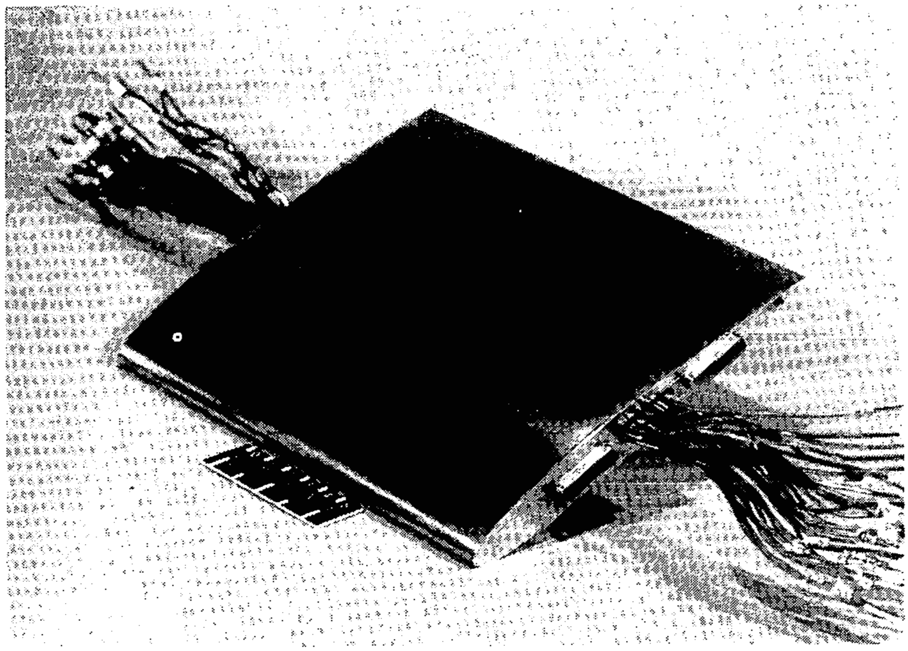


Figure 15.- NACA 0012 airfoil model with complex steel tube plumbing.

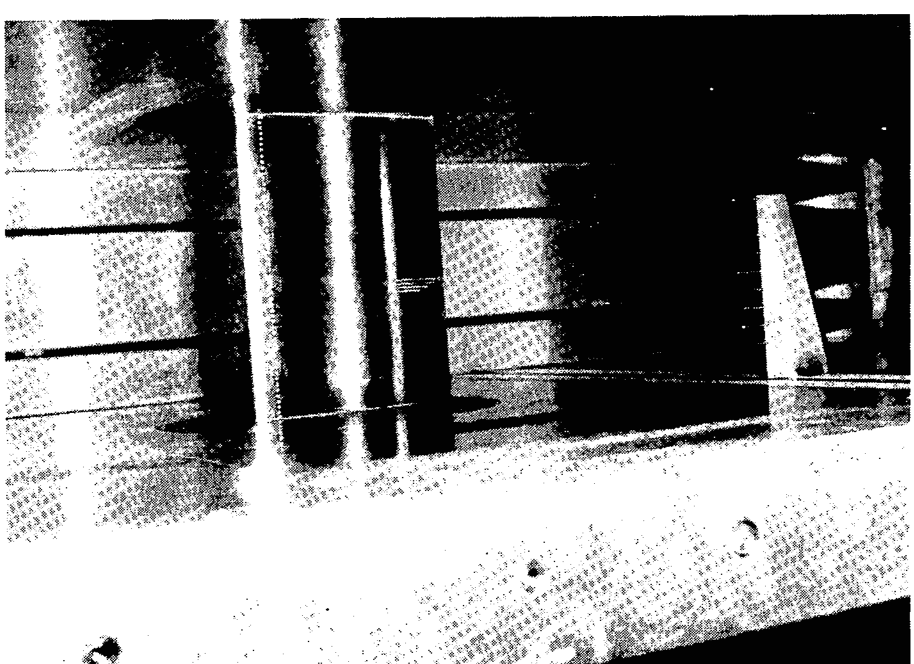


Figure 16.- Boeing airfoil model in the 0.3-m TCT.

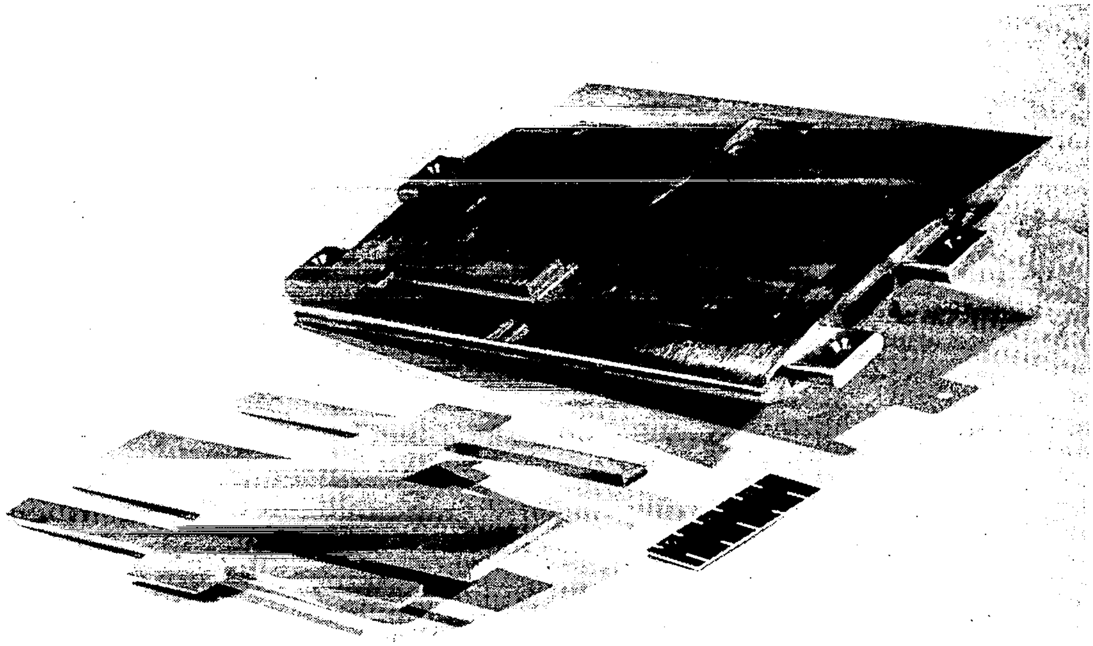


Figure 17.- Partially completed airfoil model.

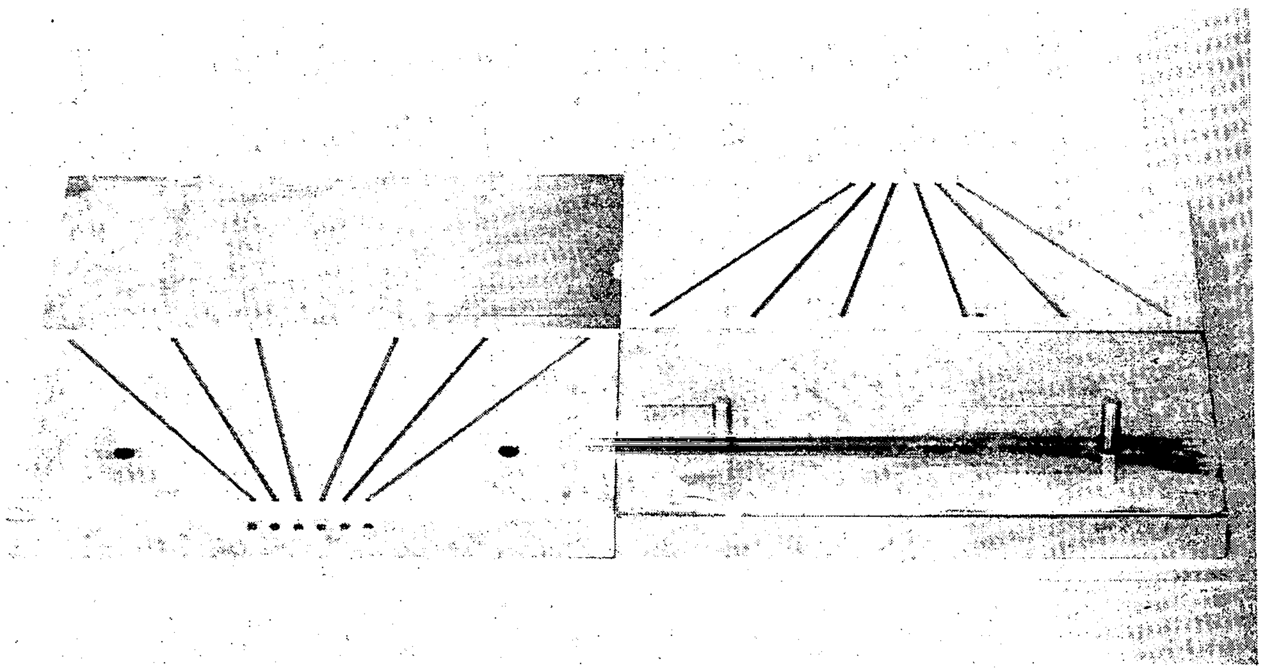


Figure 18.- Blanks used to construct sample airfoil.

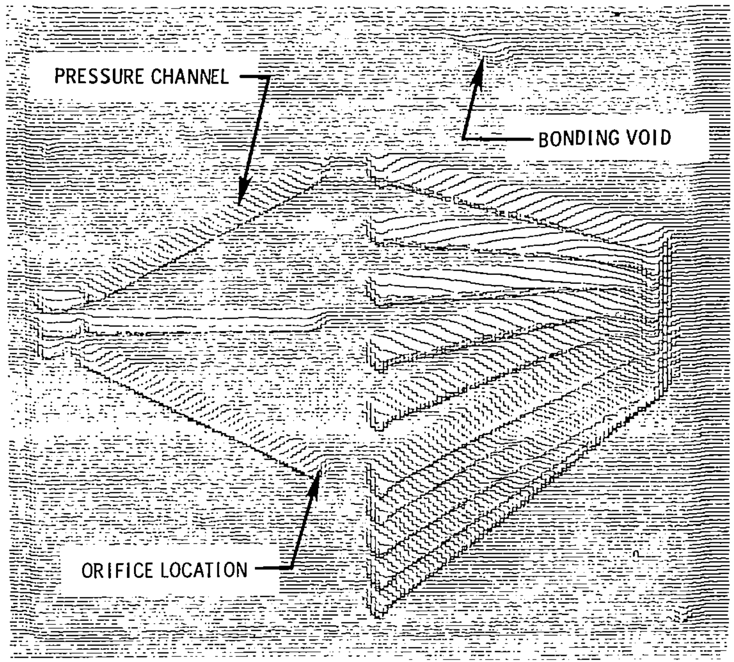


Figure 19.- Ultrasonic view of sample airfoil.

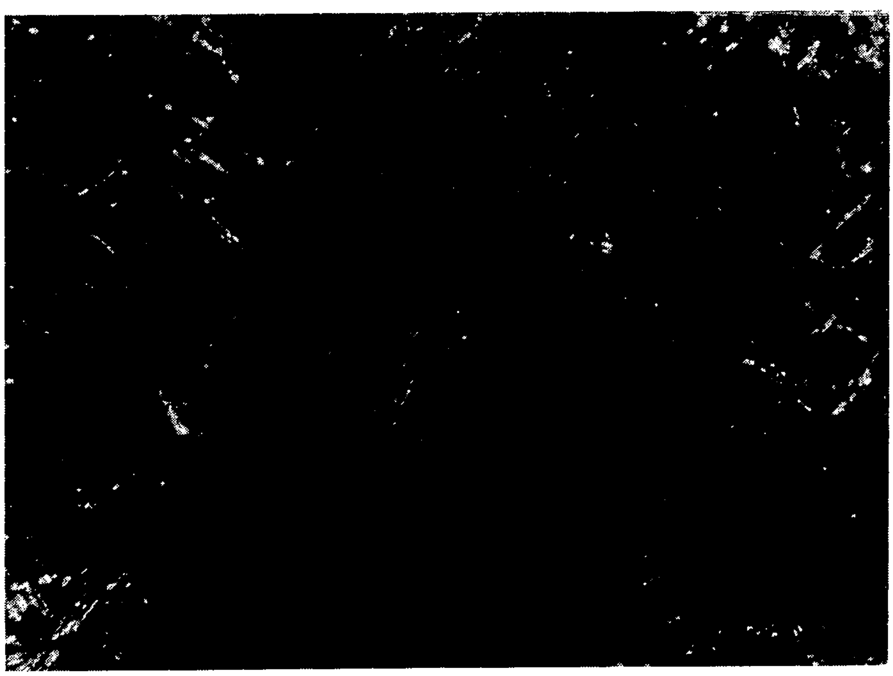


Figure 20.- Metallurgical view of diffusion brazed joint. (400x magnification)

Figure 21.- Sample model airfoil with top half finish machined.

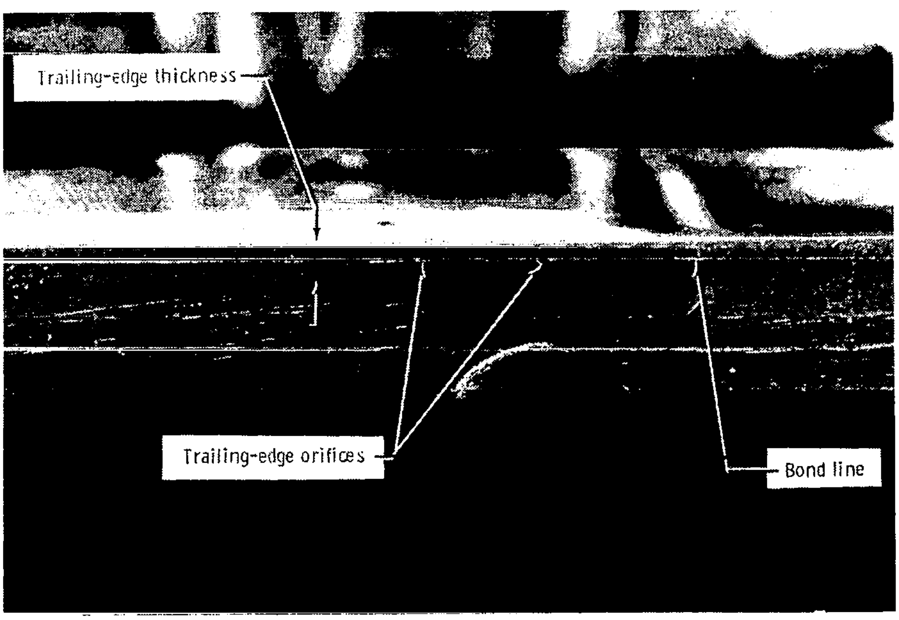


Figure 22.- Trailing-edge orifices in bond line of sample model airfoil.

INSTRUMENTATION SYSTEMS FOR THE NATIONAL TRANSONIC FACILITY

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OVERVIEW

Instrumentation and measurement systems are important elements in any complex research facility. The National Transonic Tunnel with its unique operational characteristics is clearly a complex facility and as such represents a significant challenge to wind tunnel instrument designers. This paper will briefly describe the instrument requirements imposed by the new testing environment, the instrument systems being provided for facility calibration and operation, and the research and development activities directed at meeting overall instrument and measurement requirements.

New Design Requirements

The approach taken to achieve high Reynolds number test capability in the NTF was through a combination of cryogenic operation and increased pressure. These factors singly or together affected all instrument designs and in some instances created new measurement requirements. The impact of this new operating environment is clearly illustrated by the typical NTF operating envelope shown in figure 1. As a point of reference, the shaded area at the lower left depicts the operating range of existing transonic tunnels. The vertical scale is essentially proportional to temperature and covers a range varying from ambient down to 77 K (140°R). The cryogenic environment requires that instruments be developed capable of obtaining accurate data by direct operation at cryogenic conditions or by providing instruments with thermally controlled and conditioned enclosures. Both approaches have been pursued with success. Force balances have been developed and evaluated at cryogenic temperatures with extremely good results. Pressure instrumentation, on the other hand, requires a thermally controlled environment for successful operation at reduced temperature. High tunnel operating pressure, represented by the horizontal scale on figure 1, results in significantly increased model loads. This requires that force balances be designed with extremely high capacities--up to 85 000 N (19,000 pounds) of normal force as an upper limit. The high anticipated model loads also dictate the need for a model deformation measurement system which represents both a new and very difficult measurement requirement. Instrument design is further impacted by the wide operating range achievable in NTF represented by the area of the operating envelope in figure 1. This may require, for example, that more than one balance is needed to cover a particular test program if data accuracy is to be maintained. Although only the extreme and most severe test conditions and requirements have been discussed to this point, it should be emphasized that the tunnel can and will be operated at significantly less demanding conditions resulting in a less severe instrument design problem.

NTF Instrumentation Systems

A new, complex wind tunnel such as NTF requires a high degree of instrumentation to bring the tunnel on-line and for subsequent research operation. Four basic measurement systems are provided which include facility calibration instrumentation, process monitoring instrumentation, research model instrumentation, and data acquisition. The first two are facility related and are required to bring the tunnel into operation and to monitor critical facility functions. Over 1500 individual measurements, as outlined in figures 2 and 3, have been identified for these two categories of instrumentation. A significant feature of this instrumentation is that most of it will be retained in the tunnel for subsequent tunnel calibration checks, with a significant portion to be selected and used in conjunction with research data acquisition.

The number and type of instrumentation channels for research data acquisition are shown in figure 4. Included are the familiar model-related and tunnel parameter instrumentation typically required to acquire research data. Of note is the addition of a model deformation measurement system and the use of the newly developed Electronically Scanned Pressure Measurement System (ESP) for pressure model application. This list, although rather complete, can and no doubt will change as requirements change.

Data acquisition is a central function of any complex measurement system. The NTF data system complex, illustrated in figure 5, provides subsystems for research data acquisition, process monitoring, facility control, and data management. The key features of this system include redundancy provided by use of four identical computers and switchable peripherals, transfer of data among subsystems, real-time display, on-site posttest data analysis, a communication link to the Central Data Reduction Center, and national user access for pretest entry for test planning, software development, and debugging.

Research Instrument Development

To meet the challenges imposed by the cryogenic and high pressure operation in the NTF, a comprehensive development program was required. A good portion of this work, addressing the measurement areas listed in figure 6, began about 4 years ago and is still ongoing. Considerable progress has been made to date in most areas and is reflected in separate papers covering the first five categories listed in figure 6.

Progress also has been made on the remaining items listed. Mach number and LN_2 flow measurement presented both stringent accuracy and time response requirements. Fast response digital quartz pressure transducers were selected for the Mach number system which meet control system response requirements and provide the ± 0.002 Mach number uncertainty desired. For LN_2 flow measurement, the ultrasonic flowmeter was selected and is still undergoing test and evaluation. This sensor provides the fast response, accuracy, and dynamic range needed in the design of the NTF control system. Work is currently underway aimed at selecting and developing a suitable flow visualization system for NTF. Facility design dictates that any system considered must be located within the plenum of the tunnel--which is at cryogenic temperature. Both schlieren and shadowgraph approaches are under consideration and problems of alinement, thermal effects on windows, and other system components are currently being investigated. In addition, a schlieren system has been designed for the Langley 0.3-Meter Transonic Cryogenic Tunnel which will provide for direct facility testing and system evaluation. In direct support of the instrument development activities and selection of facility instrumentation, a detailed measurement analysis was initiated. The ultimate, and ambitious, aim of the effort is to analyze the total wind tunnel measurement process--from sensor through final data. To date an analysis of critical tunnel parameter measurements has been made which directly aided in the selection of primary tunnel instrumentation.

The measurements listed in figure 6 certainly do not comprise all the anticipated NTF instrument requirements. Future activities will include development of flow diagnostic instrumentation, dynamic stability, side-wall and flow--through force balances, gust and flutter instrumentation, and fast response model pressure instrumentation. This list is certainly not complete and it is anticipated that additional new measurement requirements will be generated once the facility is operational.

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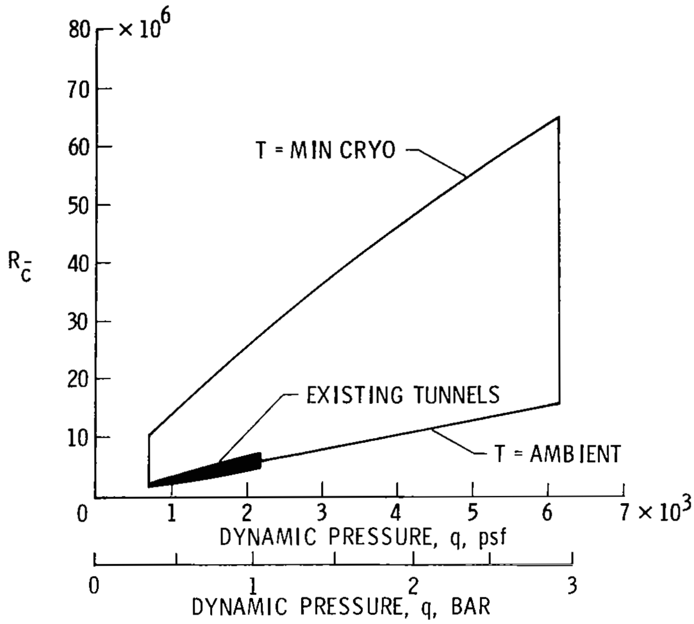

Figure 1.- Typical NTF operating envelope.
M = 0.90; c = 14.32 cm (0.47 ft).

- PRESSURE
  - WALL STATICS
  - CENTERLINE STATICS
  - CONTRACTION CONE RAKE
  - TEST SECTION RAKE
  - BOUNDRY LAYER RAKES
- TEMPERATURE
  - COOLING COIL SURVEY
  - TEST SECTION RAKE
- TURBULENCE & NOISE
- FLOW ANGULARITY

Figure 2.- Tunnel calibration instrumentation (1300 measurements).

- STRUCTURAL TEMPERATURES
- COMPONENT TEMPERATURES
- LN_2 FLOW
- PLENUM STATIC PRESSURE
- SCREEN DIFFERENTIAL PRESSURE

Figure 3.- Facility and process monitoring instrumentation (250 measurements).

- MODEL INSTRUMENTATION (CHANNELS)

• FORCE	(18)	• STRAIN	(10)
• PRESSURE	(1024)	• POSITION	(12)
• TEMPERATURE	(80)	• ACTUATORS	(12)
• AOA	(5)	• FOULING CIRCUITS	(2)

  • MODEL DEFORMATION

- TUNNEL PARAMETERS
  - MACH NUMBER
  - REYNOLDS NUMBER
  - DYNAMIC PRESSURE

Figure 4.- Number and type of instrument channels for initial operation.

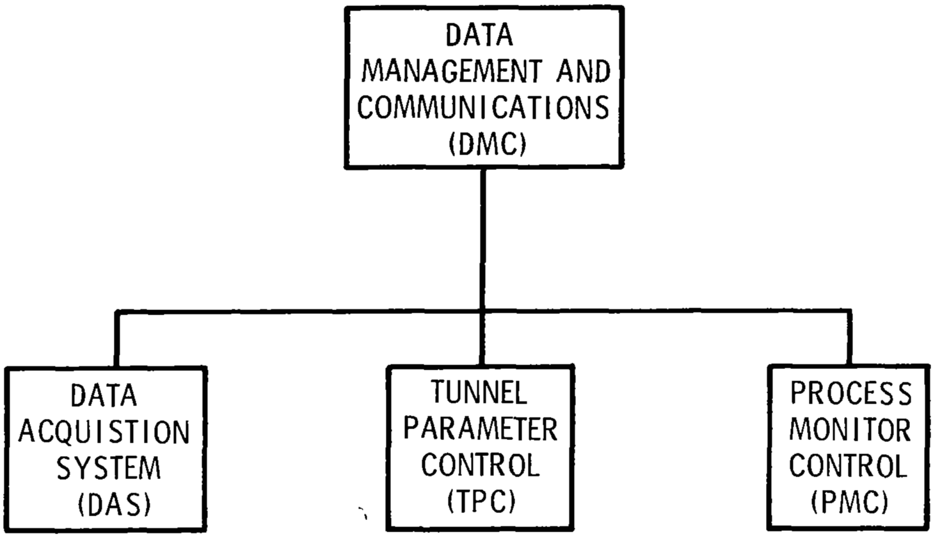

Figure 5.- Data system functional block diagram.

- FORCE BALANCES
- MODEL PRESSURE
- MODEL ATTITUDE
- TEMPERATURE
- MODEL DEFORMATION
- MACH NUMBER
- LN_2 FLOW
- FLOW VISUALIZATION
- MEASUREMENT ANALYSIS

Figure 6.- Ongoing research instrument development areas.

INSTRUMENTATION FOR CALIBRATION AND CONTROL OF A CONTINUOUS-FLOW CRYOGENIC TUNNEL*

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SUMMARY

This paper describes those aspects of selection and application of calibration and control instrumentation that are influenced by the extremes in the temperature environment to be found in cryogenic tunnels. A description is given of the instrumentation and data acquisition system used in the Langley 0.3-m transonic cryogenic tunnel along with typical calibration data obtained in a 20- by 60-cm two-dimensional test section.

SYMBOLS

c	chord of airfoil
M	Mach number
p	pressure
q	dynamic pressure
R	Reynolds number
T	temperature
$\bar{T}_t$	mean value of stagnation temperature
V	velocity

Subscripts

l	local value
ref	reference
t	stagnation value
∞	free-stream value

Abbreviations

GN_2	gaseous nitrogen
LN_2	liquid nitrogen
%	percent
2-D	two dimensional
NTF	National Transonic Facility
PRT	platinum resistance thermometer
TCT	0.3-m Transonic Cryogenic Tunnel

1. INTRODUCTION

The need for increased capability in terms of Reynolds number has been recognized since the earliest days of testing subscale models in wind tunnels. The need has been especially acute at transonic speeds where, because of the large power requirements of transonic tunnels, economic forces have dictated the use of relatively small tunnels.

In considering the various ways of increasing Reynolds number that have been tried or proposed for transonic tunnels, cooling the test gas to cryogenic temperatures (150 K or less) appears to be the best solution in terms of model, balance, and sting loads, as well as capital and operating cost.[1] In addition, having temperature as an independent test variable offers some new and unique aerodynamic testing capabilities which, in some instances, may be of equal importance with the ability to achieve full-scale Reynolds number.[2]

Personnel at the NASA Langley Research Center have been studying the application of the cryogenic wind-tunnel concept to various types of high Reynolds number transonic tunnels since the autumn of 1971 and, through extensive theoretical and experimental studies, have successfully demonstrated both the validity and practicality of the concept.[3-5]

There is, however, a price to be paid for the many advantages offered by a cryogenic wind tunnel. The price is added complexity in both the design and operation of the tunnel. Compared to an ambient temperature tunnel, the wide range of operating temperatures available in a cryogenic tunnel results in added complexity in such areas as model design and instrumentation, tunnel control, and the instrumentation used for tunnel calibration and control. This paper deals with the single area of instrumentation and associated equipment used in the calibration and control of cryogenic continuous-flow transonic pressure tunnels. Specifically, this paper addresses those aspects of instrumentation selection

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and application that are influenced by the extremes in the temperature environment to be found in cryogenic tunnels. The calibration and control instrumentation used in the Langley 0.3-m transonic cryogenic tunnel is described and examples of tunnel calibration data obtained in a 20- by 60-cm 2-D test section are given.

It should be noted that the use of trade names in this paper in no way implies endorsement or recommendation by the U.S. government.

2. INSTRUMENTATION SELECTION

In general, the process of selecting instrumentation for the calibration and control of a continuous-flow transonic pressure tunnel capable of being operated at cryogenic temperatures is no different from the process that would be followed for a similar tunnel that operates only at ambient temperatures. The steps to be taken are: (1) determine the accuracy requirements for the test parameters of Mach number M_∞, Reynolds number R, dynamic pressure q_∞, and velocity V_∞; (2) calculate the sensitivity of the test parameters to the uncertainties in the test conditions of total pressure p_t, static pressure p, and total temperature T_t; and finally, (3) select instrumentation which will meet the accuracy requirements.

2.1 Accuracy Requirements for M_∞, R, q_∞, and V_∞

The accuracy requirements of the test parameters will vary in a given tunnel depending upon the type of aerodynamic test to be performed. For most transonic applications, an accuracy in M_∞ of ±0.002 and accuracies in R and q_∞ of better than ±0.5% of reading are required. This accuracy requirement for Mach number can probably be relaxed for low-speed testing. However, larger variations in Mach number can cause significant changes in shock wave location at the higher transonic speeds. The accuracy requirements of R and q_∞ are independent of tunnel speed. In some applications, such as dynamic-stability testing, V_∞ is an important test parameter which must be determined. Here again, an accuracy of better than ±0.5% of reading is usually required.

2.2 Allowable Error in Pressure and Temperature Measurements

None of the test parameters - M_∞, R, q_∞, or V_∞ - can be measured directly with conventional instruments. However, all are functionally related to three quantities that can be measured directly, namely, stagnation pressure p_t, static pressure p, and stagnation temperature T_t. It then becomes necessary to determine the sensitivity of M_∞, R, q_∞, and V_∞ to the inaccuracies in the measured values of p_t, p, and T_t. Once this has been accomplished, the required accuracy for a given test parameter can be expressed in terms of the required accuracy for the pressure and temperature measuring instruments.

2.2.1 Allowable Error in Pressure Measurement

Calculations have been made to determine the allowable error in the measurement of stagnation pressure p_t for fixed errors in M_∞ and q_∞.[6] The results of these calculations are shown in Figure 1 as a function of M_∞. For an allowable error in M_∞ of ±0.002 and q_∞ of ±0.5% of reading, p_t must be measured to an accuracy of about ±0.25% of reading at the higher Mach numbers and to better than ±0.1% of reading at the lower Mach numbers. The sensitivity of M_∞ to errors in static pressure is the same as for stagnation pressure, but q_∞ is less sensitive, especially at the higher Mach numbers. It should be noted, however, that these are maximum errors for a single measurement. If the error analysis is based on most probable error or on standard deviation, the required instrumentation accuracy is not as great.

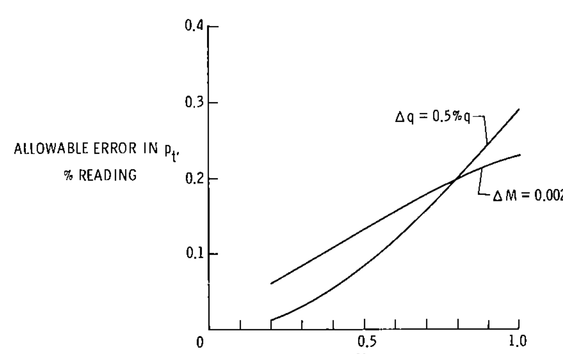


Fig. 1 Allowable error in p_t for fixed error in M_∞ and q_∞.

2.2.2 Allowable Error in Temperature Measurement

Calculations have also been made to determine the required accuracy of stagnation temperature T_t, necessary to have measurements of V_∞ and R accurate to ±0.5% reading.

Results of these calculations, in terms of allowable temperature error, are presented in Figure 2. Of the two parameters, R is the more sensitive to temperature, requiring an accuracy of ±1 K at ambient temperatures and about ±0.3 K at cryogenic temperatures. This is one area in which the accuracy requirements for instrumentation for a cryogenic tunnel are more severe than for an ambient temperature tunnel.

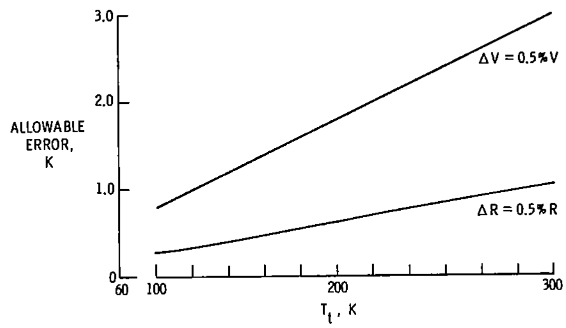


Fig. 2 Allowable error in T_t for 0.5% error in R and V_∞.

2.3 Other Considerations

The final step in the selection process is one of choosing from the available instrumentation that which will meet the accuracy requirements. However, other factors, such as the range of the test variable, response time, compatibility with the test environment, and cost and budget constraints, must also be considered.

2.3.1 Accuracy of Available Instrumentation

2.3.1.1 Pressure instrumentation.- The accuracy of three types of commercially available pressure transducers as a function of percent load is shown in Figure 3. Typical "high accuracy" strain-gage devices have an accuracy of about 0.25% of full scale. When converted to percent of reading, this increases to about 1% at 25% of full-scale load. Variable capacitance transducers have a quoted accuracy of 0.25% of reading throughout the load range but their cost is at least an order of magnitude greater than a strain-gage device. The quartz bourdon tube transducer is by far the most accurate, although it costs about 50% more than the variable capacitance transducers.

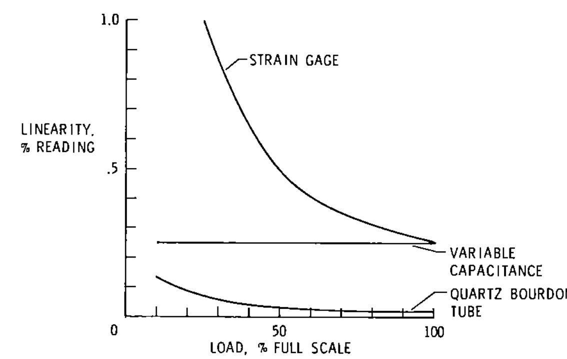


Fig. 3 Accuracy of pressure instrumentation.

Because pressure instrumentation accuracy on the order of 0.1% is required to give the desired accuracy in Mach number and dynamic pressure, the quartz bourdon tube instrument has been selected for the primary tunnel pressure instrumentation for both the 0.3-m TCT and the National Transonic Facility (NTF), the large transonic cryogenic tunnel now under construction at Langley. The quartz bourdon tube instrument is also used as the primary pressure instrumentation in other pressure tunnels at Langley. Unfortunately, the quartz bourdon tube pressure instrument has a response time that is generally much too slow for use with automatic tunnel control systems. Thus, as will be described in more detail later, less accurate but more responsive pressure instrumentation is used to provide the inputs to these systems.

2.3.1.2 Temperature instrumentation.- Two general types of temperature measurement devices are available for use at cryogenic temperature. These are resistance devices, such as the platinum resistance thermometer (PRT), and thermocouples, both of which are being used in the 0.3-m TCT. As shown on Figure 4, the PRT has an accuracy of ±0.25K which meets the accuracy requirement for measuring T_t. However, it also has a very slow response time, varying from 10 to 100 seconds, depending upon its design and size. Again, as was the case for the pressure instrumentation, the high accuracy devices do not have adequate response time for use in automated tunnel control systems. Thermocouples, which have accuracies from ±0.5 K to ±2 K, depending upon the temperature range, have response times which vary from about 0.1 to 20 seconds. This

- RESISTANCE DEVICES
  - PLATINUM RESISTANCE THERMOMETER (PRT)
  - ACCURACY ± 0.25 K
  - RESPONSE TIME 10 - 100 SECONDS
- THERMOCOUPLES
  - ACCURACY ± 0.5 K TO ± 2 K
  - RESPONSE TIME 0.1 - 20 SECONDS

Fig. 4 Temperature transducers suitable for cryogenic tunnels.

response time is a function of both the wire diameter and the type of thermocouple (bare wire, shielded, etc.), and for unshielded thermocouples in the smaller wire diameters, the response times are compatible with automatic tunnel temperature control systems.

One disadvantage of thermocouples is their very low voltage output, being on the order of a few millivolts. Another disadvantage with thermocouples is due to the fact that the net voltage output is a function not only of the base metals of the two wires but also of any material inhomogeneities which produce a parasitic voltage if located in a temperature gradient. Inhomogeneities in the wire can be produced by either a variation in the chemical composition along its length or by mechanical strain such as a kink.

The presence of defects such as these can be detected by testing lengths of thermocouple wire in a temperature gradient by using a device, such as shown in Figure 5, which runs the wire through a liquid nitrogen bath and records the voltage output resulting from the temperature gradient. However, the effects of inhomogeneities cannot be readily corrected. Typical results obtained at Langley by Germain on three types of thermocouple wire in the test apparatus shown in Figure 5 are presented in Figure 6. The voltage spikes for the type E (Chromel vs. Constantan) and K (Chromel vs. Alumel) thermocouples are typical of those recorded throughout the length of wire tested whereas the spike for the type T (Copper vs. Constantan) thermocouple was the only one found in that sample of material.

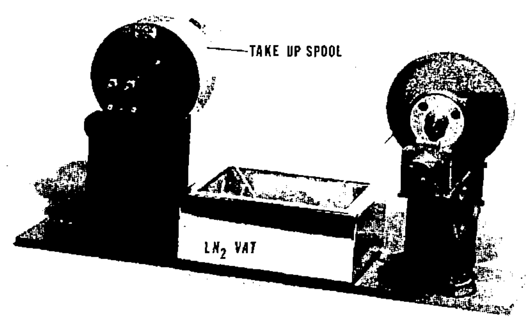


Fig. 5 Homogeneity test rig.

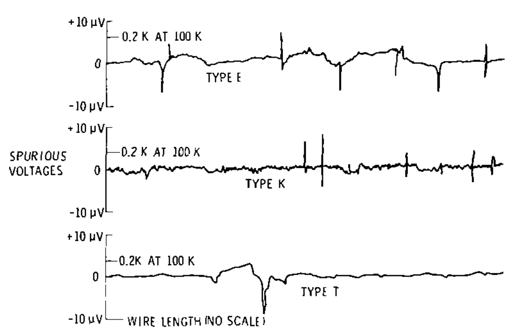


Fig. 6 Typical homogeneity test results.

Based on these results and on company literature, it appears that having one wire of pure copper makes it easier to control the overall chemical composition of type T thermocouples than either types E or K which have both wires of alloyed metal. For these reasons, type T thermocouples are being used in the 0.3-m TCT and have been selected for use in the NTF. An additional practical advantage of the type T thermocouple is that it is available in a "premium quality" wire which has a good (±1%) match to published voltage standards at cryogenic temperatures. None of the other types are so available. Our experience over the years has been that only the type T "premium" thermocouple remains within the ±1% band at cryogenic temperatures.

The standard limits of error for commercial premium grade type T thermocouple wire are presented in Figure 7 as a function of temperature. These results, from a calibration of four selected thermocouples, show their error to be only about one-half of the limit of error specification, with the largest error occurring in the cryogenic temperature region. Using calibration data such as these can increase the accuracy of a thermocouple reading, but it must be remembered that the effect of inhomogeneity cannot be accounted for.

2.3.2 Range of Test Variables

In the selection of instrumentation to meet the required accuracy, the range of the variable to be measured can be a major factor. For example, a liquid column manometer is a very accurate device for pressure measurement. However, if the pressure level exceeds two atmospheres or so, the height of the liquid column makes it impractical to use a manometer. The range of test conditions

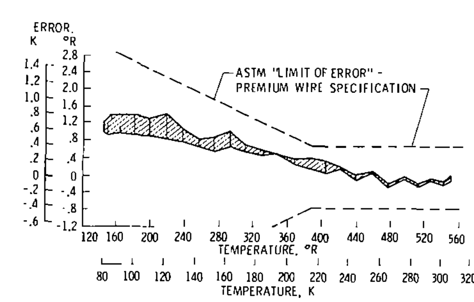


Fig. 7 Calibration of type T thermocouples.

for both the 0.3-m TCT and the NTF is shown in Figure 8 to illustrate the ranges to be encountered in these tunnels as well as to indicate the ranges likely to be encountered in other transonic cryogenic tunnels. The largest difference to be noted between these two tunnels is in stagnation pressure where the maximum pressure in the NTF is about 50 percent greater than in the 0.3-m TCT. The differences in Mach number do not affect the selection of instrumentation to any great extent.

	0.3-m TCT	NTF
MACH NUMBER, M_∞	0.1 - 0.9	0.2 - 1.2
STAGNATION TEMPERATURE, T_t, K	78 - 340	78 - 339
STAGNATION PRESSURE, p_t, atm	1.0 - 6.12	1.0 - 8.85
STATIC PRESSURE, p, atm	0.5 - 6.12	0.5 - 8.85

Fig. 8 Range of test conditions.

2.3.3 Response Time

Because of the relatively high cost of operation of high Reynolds number tunnels, fast response times are desirable for all instrumentations. Response time is especially important when an instrument is used to provide an input signal to an automatic closed-loop tunnel control system. For example, the tunnel control system inputs for the 0.3-m TCT come from instruments with as short a response time as practical, at the sacrifice of some accuracy. Separate dedicated instrumentation, which is less responsive but more accurate than that used with the tunnel controls, is used to give highly reliable readings of the test variables p_t, p, and T_t. Details of this particular instrumentation as well as further discussion of response times are given in subsequent portions of this paper.

3. INSTRUMENTATION USED IN THE 0.3-m TCT

In the 0.3-m TCT, the practice of isolating instrumentation from the cryogenic environment has been adopted when possible. With the obvious exception of temperature transducers, most instrumentation is located exterior to the tunnel pressure shell, thus completely avoiding problems related to temperature variation. If a device must be located in the cryogenic environment, it is enclosed in an insulated container which is maintained automatically at approximately 300 K by using a thermostatically controlled electric heater. Devices such as slidewire potentiometers, digital shaft encoders, and pressure scanning valves have been successfully operated within the tunnel pressure shell in this manner. However, with each of these systems there are heaters, thermostats, wires, electrical connections, electrical feedthroughs, and power supplies which add to the complexity of the system and, inevitably, reduce its reliability. On many occasions, run time in the 0.3-m TCT has been lost due to the failure of one of these simple support devices rather than due to any failure in the instrument itself. Based on our experience with the 0.3-m TCT, we must conclude that it is best to locate electronic equipment exterior to the tunnel in an ambient temperature and pressure environment if at all practical.

Some details of the instrumentation presently being used in the 0.3-m TCT are now described.

3.1 Pressure Instrumentation

For two-dimensional airfoil tests, the 0.3-m TCT is equipped to obtain static-pressure measurements on the airfoil surface, total head measurements in the airfoil wake, and static pressures on the test section sidewalls, floor, and ceiling. Static-pressure taps are also located throughout the tunnel circuit to obtain information on the performance of the contraction and diffuser sections, fan pressure rise, and pressure losses across various elements of the tunnel circuit. To measure the pressures for such a large number of points, a scanning valve system capable of operating ten 48-port scanning valves is used. Because of the large changes in dynamic pressure of the tunnel over its operational range (a factor of about 75), conventional strain-gage pressure transducers are not used. Instead, we use commercially available high-precision capacitive potentiometer-type pressure transducers.[7]

The pressure transducers are mounted in instrument racks adjacent to the test section in order to reduce response time. To provide increased accuracy, the transducers are mounted on thermostatically controlled heater bases to maintain a constant temperature and on "shock" mounts to reduce possible vibration effects. The electrical outputs from the transducers are connected to individual signal conditioners located in the tunnel control room. The signal conditioners are autoranging and have seven ranges available. As a result of the autoranging capability, the analog electrical output to the data acquisition system is kept at a high level even though the pressure transducer may be operating at the low end of its range.

Pressure transducers with a maximum range of 6.8 atm are available for model and tunnel wall pressure. More sensitive transducers, having a maximum range of 1.36 atm, are available for pressure measurements in the airfoil wake. The transducers have an accuracy of ±0.25% of reading from -25% to 100% of full scale. At present about 35 pressure transducers are available. However, the system is being expanded and 125 pressure channels will be available in early 1982.

To determine M_∞ to an accuracy of ±0.001, it is necessary to use instrumentation that is much more accurate than the capacitive potentiometer type pressure transducer. As a result, the commercially available quartz bourdon tube type of pressure gage previously described is used for the measurement of p_t, p, and the reference pressure used on other differential pressure transducers. This system has an accuracy of about 0.01% of full scale at low pressures to about 0.02% of full scale at the high end of its range.

3.2 Temperature Instrumentation

The characteristics of both thermocouples and platinum resistance thermometer (PRT) temperature sensors have been previously described. Based on response time, accuracy, and to some extent cost, it has been necessary to use both thermocouples and PRT's in the 0.3-m TCT.

Type T thermocouples are used for a fixed temperature survey rig located in the settling chamber of the tunnel. In this application, we are trying to determine the distribution of temperature and the absolute level is relatively unimportant. The various thermocouples on the rig are made from the same roll of wire and care was taken to avoid kinking the wire or introducing other possible sources of error into the system.

In addition, type T thermocouples are used in the test-section—plenum area to monitor temperature differences during and immediately after tunnel cooldown in order to avoid taking aerodynamic data with the relatively thick test section walls distorted due to thermal gradients. Although the amount of thermally induced distortion is small, even small changes in test section area have a major effect on the longitudinal distribution of M_∞ at the higher values of M_∞.

As previously mentioned, because of the need for rapid response time, the type T thermocouple is also used as the temperature sensor for the closed-loop automatic temperature control system. In this application, the rapid response requirement has been achieved at the expense of accuracy.

The reference temperature for the thermocouples is provided by a commercially available "ice point" junction which automatically maintains the reference junction at 273.2 K ± 0.025 K.

The values of T_t which are used to calculate the tunnel parameters are obtained from PRT's located just downstream of the screen section. As previously described, the higher accuracy requirements for T_t have been achieved at some sacrifice in response time by using PRT's. Although the long response times of the PRT's would introduce problems if they were used in the automatic temperature control system, the nature of the testing in the 0.3-m TCT is such that sufficient time is available for the PRT's to come to equilibrium with the stream before temperature readings are required for the accurate determination of T_t.

3.3 Mass-Flow Measurement and Control Instrumentation

One special instrumentation requirement for the 0.3-m TCT is associated with the 2-D test section sidewall boundary-layer removal system. The requirements for this system are to remove from 1% to 4% of the test section mass flow through porous plates in the test section sidewalls and to measure and control the removal rate to within 1% of the desired set point. With the large operational range of the 0.3-m TCT, the ratio of maximum to minimum removal rate is about 140 to 1. This range of mass-flow control and its measurement to the required accuracy is well beyond the capability of conventional mass-flow control and measurement devices.

Two commercially available (Process Systems Incorporated) microprocessor controlled 11 bit digital valves have been selected to meet these requirements since they have the ability to handle both the control and measurement functions. These digital valves are similar to those presently being used at the 0.3-m TCT for the control of LN_2 injection into the tunnel and the control of GN_2 exhaust from the tunnel. Although the 11 bit valve does not give the desired 1% of reading accuracy over the entire mass-flow range, it gives the desired accuracy over the most important portion of the operating envelope. The two digital valves and their associated control microprocessor are currently under operational check-out in the tunnel.

3.4 Traversing Wake-Survey Probe

A vertical traversing probe system is located on the left sidewall of the 2-D test section downstream of the turntable. The mechanism has a traversing range of 25.4 cm. The distance between the airfoil and the centerline of the probe support can be varied with the probe capable of being located either at tunnel station 26.0 cm or at tunnel station 31.1 cm. The probe is driven by an electric stepper motor and can operate at speeds from about 0.25 cm/sec to about 15 cm/sec. The stroke and speed can be remotely controlled from the operator's panel in the control room. Although the primary purpose of this system is to survey the total pressures in airfoil wakes by using a pitot tube survey rake, the probe can be equipped with other types of instruments such as thermo-

Fig. 9 View of 2-D test section showing airfoil and survey rake.

couples or hot wires. Figure 9 is a general view of the 2-D test section which shows an airfoil in place and the present pitot-tube survey rake located in its most forward position in the test section. For this configuration, the pitot tubes are located 0.88 chord downstream of the airfoil trailing edge.

Details of the multitube pitot probe are shown in Figure 10. Three disc type static probes as well as six pitot probes are mounted on the assembly. Tunnel sidewall static pressure taps are also provided in the plane of the pitot probes for use in the determination of the airfoil drag coefficient. Individual transducers are used for each tube on the probe assembly in order to keep pressure response time low. The vertical position of the probe is recorded on the data acquisition system using the output from a digital shaft encoder geared to the probe drive mechanism.

3.5 Flow-Quality Instrumentation

During the calibration of a wind tunnel it is highly desirable to obtain information on the flow quality as well as on the Mach number distribution. The flow quality is generally determined by measurements of sound pressure level and velocity fluctuations in the different planes.

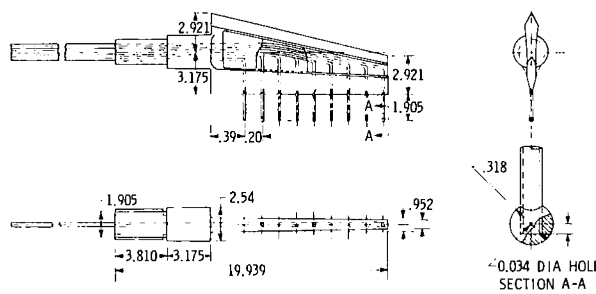


Fig. 10 Details of wake-survey probe.

The sound pressure level can be measured by the use of piezoelectric transducers. One such device has been used in the test section of the 0.3-m TCT at Mach numbers up to 0.9 for stagnation pressures to 5 atm.[5] During these tests, T_t was varied from ambient to near liquid nitrogen temperatures. Because this particular transducer has a sensitivity to temperature of about 0.05% per 1 K, it must be calibrated dynamically throughout its operational temperature range if accurate measurements are required. This type of calibration is difficult to obtain and little has been done to date.

Velocity and temperature fluctuations can be measured by using hot-wire probes. To date, hot wires of 5 micron diameter and a length-to-diameter ratio of about 250 have been successfully used in the 0.3-m TCT only to Mach numbers of about 0.1 for stagnation pressures up to 5 atm under cryogenic temperature conditions. Further tests are planned in the near future using 3.8 micron diameter wire at higher Mach numbers. The limit to which hot-wire probes may be used has not yet been determined.

Laser doppler velocimeter (LDV) systems can also be used to determine fluctuating velocities in a tunnel of this type. A simple LDV setup has been used to make preliminary tunnel empty measurements. Improvements to the system are now being made and further testing will be done in the near future.

4. 0.3-m TCT DATA ACQUISITION SYSTEM

Data from the 0.3-m transonic cryogenic tunnel are recorded on magnetic tape at the Langley central data recording system. This system, which was installed in the late 1950's, serves several wind tunnels and other test facilities on a time-share basis. A total of 99 analog channels of recording capability are available at the tunnel with a maximum range of 100 mV and a resolution of 1 part in 10,000. In a continuous mode, all data channels can be scanned at a sample rate of 20 per second while in a single scan mode the maximum rate is about 4 scans per second. All analog data are filtered with a 4 Hz low pass filter.

A small computer is used to sequence the data acquisition system, provide timing input signals for the scanning-valve drive system, provide real-time visual displays and plots, and control the angle-of-attack and wake-survey-probe drive systems. The computer is programmed to allow the recording of from one to nine single frames of data for each port on the scanning valves. The time between each frame of data and the dwell time on the port before the first frame of data is taken are both variable inputs.

All inputs to the computer are made through a teletype keyboard. An X-Y plotter is used to produce real-time plots of pressure distribution over the airfoil and total head loss through the airfoil wake. Other real-time displays include digital readouts of Mach number and Reynolds number. Angle-of-attack and wake-survey-probe drive commands are also entered through the teletype and are transmitted through the computer to the respective drive system.

The wake-survey drive can operate in either of two modes. In the first or manual mode, the initial and final location of the probe in the tunnel as well as the number of steps between are the input parameters. In the second or automatic mode, the computer determines the upper and lower boundaries of the airfoil wake automatically by first making a continuous sweep of the survey probe through the tunnel before the data recording sequence begins. The wake-survey probe is synchronized with the scanning valves so that the probe is moved to a different vertical location each time the scanning valves are advanced to a new port. If more survey probe points are desired than scanning valve ports, the probe will continue on its traverse after the scanning valves have reached their last port.

5. TYPICAL CALIBRATION RESULTS FROM THE 0.3-m TCT

A sketch of the Langley 0.3-m TCT is presented in Figure 11. In this view the flow is clockwise. An array of thermocouples is located at the upstream end of the screen section to measure vertical and lateral temperature distributions. The primary pressure and temperature sensors are located in the contraction section, just downstream of the screen section. Details of the plenum and 2-D test section of the 0.3-m TCT are shown in Figure 12. Pressure taps are located in

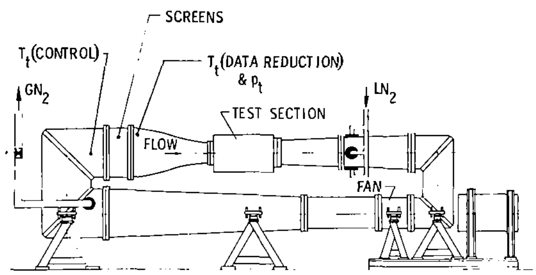


Fig. 11 Sketch of Langley 0.3-m TCT.

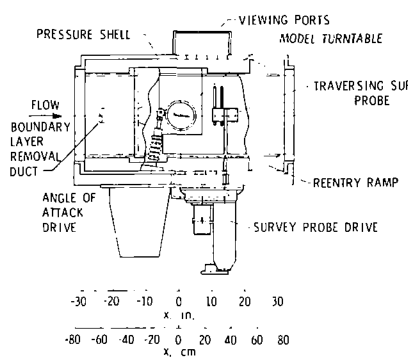


Fig. 12 Sketch of 2-D test section.

longitudinal rows along the centerline of both top and bottom slotted walls and along the centerline of one solid sidewall. Both left and right turntables are also instrumented with pressure taps for tunnel empty calibration purposes. The reference static pressure is measured on a manifold which connects four orifices, two on each sidewall located as far upstream of the turntable as practical.

5.1 Mach Number Distribution

Typical results of the longitudinal distribution of local Mach number along the centerline of the top slotted wall are presented in Figure 13. This distribution is for a stagnation pressure of 3.1 atm and a stagnation

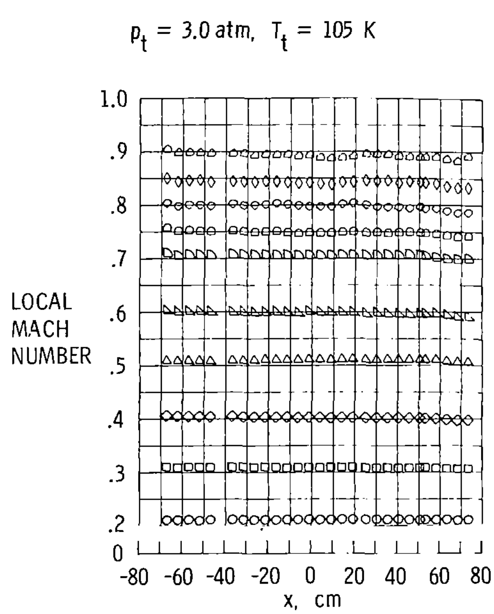


Fig. 13 Typical local Mach number distribution along top slotted wall.

temperature of 105 K. The top and bottom slotted walls were diverged about 0.5° from the centerline whereas the two solid sidewalls were parallel to the centerline. The data show that for Mach numbers below about 0.75, the distribution is relatively uniform to about 50 cm downstream of the center of the turntable where the speed begins to drop off. This location roughly corresponds to the location in the test section where the slots begin to open rapidly. At higher Mach numbers, a negative gradient is observed, indicating the walls were diverged slightly more than necessary.

Figure 14 shows the local Mach number distribution on one of the model turntables for the same conditions of 3.1 atm and 105 K. As shown on the sketch, there are 5 rows of pressure taps on the turntable, two horizontal, two diagonal, and one vertical, numbered 1 through 5. This figure again shows the longitudinal gradient in rows 1 through 4, but the data from row 5 indicate essentially no vertical gradient.

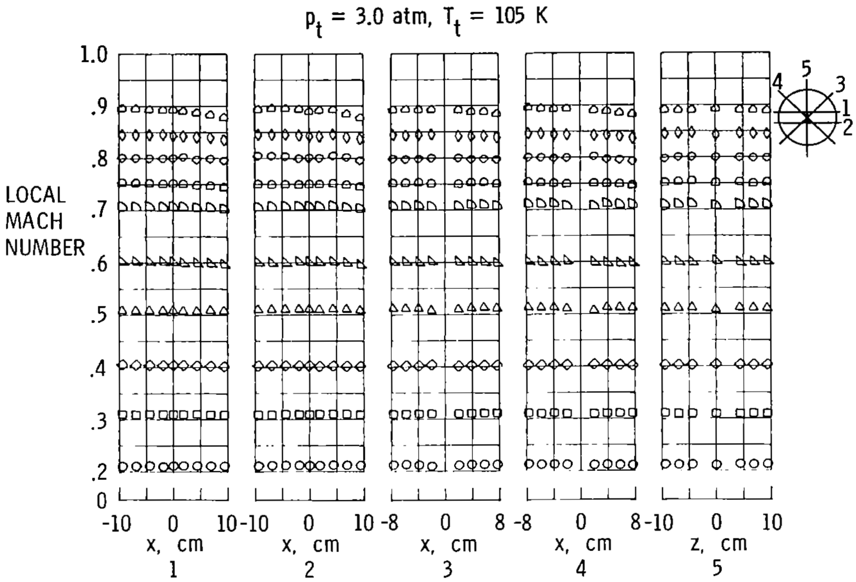


Fig. 14 Typical local Mach number distribution on model turntable.

The test section Mach number M_∞ has been defined based on the average of the 36 pressure taps located on the turntable. The calibration factor ΔM is defined as the difference between the calculated reference Mach number M_{ref} and the test section Mach number M_∞. A plot of the calibration factor is presented in Figure 15 as a function of M_{ref}. Data are presented for four stagnation pressures and at three stagnation temperatures at each pressure. These values of temperature and pressure include combinations which are close to the operational extremes of the tunnel. Although a large increase in the calibration factor occurs at Mach numbers above approximately 0.75, due to the previously discussed longitudinal gradient, it should be noted that there is little effect of temperature and pressure on the calibration factor. The scatter of the data is generally within an error band of about 0.001 which, it should be noted, is within the accuracy of the test instrumentation.

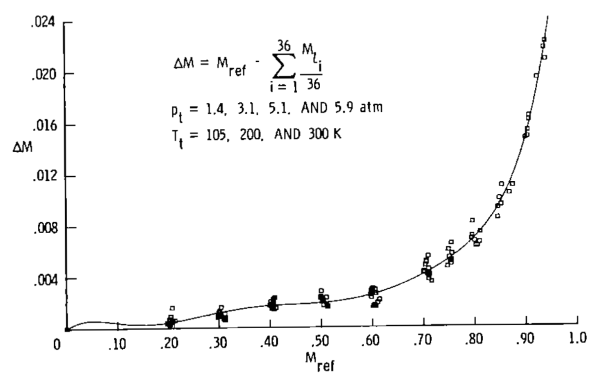
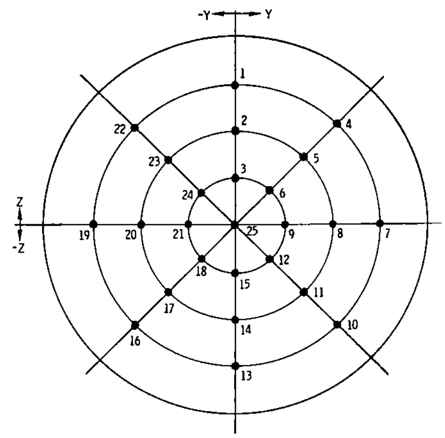


Fig. 15 Mach number calibration for 2-D test section of 0.3-m TCT.

5.2 Temperature Distribution

A typical lateral temperature distribution in the screen section of the tunnel is presented in Figure 16 for a Mach number of 0.70 and a stagnation pressure of 5.9 atmospheres. As shown on the figure, there are eight radial rows of instrumentation with three thermocouples on each row, spaced equidistant between the centerline and the wall. There is also a single thermocouple located on the centerline. Each temperature shown on the figure is the average of 185 readings of the thermocouple during a time span of 72 seconds. During this time, the tunnel was at a steady state condition and pressure data were being obtained using scanning valves. For this case, which is near the maximum Reynolds number capability of the tunnel, the average of the 25 thermocouples is 105.9 K and the standard deviation, σ, is 0.20 K. As mentioned previously, the absolute value of this temperature may be slightly in error due to the accuracy of the thermocouple wire, but the differential temperatures should be reliable.



T_1 106.0 K	T_6 106.2 K	T_{11} 106.0 K	T_{16} 105.4 K	T_{21} 105.8 K
T_2 106.0	T_7 106.0	T_{12} 105.9	T_{17} 106.0	T_{22} 105.8
T_3 106.2	T_8 106.0	T_{13} 105.7	T_{18} 105.6	T_{23} 105.6
T_4 106.1	T_9 106.2	T_{14} 105.7	T_{19} 105.7	T_{24} 106.0
T_5 106.0	T_{10} 105.8	T_{15} 105.8	T_{20} 105.7	T_{25} 105.9

Fig. 16 Typical lateral temperature distribution.

Data such as presented in Figure 16 have been obtained throughout the operational range of the tunnel during the tunnel empty calibration process. Results, in the form of standard deviation, are presented in Figure 17 for selected Mach numbers at three different stagnation temperatures and for near minimum and maximum stagnation pressures. The results show that the variations in standard deviation of stagnation temperature in the screen section of the tunnel are less than about 0.5 K and that the results do not vary in any systematic way with either Mach number or stagnation temperature. The value of the deviation does, however, tend to be lower at the higher stagnation pressure.

	T_t, K: 105		200		300	
M_∞ \ p_t, atm	1.4	5.9	1.4	5.9	1.4	5.9
0.20	0.51 K	0.16 K	0.49 K	0.23 K	0.12 K	0.05 K
.30	.43	.16	.50	.15	.13	.09
.40	.38	.14	.41	.12	.11	.12
.50	.31	.19	.50	.13	.24	.24
.60	.27	.21	.41	.16	.19	.31
.70	.21	.20	.35	.21	.50	.30
.80	.18	.18	.27	.19	---	---
.90	.17	---	.37	---	---	---

Fig. 17 Standard deviation of stagnation temperature survey.

6. CONCLUSIONS

Some of the aspects of selecting and using instrumentation for the calibration and control of continuous-flow cryogenic wind tunnels have been discussed and examples of instrumentation used in the Langley 0.3-m TCT have been given in this paper. In addition, typical calibration results from the 0.3-m TCT have been described. The main conclusions to be drawn from our experience at Langley are:

(1) Adequate pressure and temperature instrumentation is commercially available to meet the requirements for calibration and control of continuous-flow cryogenic wind tunnels.

(2) The response time of high accuracy instrumentation is usually much too slow for use with automatic tunnel control systems. Thus, it may be necessary to use one set of less accurate but highly responsive instrumentation for control purposes and a separate set of highly accurate but less responsive instrumentation for the determination of the test conditions.

(3) Thermal isolation of the instrumentation from the cryogenic environment is the best way of avoiding problems related to temperature variation. However, the ancillary equipment needed for thermal isolation inevitably results in reduced reliability.

(4) Conventional flow-quality instrumentation, such as hot-wire probes, has yet to be used successfully except at extremely low speeds in the 0.3-m TCT.

(5) The use of a piezoelectric transducer to measure sound pressure levels has been demonstrated. However, because these transducers are sensitive to changes in temperature and are exposed to the cryogenic environment, they must be dynamically calibrated over the entire operational temperature range of the tunnel if high accuracy measurements are desired.

(6) For applications such as sidewall boundary-layer removal systems, measuring and controlling mass flow can be realized to any desired degree of accuracy by using commercially available "digital" valves.

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8. ACKNOWLEDGEMENT

The authors are grateful to Mr. E. F. Germain of the Instrument Research Division of the Langley Research Center for providing much of the information relative to thermocouple selection.

ONSET OF CONDENSATION EFFECTS IN CRYOGENIC WIND TUNNELS

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ABSTRACT

The onset of condensation effects in cryogenic wind tunnels limits their minimum operating temperatures. If this onset of effects occurs below saturation temperature, then the tunnels may be operated at the lower temperatures and additional benefits to cryogenic tunnel operation, such as increased Reynolds number capability and reduced operating costs, will result. Both homogeneous and heterogeneous nucleation processes are discussed as they pertain to continuous-flow cryogenic wind tunnels. Examples from condensation experiments in the Langley 0.3-m Transonic Cryogenic Tunnel are also reviewed.

INTRODUCTION

The subject of this paper is the minimum operating temperatures of cryogenic wind tunnels as determined by the onset of effects due to condensation in aerodynamic test data. The goals and objectives of this research effort are illustrated in figure 1. Unit Reynolds number and drive-fan power requirements are shown over a total temperature range from 322 K down to a total temperature fixed by saturation, where saturation represents those flow conditions where the static pressure and temperature at a given Mach number fall on the vapor-pressure curve for nitrogen. As seen in figure 1, there is a dramatic factor of 6 increase in Reynolds number capability and a significant reduction in drive power for operation at a constant value of tunnel total pressure of 1 atm when varying temperature down to its saturation value. (That nitrogen is a suitable test gas over the temperature range shown and for total pressures up to 9 atm has been discussed in references 1 to 3.) The objective of the condensation work is to determine for a range of operating pressures the temperatures at which the onset of condensation effects on aerodynamic test parameters actually occurs. If, as is expected, the minimum operating temperature boundary is below the saturation temperature, then additional increases in Reynolds number and decreases in drive power beyond those seen in figure 1 will be possible.

Any additional increase in Reynolds number due to the temperature reduction below saturation may be utilized in several ways. First, by operating at the maximum tunnel total pressure, it will be possible to maximize the available unit Reynolds number. Second, if a given model is limited because of structural constraints to testing at some reduced total pressure, it will be possible to maximize the chord Reynolds number for the model at the structural total pressure limit. Finally, it would also be possible to use the additional Reynolds number capability due to temperature reduction to reduce the tunnel total pressure required for operating at a fixed value of Reynolds number that falls within the model or tunnel operating limits. In other words, if maximizing Reynolds number is not necessary, then this additional capability due to temperature reduction can be used to reduce the operating total pressure necessary to achieve the fixed value of Reynolds number.

Reduced operating total pressure reduces drive power required, which in turn leads to reduced liquid nitrogen required for absorbing the heat added to the stream by the drive fan. For operating at a fixed value of unit Reynolds number, figure 2 shows that there is approximately a 2 percent decrease in required liquid nitrogen consumption to absorb the heat of the drive fan for a 1° reduction in total temperature while operating at temperatures close to saturation. Since liquid nitrogen costs can dominate operating expenses, there are potentially worthwhile savings in operating costs if operation below saturation temperatures is possible.

SYMBOLS

C_p	static pressure coefficient, $\frac{p - p_\infty}{q_\infty}$
c	airfoil chord, m
M	Mach number
p	pressure, atm (1 atm = 101 kPa)
q	dynamic pressure, atm
R	Reynolds number
r	radius of droplet, m
r_c	critical radius of droplet, m
T	temperature, K
x	linear dimension along airfoil chord line, m
α	airfoil angle of attack, degrees

Subscripts:

L	local conditions
t	total conditions
∞	free-stream conditions

TYPES OF NUCLEATION PROCESSES

There are two distinct processes that can occur in cryogenic wind tunnels that may lead to the onset of condensation effects. The primary difference between the processes is related to the availability of seed particles. If there are not enough pre-existing seed particles, or nuclei, present in the flow to cause condensation then the gas must form its own seed particles through the formation of liquid droplets and thus undergo what is called homogeneous nucleation. If there is a sufficient number of seed particles already present in the flow, then condensation can occur directly on these seeds without having to wait for the formation of liquid droplets out of the gas itself. This second process is called heterogeneous nucleation, where the prefix "hetero" refers to the mixture of gas molecules and the pre-existing seed particles.

The formation of liquid droplets, which is so essential to homogeneous nucleation, has an energy barrier associated with it. Droplets have surface tension and this surface tension results in a surface energy. This required surface energy results in an energy barrier to liquid droplet formation. As shown in figure 3, in minimizing energy of formation, all drops smaller than the critical radius r_c tend to evaporate while all droplets larger than r_c tend to grow. At saturation temperature the value of r_c is infinite but gets progressively smaller as the local values of p and T go further below the vapor-pressure curve. The onset of condensation effects due to homogeneous nucleation occurs when the value of r_c is small enough that enough droplets of radius greater than r_c are being formed by random collisions of the gas molecules to affect the aerodynamic data.

Homogeneous nucleation can be predicted by classical liquid droplet theory as discussed in reference 4. One investigation of the onset of condensation effects due to homogeneous nucleation in nitrogen was reported by Sivier in reference 5 and is summarized in figure 4. This figure summarizes the results of a computer study in which Sivier predicted the onset of condensation effects in a one-dimensional nozzle for a variety of different nozzle expansion rates and for a variety of different total conditions. The onset of effects was found to fall within the two lines of figure 4 to the left of the nitrogen vapor-pressure curve. Since a flow isentrope would follow a path from the upper right of the figure toward the lower left, it can be seen that the distance below the vapor-pressure curve that one may go before the predicted onset of homogeneous effects may be a much stronger function of onset pressure than of factors such as expansion rates. The experimental data used for comparison by Sivier were all taken in hypersonic wind tunnels, where the static values of p at onset were 0.01 atm or less. However, in the case of the Langley 0.3-m Transonic Cryogenic Tunnel (TCT) or the National Transonic Facility (NTF), p at onset may be on the order of 3 or 5 atm. Consequently, one goal of the condensation research at Langley has been to verify or correct the onset band of Sivier in the pressure range of interest to high-pressure, transonic, cryogenic tunnels. To this end, research is still continuing.

In the case of heterogeneous nucleation, where condensation occurs on pre-existing seed particles, it is not necessary to overcome an energy barrier to droplet seed formation since the seed particles already exist. Consequently, growth can be expected as soon as the flow becomes saturated. Of course, whether or not the onset of condensation effects will occur depends upon the seed particles. To cause effects there must be enough pre-existing seed particles to allow a sufficient amount of gas molecules to condense and thermodynamically influence the flow. As explained in reference 6, the occurrence of condensation effects depends on the size and number density of the pre-existing seed particles, how far below the vapor-pressure curve the values of p and T are, and the length of time the flow is exposed to these temperatures below the vapor-pressure curve. Therefore, once a flow field can be approximated, the most important quantity for predicting condensation effects due to heterogeneous nucleation is the initial size and number density of the pre-existing seed particles.

The most likely source of seed particles in continuous-flow cryogenic tunnels is the liquid nitrogen injection process, which is used to absorb the heat of the drive fan. First, the liquid that is injected into the tunnel is broken up by the injecting spray nozzles and the aerodynamic forces arising from the tunnel flow past the injection station (see ref. 6). Whether or not the resulting liquid droplets completely evaporate before they are swept by the tunnel flow from the injection station to the test section depends on the temperature of the tunnel (see fig. 5 for the geometry of the 0.3-m TCT). Certainly at some low value of temperature the bulk of the injected liquid will not evaporate and may provide a sufficient number of seed particles in the test section to trigger condensation effects due to heterogeneous nucleation. Second, the injected liquid nitrogen itself may contain a sufficient number of solid or liquid impurities with higher boiling temperatures than nitrogen to provide a background of pre-existing seed particles sufficient to cause effects. In contrast to the number density of unevaporated nitrogen droplets, the number density of the impurities may be relatively insensitive to tunnel operating temperature.

EXPERIMENTS IN THE 0.3-M TCT

To answer the questions regarding the onset of effects due to homogeneous and heterogeneous nucleation, a number of experiments have been conducted in the 0.3-m TCT at Langley. They have primarily involved an NACA 0012-64 airfoil, total pressure probes, and an NPL-9510 airfoil. This work is referred to in detail in references 7, 8, and 6, respectively, and is summarized in this section.

The first experiment involved the NACA 0012-64 airfoil, which is shown in cross section in figure 6. The airfoil was instrumented with pressure orifices at 5 percent chordwise intervals. The airfoil was tested in such a manner that total temperature was reduced while both Mach number and Reynolds number were held constant. Consequently, with both Reynolds and Mach numbers constant, any difference in one of the orifice pressure coefficients, C_p, could be attributed to the onset of condensation effects. An example of the total temperatures and pressures at which the onset of effects was actually observed is shown in figure 7. The airfoil was mounted at zero angle of attack, which resulted in a maximum local Mach number of 1.20 for $M_\infty = 0.85$. The total temperatures and pressures corresponding to saturation at the maximum local Mach number, the free-stream Mach number, and the nearly zero Mach number in the settling chamber are all shown in figure 7. As is apparent in the figure, effects occurred at temperatures not only below local saturation over the airfoil but below free-stream saturation. Furthermore, the nearly constant temperature difference between free-stream saturation and the onset of condensation effects was the first indication that the effects detected were resulting from condensation on pre-existing seed particles upstream of the model in the test section. That these effects were indeed the result of heterogeneous nucleation was verified by removing the airfoil from the tunnel and looking at the free-stream flow by means of total pressure probes. Indeed, the total pressure probes detected the onset of effects at the same general temperatures and pressures as did the NACA 0012-64 airfoil.

Also included in figure 7 is the line corresponding to Sivier's predicted onset of homogeneous nucleation for the maximum local Mach number of 1.20 over the airfoil. (See ref. 6 for details of how Sivier's analysis was applied.) As can be seen, that homogeneous nucleation effects were not detected during the experiment is consistent with Sivier's analysis because his calculations show homogeneous nucleation occurring at temperatures below those where heterogeneous nucleation effects began.

In order to get a better chance to detect homogeneous nucleation at temperatures above those where heterogeneous nucleation would be expected in the free stream, it was decided to test an airfoil capable of generating relatively high local Mach numbers. A British NPL-9510 airfoil was chosen for the experiment and was mounted in the two-dimensional turntable of the 0.3-m TCT, as shown schematically in figure 8. The primary data source during this experiment was visual detection of condensate, or fog, as seen by viewing through the "D" shaped windows over the model. In this manner visual differentiation could be made between fog occurring locally in the region just above the airfoil, fog of a light and unsteady nature appearing throughout the test section itself, and fog of a moderate and steady nature occurring in the test section. An example of the results obtained from this test is shown in figure 9, where for a free-stream Mach number of 0.75 and an angle of attack of 6°, the maximum local Mach number reached a value of 1.70. The visual detection of local fog is shown by the circles and compares very closely with the line corresponding to Sivier's prediction for the onset of effects due to homogeneous nucleation in the region of maximum local Mach number. Unfortunately, as seen in figure 8, the field of view over the airfoil did not extend down to the surface of the model. Consequently, local fog may have occurred several degrees higher in temperature before it was observable in the field of view. Nevertheless, the trend with pressure of the observed local detection does agree with Sivier's prediction.

The onset of light, unsteady fog in the test section is represented by the squares in figure 9 and occurs close to free-stream saturation temperature over the total pressure range. The onset of steady, brighter fog, represented by the triangles, occurred a degree or two below free-stream saturation total temperature. The onset of the steady fog appears to correlate quite closely with the onset of pressure effects for both the NACA 0012-64 airfoil and the total pressure probes.

That the appearance of visual fog in the test section correlates very closely with the free-stream saturation temperature strongly suggests that the pre-existing seed particles responsible for the fog are the result of impurities in the tunnel rather than the result of unevaporated liquid nitrogen droplets remaining from the injection process. The main reason for this conclusion is that many of the processes associated with liquid nitrogen droplet atomization, evaporation, and re-entrainment downstream of flow obstacles are pressure dependent. For example, increased pressure would be expected to result in more rapid evaporation of the droplets and much smaller droplets after collisions with turning vanes and other flow obstacles. Consequently, increased tunnel pressure should minimize the temperature, relative to some saturation boundary, at which the unevaporated nitrogen droplets reach the test section. The data in figure 9, however, show absolutely no influence of tunnel pressure with regard to the appearance of fog in the test section compared to free-stream saturation temperature.

CONCLUDING REMARKS

The following remarks appear to be appropriate in light of the minimum operating temperature studies at Langley up to this time.

1. Cryogenic wind tunnels should be operated as cold as possible to maximize Reynolds number capability and to reduce the amount of liquid nitrogen required to absorb the heat of the drive fan.

2. As a result of the simulation studies by Adcock and others, nitrogen appears to be an acceptable test gas at both the high pressures and low temperatures of interest to the NTF and other transonic cryogenic wind tunnels using nitrogen as a test gas. Consequently, the onset of condensation effects appears to be the limiting factor with regard to minimum operating temperatures of the NTF and other cryogenic tunnels.

3. There are two distinct types of nucleation processes - homogeneous and heterogeneous - that can, either separately or in combination, affect the aerodynamic test data.

4. Homogeneous nucleation, by its nature of having to form its own seed particles, or nuclei, will be most likely to occur in regions of high Mach number where the resulting pressures and temperatures will be appreciably below the vapor-pressure curve. Consequently, tests in which there are very high local Mach numbers over a wing will be particularly susceptible to homogeneous nucleation effects. The onset of these effects has been detected at temperatures above those corresponding to free-stream saturation.

5. Heterogeneous nucleation is the process by which condensation takes place on pre-existing seed particles in the flow. The crucial factor in this case is knowing the size and number density of these seed particles. The most likely areas of effects would be in regions where the flow is at temperatures below saturation for a relatively long time - in the test section itself or over relatively large chord models. While heterogeneous nucleation in the 0.3-m TCT has only been detected at temperatures below free-stream saturation at this point in

the experimental program, it may also cause condensation effects locally over models at temperatures above free-stream saturation.

6. More work is required for both types of nucleation before the minimum operating temperatures of cryogenic wind tunnels can be confidently predicted.

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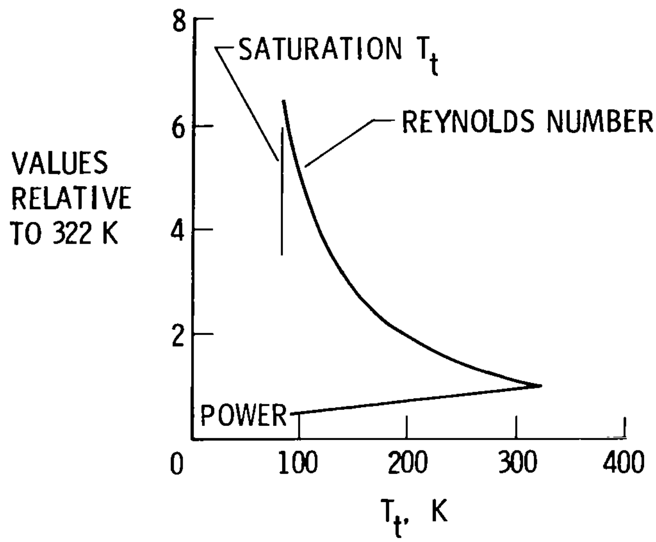

Figure 1.- Effects of temperature reduction. $M_\infty = 1.0$; $p_t = 1.0$ atm; nitrogen gas.

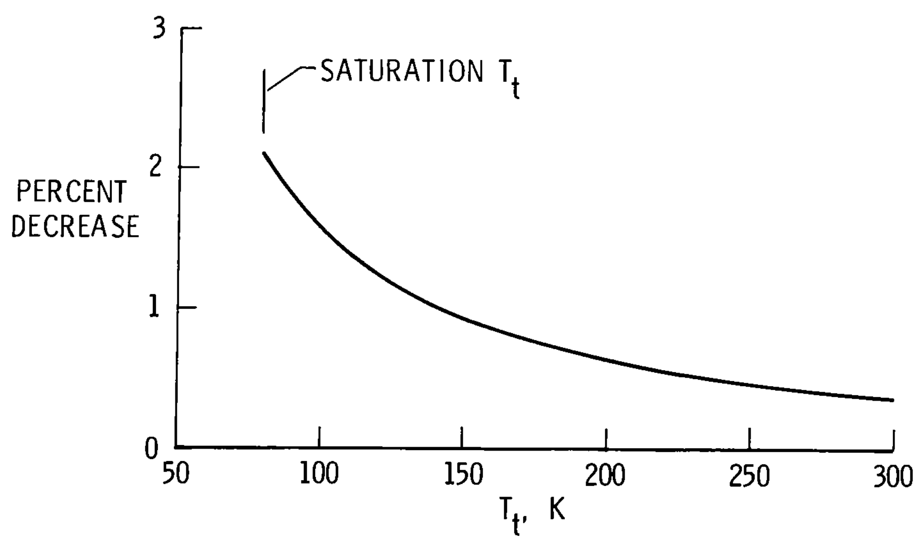

Figure 2.- Change in liquid nitrogen consumption per 1 K decrease in T_t. $M_\infty = 1.0$; $R/m = 1.2 \times 10^8$.

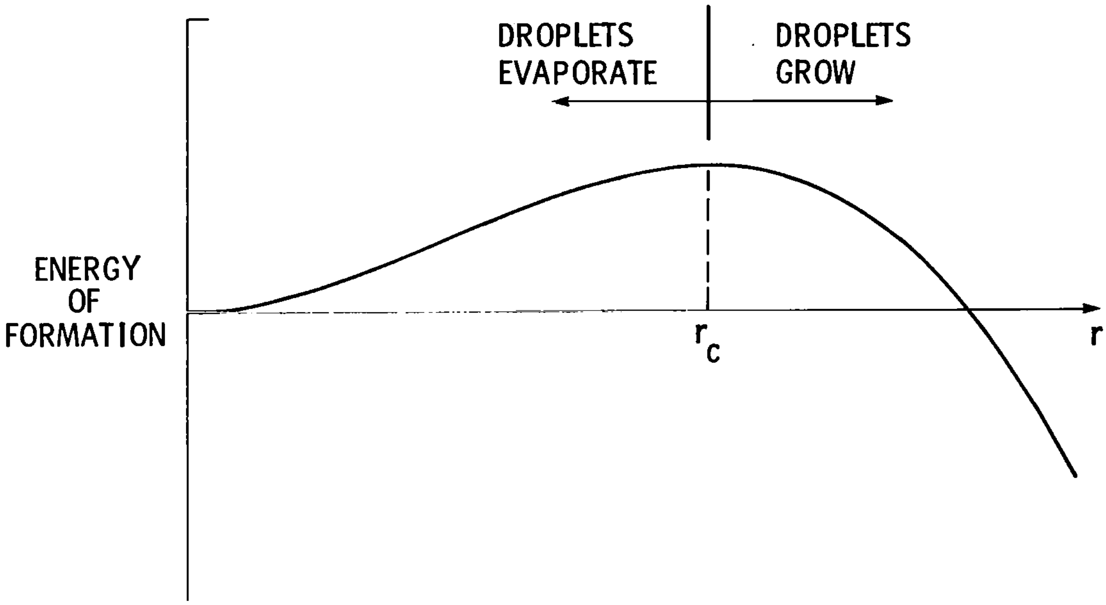

Figure 3.- Energy of formation of a classical liquid droplet.

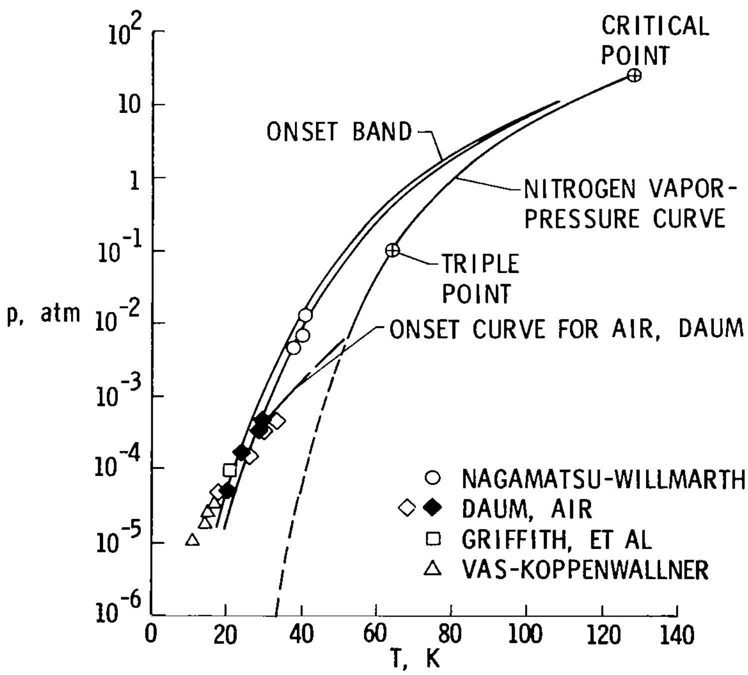

Figure 4.- Onset band of Sivier for homogeneous
nucleation with nitrogen.

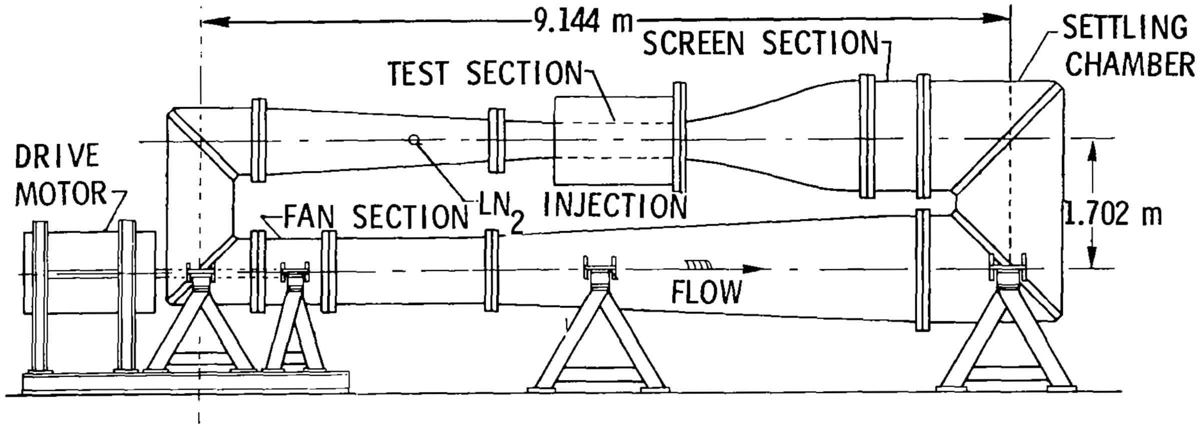

Figure 5.- Schematic of Langley 0.3-meter Transonic Cryogenic Tunnel (TCT).

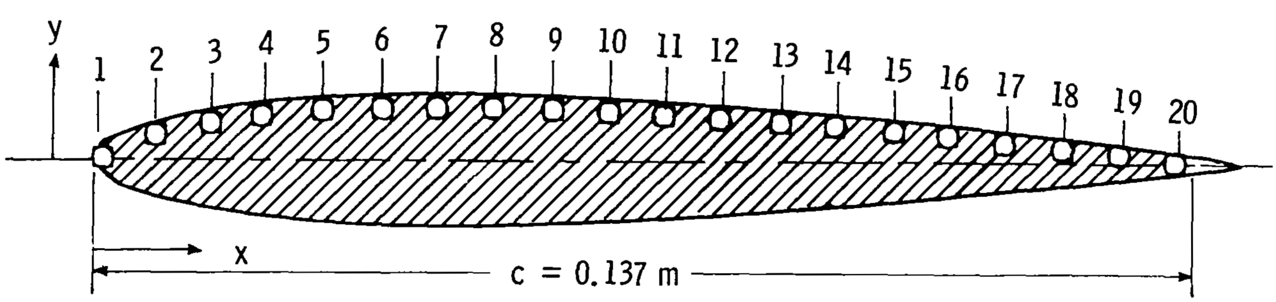

Figure 6.- NACA 0012-64 airfoil.

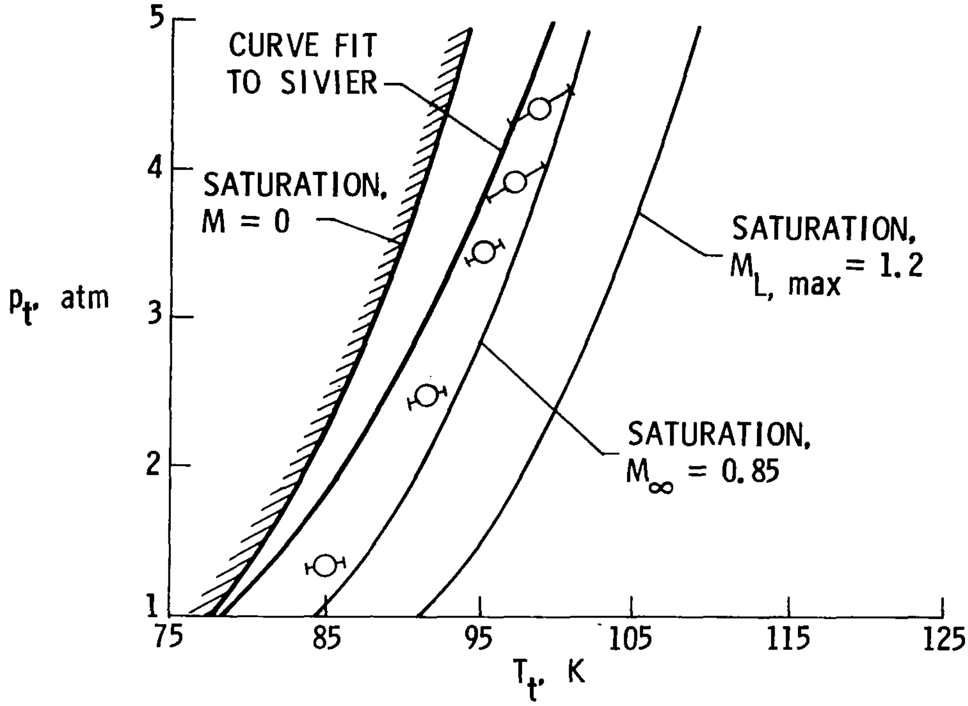

Figure 7.- Onset of effects with the NACA 0012-64 airfoil compared to predicted onset of homogeneous nucleation.

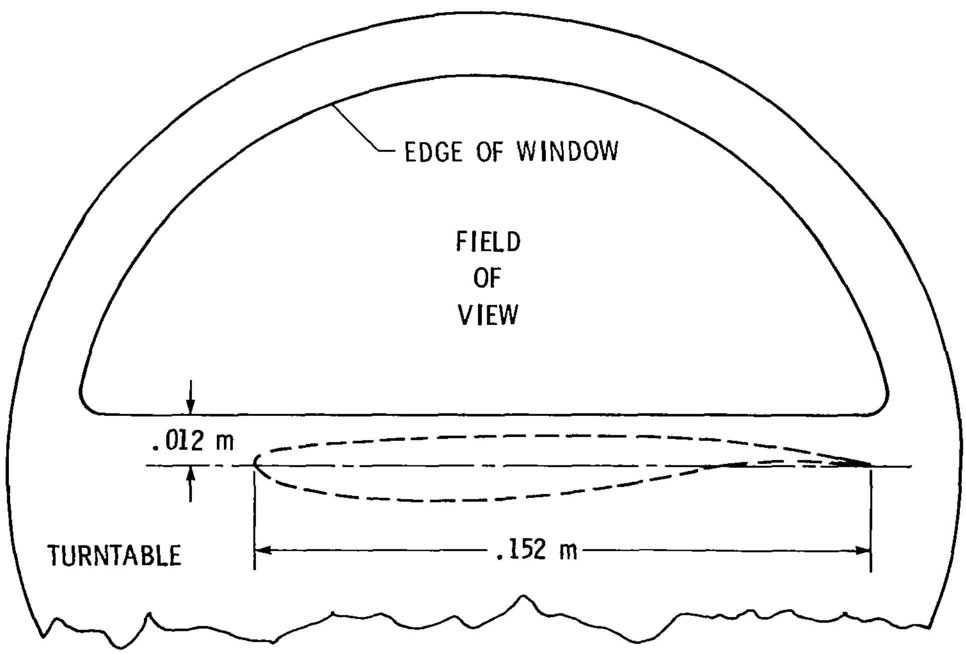

Figure 8.- Field of view over the NPL-9510 airfoil.

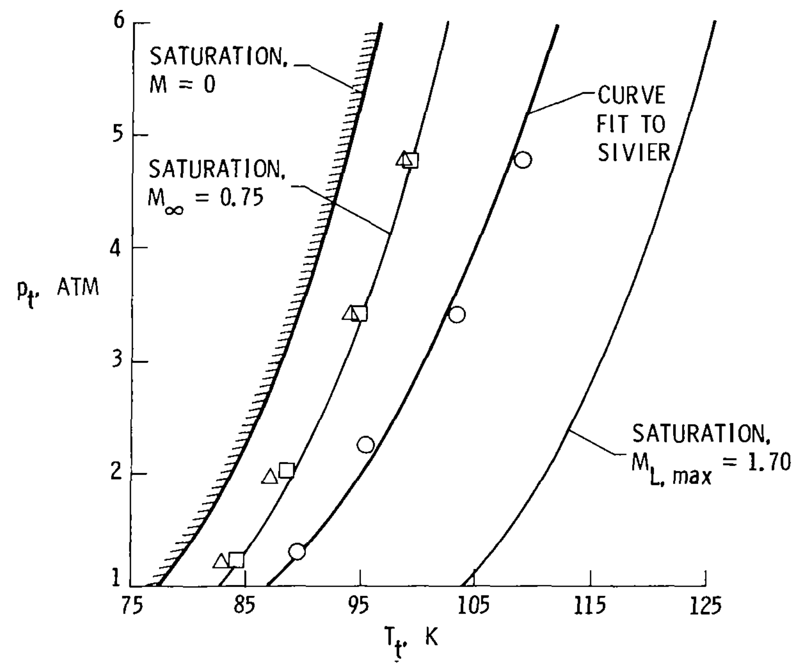

Figure 9.- Onset of condensation effects for NPL-9510 airfoil. $M_{\infty} = 0.75$; $\alpha = 6^{o}$; $M_{L,max} = 1.70$.

FLOW QUALITY MEASUREMENTS IN TRANSONIC WIND TUNNELS AND PLANNED CALIBRATION OF THE NATIONAL TRANSONIC FACILITY

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SUMMARY

The need for mean flow and dynamic flow quality measurements was considered for the National Transonic Facility (NTF). Past experience in making flow quality measurements in transonic flows and at cryogenic temperatures was used to guide the selection of methods to be used in the NTF. It appears that suitable instrumentation will be available and adequate experience has been obtained to insure that the proper calibration of the NTF can be made.

INTRODUCTION

The apparent ever-escalating cost of fuel has resulted in severe design requirements for future aircraft to insure a substantial improvement in fuel economy. This, in turn, has placed stringent requirements on wind tunnel data obtained with models used to predict the performance of full scale aircraft. In addition to more nearly simulating mean flow flight conditions, wind tunnels must have a flow quality in the test section that is good enough to insure that perturbations in the flow will not adversely influence the aerodynamic characteristics of models.

In the past, good flow quality was based almost entirely on mean flow measurements. At the present time, it is believed that the added accuracy required from wind tunnel data will also require the dynamic flow quality to be good (ref. 1). That is, fluctuations of velocity, density, and total temperature (or vorticity, sound, and entropy) must be small enough to insure that dynamic fluctuating flow effects will not adversely influence measured results on models.

As a result of these requirements, the Subsonic-Transonic Aerodynamic Division has embarked on an extensive program to measure and document the dynamic flow quality of its subsonic and transonic wind tunnels (ref. 2). In addition, extensive planning is underway to insure that the mean and dynamic flow quality in the NTF will be adequately measured and documented (refs. 3 and 4). This report will discuss briefly some of the experiments that have been made to measure the dynamic flow quality in several facilities, present some of these results, and describe plans for measurements to be made in the NTF.

SYMBOLS

$A_1 - A_8$	constants in equation (7)
E	mean voltage across wire
e'	instantaneous voltage across wire
f_{max}	maximum frequency
M_∞	free-stream Mach number
p	free-stream static pressure
p_w	static pressure at wall
R_{hot}	hot resistance of wire
R_{cold}	cold resistance of wire
R_∞	free-stream unit Reynolds number
$S_u = \left(\frac{\partial \ln E}{\partial \ln u}\right)_{\rho, T_o, T_w}$	velocity sensitivity of wire
$S_\rho = \left(\frac{\partial \ln E}{\partial \ln \rho}\right)_{u, T_o, T_w}$	density sensitivity of wire
$S_{T_o} = \left(\frac{\partial \ln E}{\partial \ln T_o}\right)_{u, \rho, T_w}$	total temperature sensitivity of wire
T_o	free-stream total temperature
T_w	wire temperature
u	free-stream velocity
ρ	free-stream density

Superscripts

$'$	instantaneous values
$\sim$	rms values

FLOW QUALITY - GENERAL

In general, mean flow quality measurements are made only in new facilities or facilities in which major alterations have been made that can change the mean flow in the test section. These measurements are usually called tunnel calibrations and consist of measuring Mach number and flow angle variation within the test volume of the test section. If larger than acceptable variations in these quantities occur, additional measurements are required throughout the circuit of the tunnel to determine the causes of the variations. These additional measurements might be made in the high speed diffuser, settling chamber, the contraction, plenum, test section, and fan section, and across coolers and turning vanes. In some facilities the mean total temperature distribution in the test volume also requires measuring.

Dynamic flow quality is seldom routinely measured in subsonic and transonic wind tunnels. These measurements are usually only made by researchers investigating phenomena known to be significantly influenced by fluctuations in the flow. The determination of the transition Reynolds numbers of laminar boundary layers is an example of these types of investigations.

Flow fluctuations are usually measured using a hot wire anemometer; however, at compressible subsonic, transonic, and low supersonic Mach numbers, the mean voltage measured across a hot wire is a function of the mean velocity, density, and total temperature (ref. 5). Because of this complexity, dynamic flow quality measurements in these Mach number regimes are almost nonexistent, and the measurements that have been made required several assumptions in order to calculate the desired fluctuating quantities.

MEASUREMENT OF MEAN FLOW QUALITY

Since mean flow quality is usually reported for wind tunnels in the form of tunnel calibrations, a discussion of past measurements will not be made. Instead, this section will describe the mean flow measurements to be made in the National Transonic Facility (NTF).

Figure 1 shows a diagram of the circuit of the NTF and indicates the locations where mean flow measurements are to be made. At the present time plans are being made to make most of the mean flow measurements in the test section and, unless major flow problems are encountered, only a limited number of mean flow measurements will be made at other locations. Therefore, the present discussion will be limited to a description of measurements to be made in the test section.

The location of static pressure orifices in the test section walls is illustrated in figure 2. There are 76 orifices located in both the top and bottom surfaces of the test section and 74 on the "far" wall. Measurements from these static pressure orifices along with the measured total pressure will be used to calculate the local Mach numbers at the wall along the test section. The upper and lower surfaces of the test section are moveable in order to establish a uniform Mach number for the length of the test section. Because of the large Reynolds number range of the NTF, it might be necessary to locate the upper and lower surfaces at different positions as the Reynolds number is changed to insure that the Mach number is uniform over the test section length for all test section flow conditions. The upper and lower surfaces at the entrance to the high speed diffuser are also moveable. These surfaces will be positioned to provide maximum efficiency for the high speed diffuser.

After a uniform Mach number has been established over the test section based on measurements at the wall, a constant diameter tube will be installed in the test section as shown in figure 3. This tube is 7.6 cm in diameter and 10.7 m long, mounted into a fixture on the angle of attack sector. The tube can be placed along the centerline of the test section and on a 0.61-m radius around the centerline. The tube has 300 static pressure orifices along its length. The measurement of the static pressures plus the total pressure will be used to calculate the Mach numbers along the tube at the centerline and at several positions around the 0.61-m radius away from the centerline to insure that the Mach number distribution in the interior of the test section is also constant over its length. During these tests, measurements along the walls will also be made, recorded, and compared with those measured at the interior of the test region.

A rake (fig. 4) will be used to measure the total pressure and total temperature distribution over the test section volume. The rake is 1.52 m long, has 20 total pressure and 10 total temperature probes, and can be rotated about its sting and moved fore and aft in the test section. These movements permit the mapping of the total pressures and total temperatures throughout the volume of the test section.

A yaw probe will be used to measure the flow angles throughout the test volume, and a candidate probe is illustrated in figure 5. The surfaces on which measurements will be made are portions of a square pyramid into which are drilled four static pressure orifices normal to each surface. This probe, or one of similar configuration, will be used to measure the mean flow angles in the vertical and horizontal planes over the test volume.

The mean flow measurements will be made over the Mach number, total pressure, and total temperature operating range of the NTF. Because of the large Reynolds number range of the NTF, it is possible that some adjustment of the test section walls, the reentry flaps, and the diffuser inlet ramps will be required as the operating conditions are changed. From the above discussions, it appears that adequate provisions have been made to measure the mean flow conditions in the NTF to insure that the mean flow quality is adequate to insure data of sufficient accuracy to permit the successful testing of models which can be used to guide the design of efficient highspeed aircraft.

DYNAMIC FLOW QUALITY

In recent years there has been renewed interest in the effect of dynamic flow disturbances (vorticity, sound, and entropy fluctuations) on test results obtained on models in wind tunnels and the effect of these disturbances on the ability to extrapolate wind tunnel data to flight conditions (ref. 1). The Subsonic-Transonic Aerodynamic Division has an on-going program to measure and document the dynamic flow quality in its wind tunnels. In addition, research is underway to insure that adequate instrumentation is available to make the necessary measurements over the Mach number, total pressure, and total temperature ranges of the facilities.

For example, flow quality measurements have been made in the following wind tunnels:

8-Foot Transonic Pressure Tunnel (8' TPT)

Low Turbulence Pressure Tunnel (LTPT)

4- by 7-Meter Tunnel

0.3-m Transonic Cryogenic Tunnel

Measurements are planned for the Langley High Speed 7- by 10-Foot Tunnel and the NTF.

LOCATION OF DYNAMIC FLOW QUALITY MEASUREMENTS IN THE NTF

The initial dynamic flow quality measurements and most of the later measurements in the NTF will be made in the test section since the major interest is in the flow quality there. However, if problems are encountered with higher than acceptable fluctuations in the test section, measurements will be made at other locations to determine the origin of the disturbances. A prime source for high level disturbances in the test section in the form of sound is the high speed diffuser. Therefore, provisions are being made to measure fluctuations in the diffuser.

Other locations might be responsible for generating disturbances that can enter the test section, and provisions are being made to measure dynamic data in the following locations (see fig. 1):

1. Settling chamber

2. Upstream and downstream of screens in settling chamber

3. Upstream and downstream of cooler

4. Rapid diffuser at entrance to settling chamber

5. Across turning vanes

6. Fan section

INSTRUMENTATION

Several different types of instrumentation have been used to make dynamic flow quality measurements in transonic wind tunnels. These instruments include hot wire anemometers, fluctuating static pressure probes, fluctuating static pressure transducers mounted flush with the walls in the test section, and accelerometers.

In recent years there has been renewed interest in applying the hot wire anemometer technique to transonic flow (ref. 6). However, at transonic speeds the hot wire is sensitive to three coexisting fluctuations: velocity, density, and total temperature (or vorticity, sound, and entropy). Because of this complexity and the methods used to separate the three coexisting fluctuations, there exists some controversy concerning the accuracy or generality of recent results. Because of this, an attempt has been made to develop a hot wire anemometer technique which will be adequate for compressible subsonic, transonic, and low supersonic Mach number regimes. A photograph of the probe under development is shown in figure 6. The probe has three wires mounted normal to the flow. By using a proper calibration and data reduction technique, it is believed that the results from this probe can separate the velocity, density, and total temperature fluctuations in the three flow regimes noted above. A brief discussion of the concepts of this probe and the calibration and data reduction technique is given in the appendix.

Static pressure probes have been used in transonic flows to measure fluctuating static pressures (ref. 2) in test sections. The results from these probes compare favorably with results obtained with hot wire anemometers. Therefore, probes similar to these will continue to be used for flow quality measurements. Transducers are available which can operate at cryogenic temperatures and probes have been designed and built for these transducers.

It is relatively easy to measure the fluctuating static pressures at the surface of walls in the test section of transonic wind tunnels, and often these measurements are adequate for some purposes. These types of measurements have been made in several facilities (ref. 2) and it is planned to continue these measurements if circumstances dictate.

THE MEASUREMENT OF DYNAMIC FLOW QUALITY

Extensive flow quality measurements were made in the 8' TPT and these results were reported in references 2, 7, and 8. These measurements were made to guide the alteration of the tunnel for future Laminar Flow Control Experiments (ref. 9). Additional measurements will be made after the alterations have been completed to determine the improvement in the flow quality resulting from the alterations.

An example of flow quality measurements made in the 8' TPT using a hot wire anemometer is presented in figure 7. These results indicate that the normalized velocity fluctuations vary significantly with Mach numbers but modestly with unit Reynolds number. The velocity fluctuations are about 0.10 percent at $M_\infty = 0.2$ and increase to 0.30 to 0.40 percent at $M_\infty = 0.8$.

Figure 8 shows some data obtained with a fluctuating static pressure probe in the 8' TPT. There is a very large variation in the normalized fluctuating static pressure with Mach number but only a modest variation with unit Reynolds number. At $M_\infty = 0.2$ the fluctuating static pressures are only about 0.01 percent but increase to 0.60 percent at $M_\infty = 0.9$.

Some very preliminary flow quality measurements were made in the 0.3-m Transonic Cryogenic Tunnel. These measurements were made primarily to develop instrumentation that must not only operate properly in transonic flows, but must also operate properly at cryogenic temperatures. Data obtained with the three-wire hot wire probe described above is presented in figure 9. These results show that the normalized velocity and density fluctuations are nearly equal and vary from about 0.4 to 2.0 percent over the velocity range from 70 to 200 m/sec. The total temperature fluctuations varied from about 0.1 to 0.25 percent over the same velocity range.

An example of the fluctuating static pressures measured at the wall of the test section using a flush-mounted transducer is shown in figure 10. These results also show a significant increase in the fluctuating static pressures with increasing Mach number but a modest variation with unit Reynolds number. The variation with total temperature was small (and not shown). Any apparent variation of the fluctuating pressure with total temperature can be accounted for by using the unit Reynolds number.

Considerable problems were encountered in making hot wire measurements in the 0.3-m Transonic Cryogenic Tunnel. These problems included:

1. Wire resistance

2. Tunnel control

3. Liquid nitrogen

4. Wire breakage

A hot wire probe that has sufficient wire resistance at room temperature can have a very low resistance at low temperatures. This, in turn, can result in excessive current flow from the anemometer. The wire resistance can be increased either by increasing the length of the wire or by decreasing its diameter. However, both of these actions will result in increased wire stress and increased wire breakage. Therefore, some compromise must be made between wire resistance and wire strength.

The calibration of a hot wire probe at transonic Mach numbers requires a very good control of velocity, density, and total temperature in the wind tunnel. During recent tests, the controls of the 0.3-m Transonic Cryogenic Tunnel were only fair. After the tests, it was found that one of the digital valves controlling the mass flow of liquid nitrogen to the tunnel was faulty.

This valve has been repaired and now the controls of the tunnel appear to hold velocity, density, and total temperature constant to good accuracy.

The data from the hot wire probe indicated the possibility of liquid nitrogen in the test section at low liquid nitrogen flow rates. The tunnel has only four nozzles to inject liquid nitrogen into the circuit and these nozzles must be sized to pass the high flow rates required at high Mach numbers and high total pressures. As a consequence of this, at low pressures and low Mach numbers, the flow rate of liquid nitrogen required is very low, and it is suspected that under these conditions the four nozzles cannot adequately atomize the liquid nitrogen. This inadequate atomization makes it possible for drops of liquid nitrogen to reach the test section. The controls have now been changed to permit the cutting off of three nozzles at low flow rates to improve the atomization of liquid nitrogen under these conditions.

For the initial test of the hot wire probe, wire breakage was excessive. This was due to several factors. Under steady state conditions, the liquid nitrogen problem noted above was probably the major cause of wire breakage. If the tunnel conditions were changed too rapidly, liquid nitrogen would not evaporate before it entered the test section. This also resulted in excessive wire breakage. Because of this, tunnel conditions were changed slowly and wire breakage was reduced significantly.

CONCLUSIONS

From past experience gained in making mean flow and dynamic flow quality measurements in transonic and cryogenic wind tunnels, it appears that the following conclusions can be made:

1. Mean flow quality measurements in transonic flow and cryogenic wind tunnels are rather routine and it appears that adequate provisions have been made to insure that suitable mean flow quality measurements can be made in the NTF.

2. Even though dynamic flow quality measurements are not routine, past experiences indicate that adequate dynamic flow quality measurements can be made at transonic speeds at cryogenic temperatures, and adequate provisions have been made to make suitable dynamic flow quality measurements in the NTF.

APPENDIX

A PROBE AND DATA REDUCTION TECHNIQUE FOR OBTAINING HOT WIRE DATA AT TRANSONIC SPEEDS

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In general, the mean voltage measured across a hot wire can be expressed as follows (ref. 5):

$$E = f(u, \quad \rho, \quad T_o, \quad T_w, \ . \ . \ .) \tag{1}$$

where u, ρ, T_o, and T_w are the mean velocity, density, total temperature, and wire temperature (the quantities of interest at the present time). The total change in E due to a change in u, ρ, T_o, and T_w is:

$$dE = \left(\frac{\partial E}{\partial u}\right)_{\rho,T_o,T_w} du + \left(\frac{\partial E}{\partial \rho}\right)_{u,T_o,T_w} d\rho + \left(\frac{\partial E}{\partial T_o}\right)_{u,\rho,T_w} dT_o + \left(\frac{\partial E}{\partial T_w}\right)_{u,\rho,T_o} dT_w \tag{2}$$

If a constant temperature hot wire anemometer is used, $dT_w = 0$ within the limits of the feedback amplifier. Therefore, up to some frequency, f_{max}, $dT_w = 0$ and equation (2) can be written as:

$$\frac{e'}{E} = \left(\frac{\partial \ln E}{\partial \ln u}\right)_{\rho,T_o,T_w} \left(\frac{u'}{u}\right) + \left(\frac{\partial \ln E}{\partial \ln \rho}\right)_{u,T_o,T_w} \left(\frac{\rho'}{\rho}\right) + \left(\frac{\partial \ln E}{\partial \ln T_o}\right)_{u,\rho,T_w} \left(\frac{T_o'}{T_o}\right) \tag{3}$$

or

$$\frac{e'}{E} = (S_u) \quad \left(\frac{u'}{u}\right) + (S_\rho) \quad \left(\frac{\rho'}{\rho}\right) + \left(S_{T_o}\right) \quad \left(\frac{T_o'}{T_o}\right) \tag{4}$$

Attempts have been made to solve the above equation in terms of its mean square value. However, when the equation is squared, six unknowns result. Operating a single wire at six "overheats" gives, in principle, a method for solving the system of six equations with six unknowns. However, when this was tried, difficulties were encountered in inverting the 6 × 6 matrix (ref. 10).

The present proposal suggests that a solution for u', ρ', and T_o' can be obtained using equation (4) without squaring. This concept, however, results in one equation with three unknowns. One way to obtain a solution would be to make a probe with three wires placed normal to the flow. Then equation (4) can be written for each of the wires as follows:

$$\left(\frac{e'}{E}\right)_1 = (S_u)_1 \left(\frac{u'}{u}\right) + (S_\rho)_1 \left(\frac{\rho'}{\rho}\right) + \left(S_{T_o}\right)_1 \left(\frac{T_o'}{T_o}\right)$$

$$\left(\frac{e'}{E}\right)_2 = (S_u)_2 \left(\frac{u'}{u}\right) + (S_\rho)_2 \left(\frac{\rho'}{\rho}\right) + \left(S_{T_o}\right)_2 \left(\frac{T_o'}{T_o}\right) \tag{5}$$

$$\left(\frac{e'}{E}\right)_3 = (S_u)_3 \left(\frac{u'}{u}\right) + (S_\rho)_3 \left(\frac{\rho'}{\rho}\right) + \left(S_{T_o}\right)_3 \left(\frac{T_o'}{T_o}\right)$$

Equation (5) is the three equations that must be solved for the three unknowns after the sensitivity coefficients S_u, S_ρ, and S_{T_o} have been determined from mean flow calibrations and e' and E have been measured.

A problem was encountered in the calibration of the wires to obtain values for S_u, S_ρ, and S_{T_o}. The above equations state that S_u must be evaluated by varying u while keeping ρ, T_o, and T_w constant. The same relative situation exists for the other quantities, S_ρ and S_{T_o}. This requires a facility which can independently vary u, ρ, and T_o and hold them constant for the time required to make the necessary measurements. Once S_u is obtained for one value of ρ and T_o, the quantity ρ must be changed and the process repeated. This continues until a sufficient amount of data has been obtained to define S_u as a function of u and ρ. Then T_o must be changed and the process repeated; this becomes a very lengthy process.

Now to insure that the 3 × 3 matrix is well-behaved, the sensitivity coefficients $(S_u)_1$, $(S_u)_2$, $(S_u)_3$, etc., must be sufficiently different to provide a stable solution. This can be assured by operating the three wires at different "overheats" or different wire temperatures. This was done as follows. The tunnel was operated at a given total temperature and at the maximum density and velocity for which the calibration was to be conducted. The cold resistance of each wire was measured and multiplied by an "overheat" value. For example:

$$\left(R_{hot}\right)_1 = \left(R_{cold}\right)_1 \times 1.6$$

$$\left(R_{hot}\right)_2 = \left(R_{cold}\right)_2 \times 1.8 \tag{6}$$

$$\left(R_{hot}\right)_3 = \left(R_{cold}\right)_3 \times 2.0$$

These hot resistances were held constant for the remainder of the calibration as required by the math used to obtain expressions for the sensitivity coefficients. [Herein might be a problem that requires additional investigation. For example, are the "overheats" sufficiently different to produce the required differences in the values of $(S_u)_1$, $(S_u)_2$, $(S_u)_3$, etc.]

After the sensitivities have been evaluated and the instantaneous fluctuating voltages recorded on tape, these data are digitized at a sufficient rate to produce valid data up to the maximum frequency required. Equation (5) is then solved at every instant to produce an array of data for u', ρ', and T_o' as a function of time. In addition, the cross product terms can also be calculated at the same time — namely, $u'\rho'$, $u'T_o'$, $\rho'T_o'$, etc. A tape of these quantities u', ρ', T_o', $\rho'T_o'$, etc., can be used to analyze these data using standard random data analysis techniques to obtain mean values, RMS values, autocorrelations, spectra, etc.

In the past, difficulties were encountered in obtaining S_u, S_ρ, and S_{T_o} for the three wires using the mean flow measurements. At the present time a multiple regression technique is being used to obtain the sensitivities. The equation that appears to be adequate is:

$$\log E = A_1 + A_2 \log u + A_3 \log \rho + A_4 \log T_o + A_5 \log u \log \rho$$
$$+ A_6 \log u \log T_o + A_7 \log \rho \log T_o + A_8 \log u \log \rho \log T_o \qquad (7)$$

[Note that this technique does not require the constant condition constraints on the wind tunnel noted above, but the constant conditions are still desired to help formulate equation (7).]

Equation (7) implies that the following are true:

$$S_u = \left[\frac{\partial \log E}{\partial \log u}\right]_{\rho,T_o,T_w} = A_2 + A_5 \log \rho + A_6 \log T_o + A_8 \log \rho \log T_o$$

$$S_\rho = \left[\frac{\partial \log E}{\partial \log \rho}\right]_{u,T_o,T_w} = A_3 + A_5 \log u + A_7 \log T_o + A_8 \log u \log T_o \qquad (8)$$

$$S_{T_o} = \left[\frac{\partial \log E}{\partial \log T_o}\right]_{u,\rho,T_w} = A_4 + A_6 \log u + A_7 \log \rho + A_8 \log u \log \rho$$

The calibration data seem to agree with these equations to the extent that S_u was a constant for given values of ρ and T_o for the wire Reynolds numbers of the present test. This was also true for S_ρ and S_T. Equation (7) might not be general enough for very low wire Reynolds numbers.[o] For these conditions a second degree equation in u, ρ, and T_o might be required.

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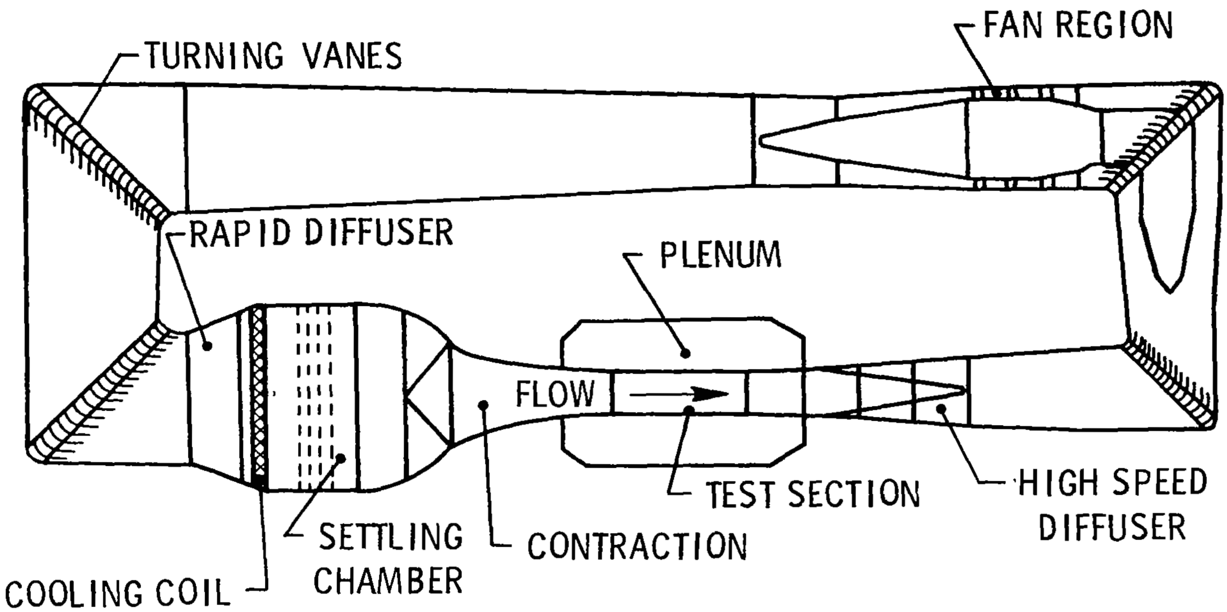

Figure 1.- NTF circuit and locations where flow quality measurements can be made.

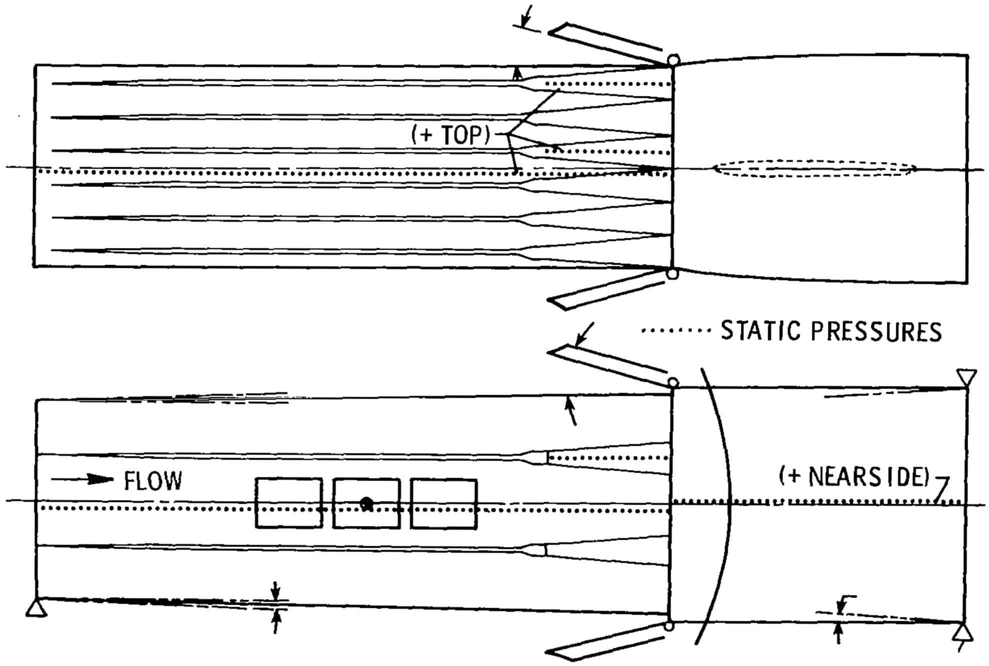

Figure 2.- Location of static pressure orifices in test section wall of NTF.

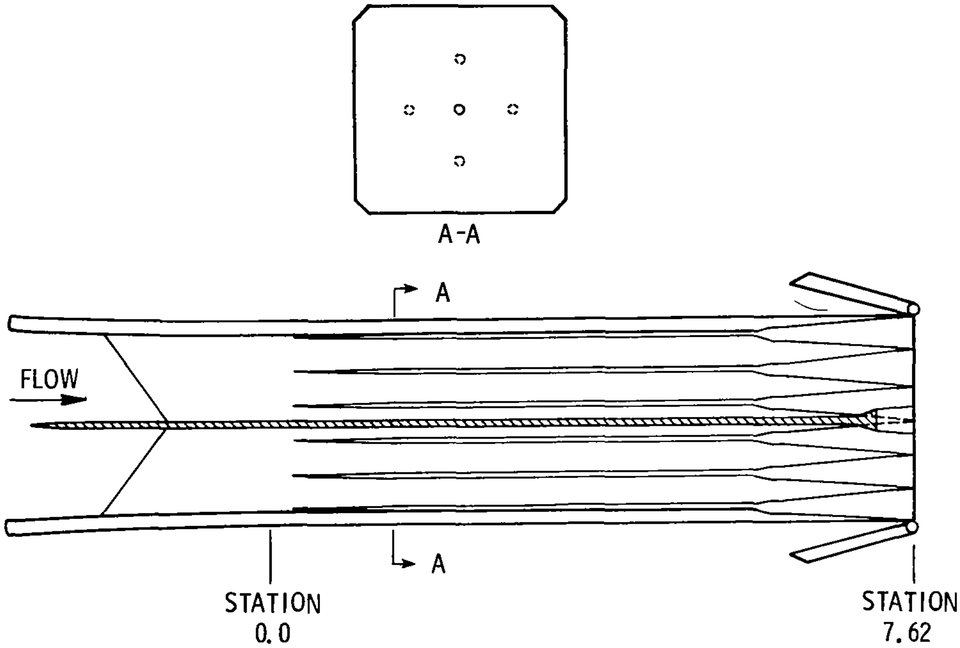

Figure 3.- Test section static pressure calibration tube.

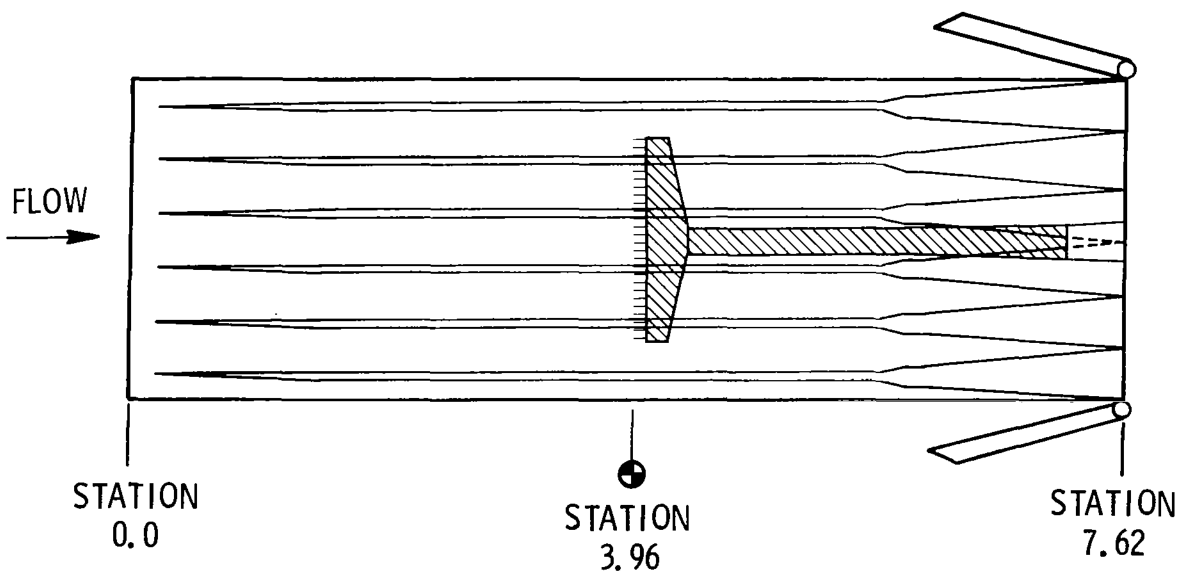

Figure 4.- Total pressure and total temperature rake for test section.

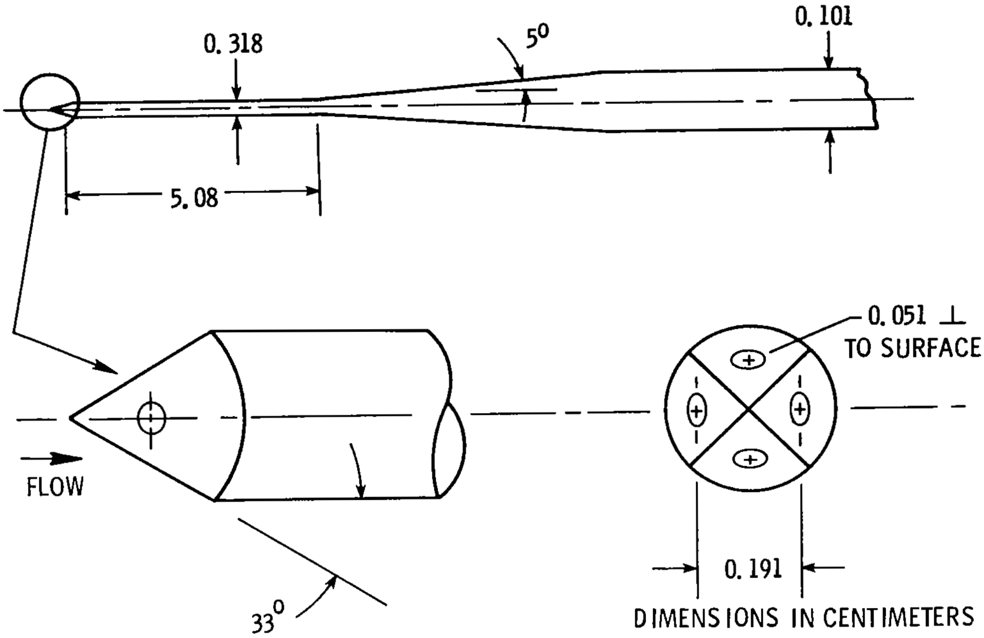

Figure 5.- Candidate yawmeter for test section.

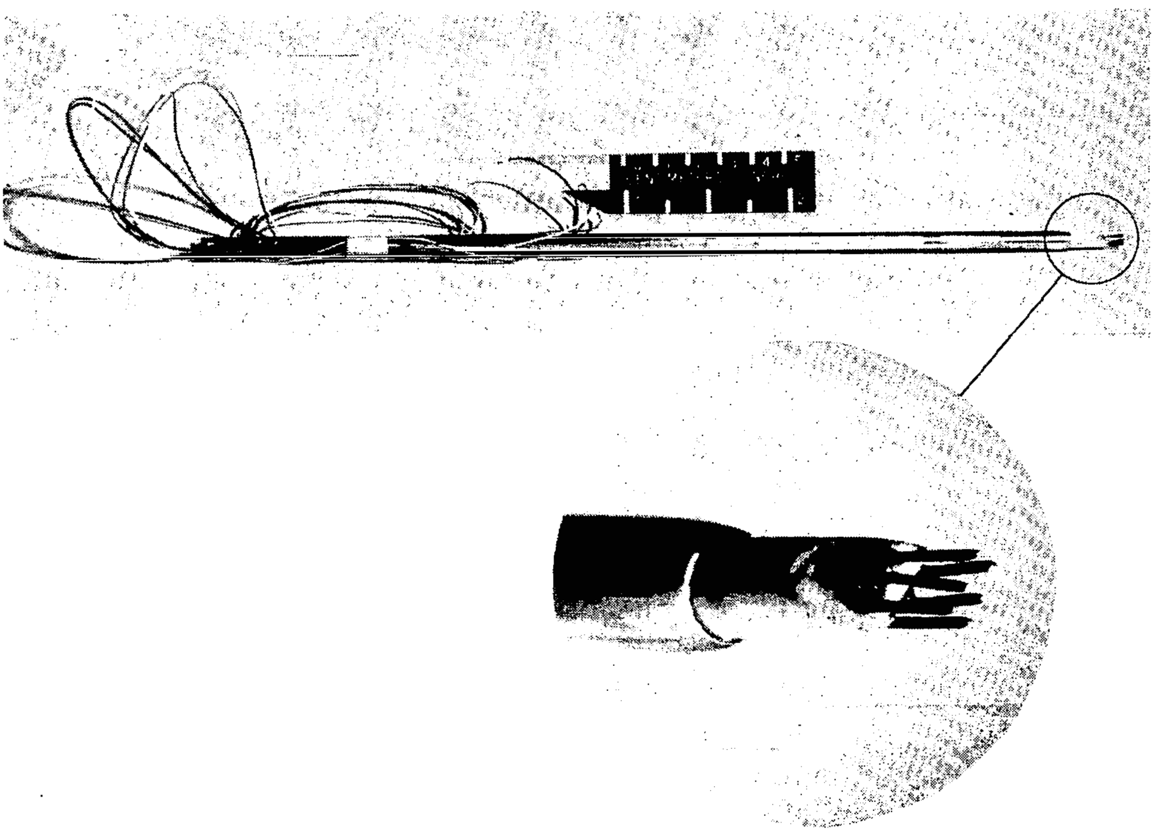

Figure 6.- Three-wire hot wire probe.

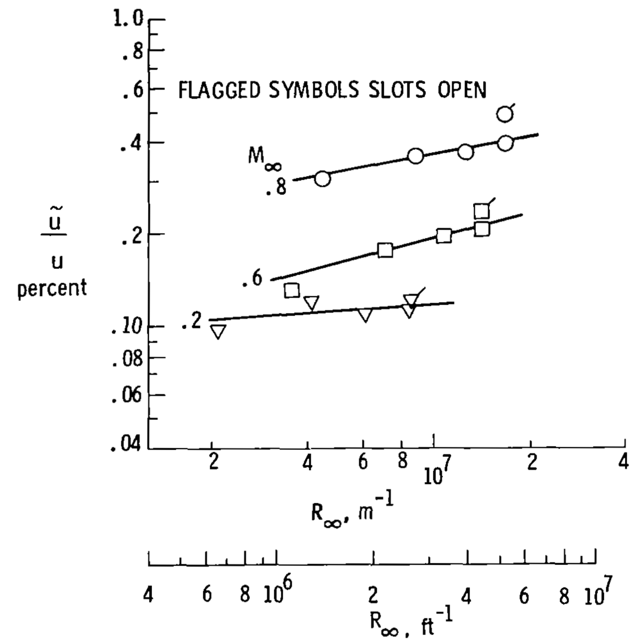

Figure 7.- Velocity fluctuations in test section of 8' TPT (ref. 2).

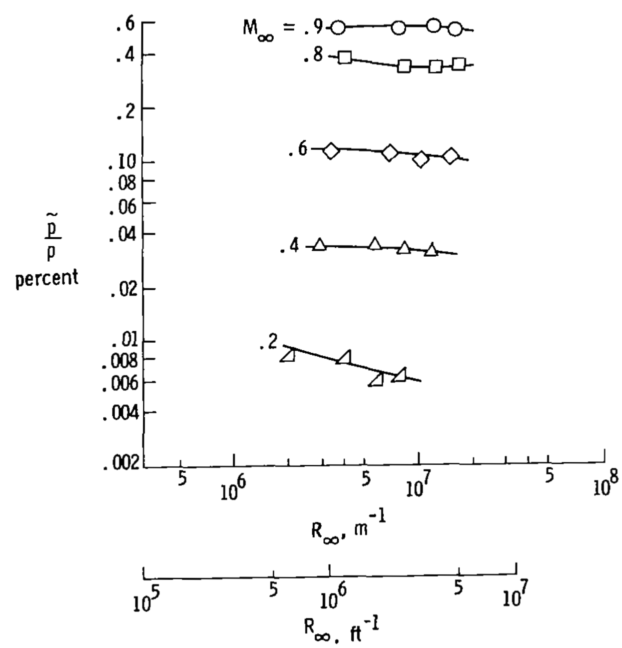

Figure 8.- Pressure fluctuations in test section of 8' TPT (ref. 2).

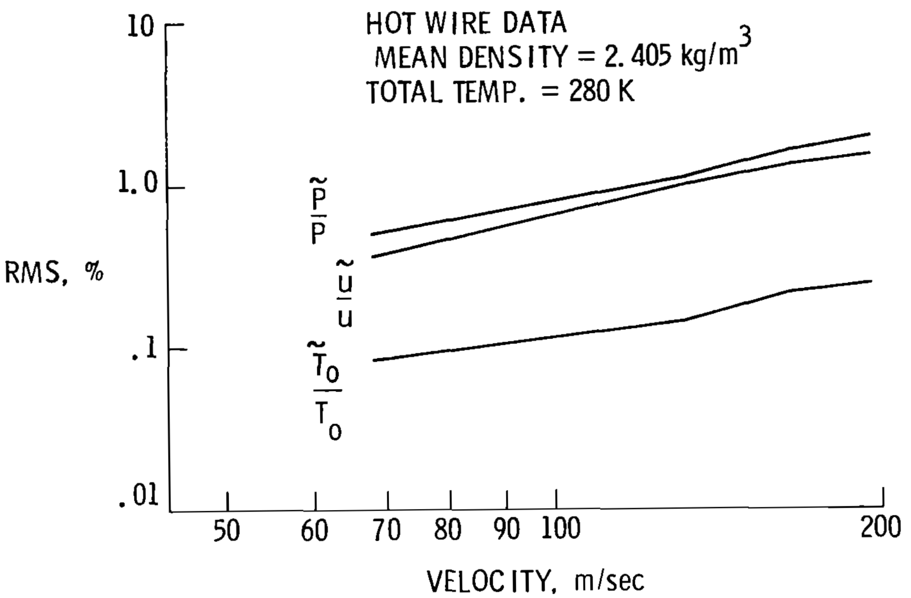

Figure 9.- RMS of fluctuating quantities measured with a hot wire in test section of 0.3-m Cryogenic Wind Tunnel. $\dot{m}$ = 2.4 kg/m^3; T_o = 280 K.

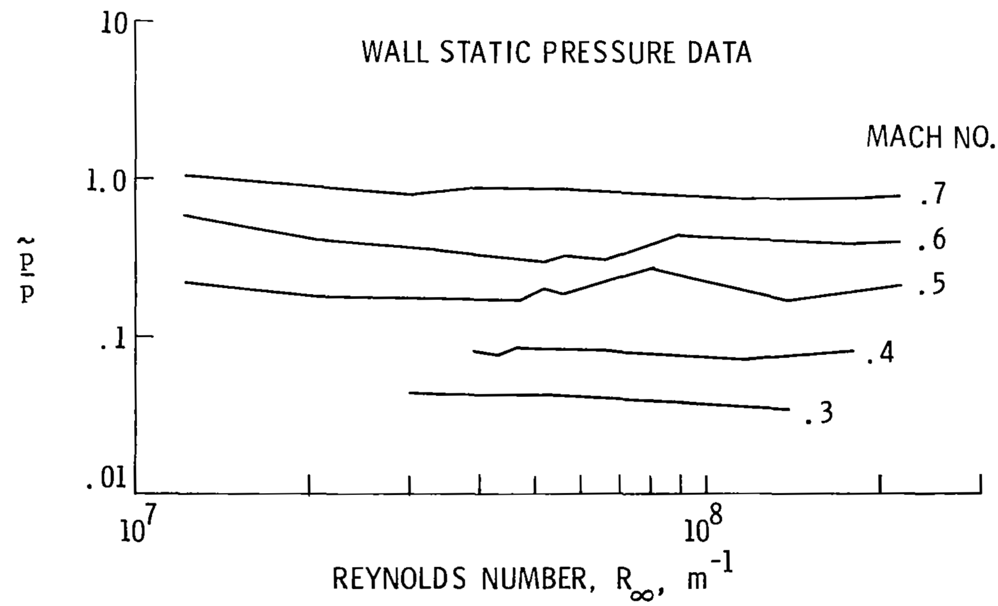

Figure 10.- Fluctuating static pressure at the test section wall of 0.3-m Cryogenic Wind Tunnel. 100 $\leq T_o$; K $\leq$ 350.

WALL-INTERFERENCE EFFECTS: STATUS REVIEW AND PLANNED EXPERIMENTS IN NTF

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INTRODUCTION

The possibility of simultaneously controlling and matching both Mach number and Reynolds number over a wide range for wind tunnel tests in the National Transonic Facility (NTF) requires a closer look at other free-flight conditions for which lack of simulation in the tunnel is a potential source of error. Even though this new independent control over both Mach and Reynolds numbers should allow for better experimental assessment of some remaining error sources for the NTF more precisely than has been possible for other test facilities, the need for more accurate assessment or elimination of these error sources assumes greater importance.

An expanded NASA Langley research effort related to transonic wind-tunnel-wall interference was briefly reviewed (ref. 1) at the last NTF High Reynolds Number Workshop in 1976. The present paper will review the status of wall-interference technology in terms of that incorporated into the NTF design (hardware) and the emerging transonic wall-interference assessment correction procedures (software) to be employed when the NTF becomes operational. It is envisioned that all of the early experiments in NTF will provide data relevant to wall-interference effects. Use of such data is discussed in the last section. An attempt is made herein to convey these wall-interference ideas without getting into mathematical detail. In so doing, we have likewise quoted only a few sample references; it was not our intention to review the extensive wind-tunnel-wall-interference literature.

SYMBOLS

c	airfoil chord length
C_ℓ	airfoil or section lift coefficient
C_p	pressure coefficient
M	Mach number
u,v	nondimensional perturbation velocity components
x,y	Cartesian coordinates; x is streamwise
α	angle of attack

θ flow direction angle

Δq correction to quantity q

$[\![q]\!]$ jump discontinuity in quantity q

WIND-TUNNEL SIMULATION OF TRANSONIC FLIGHT CONDITIONS

In wind-tunnel simulation of free flight, the relative importance of various conditions required to be matched depends upon the flight regime and specific investigation. Throughout the transonic flight range, however, many of the conditions which one would like to simulate are not realized independently of one another due to the basic nonlinearity of the flow and interference due to the boundedness. For a wind-tunnel simulation, one must either correlate data and then extrapolate to free-flight conditions or match the conditions. Traditionally, one used theory and computers for the correlation/extrapolation approach; whereas, a hardware device has been used to effect a match or true simulation of flight conditions.

A list of important conditions which one would like to see simulated would include:

Mach number
Reynolds number
Unboundedness
Flow quality
Model fidelity

These roughly trace the continuing development of wind-tunnel simulation of free flight. A number of these aspects and their relation to NTF have already been discussed in previous papers of this present volume. Simulation of the unboundedness condition of free flight is hampered not only by the wind-tunnel walls but also by model supports and measurement probes. We simply note here that for transonic flow, these conditions and the devices used to attempt a match interact critically. For example, if one increases the pressure or decreases the temperature in order to increase the Reynolds number, one generally pays for it by adversely affecting the flow quality and/or model fidelity. Likewise, the tunnel-wall treatment to simulate unboundedness also influences how or how well the Mach number may be simulated. If one is to use a device to perform a match of conditions over a wide range, then that device will, in all probability, be active and thus need to be controlled. One must therefore interact theory and computers with hardware devices in order to achieve a match, particularly through the transonic flow regime.

The discussion here concerns wind-tunnel-wall interference; it is the primary consequence of not simulating the unboundedness condition of free flight. In the past, neither experimental nor analytical solutions have been satisfactory at transonic flow speeds where ventilated test sections are used. Several recent concepts have been proposed and technology is now being developed to implement them. These are discussed here in relation to the NTF.

WALL-INTERFERENCE TECHNOLOGY IN NTF

In many programs involving extensive hardware and new technology, certain basic design decisions must be frozen early in the development. Some time later, when the device becomes operational, parts of it may not contain the latest technology in some areas. Some may view the NTF test section this way, so we digress for the moment to look at wall-interference technology in the NTF historical context.

Historical Context

In the 1975-76 time frame certain basic design decisions had to be made for the NTF. We have tried to graphically illustrate, in figure 1(a), some milestones in wall-interference and transonic testing technologies which led to the basic decisions which were made. In figure 1, time proceeds from left to right. Classical wall-interference theory began in 1919 with Prandtl (ref. 2), and has since grown through many geometric extensions and refinements. It is based upon linear theory and predicted wall interference satisfactorily in the open-jet and solid-wall wind tunnels used for subsonic testing. Introduction of the ventilated wall test section around 1948 (ref. 3) allowed for successful tunnel testing through the transonic range. Its impact upon wall-interference theory, however, was that the homogeneous wall boundary condition became suspect and even the derivation of an effective one involved consideration of the nonlinear viscous effects on the wall. In addition, the inviscid transonic flow is also nonlinear so there was no a priori reason to believe that the classical linear superposition techniques should even be applicable. Attempts to obtain (refs. 4-6) and successfully use homogeneous boundary conditions appropriate to ventilated transonic test-section walls still continue to be made (see, for example, refs. 7 and 8). The advent of practical transonic computational fluid dynamic calculations around 1970 (ref. 9) has allowed one to perform numerical experiments related to tunnel-wall effects. The general conclusions which have emerged are that: (1) somewhere in the transonic regime linear superposition does break down (ref. 10) and (2) wall characteristics are very nonlinear at transonic conditions and dependent upon the model pressure field (ref. 11) even for solid walls (ref. 12).

In 1972, the concept of obtaining high Reynolds number by operating a wind tunnel at cryogenic conditions was shown to be feasible (ref. 13). It is interesting to note that initial studies at Langley of the cryogenic concept were for a small tunnel in support of work on the magnetic suspension/balance concept; that is, a device related to simulating one aspect of the unboundedness condition of free flight. The concept of simulating another aspect of unboundedness by adapting the wind-tunnel walls as the test proceeds was independently put forward (refs. 14 and 15) around 1973. By 1974, the transonic cryogenic tunnel concept had been demonstrated to be feasible in a pilot facility (refs. 16 and 17). The transonic adaptive wall technology for 3-D applications has been actively pursued by several groups but had not been demonstrated, even for 2-D, as a practical means of achieving the stated NTF objectives (ref. 18) by 1975-76, the basic decision time. These basic decisions then, as given in reference 18, were to use the newly demonstrated cryogenic concept in order to obtain the high Reynolds number condition and to use a slotted test section in order to operate over the required Mach number range. Both of these concepts were compatible with the additional requirements concerning continuous operation and high productivity (ref. 18). This was the situation at the time of the first NTF High Reynolds Number Workshop in 1976.

The post-1976 NTF wall-interference technology picture shown in figure 1(b) resulted from the basic design decisions of 1975-76. The structural layout was required in 1977 and it was not inconsistent with a future installation of an adaptive wall test section. As indicated in an earlier paper in this volume (ref. 19) the test section is flanged into the rest of the tunnel circuit allowing for later removal and replacement. However, it should be noted that this test section is massive; and, in terms of time and cost, such an undertaking would be a major change. Slot shape details were required this year and the improvements introduced there were based on accumulated experiences (art) and a computer design method (theory) for low supersonic flow based on the method of characteristics (ref. 20).

The third element required is the timely development of an adequate transonic Wall-Interference Assessment Correction Procedure (WIAC). As indicated in figure 1(b), this endeavor was begun in 1974 and is based upon ideas drawn from classical wall-interference theory, transonic computational fluid dynamics, and the adaptive wall concept. The basic idea of WIAC was summarized in reference 1 and will be reviewed in a subsequent section.

Test Section

The NTF slotted-wall test section design is an update (hopefully improved) of that in the NASA Langley 8-Foot Transonic Pressure Tunnel (TPT). Figure 2 is a sketch of the NTF test section which indicates the areas where improvements in slotted-wall technology have been incorporated. As already mentioned, the slot shape has been designed numerically for better flow at low supersonic flow conditions (ref. 20). Slot shapes for the 8-Foot TPT were determined experimentally by testing a number of wooden inserts before arriving at the final design which was then fabricated and permanently installed. The present design incorporated the insert concept; hopefully, the numerical design will circumvent a long empirical determination of the slot shapes.

Experience with current NASA slotted-wall transonic tunnels indicates that, for many test conditions, the slotted-wall test sections should be longer in order to lessen the wall interference. Thus, the NTF test-section slots are relatively longer than those of the 8-Foot TPT. Experience has also shown that a variable test section and diffuser divergence angle would provide some possibility for improved control of test conditions. Since this may be of greater concern with independent control over wide ranges of both Reynolds and Mach numbers, this added variability has been included in the NTF design. Finally, very early three-dimensional transonic calculations for a wing in a simulated tunnel (ref. 21) indicated a need for some sidewall relief. The two slots incorporated on each sidewall provide this relief and are compatible with the mechanical and optical requirements on the NTF test section.

WALL-INTERFERENCE ASSESSMENT CORRECTION PROCEDURE

As indicated earlier, the WIAC concept for transonic flow conditions was initiated around 1974 in order to support the NTF technology. A number of other groups are also actively working on the concept since it should improve the productivity and extend the usefulness of existing facilities which will not be retrofitted with an adaptive or partially adaptive wall. Even for those facilities which will have an adaptive wall capability, one may require an assessment/ correction procedure in order to achieve an acceptable level of productivity. This general discussion of the WIAC procedures will first contrast the concept with that of the more easily understood adaptive wall, then show a recent 2-D result, and finally indicate the current status of both 2-D and 3-D procedures.

Contrast With Adaptive Wall

The 2-D adaptive wall concept is depicted graphically in figure 3. There are two basic elements: the tunnel which "solves" an internal flow problem and the computer which solves an "external-flow" problem. The solutions produced by these two devices are iterated by changing boundary conditions on the interface between the solution domains (that is, the measurement surface) until some specified convergence criterion (which indicates a match) has been met. Measurement is required of the distributions of two independent parameters along the measurement surface or the equivalent, that is, $C_p(x)$ and $\theta(x)$ or $u(x)$ and $v(x)$ or $C_p(x)$ and $y_{wall}(x)$ along a single surface or one parameter along two noncoincident surfaces depending on the particular implementation. One is used as an input boundary condition for the (fictitious) external flow problem whereas the second is compared with the prediction of the external flow calculation in order to obtain an input to the wall adaptation logic. This procedure is iterated until the wall adaptation meets the specified tolerance criterion. From the numerical point of view, two coupled boundary value problems are being solved, one on an analog machine (the tunnel) and the other on the digital machine (the computer).

The adaptive wall of the tunnel is an active element and it must be automatically controlled and adjusted in a reasonable time while the tunnel is operating in order to achieve any practical testing productivity. This will involve extensive hardware and software. In addition, the required data acquisition and processing tasks are greatly expanded over those associated with conventional tunnels. This technology is being pursued by a number of groups. Conceptually, it allows for transonic testing with minimal interference since consistently matched internal and external flows are obtained before model data are taken. Some of the tasks to be accomplished in a successful adaptive wall procedure have similar counterparts in the WIAC.

The 2-D WIAC concept is depicted graphically in figure 4. There are likewise two basic elements: the tunnel which generates the flow in which measurements are made, and the computer which now solves two related flow problems. The tunnel is a passive device where one takes more measurements than in conventional tunnel tests but less than in an adaptive wall tunnel. In the full nonlinear correction procedure two transonic flow problems are solved on the computer. The first is an equivalent inviscid tunnel flow where measured pressures near the wall and on the model are used as boundary conditions. The result of this first calculation is an equivalent inviscid model defined in terms of either its shape or its distribution of singularities. The second problem to be solved on the computer is a sequence of inviscid transonic calculations using free air outer boundary conditions and the equivalent model as the inner boundary and perturbing the free-stream Mach number and α in order to satisfy a best fit criterion at the model. In reference 22, the equivalent model boundary condition is input as the jump in perturbation velocities across the model (that is, $[\![u(x)]\!]$ and $[\![v(x)]\!]$) and the velocity vector at a point on the model just ahead of a shock is matched to that in the tunnel flow. In reference 23, the equivalent model boundary condition is the conventional shape specification of $v(x)$ for both upper and lower surfaces, while the matching criterion for M and α is specified as a best-fit with the measured model pressures in a least-squares sense. Two results are then obtained from these computer calculations: corrections to the free-stream conditions M and α and a measure of residual interference. If the residual interference is deemed to be large, then corrections to M and α alone are not satisfactory. It is suggested in reference 24 that this residual interference could be reduced to acceptable levels by a very simplified application of the adaptive wall concept. It should be noted that the measured model pressure data are customarily reduced to coefficient form using a reference Mach number; therefore, one must take into account the effect of the Mach number correction on these coefficients.

An Example of 2-D WIAC Results

A recent example to which the 2-D transonic WIAC procedure of reference 22 has been applied is for the supercritical BGK-1 airfoil of 10-inch chord which was extensively tested in the Ottawa NAE 15" × 60" Tunnel. Wall-interference results are illustrated herein for two cases at a Mach number of about 0.77 with different wall porosities. The case for 20 percent porosity is one for which other versions of correction procedures have also been applied by different groups (see, for example, ref. 25).

Wall perturbation velocities Δu and Δv along the model mean plane are shown in figure 5 for wall porosities of 20 percent and 6 percent. The streamwise location of the model is denoted by a heavier line and the distribution of pressure coefficients on the upper and lower model surfaces is plotted above it. Corrections to Mach number and angle of attack, ΔM and $\Delta\alpha$, should produce constant Δu and Δv so that departure from a horizontal line on these plots is a measure of the residual interference. In this sense then, it can be seen from figure 5 that one may well have less residual interference with larger corrections ΔM and $\Delta\alpha$. That is, one may not want to adjust even gross wall parameters so as to minimize ΔM and $\Delta\alpha$ in the context of the WIAC procedure. It can also be seen that there are several length scales to this residual interference: the very small length scales appear to be those of the individual large pressure gradients which bound the supercritical region on the airfoil upper surface and these, in turn, appear to be superimposed upon the larger model chord scale.

Figure 5 shows the residual interference for the 6-percent porous wall to be much larger than that for the 20-percent porous wall case. Figure 6, however, shows that even for the 6-percent porous wall case, the agreement between airfoil surface pressure coefficient distributions from the experimental data (corrected for ΔM) and from the calculated free-air flow (for the equivalent shape at corrected M with C_ℓ matched) is very good and the effect of the nonuniform wall perturbation is very small. Results from either wall porosity therefore appear amenable to correction by the WIAC procedure.

In obtaining the $\Delta\alpha$ correction, it appears that one absolute measurement of the flow angularity needs to be made somewhere in the test section. For the results presented in figures 5 and 6, a value was assumed upstream of the model.

Status of 2-D WIAC

As one might expect, the 2-D WIAC procedures are much further along than any for the 3-D case. There are inherent as well as practical differences and these will be discussed from the 3-D point of view in the next section. For the 2-D case, several methodologies have been published and both method validation and technique development continue to be investigated by a number of groups. Excluded from WIAC methodologies in the sense considered herein are those direct calculations which employ the homogeneous wall boundary conditions (refs. 10, 20, 21, 26, 27, and 28, for example). However, by imposing experimentally measured wall (or reference plane) and model boundary conditions in them, one generates assessment correction codes of interest.

Linearized 2-D WIAC methods applied to transonic flows or perturbations from transonic flows have been reported by groups in the USA (refs. 24, 29, and 30), France (ref. 31), Canada (refs. 25, 32, 33, and 34), Japan (ref. 35), Germany (ref. 36), and the Netherlands (ref. 37). Nonlinear 2-D transonic methodologies have been developed and reported in references 22 and 23. Even though details of these various implementations differ, corrections for one sample transonic test case as determined from five of the above methods (refs. 22, 25, 31, 35, and 37) were essentially the same. These results (as yet unpublished) were presented at the 1980 meetings of the AGARD Fluid Dynamics Panel Conveners Group on Transonic Test Sections.

Method validation and technique development continues using data bases in which pressure data (and also velocity data for some procedures) are measured not only on the model but also near the tunnel wall. Such data bases are being generated by groups interested in WIAC and also adaptive wall applications. Both method validation and technique development are guided by the specific application one has in mind. Fast linearized methods could conceivably be used on-line for test guidance while more exact nonlinear formulations might be used in formal data reduction. It is not yet clear what the range of applicability is for either formulation; surely there is some difference between that of the linear and nonlinear cases. One conclusion which can be drawn, however, is that at least one flow angularity measurement needs to be made somewhere in the test section; that is, in the region where pressure measurements are being made. This flow angularity value corresponds to an integration constant in these formulations. Without it, one can only assume some value for a flow direction, generally taken at the upstream extent of the pressure data measurements.

Status of 3-D WIAC

Several aspects of extending the 2-D WIAC to 3-D need to be discussed. First, a straightforward direct extension of the 2-D procedure (as outlined earlier) to 3-D is certainly possible from the software or computer point-of-view. However, an excessive amount of pressure data would be required both near all walls and on the model. Tests will continue to be conducted on many models where pressure data are not taken on the model and this then impacts how or whether one will solve the inviscid tunnel flow to obtain a computational model of the equivalent body. These two aspects of how much and what kind of data are needed as well as where it is to be taken will be governed by the hardware or tunnel/model aspects. Our current thinking is that representative pressure data near the walls (perhaps several streamwise traces as opposed to sparse 2-D arrays for each wall) and model force and moment data are all that one could reasonably expect to get in most tests.

Another aspect is related to the inherent difference between 2-D and 3-D flows. The transonic relief effect afforded by the third dimension compresses the Mach range of nonlinear (transonic) effects. Thus, the linear/nonlinear methodology mix which will be required for an adequate 3-D WIAC may differ from the 2-D case. As already indicated, this issue has not yet been resolved for 2-D; we expect that the 3-D case might be more tolerant of linear aspects except very near Mach 1. The case of transonic axisymmetric flow which exhibits 3-D relief, but is mathematically 2-D, may allow some insight into these aspects; we have therefore included it in this discussion.

Just as in the 2-D case, homogeneous tunnel-wall boundary conditions have been used in the earliest direct transonic axisymmetric (refs. 38 and 39) and 3-D wing (ref. 21) codes. Again, with changes to allow for measured data as boundary conditions, one could base a WIAC procedure upon these codes. In this regard, a nonlinear (transonic) axisymmetric code has been modified to take measured pressures as outer flow boundary conditions and results have been published in reference 40. These results are encouraging since the agreement for model pressure distribution was good even though the equivalent body model was taken as the solid geometric shape. Likewise, several transonic 3-D wing alone and wing-body codes have been modified to accept measured pressures as outer field boundary conditions (refs. 41 and 42). In reference 41, experiments were tailored to produce transonic data to be used in 3-D transonic code evaluations. In order to account for the transonic wall interference, the tested codes were modified to accept measured pressures as outer field boundary conditions. The good agreement between experimental and numerical (with experimental pressure values for outer flow boundary data) wing surface pressures is encouraging for a 3-D WIAC. It was seen, however, that for agreement over the aft portion of the wings, an account of the boundary layer was needed. That is, one may not be able to use only the solid geometric shape as the equivalent model shape in a WIAC. The 3-D wing-body code of reference 42 was developed in order to model the tunnel in numerical simulations of adaptive wall control in support of the USAF/AEDC program. This code allows for several "wall" models, one of which is for pressure data prescribed. It is just now beginning to be exercised.

An NASA study contract (NAS1-16262) was competitively won by Flow Research Company (FRC) and initiated in September 1980 in order to examine the aspects noted above. It is envisioned that they will develop pilot 3-D codes which will be available when the NTF comes on line in 1982. A 3-D version of the 2-D linear method outlined in reference 25 has been coded by FRC but not yet exercised nor published. It has come to our attention that the Canadians also have a 3-D version of their 2-D method (ref. 25) but we have not found the publication of it.

WALL-INTERFERENCE EXPERIMENTS

It is envisioned that all of the early experiments in NTF will provide data relevant to wall interference effects and should, in the broadest sense, be considered as wall-interference experiments. The independent matching of both Mach and Reynolds number on the one hand places added emphasis on properly accounting for wind-tunnel-wall effects while on the other hand it provides the possibility of a better experimental assessment of these effects (in NTF) than has been possible for other facilities. Since one of the purposes of this workshop is to provide input for impact upon early NTF use, we will discuss wall-interference experiments in terms of general ideas about tests already planned which involve two different model sizes and, more specifically, for a proposed body-of-revolution drag-rise experiment.

Independent Experimental Wall-Interference Assessment

An independent experimental assessment of the symptoms of wall interference can be made by testing two different model sizes in the same tunnel. In addition, if one can independently adjust Mach and Reynolds numbers then the wall-interference effect can be uncoupled from the Reynolds scaling due to model size. Several experiments are planned where two model sizes are envisioned for reasons in addition to wall-interference studies. Figure 7 depicts (not to scale) three such studies. For example, the large SCR model is to be used in low-speed, high-lift tests while a smaller size model will be required for high-speed testing. Two model sizes are also planned for the Pathfinder I EET model. Drag-rise experiments will be done on bodies of revolution and here at least two model sizes will be used. In earlier NASA Langley experimental work (ref. 43) a number of model sizes were tested in order to understand tunnel influence upon drag. It is felt that these drag-rise experiments on bodies of revolution, very near Mach 1, represent a situation in which an assessment of wall interference may be essential for interpreting the outcome.

Body of Revolution

The interpretation of many experiments in NTF may depend upon being able to unscramble the tunnel wall interference. A proposed body-of-revolution drag study through Mach 1 certainly falls into this category and we propose here to design the experiment and procedures so that data are taken to allow an early application of WIAC procedures to the NTF test section. For axisymmetric flow, the 2-D mathematics allow for ease in studying different WIAC approximations while the flow itself exhibits some of the 3-D phenomena. Some suggested aspects of such an experiment are depicted in figure 8. As indicated, the emphasis would be about $M = 1$; we would use axisymmetric WIAC procedures in attempting to assess wall interference for the experiment while at the same time using the experimental data for a validation of the adequacy of the WIAC procedure. At least two model sizes, perhaps scaled for similar flow between the larger one in the NTF and the smaller one in the 0.3-m TCT, would be tested. The smaller one would, of course, be available for testing in the NTF. Pressure measurements would be taken on the model, sting, tunnel walls, and at several places in the flow field (at least for the NTF). One would like to ascertain that a region of axisymmetric flow does exist so that results could be obtained prior to the development of a nonaxisymmetric WIAC procedure.

NTF WALL-INTERFERENCE TECHNOLOGY OUTLOOK

The near-term aspects of the NTF wall-interference technology are shown schematically in figure 9. Several of the early tests in NTF will involve two model sizes. Data from such pairs of tests will show the symptoms of wall interference, perhaps parameterized over Mach and Reynolds number ranges in a way which was not possible in the past. This would represent an independent experimental assessment of wall interference. At the same time, these data provide a base for validation of WIAC procedures since pairs of tests, with most other conditions matched, will have different wall interferences. All tests in NTF will contribute to generation of a statistical data base for wall interference. The near-term use as indicated in figure 9 would be for predicting corrections, defining the range of Mach number, lift, model size, etc., where the data are correctable and hopefully to guide future wall and procedure improvements.

Far-term aspects will certainly follow progress in the related technologies of computational fluid dynamics, tunnel-wall boundary-layer control, data acquisition, and adaptive tunnel walls. As indicated earlier, the major simulations not yet achieved are related to matching the unboundedness condition of free flight. The prospect of eliminating interferences due to tunnel walls, model support, and measurement probes via hardware devices has been discussed in this workshop and is being pursued and funded here at NASA Langley. The long-range plan for NTF includes consideration of an adaptive wall. From the productivity standpoint, it is not yet clear just how adaptive though. Conceptually, it seems at this time to be the proper way to go. The experience to be gained by use of WIAC in the present NTF test section should, however, be used to optimize the present test section and its use in conjunction with WIAC should then be evaluated to determine the need for adaptive features.

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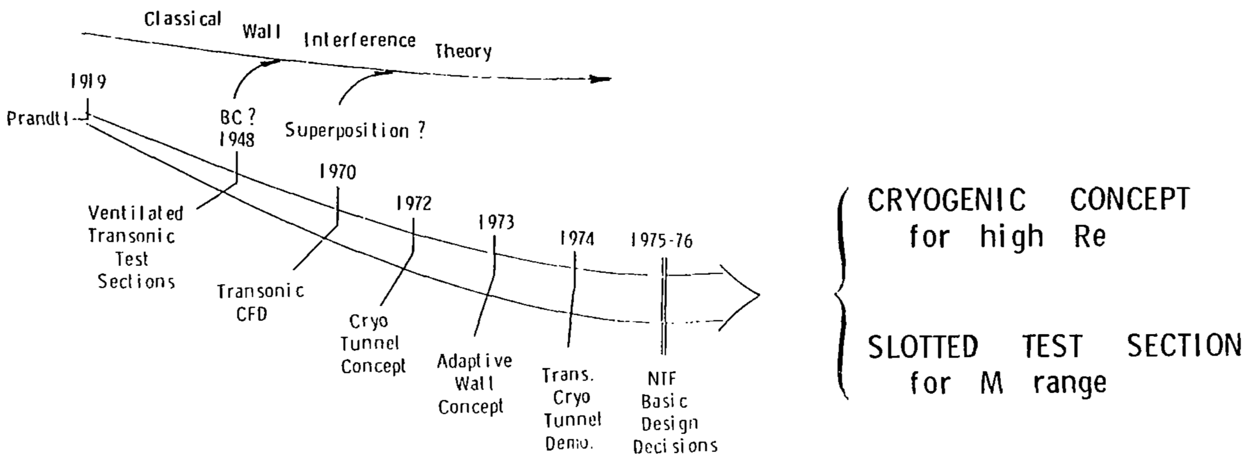
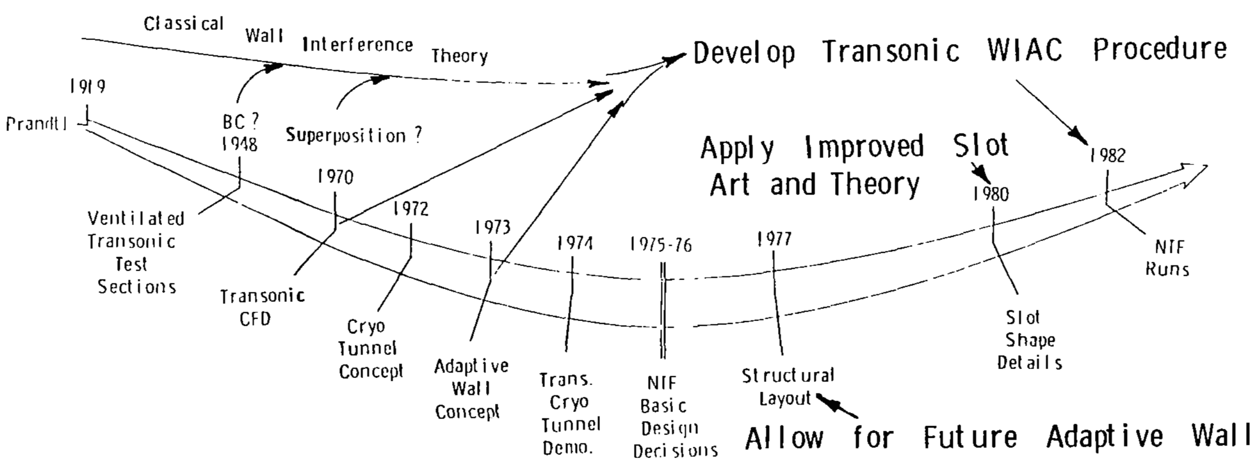


(a) Pre-1976.



(b) Post-1976.

Figure 1.- Wall interference technology in NTF Historical context.

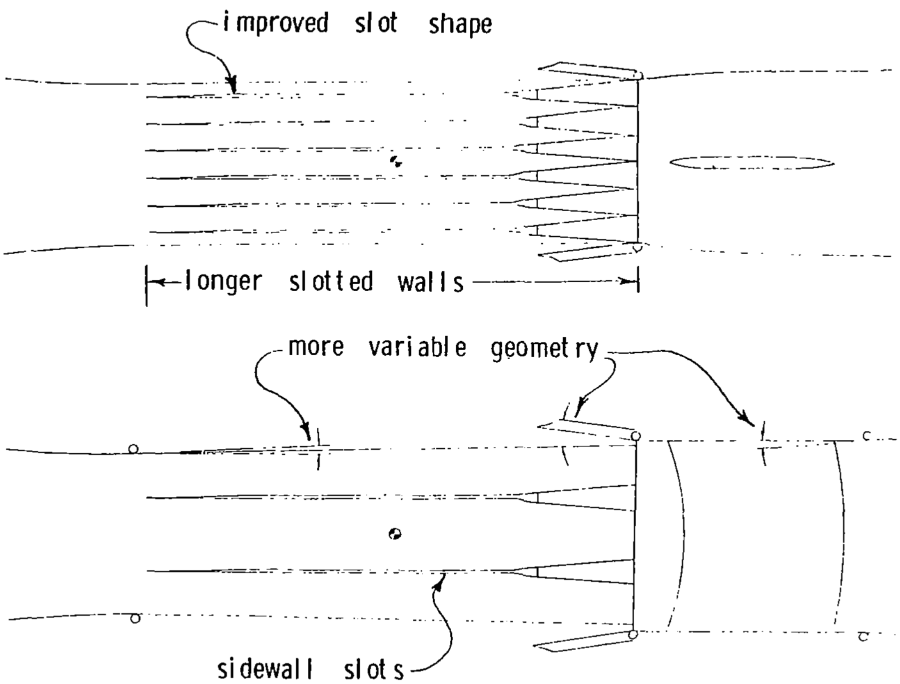


Figure 2.- Sketch of NTF slotted-wall test section.

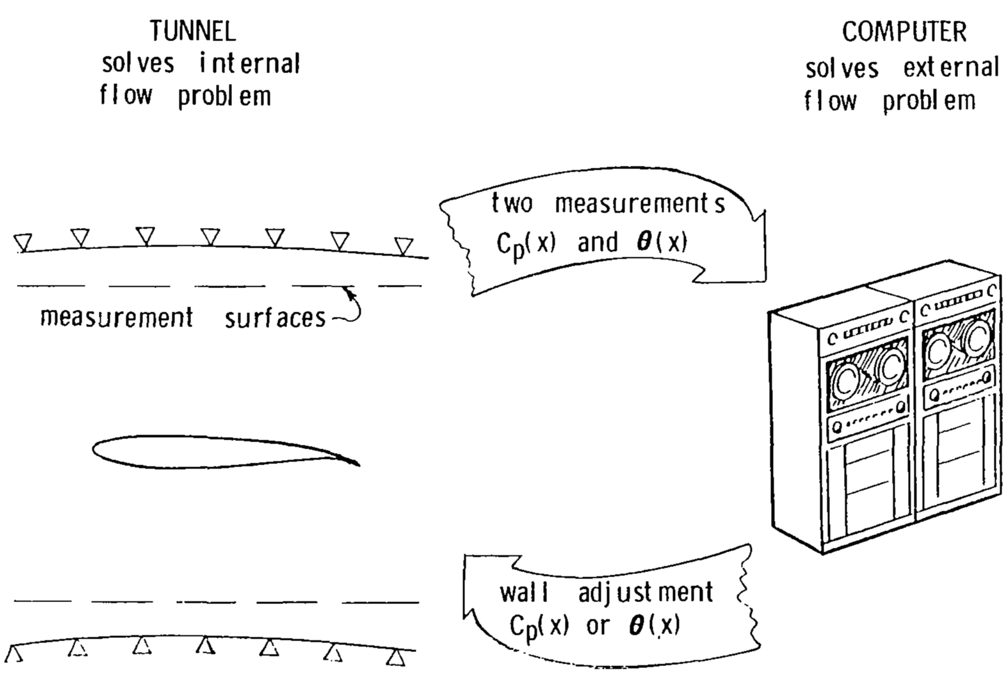


Figure 3.- Adaptive wall concept.

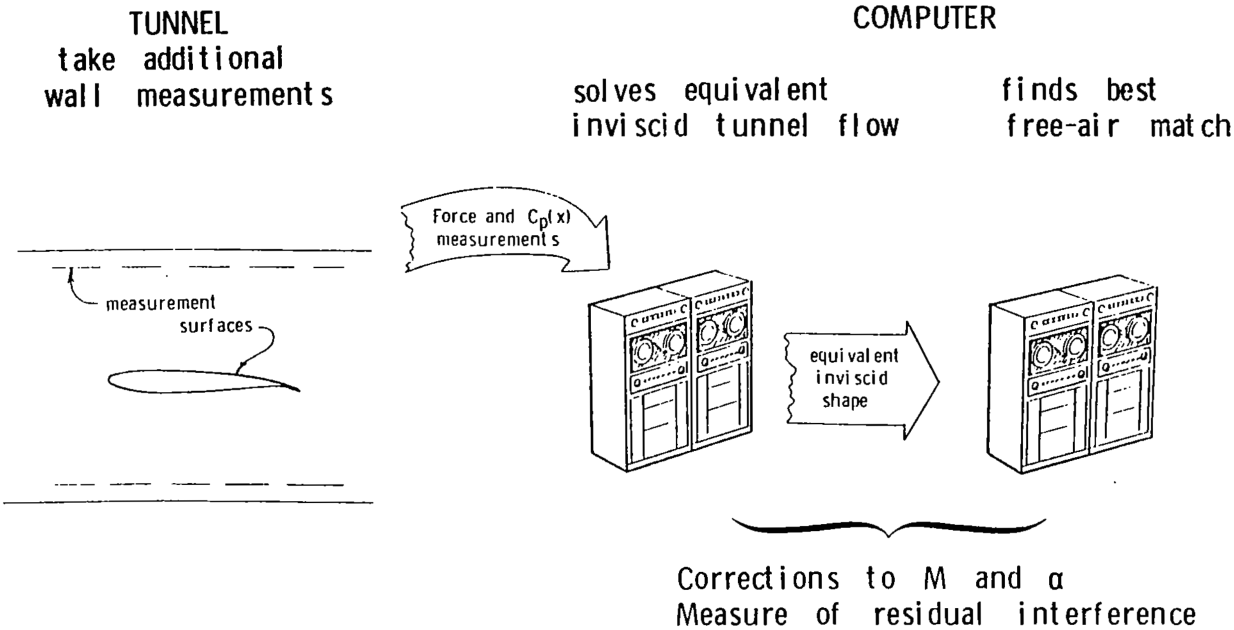


Figure 4.- Wall interference assessment correction (WIAC) concept.

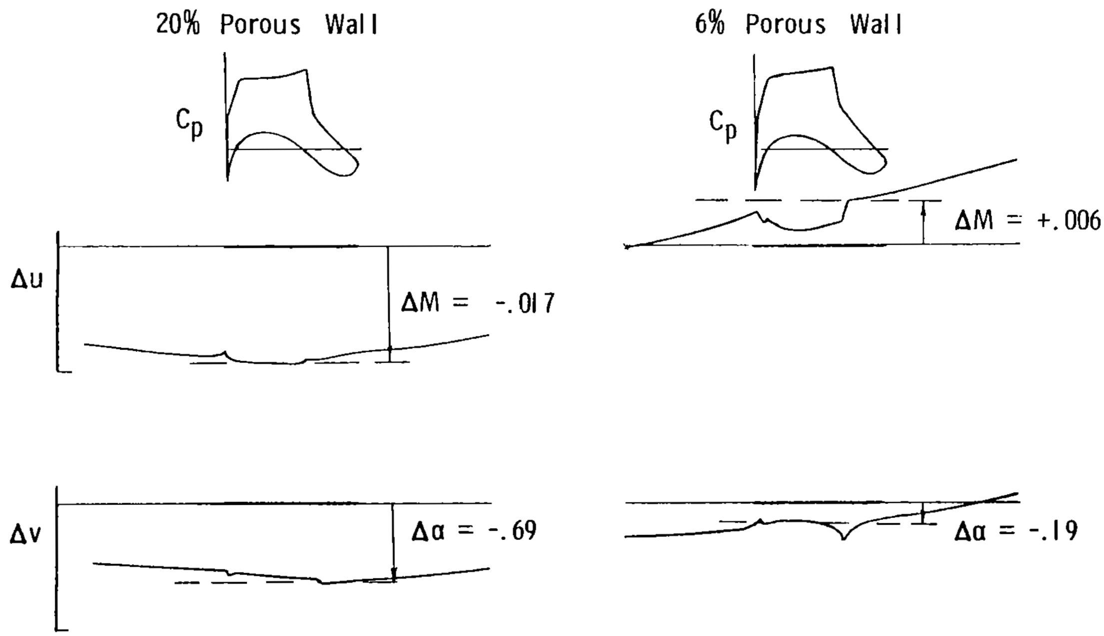


Figure 5.- Wall perturbation velocities calculated from 2-D WIAC procedure.

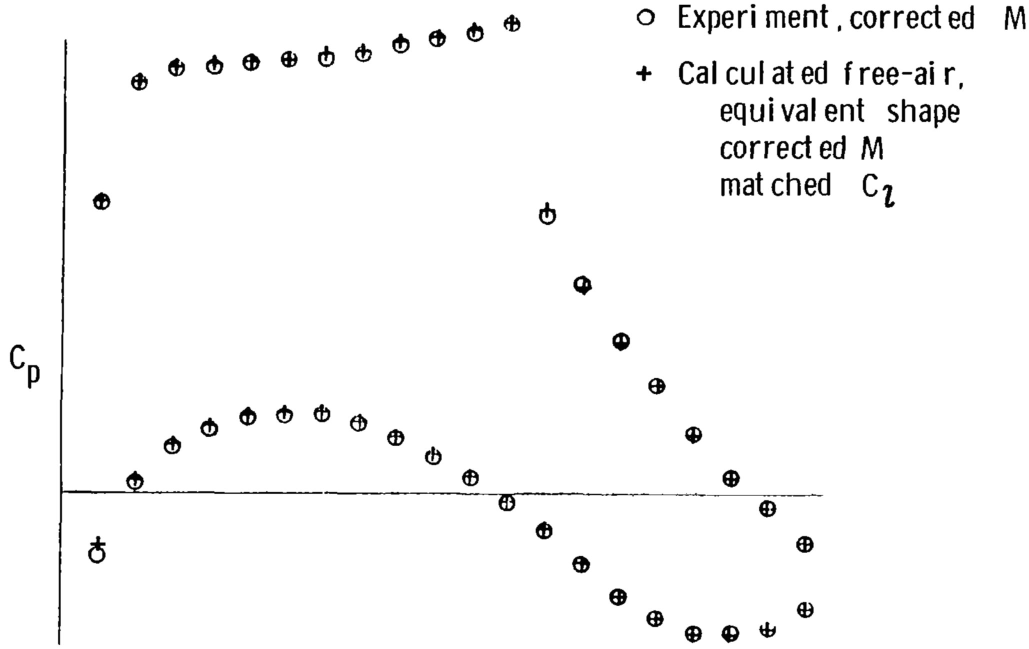


Figure 6.- Comparison of experimental and calculated airfoil surface pressures. NAE 15" x 60" tunnel; BGK-1 airfoil; c = 25cm; M ≈ 0.77; 6% porous wall.

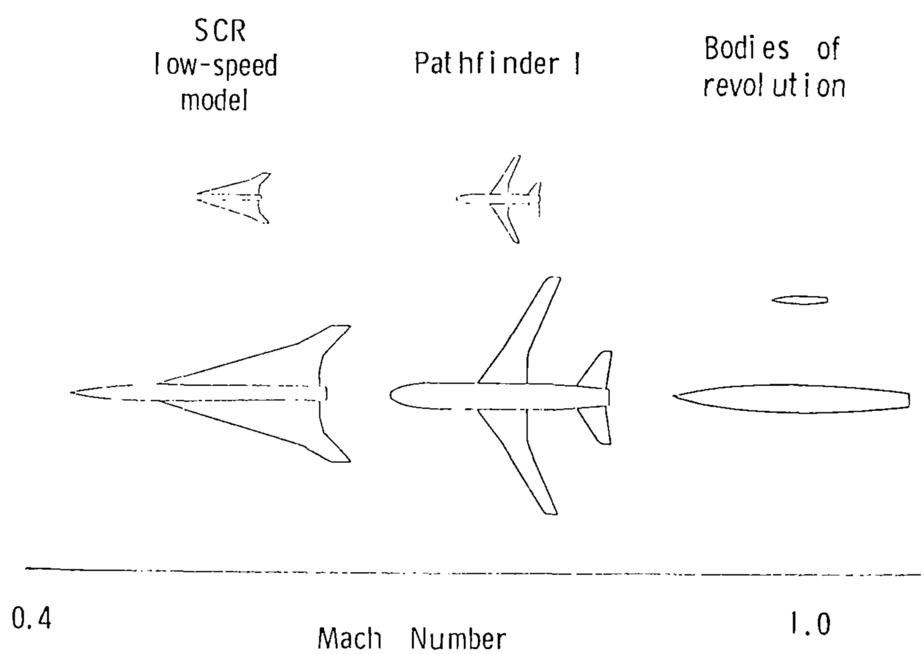
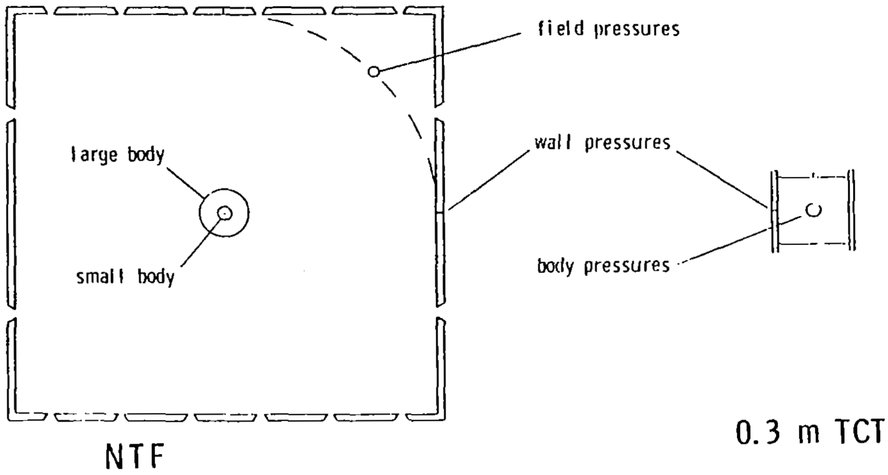
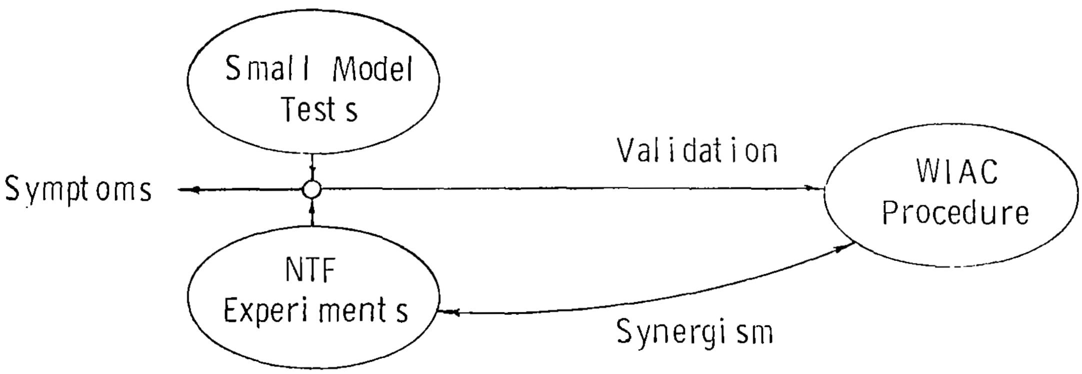


Figure 7.- Some planned early NTF independent wall interference assessment experiments.



- Emphasis on M ≈ 1
- Use axisymmetric WIAC procedure
- Data for 3 - D WIAC procedure

Figure 8.- Bodies-of-revolution drag-rise experiments.



GENERATE STATISTICAL DATA BASE

Predict corrections
Define correctable range
Mach, lift, model size
Guide wall improvements

Figure 9.- Near-term NTF wall interference technology outlook.

COMMENTS ON REYNOLDS NUMBER EFFECTS AND THE ROLE

OF NTF IN THE DEVELOPMENT OF AIR VEHICLES

A. L. Nagel
Boeing Commercial Airplane Company

The advent of the National Transonic Facility (NTF) with its capability for matching the full scale Reynolds numbers of all but the largest airplanes is a major advance in testing capability. Boeing welcomes the opportunity to help formulate development plans for this unique new facility by participating in this second high Reynolds number workshop. Conversion factors to enable calculation of SI-unit equivalents for all U. S. units used in this paper are listed in Table 1.

TABLE 1

To convert from	to	Multiply by
inch	centimeter	2.54
foot	meter	0.3048
microinch	micron	2.54×10^{-2}

REYNOLDS NUMBER RANGE OF INTEREST TO BOEING

Figure 1 illustrates the Reynolds number range of current and forseeable Boeing products. It is seen that the NTF will encompass much but not all of the region of interest. In the past, Boeing Company products have been developed primarily in the Boeing Transonic Wind Tunnel (BTWT), an atmospheric wind tunnel with an 8- by 12-foot test section. Because of the large differences shown between the Reynolds numbers available in BTWT and flight, Boeing has made extensive use of other facilities, including those of NASA, the Air Force, and private organizations. Using data from several facilities, analytic methods, and an extensive backlog of flight test data, Boeing has been successful in developing competitive aircraft in the relatively low Reynolds numbers available in BTWT. Although there has not yet been a case of a major discrepancy between the corrected wind tunnel data and the characteristics of the full scale airplanes, Boeing will be quick to take advantage of the better simulation offered by NTF. As shown in the figure, NTF will offer the capability of obtaining data at full scale Reynolds numbers in the cruise condition for most of the forseeable Boeing products, and will be much closer than previous tunnels to full scale Reynolds number for the operating envelopes. Great care will still be required in applying the data, however, because it is primarily on the operating envelope that Reynolds number effects are most important and least predictable.

BOEING CRYOGENIC AIRFOIL MODEL AND DATA

In June 1980 Boeing became the first of the airplane industry to conduct an airfoil test in the Langley 0.3-Meter Transonic Cryogenic Tunnel (0.3-m TCT). Figure 2 shows the model that was used for this test: an 8-inch span, 6-inch chord, 2-dimensional wing having a proprietary Boeing airfoil section. The model was smoothed to an RMS reading of 4 to 6 microinches and all of the coordinates were within .0012 inches. The smoothness requirement was based on classical skin friction data. The surface contour requirements were evaluated by analyzing contour error effects with transonic airfoil codes.

Figure 3 shows the comparison of the data obtained in this test with theoretical methods. The data and theory are normalized by the theoretical turbulent flat plate skin friction. The data of this figure are taken at a Mach number of 0.6 so as to be free from shock-boundary layer interaction effects. This condition is considered to be the basic starting point for evaluating Reynolds number effects. Although no transition detection method was available for this test, it was established by comparing free transition data with tripped data that at the lowest Reynolds number shown transition occurred at approximately 10 percent chord. When the low Reynolds number data are corrected for this initial laminar run the agreement with the theory is very good. At higher Reynolds numbers where the transition point is unknown, the data theory comparison cannot be so discriminating. An upper bound for the effect of the laminar run was estimated by assuming 10 percent laminar flow at all Reynolds numbers. A lower bound was estimated by assuming a constant transition Reynolds number, which appears to be more realistic. The difference between the two methods of correcting for laminar run is indicated by the shaded band in the figure. The data agree well with the theory, but for the case shown the various theories are all in quite close agreement. The actual curve shown here is based on the method of Melnik,[1] but other methods are also in close agreement with the data, as are the RAE data sheets (ref. 1).

The largest source of uncertainty in this comparison is the unknown extent of the laminar flow, which may also be a difficult problem for future NTF testing. The location of boundary layer transition must be expected to affect all Reynolds-number-dependent phenomena, and so must be determined. However, at the very high unit Reynolds numbers of NTF, flow visualization materials may affect transition locations. On the other hand, surface instrumentation techniques such as thin film gauges have the disadvantage that for the irregular patterns of transition that often occur on complicated shapes, an impractically large number of gauges may be required.

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[1]R. E. Melnik; R. R. Chow; H. R. Mead; and A. Jameson: An Improved Viscid/Inviscid Interaction Procedure for Transonic Airflow Over Airfoils. Grumman Aerospace Corporation, prepared under contract no. NAS1-12426, February 1980.

DRAG RISE VARIATION WITH REYNOLDS NUMBER

Figure 4 illustrates a more complicated situation. Here we present the drag increase above a "subcritical" Mach number at which no shock interaction effects occur. For different classes of airfoils this onset of transonic flow effects occurs at slightly different Mach numbers. The data shown in this figure are obtained from proprietary Boeing tests.

The point illustrated by figure 4 is that Reynolds number effects may be in opposite directions for different airfoils. We do not know of a theoretical method that can successfully predict the different effect of Reynolds number on drag rise observed on these two airfoils. Hence, even for this relatively simple two-dimensional flow condition, reliable wind tunnel data are required. For more complicated situations involving three-dimensional wing-body combinations, we must anticipate that high quality wind tunnel data will be required to determine Reynolds number effects for the forseeable future.

ADDITIONAL SIMULATION REQUIREMENTS

Reynolds number is only one of many possible sources of differences between the wind tunnel data and full scale data. Other sources include wall effects, model support effects, model and model support deflections, flow quality, model surface smoothness, and model surface temperature.

The effects of these variables will not disappear as full scale Reynolds numbers are achieved in the wind tunnels. Some, such as wall effects, are themselves Reynolds number dependent. If the tunnel dynamic pressure varies with Reynolds number, deflections of the models and supports under airloads may distort or even reverse the apparent effect of Reynolds number. The new capability of NTF to isolate such effects by varying dynamic pressure and Reynolds number independently may be almost as valuable as the Reynolds number capability itself.

Flow qualities include angularity, swirl, noise, and turbulence. Flow angularities introduce uncertainties in resolving balance measurements into pure lift and drag. Noise and turbulence affect boundary layer transition, which may therefore occur at different locations than on the full scale aircraft, even at identical Reynolds numbers. The result may be differences in shock-boundary layer interactions and separation, leading to differences in drag, drag rise Mach number, buffet, and stall behavior.

Model surface smoothness cannot be expected to simulate that of the full-scale airplanes. Scaled excrescences and roughnesses would be impractically small. In a cryogenic wind tunnel of short duration flow the wall temperature ratio will tend to be higher than at the full scale flight condition, unless effective model cooling methods are devised. Theoretically, at least, wall temperature ratios and temperature gradients affect both boundary transition and the turbulent boundary layer profiles, and could lead to the separation effects discussed above.

CONCLUDING REMARKS

The NTF will offer a major improvement in test capability. A preliminary analysis of data from the 0.3-Meter TCT indicates that valid high Reynolds number data can be obtained in a cryogenic environment. The largest source of uncertainty found so far is in the unknown extent of laminar flow. Research into transition detection and transition effects in NTF should receive high priority.

In a previous communication (letter B-8020-00-05, dated 2-4-75, to W. B. LaBerge, Dept. of Air Force, from H. W. Withington) Boeing has indicated that "use of NTF would only be for verifying or fine tuning particular configurations that are expected to be sensitive to Reynolds number effects." Even if cost and logistics were not important considerations, the total NTF testing time available to any one user would preclude its use for product development testing. For this reason it is recommended that research be conducted in both the 0.3-Meter TCT and the NTF with the object of developing improved simulation techniques and methods for extrapolating model data to full scale interference-free conditions. The development of such techniques would significantly enhance the great national investment in many excellent existing wind tunnels.

REFERENCE

1. Engineering Science Data, Aerodynamics Sub Series, Vol. 2a, published by Engineering Sciences Data Unit, London.

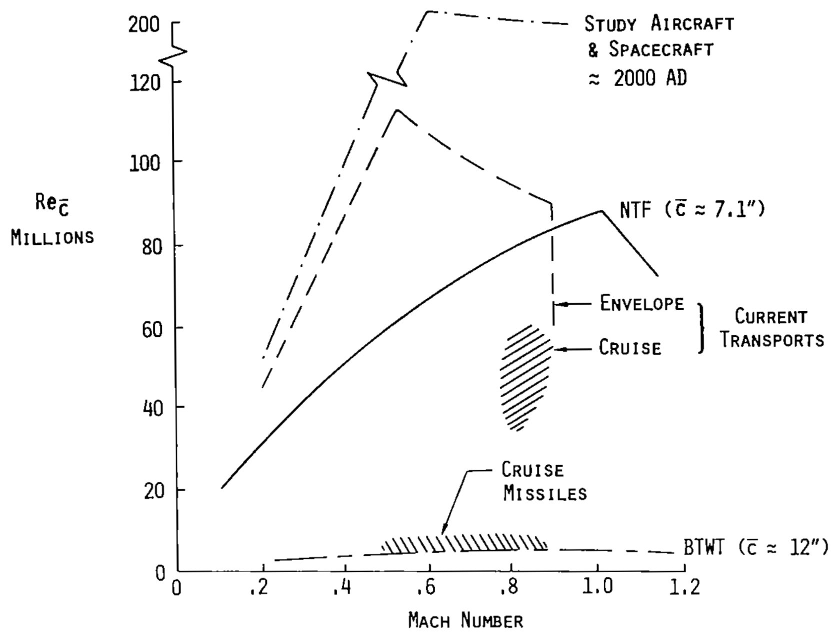

Figure 1.- Reynolds numbers of interest to Boeing.

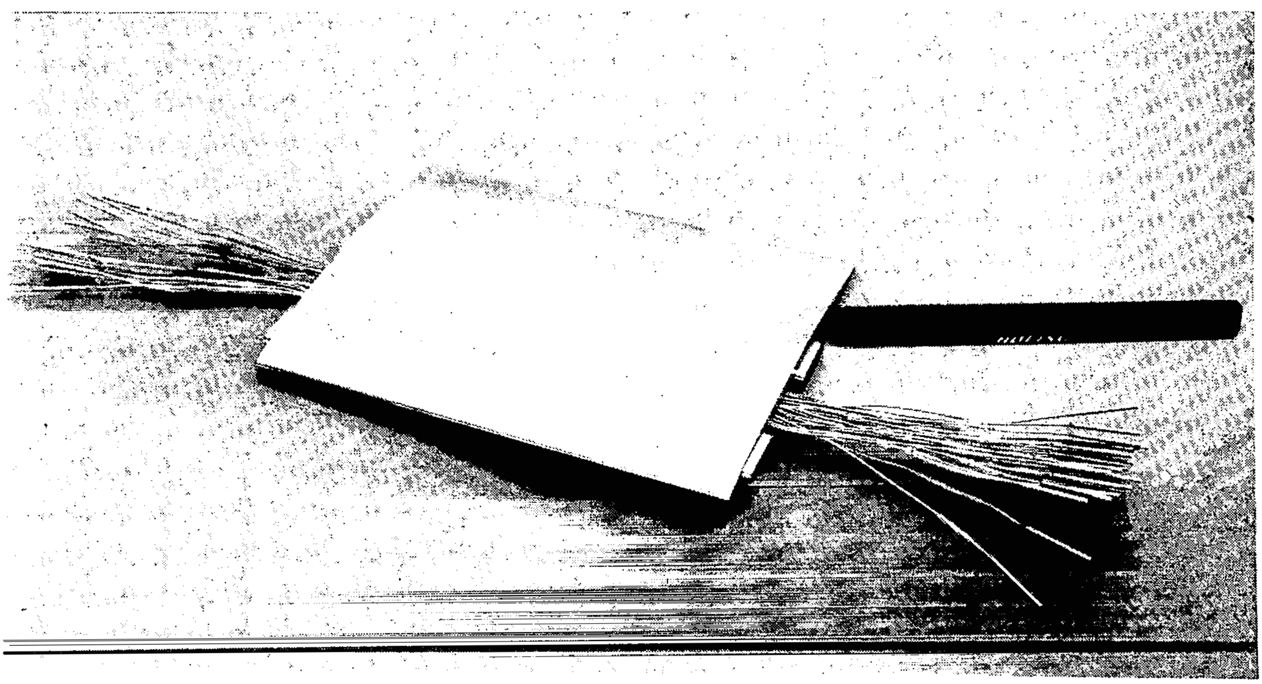

Figure 2.- Boeing-developed 15.24-cm-chord (6-in-chord) airfoil recently tested in the 0.3-m Transonic Cryogenic Tunnel.

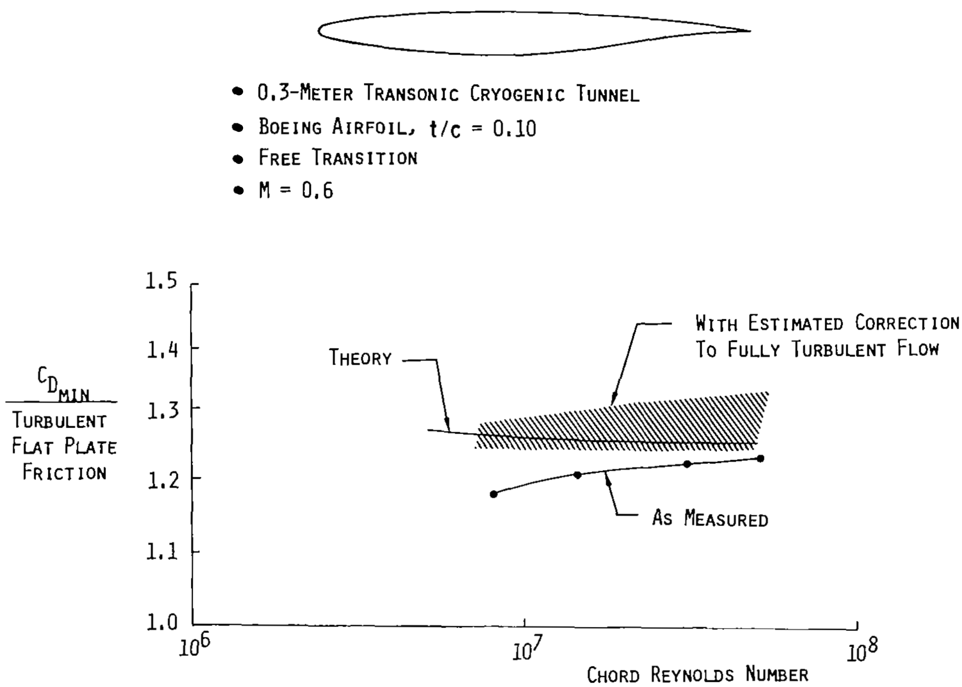

Figure 3.- Reynolds number effect on minimum drag.

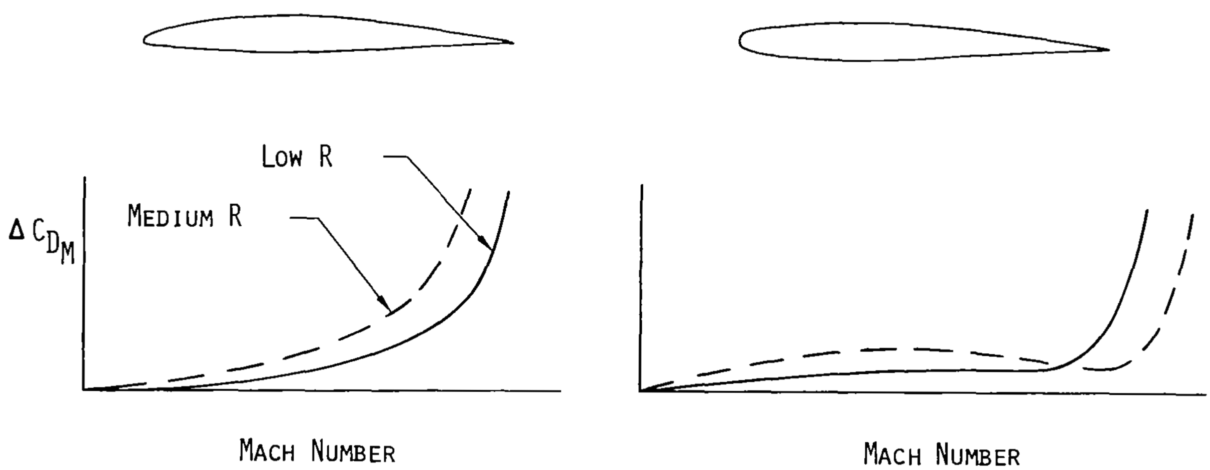

Figure 4.- Reynolds number effect on drag rise.

AIRCRAFT DESIGN USING THE NATIONAL TRANSONIC FACILITY

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North American Aircraft Division, Rockwell International

INTRODUCTION

The approaching availability of the 2.5-meter cryogenic high Reynolds number test facility necessitates evaluation of its uses by prospective groups. An aircraft manufacturer's views are presented in the context of production tunnel status. Facility calibration and wall and support system corrections are essential precursor activities. Early flight test correlation of local characteristics is required to foster high user confidence.

GENERAL

Greatest use of the NTF for new system development is antitipcated in the areas of 1) developing reliable attached and separated flow scaling rules for use in correcting existing test facility measurements to full scale Reynolds number and 2) estimation of the aerodynamic characteristics of well designed arrangements which have undergone a series of evaluation and refinement tests in large high density conventional facilities. This latter activity also supports the important function of validating advanced computational design methods and defining areas requiring further development.

The envisioned interfaces between the NTF and conventional wind tunnels are summarized in figure 1. It is tacitly assumed that either company or government test facilities with Reynolds test capability of $R_{\bar{c}} \sim 5 \times 10^6$ with angle of attack ($R_{\bar{c}} \sim 10 \times 10^6$ for load limited testing) are available for development efforts in order to reduce the impact of scale effects on configuration design.

DESIGN CONSIDERATIONS

Aerodynamic performance estimation, numerical wing design verification, and high Reynolds number theoretical extensions are specific areas of NTF use that are important for advanced system development.

Rules for extrapolating viscous drag to flight Reynolds number need to be validated for both hydraulically smooth and distributed rough surfaces since many aircraft have important flight points which span both conditions. Development of test techniques to control or measure transition is a necessary part of such a correction procedure. Two- and three-dimensional boundary layer analysis as well as flat plate skin friction algorithms should be considered since they systematically treat variable pressure gradient effects and associated displacement thickness pressure drag.

A systematic accounting for the boundary layer developing on the wing is required as a result of the modern practice of employing wing sections developing controlled supercritical flow in conjunction with steep but well managed adverse pressure gradients. This has given rise to the use of Reynolds number incremented geometry (undercutting of potential design coordinates) or transition strips located well back from the leading edge to simulate full scale displacement thickness approaching the trailing edge. The ability of these approaches to simulate desired flows at flight Reynolds number should be demonstrated.

The use of adaptive wing geometry (e.g. variable camber, aeroelastic twist, etc.) to expand the high aerodynamic efficiency operating envelope results in boundary layer interactions which are not fully controlled and consequently are of the strong (at least locally) variety. The wing performance for such conditions is subject to local separated flow scale effects which are generally diminished as the Reynolds number is increased.

In the area of high Reynolds number theoretical extensions, validation of attainable potential leading edge suction is required to support emerging wing planform design studies based on such concepts. Present knowledge is based on limited test results for $R \sim 6 \times 10^6$.

Component estimates of state-of-the-art integral and finite difference boundary layer analysis use various natural transition and separation criteria. In general, this entire methodology requires validation at high Reynolds number and in all probability will need revision to bring it into conformity with new low turbulence test information.

The foregoing considerations are summarized in figure 2. The topics are not considered to be exhaustive but represent important areas for which the NTF should markedly improve the state of knowledge.

USE PRIORITY

Ranking of NTF uses from an airframe development viewpoint is presented in figure 3. First priority is assigned to activities 1) providing estimates of new system six-component aerodynamic force and moment characteristics at full-scale R and 2) establishing conventional tunnel limits and Reynolds number scaling procedures. A slightly lesser priority is assigned to the validation of numerical design methodology and support of theoretical extension concerned with correcting identified deficiences.

It is recognized that the indicated ranking is quite different from that required for basic research, fundamental methods development, etc.

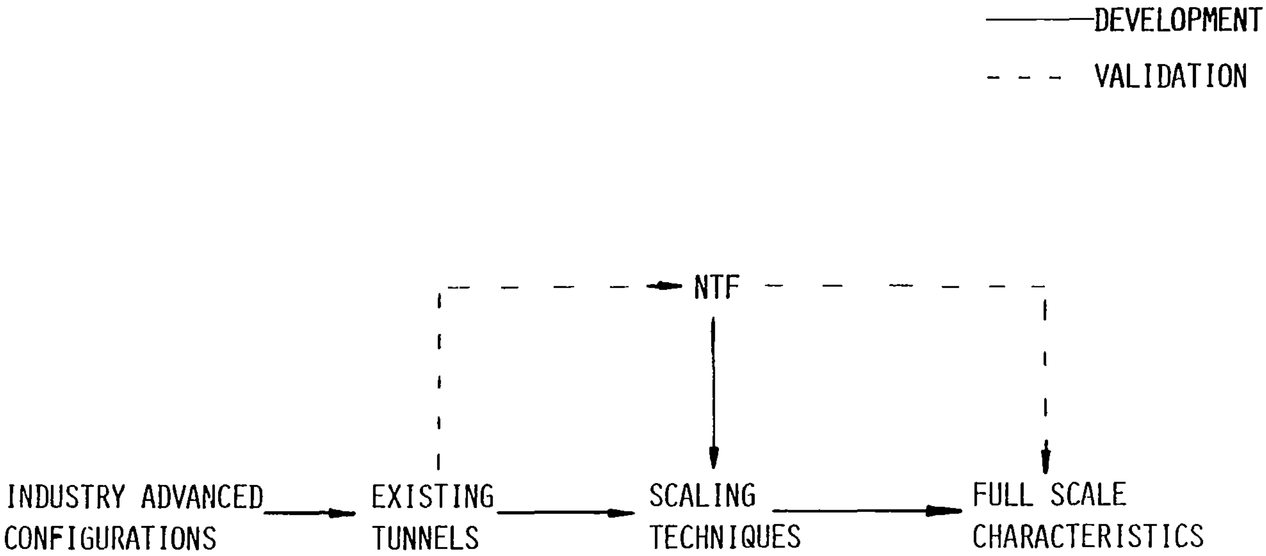

Figure 1.- Envisioned NTF role in new configuration development.

- PERFORMANCE VALIDATION
  - A) VISCOUS SCALING
  - B) ROUGHNESS EFFECTS/CUTOFF BEHAVIOUR
- NUMERICAL WING DESIGN VALIDATION
  - A) REYNOLDS NUMBER INCREMENTED GEOMETRY
  - B) VARIABLE CAMBER PERFORMANCE
  - C) ELASTIC AERODYNAMIC CHARACTERISTICS
- HIGH RN THEORETICAL EXTENSIONS
  - A) ATTAINABLE EDGE SUCTION PREDICTION
  - B) TRANSITION PREDICTION
  - C) SEPARATION/REATTACHMENT CRITERIA

Figure 2.- Specific NTF uses in new configuration development.

- FIRST PRIORITY

  1. NEW SYSTEM AERODYNAMIC VALIDATION
  2. ESTABLISH LIMITS WITHIN WHICH CONVENTIONAL TUNNELS CAN BE EFFECTIVELY AND RELIABLY USED
  3. VALIDATE/DEVELOP MODERATE REYNOLDS NUMBER SCALING PROCEDURES FOR ATTACHED AND SEPARATED FLOWS

- LESSER PRIORITY

  1. VALIDATE ADVANCED THREE DIMENSIONAL NUMERICAL DESIGN TOOLS
  2. SUPPORT NEW THEORETICAL DEVELOPMENTS

Figure 3.- NTF use priority from an aircraft design viewpoint.

PROPOSED AEROELASTIC AND FLUTTER TESTS FOR THE NATIONAL TRANSONIC FACILITY

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Rockwell International, North American Aircraft Division

INTRODUCTION

Based on our experience in aeroelasticity and flutter at the North American Aircraft Division of Rockwell International, a number of tests come to mind that can either exploit the unique high Reynolds number capabilities of the National Transonic Facility and the 0.3-m Transonic Cryogenic Tunnel or lead to improvements that could enhance testing in these tunnels. These proposed tests are summarized in figure 1. Conversion factors to enable calculation of SI-unit equivalents for all U.S. units used in this paper are listed in table 1.

TABLE 1.

To convert from	to	Multiply by
inch	centimeter	2.54
foot	meter	0.3048
mile per hour	meter per second	0.447
pound per square foot	Newton per square meter	47.88

SHOCK-INDUCED OSCILLATION

The shock-induced, self-excited airfoil bending oscillation has been identified in HiMAT wind tunnel tests, B-1 flight tests, and B-1 flutter model tests. This is believed to be a new phenomenon that can occur on swept airfoils at or near critical Mach number conditions. Conditions of occurrence did not significantly impact flight mission requirements in these instances but could affect future flight systems. The oscillations were divergent, angle-of-attack dependent and not classical flutter.

Figure 2 shows an unstable oscillation of the canard of the HiMAT aeroelastic model encountered in the first symmetrical canard bending mode at a dynamic pressure less than half of the calculated dynamic pressure for classical flutter. The oscillation was self-initiated and increased in amplitude when the sting angle was decreased to -3.8°, and damped out when the sting angle was reduced toward zero. In one case the amplitude was allowed to diverge to a magnitude that, while it did not cause failure of the model with its solid steel spar located well within the mold line, would be expected to result in failure for a typical full scale structure with structural skins just inside the mold line.

Figures 3 and 4 describe the theoretical mechanism for shock-induced oscillations of a swept airfoil. Figure 3 shows the HiMAT aeroelastic model canard first bending vibration mode, typical of aft-swept airfoils. In this mode it can be seen that streamwise airfoil sections in the outer portion of the span oscillate in pitch about node line axes ahead of the leading edge. The left hand portion of figure 4 illustrates the oscillation of an outboard streamwise section of the HiMAT canard about a forward node line in the fundamental bending mode with the canard at a negative mean angle of attack α_o. The shock on the lower surface moves aft for increasing $(-\Delta\alpha)$ angle of attack and forward for decreasing $(+\Delta\alpha)$ angle of attack. Since the pressure P_2 downstream of the shock is greater than the pressure P_1 upstream of the shock, it can be seen that the prescribed shock motion with $\Delta\alpha$ results in a ΔL variation with $\Delta\alpha$ that is out of phase with the vertical displacement Z. This phase relation of ΔL with Z with no lag is shown in the Argand diagram on the right side of figure 4. Since the speed with which the shock can change its location when the angle of attack changes is limited by the speed of sound, a phase lag ϕ must be introduced. This results in a component of the restoring force, $\Delta L \sin \phi$, in phase with the velocity of motion $\dot{Z}$. It is this component that does positive work on the motion. However, for the oscillation to be divergent, $\Delta L \sin \phi$, must exceed the damping force, F_D, which includes the structural damping and the aerodynamic damping that exist without consideration of the effects of shocks. In this case it can be shown that on the upper surface a forward moving shock with increasing (negative) angle of attack would be destabilizing. Also for aft swept airfoils for angle-of-attack ranges where the shocks move in the opposite direction to the above with increasing angle of attack (positive or negative), the effect would be stabilizing. For forward-swept airfoils it can be shown that the criteria for stable or unstable shock motions with angle of attack are reversed as compared with aft-swept airfoils.

Figure 5 shows an oscillation obtained on the B-1 wing at limit load during a windup turn. The pilot allowed the amplitude to build up to ±15 inches at the wing tip before he reduced the angle of attack. Figure 6 shows a small and short-duration oscillation that occurred in the second symmetrical bending mode when the angle of attack exceeded +5° in a pitch doublet maneuver with the wings at 25° sweep. It should be emphasized that the conditions where the HiMAT and B-1 oscillations occurred do not significantly impact flight mission requirements. Figure 7 shows B-1 wing outer panel critical Mach number conditions for several sweep angles. It can be seen that the B-1 flight oscillations correlate closely with outer panel critical Mach number conditions. Also shown are flutter model test results where the oscillations could be obtained for a sweep angle of 55° only. Lack of good correlation of model and full scale data may be due to the low (.022) ratio of model to full-scale Reynolds number. Another factor influencing shock phenomena is model surface roughness compared with full scale. For the further investigation of shock-induced oscillation phenomena, the cryogenic tunnel can not only provide high Reynolds numbers but also allow the design of dynamically scaled (machined metal) models with aerodynamic smoothness similar to that attainable on high quality aerodynamic force models.

SUPERSONIC SINGLE-DEGREE CONTROL SURFACE FLUTTER

Figure 8 presents Rockwell-developed criteria for the prevention of single-degree-of-freedom control surface rotation flutter. The criteria are based on linear unsteady aerodynamic theories correlated with only a small amount of experimental data. The impact of these stringent criteria on weight, actuator space requirements, design complexity, and cost can be significant, and Reynolds number may be an important variable in determining flutter boundaries in specific cases. In view of the above, model tests in the cryogenic tunnel at flight Reynolds numbers would be useful to investigate the effects of parameters such as control surface rotational frequency, Mach number, Reynolds number, airfoil shape, control surface airfoil shape, damping, angle of attack, control surface deflection, airfoil sweep, etc.

TRANSONIC FLUTTER SPEED DIP AS A FUNCTION OF R

Investigating the effect of Reynolds number on the magnitude of the important characteristic transonic flutter speed dip (fig. 9) is another area in which the cryogenic tunnels could provide useful data.

HONEYCOMB VS. SCREENS TO SMOOTH TUNNEL FLOW

At Rockwell the relative merits of screens and honeycomb in the settling chamber to smooth tunnel test section flow have been evaluated. The investigation started in our 8- × 11-foot low speed wind tunnel in 1958 after initial tests of an XB-70 flutter model displayed excessive dynamic response to tunnel turbulence on its low frequency "free-flight" suspension system. Model response amplitude in the vertical direction was as large as ± 0.5 feet. The XB-70 model was evidently a sensitive indicator of tunnel turbulence due to its relatively large wing area.

Measurements were then made in the test section flow with an available small weather vane type of flow angle transducer. These measurements indicated the presence in both vertical and lateral directions of what was referred to as "dynamic flow angularity" at several frequencies starting at the low end of the spectrum at approximately one Hz.

To investigate the effectiveness of honeycomb in reducing the dynamic flow angularity, several test specimens measuring 3 × 3 feet and containing honeycomb of various cell sizes and cell width-to-length ratios were fabricated. These were placed on the tunnel floor in the settling chamber normal to the flow direction and the tunnel was run up to 200 mph giving a settling chamber velocity of 25 mph. The vane transducer was placed out in the settling chamber stream and then behind each honeycomb section. It was found that the honeycomb was very effective in reducing dynamic flow angularity and that the honeycomb design parameters were not critical. All honeycomb specimens reduced the dynamic flow angularity by approximately an order of magnitude.

The first fix installed in the tunnel was a screen at the downstream end of the settling chamber. This screen had a 50% porosity. While the screen no doubt improved the test section streamwise velocity distribution and reduced high-frequency turbulence, vane measurements showed that it had a negligible effect in reducing low-frequency, large-scale dynamic flow angularity. Honeycomb was then installed in the settling chamber upstream of the screen. The honeycomb had a 6-inch-square cell, a 36-inch streamwise length and spanned the 24-foot height and 24-foot width of the settling chamber. Test section measurements of dynamic flow angularity indicated that the honeycomb reduced the dynamic flow angularity by a factor of from 5 to 10 depending on the frequency. Also, fan swirl was completely eliminated and there was no appreciable effect on power required. Of even more importance, model "bounce" on the free-flight suspension system was virtually eliminated. The flow was so smooth that the approach to flutter could be more readily detected by viewing a highly amplified strip chart strain gage signal than by direct observation of the model.

When the same dynamic flow angularity problem showed up in the Rockwell 7- × 7-foot Trisonic Blowdown Tunnel in attempts to test "free-flight" B-70 flutter models, our experience in the low speed tunnel pointed toward the solution. The Trisonic tunnel had 6 settling chamber screens but no honeycomb in the original design. The honeycomb installed at that time in the settling chamber had a 6-inch-square cell, 48-inch streamwise length and 24-foot diameter. The vane pickup indicated the same large reduction factor for dynamic flow angularity as for the low speed tunnel. The flow was made so devoid of low-frequency, large-scale turbulence that complete aircraft flutter models on low-travel internal sting-mounted free-flight suspensions have been tested through entire test programs without ever illuminating the stop limit lights. Evaluations of honeycomb in the 0.3-meter tunnel may be appropriate to provide data for a decision to install honeycomb permanently in both tunnels.

RAPID ACTING TUNNEL q REDUCER TO PREVENT FLUTTER MODEL DESTRUCTION

A rapid means of reducing the test section dynamic pressure is a necessity for wind tunnel flutter model testing to eliminate or minimize the flutter destruction of the models. Figure 10 illustrates the performance attained in a q reducer developed for the Rockwell 8- × 11-foot low speed tunnel. The graph shows an initial q reduction of 15% in only 0.2 seconds and a continued reduction thereafter. This system has proved to be so effective as to permit the attainment of over 100 high frequency flutter points on only one model for various configurations without loss of the model. The system consists of floor and ceiling mounted tabs just downstream of the test section. The tabs are 11 feet long with an 8-inch chord and run the full width of the test section. They are hinged at their downstream edge and are normally parallel to the flow. Leading edge latches are automatically released at predetermined values of flutter stresses in the model or manually released by a switch. Tunnel flow rotates the tabs normal to the flow where stops prevent further rotation. At the same time power to the tunnel fan motor is cut off.

The Langley 0.3-meter tunnel appears to be the appropriate place to develop an effective q reducer for application to both tunnels.

- 2.5 - METER TUNNEL
  - SHOCK-INDUCED OSCILLATION
  - SUPERSONIC SINGLE-DEGREE CONTROL SURFACE FLUTTER
  - TRANSONIC FLUTTER q DIP AS FUNCTION OF R

- 0.3 - METER TUNNEL
  - HONEYCOMB VS SCREENS IN SETTLING CHAMBER TO SMOOTH FLOW
  - RAPID ACTING TUNNEL q REDUCER TO PREVENT FLUTTER MODEL DESTRUCTION

Figure 1.- Proposed NTF aeroelasticity and flutter tests.

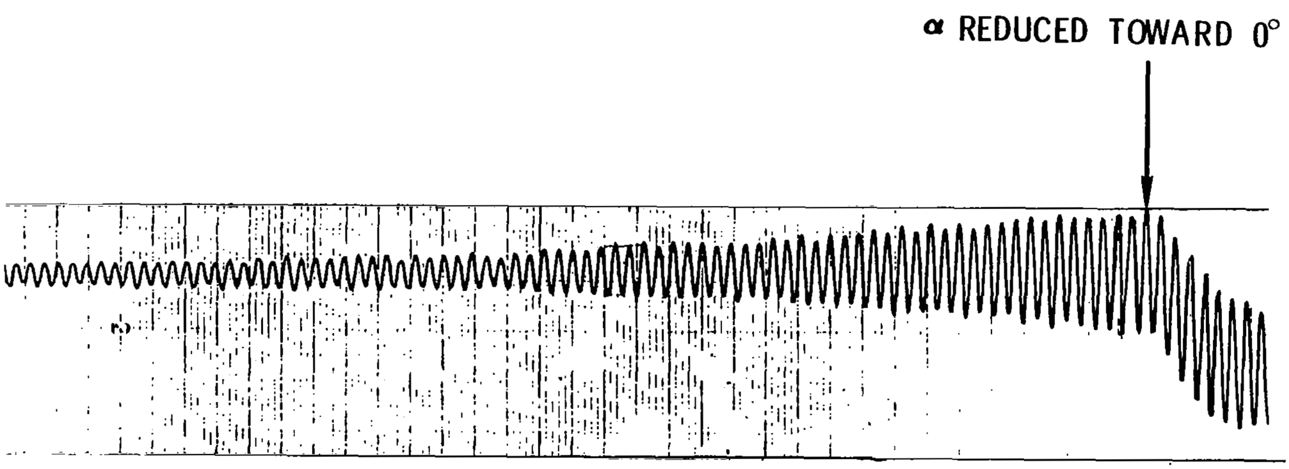


Figure 2.- Canard oscillation.
HiMAT aeroelastic model — no tip mass. Model scale = 0.22.
1st symmetrical canard bending mode, f = 56.0 Hz; M = 0.95; q = 1070 psf; $\alpha_{sting} = -3.8^{o}$; unstable damping = .009 g.

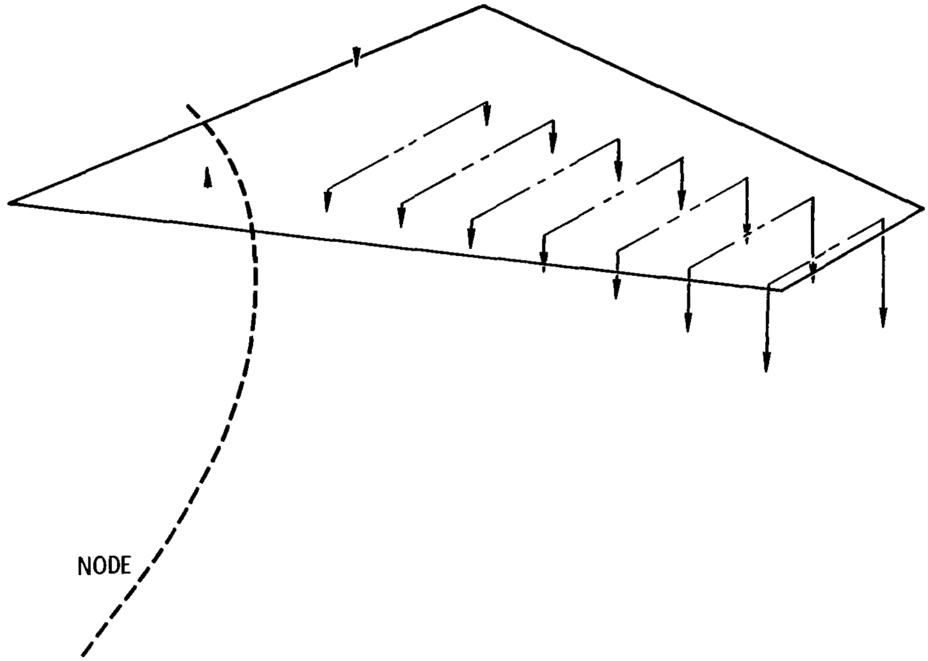


Figure 3.- Measured first bending mode.
HiMAT canard aeroelastic model — no tip mass. Frequency = 41.0 Hz.

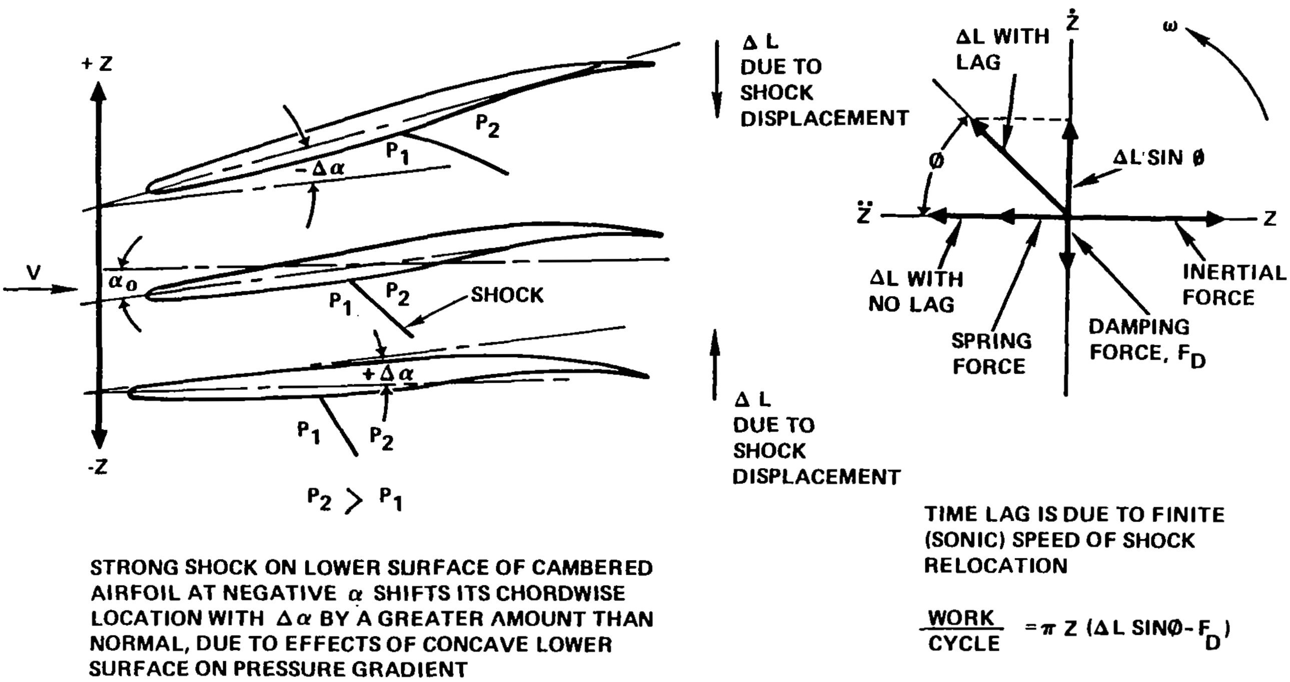
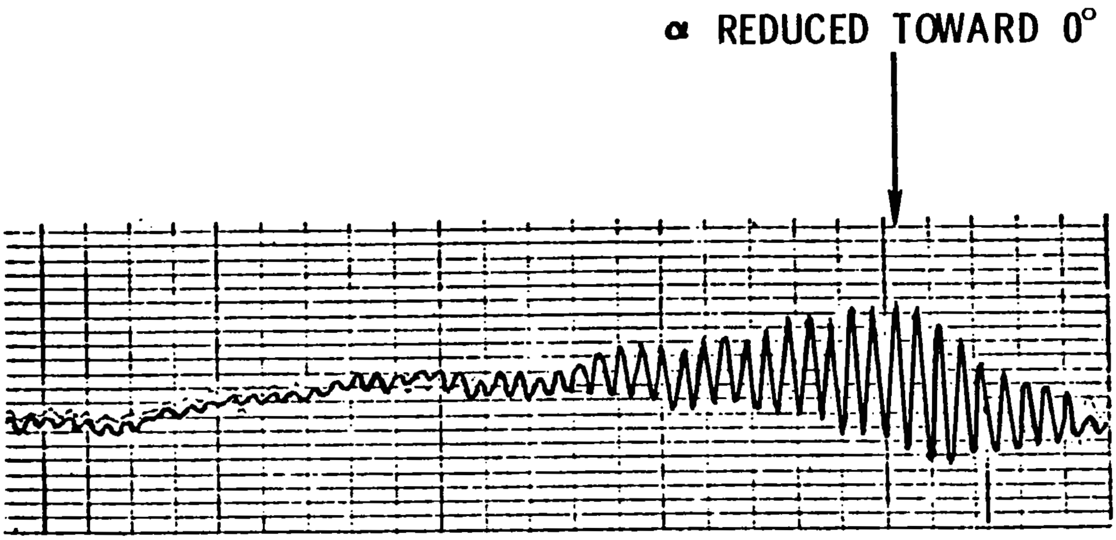


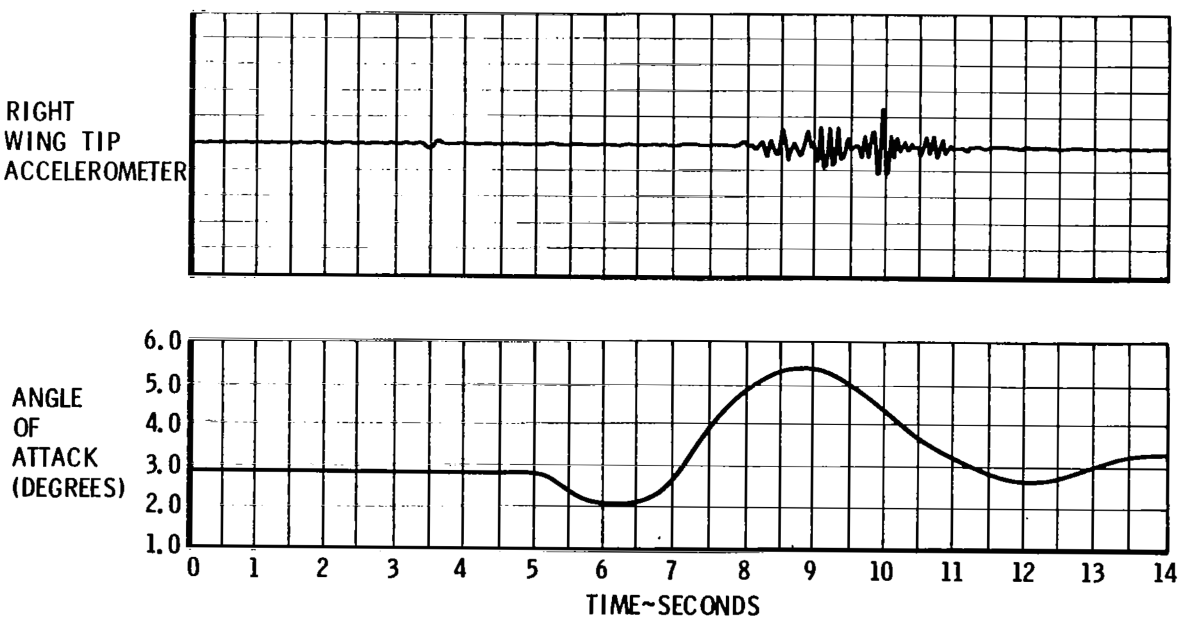
Figure 4.- Theoretical mechanism for shock-induced self-excited bending oscillation of a sweptback airfoil.



MAX WING TIP DEFLECTION = ±15 INCHES

UNSTABLE DAMPING = .034 g

Figure 5.- B-1 wing oscillation.
1st symmetrical bending mode. $\Lambda = 67^{o}$; M = 0.94;
altitude = 32 000 ft; $\alpha = +10^{o}$; f = 2.1 Hz.



MAX WING TIP DEFLECTION = ±0.5 INCHES

Figure 6.- B-1 wing oscillation.
2nd symmetrical bending mode. $\Lambda = 25^{o}$; M = 0.73;
altitude = 33 500 ft; f = 7.6 Hz.

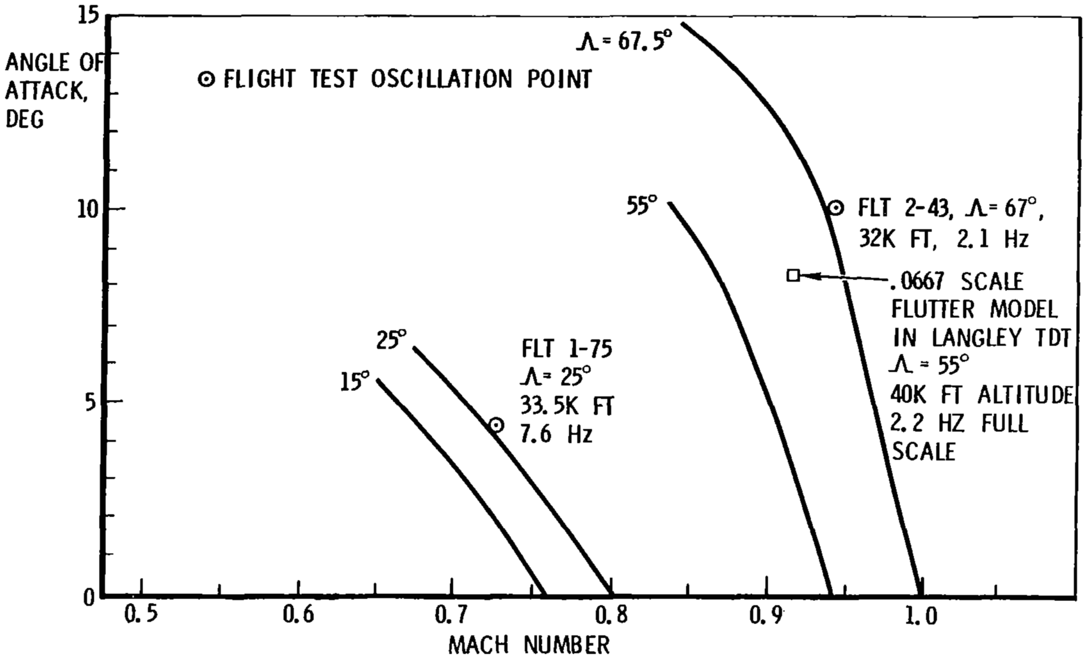


Figure 7.- B-1 wing outer panel critical Mach number and test oscillation conditions.

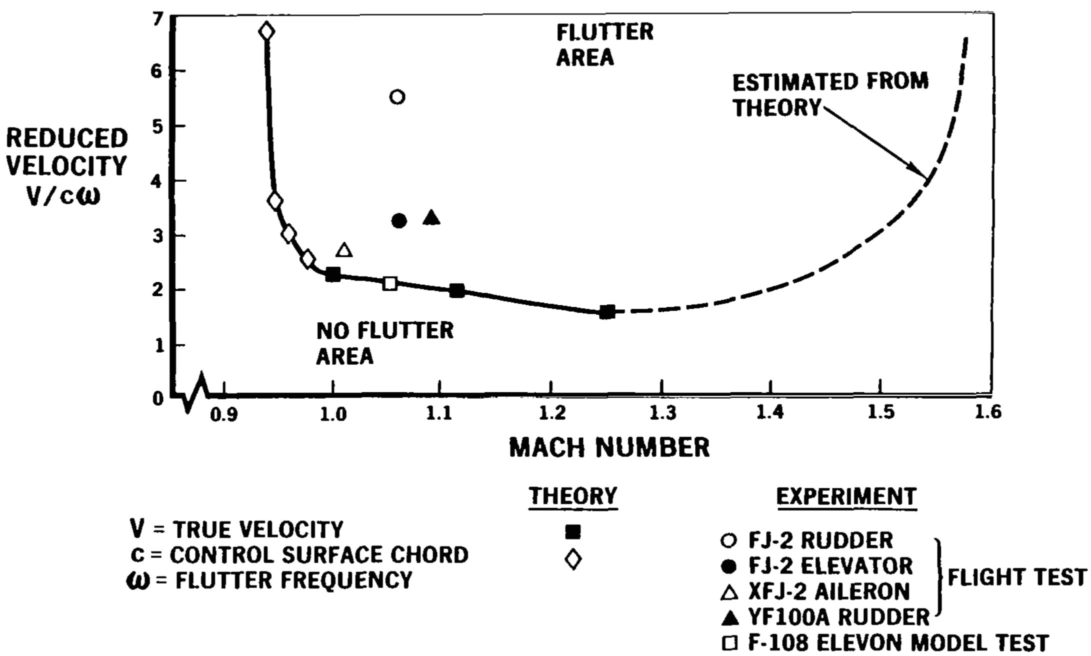


Figure 8.- Single-degree-of-freedom control surface rotation flutter criteria.

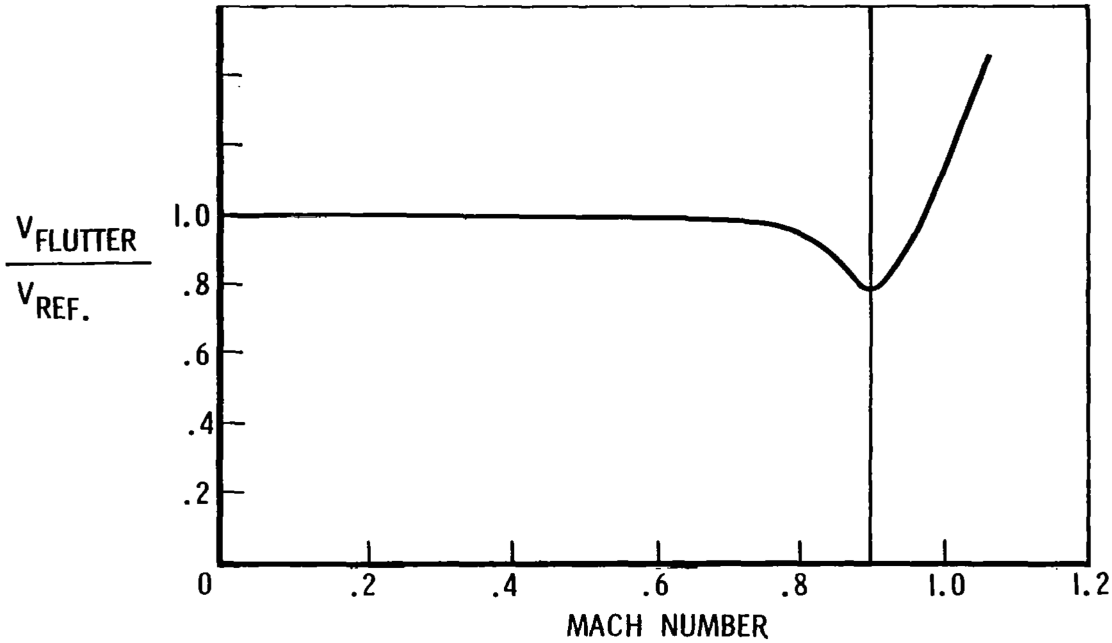


Figure 9.- Determine transonic flutter speed dip as a function of R for representative configurations. The magnitude of the important characteristic transonic dip in flutter speed due to Mach effects may be significantly affected by Reynolds number.

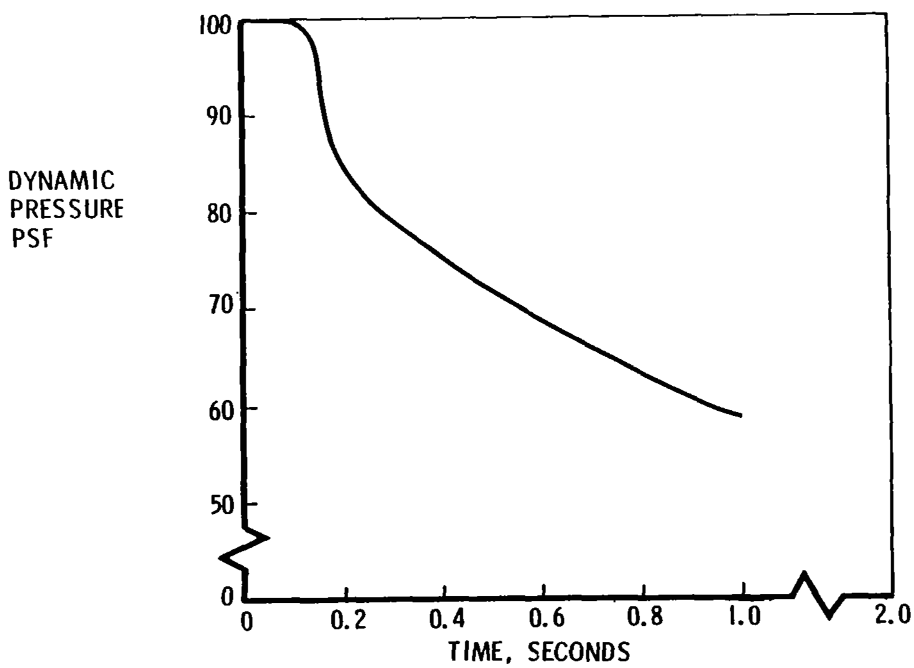


Figure 10.- Rockwell NAAL 8- × 11-foot low-speed tunnel q reducer performance.

PRELIMINARY USER PLANNING FOR THE

NASA NATIONAL TRANSONIC FACILITY

J. D. Cadwell
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Aerodynamics Subdivision
Douglas Aircraft Co., McDonnell Douglas Corporation

The completion of the construction phase of the NTF in late 1981 with a mid 1982 target to start the shakedown and calibration of the cryogenic wind tunnel indicates that the reality of a facility capable of simulating full scale Reynolds numbers will soon be fulfilled.

MDC UTILIZATION OF NTF

Figure 1 indicates how MDC will utilize the NTF during the initial years of operation. The first order of priority is to build our confidence level in the test results obtained in the cryogenic tunnel. This could best be accomplished in the NTF by testing a known configuration through the broad spectrum of Reynolds numbers - from those comparable to what can be obtained in current facilities to those achieved in flight. The correlation type tests will not only build confidence in the NTF but can also be used to assist the aerodynamicist in extrapolating data from current facilities, where the majority of his test data will continue to be generated, to the full scale or flight Reynolds number. At this time specific research and development testing scoped for NTF has not been identified by the Douglas Aircraft Company. The R&D programs where possible will be directed to the 0.3-meter facility rather than the NTF in order to minimize cost. For those cases requiring a larger facility it would be hoped that a cooperative program with NASA could be arranged. Once an acceptable level of confidence in data generated in the NTF has been reached, the cryogenic tunnel would be used early in the development of a new configuration where questions regarding Reynolds number effects were considered critical. Figure 2 shows a case where increased insight into the Reynolds number effects would directly affect the evaluation of a configuration. Although the pitch-up shown is delayed as the Reynolds number is increased the designer would normally expend considerable effort to assure a configuration that would have acceptable longitudinal characteristics at the stall. An early evaluation of a configuration at full scale Reynolds number would therefore be a key use of the NTF. A complete evaluation of the aerodynamic characteristics of the airplane to be used for final configuration refinement would also be a key milestone in a new aircraft program.

AREAS OF INVESTIGATION

The aerodynamic areas of investigation where Reynolds number effects are known to be significant are shown in figure 3 for both high speed (Mach numbers above 0.5) and low speed (those Mach numbers in the 0.2 range). The propulsion system and airframe integration investigations will require significant test technique development in order to provide the power simulation necessary to optimize the nacelle-pylon-wing configuration at high speed. The dominant challenge to obtaining representative data in low speed lies in the manufacture of the model including the high lift devices such as flaps, slats, etc.

TYPES OF DATA DESIRED

The cryogenic tunnel does not introduce new requirements for the types of data required at high Reynolds numbers, but it does present a formidable challenge to obtain the types of data shown in figure 4 under very cold temperature conditions. The primary requirement, force and moment data, appears to be well in hand with the apparent success that Langley is having in making a cryogenically compatible six component force and moment balance. Surface pressures, also an important data requirement, should not be a particular problem as long as the electronic pressure sensor modules can be maintained in a controlled temperature environment.

The wide variation in Reynolds number increases the need for rapid and reliable detection of the location of boundary layer transition. Flow visualization is considered to be an essential diagnostic tool to investigate flow conditions where a qualitative understanding of aerodynamic phenomena is required.

As indicated earlier, power effects simulation is a definite requirement and will require a major effort to enable operation of turbine-powered simulators at cryogenic temperatures. The root bending and unsteady measurements are desired data for the determination of buffet onset. Hinge moments from aerodynamic control surfaces are desired since they are generally sensitive to Reynolds number changes. Hopefully, the technology improvements that are being demonstrated in the cryogenically compatible internal strain gage balances can also be applied to hinge moments measurement devices.

The wing deformation measurements, although relatively low on our priority list of data desired, continue to be important in analyzing data obtained on high speed wind tunnel models. It would appear that work underway at Langley directed at making model deformation measurements in the NTF is well advanced and a workable system will be available when the facility becomes operational.

In conclusion the McDonnell Douglas Corporation is looking forward to the completion, calibration, and verification of the NTF as a full scale Reynolds number development tool. As the NTF operational experience increases and the effects of Reynolds number are demonstrated on known configurations, the requirements for increased utilization of the cryogenic wind tunnel for on-going programs will increase.

IT IS ENVISIONED THAT MDC WILL UTILIZE THE NATIONAL TRANSONIC FACILITY IN THE FOLLOWING FASHION:

- CORRELATION TEST - INITIAL ENTRY TO PROVIDE CORRELATION WITH TEST RESULTS ON A SPECIFIC CONFIGURATION TESTED IN OTHER FACILITIES AND FLIGHT.

- R&D TESTING - PROBABLY ACCOMPLISHED ONLY ON A CO-OP BASIS WITH NASA DUE TO HIGH COST.

- CONFIGURATION DEVELOPMENT - TYPICALLY WOULD BE FOR FINAL CONFIGURATION REFINEMENT WHEREIN REYNOLDS NUMBER EFFECTS ARE IMPORTANT. COULD POSSIBLY INCLUDE PRELIMINARY CONFIGURATION TESTING WHERE REYNOLDS NUMBER EFFECTS COULD BE A MAJOR CONCERN.

Figure 1.- MDC utilization of NTF.

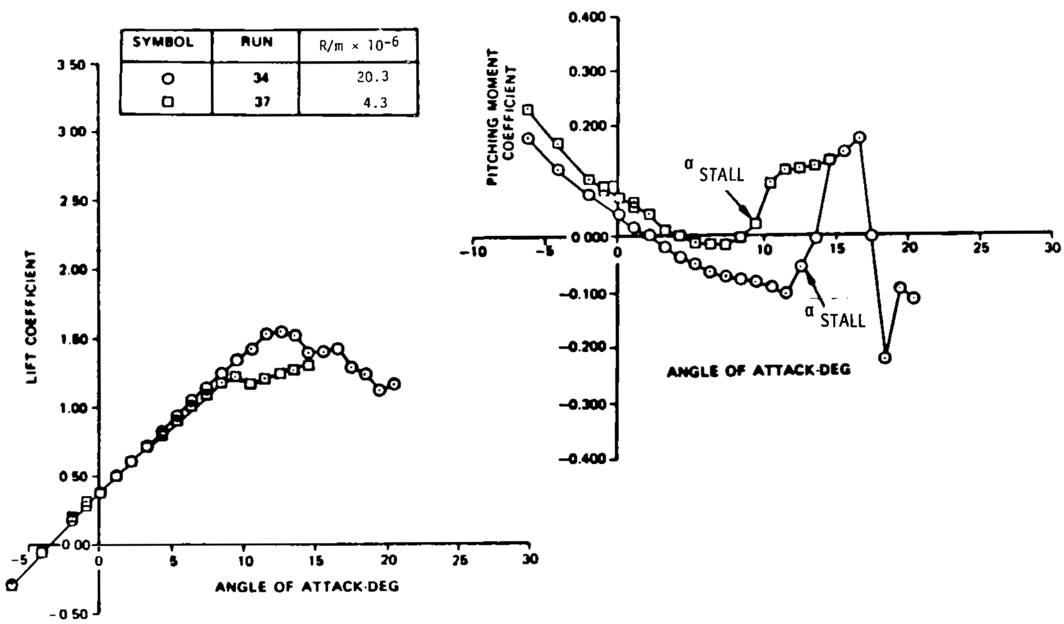

Figure 2.- Effect of Reynolds number on the low-speed high-lift characteristics on a high-aspect-ratio supercritical wing. Cruise wing; Mach = 0.20; nacelle and pylon on; $i_H = 0^{\circ}$.

HIGH SPEED

- PROPULSION SYSTEM - AIRFRAME INTEGRATION
- PITCH CHARACTERISTICS
- COMPONENT INTERFERENCE EFFECTS
- BUFFET
- HIGH ANGLE OF ATTACK WING LOADS
- STABILITY AND CONTROL CHARACTERISTICS

LOW SPEED

- MAXIMUM LIFT
- NACELLE INTERFERENCE EFFECTS
- PITCH CHARACTERISTICS

Figure 3.- Aerodynamic areas of investigation.

- SIX COMPONENT FORCE AND MOMENT DATA
- SURFACE PRESSURES
- TRANSITION DETECTION
- FLOW VISUALIZATION
- REYNOLDS NUMBER SWEEPS
- MACH NUMBER SWEEPS
- POWER EFFECTS SIMULATION
- ROOT BENDING MOMENT
- DYNAMIC MEASUREMENTS FOR BUFFET
- HINGE MOMENTS
- WING DEFORMATION MEASUREMENTS

Figure 4.- Types of data desired.

FLUID DYNAMICS PANEL

Chairman	Percy J. Bobbitt NASA LaRC
Technical Secretary	William B. Igoe NASA LaRC

Panel Members

Jerry Adcock	NASA LaRC
Seymour M. Bogdonoff	Princeton University
J. Christopher Boison	Wright-Patterson AFB
Richard P. Boyden	NASA LaRC
Eugene E. Covert	Massachusetts Institute of Tech.
Michael F. Fancher	Douglas Aircraft Company
Alfred Gessow	University of Maryland
Blair B. Gloss	NASA LaRC
Robert M. Hall	NASA LaRC
Eugene G. Hill	The Boeing Company
Charles Johnson	NASA LaRC
Robert H. Korkegi	The George Washington University
John D. Lee	Ohio State University
Eli Reshotko	Case Western Reserve University
P. Calvin Stainback	NASA LaRC
John L. Whitesides	The George Washington University
Jain Ming Wu	University of Tennessee Space Inst.

REPORT OF THE PANEL ON FLUID DYNAMICS

Percy J. Bobbitt
NASA Langley Research Center

INTRODUCTION

Fluid dynamics areas that would benefit from the unique capabilities of the National Transonic Facility (NTF) were identified as a necessary part of justifying its construction. Since that time these areas have been further expanded with a detailing of the associated research needs as well as specific experiments and model requirements. From time to time additional experiments have been added and the details of many of the original candidates enhanced. Some of the enhancements resulted from the recommendations of the 1976 high Reynolds number workshop (ref. 1), and further improvements have been made by Langley researchers as part of a continuing effort. With completion of the NTF now just a year away the need is evident for the research program, particularly that part to be undertaken during the first months of operation, to take on its final form. This workshop provided, then, the last opportunity for most of the participants to affect the program makeup and to insure that the selected experiments provide the highest scientific yield.

Some of the panel reports of the 1976 workshop proposed specific experiments, including details of the models and instrumentation required; in others only the problems were outlined and/or illustrated by data. These inputs along with those of the Langley Research Center staff were assessed and, as already noted, the existing research program modified accordingly. The fluid dynamics panel of four years ago chose not to propose specific experiments (some of those put forward by other panels were relevant to fluids as well as to their own area). Consequently the panel representing fluid dynamics in the present workshop was charged with the responsibility of formulating experiment descriptions for the topics considered by the previous panel as well as any new ones deemed to be appropriate. There was no requirement that the experiments be carried out in the NTF - experiments for the 0.3-meter Transonic Cryogenic Tunnel (0.3-m TCT) which would shed light on high Reynolds number flow were encouraged. The end result was a mixture of NTF and 0.3-m TCT experiments. In the subsequent discussion some experiments are described in relative detail while others are only given a name because time did not permit the panel to explore all possibilities equally.

The panel also gave considerable attention to theory validation, test techniques, and the importance of tunnel flow quality and calibration of the NTF. These items will be discussed in this report but the depth of detail provided on any given item is not necessarily proportional to the importance attached to it by the panel. A discussion of the technology advances since the previous workshop, presented prior to the panel's deliberations, is also given. This provides a better overall view of what the problems are and the progress being made in solving them.

SYMBOLS

c	chord of airfoil
$\bar{c}$	mean chord of wing
C_B	bending moment coefficient $(= \frac{\text{bending moment}}{q_\infty S_{ref} \bar{c}})$
C_d	section drag coefficient $(= \frac{\text{drag force}}{q_\infty c})$
C_n	section normal-force coefficient $(= \frac{\text{normal force}}{q_\infty c})$
f	frequency
k_a	admissible roughness height
L_p	sound pressure level
M_∞	free stream Mach number
p'	fluctuating pressure
p	pressure
q_∞	free-stream dynamic pressure
R	Reynolds number based on a unit length $(= \frac{\rho_\infty V_\infty}{\mu_\infty})$
$R_{\bar{c}}$	Reynolds number based on mean aerodynamic chord $(= \frac{\rho_\infty V_\infty \bar{c}}{\mu_\infty})$
R_c	Reynolds number based on airfoil chord $(= \frac{\rho_\infty V_\infty c}{\mu_\infty})$
S_{ref}	exposed wing area of wall-mounted model
T	temperature
u'	fluctuating longitudinal velocity
v'	fluctuating lateral velocity
V_∞	free-stream velocity
α	angle of attack
μ_∞	free-stream viscosity

ρ_∞	free-stream density

Subscripts:

t	total conditions
L	local
w	wall
aw	adiabatic wall

RECENT ADVANCES

Before describing the results of the panel's deliberation it is important to note that since the 1976 high Reynolds number workshop a number of advances have been made in the understanding of high Reynolds number flows. They have come from a number of investigations, both here and abroad, and in theory and experiment. A wide variety of Langley in-house and sponsored research programs are aimed at this area and many significant results have been obtained over the past few years. Some typical examples were presented to the workshop to insure that everyone had the same frame of reference and to solicit constructive comment on future emphasis.

Transition Strips

One research area that deserves attention in high Reynolds number facilities is how best to use transition strips at low Reynolds number to simulate high Reynolds number flows. A recent test in the 0.3-m TCT of an advanced supercritical airfoil provides some indication of the type of results one might expect.

Figure 1 shows both fixed and free transition data for drag coefficient C_d as a function of chord Reynolds number for several values of the normal force coefficient. Visual comparison of the two sets of data clearly shows a large difference at a Reynolds number of approximately 4×10^6, while at Reynolds numbers approaching 10^7 the results are about the same. For still higher Reynolds numbers it appears that the transition strip itself is starting to contribute to the drag. Finally there is a large difference between the low and high Reynolds number C_d's for all three C_n's.

Additional tests similar to those just described are planned; they too will be carried out in the 0.3-m TCT. The features of one such test are given in figure 2. This test differs from that which yielded the results shown in figure 1 in that diagnostic measurements will be made as well as those necessary to obtain lift and drag. Looking further into the future one can expect that the Pathfinder models will be used in the NTF to further the understanding of trip strips for wings and bodies.

Surface Roughness

Since drag estimates of full-scale aircraft are made by adding the aircraft manufacturing roughness drag to the wind-tunnel model drag measured on a smooth wind-tunnel model, skin friction penalties associated with the wind-tunnel model surface roughness are undesirable. As the Reynolds number at which a model is being tested increases, the model boundary layer becomes thinner and the admissible surface roughness height (the maximum roughness height which results in no skin friction penalty) decreases, as illustrated in figure 3 (see ref. 2). In addition, increased skin friction can result in early boundary-layer separation or erroneous shock location; either of these conditions can potentially produce large errors in lift, drag, and pitching moment. The data in figure 3 show the variation of admissible roughness height, k_a, in a zero-pressure-gradient turbulent boundary layer with Reynolds number $R_{\bar{c}}$, where the mean chord, $\bar{c}$, is taken as 0.20 m (0.65 ft). This mean chord is representative of a transport model sized for the NTF. Shown on figure 3 for reference are the maximum NTF Reynolds number, the Boeing 747 cruise Reynolds number, and the maximum Reynolds number for current tunnels. At a given Reynolds number, any roughness height falling below the admissible roughness curve in figure 3 will produce no skin friction penalty. The shaded band on figure 3 is the range of typically specified and routinely achievable surface finishes for current transonic models. Since the NTF Reynolds number range, based on a chord of 0.20 m (0.65 ft), is approximately 0.5×10^6 to 95×10^6, the current specified model surface finishes appear to be compatible with a significant part of the NTF Reynolds number range; the surface finishes required for high Reynolds number testing in the NTF are achievable, but of course at extra cost. However, as is noted in figure 3, the admissible roughness curve is for a surface with uniformily distributed three-dimensional particles affixed to it. An experimental program is planned to determine the equivalent distributed particle roughness for typical NTF model surfaces. In order to carry out this experimental program, a good definition of the topography of a typical NTF model surface is needed.

The instrumentation which is almost universally used to measure model surface roughness is the stylus profilometer type equipment. However, there are at least two potential problems associated with the stylus profilometer. Figure 4 depicts these two problem areas: (1) roughness slope is too steep, and (2) roughness frequency is too high. It should be noted that the stylus radius is typically 2.5 μm (~ 100 μin). Since there are no published data which verify that the stylus profilometer accurately determines surface topography data on surfaces typical of NTF models, the National Bureau of Standards (NBS) is in the process of comparing the topography of a surface typical of NTF models as measured by a stylus profilometer and a stereo scanning electron microscope. Figure 5 shows an area mapped by the stylus equipment. (The vertical dimensions in this map are not the same scale as the inplane dimensions; that is, the roughness is exaggerated). The four rather large lines of roughness shown in this figure are scratches put in this stainless steel surface by an inexpensive shop type profilometer, thus exemplifying that high quality stylus profilometers are required to obtain the best possible stylus data from a specimen without damaging the specimen. In addition, the stylus profilometer has great difficulty measuring surface finishes on curved surfaces similar to the leading edge region of wings. The leading edge region of the wing is the area where the boundary layer is the thinnest, and therefore local skin friction is the most sensitive to surface roughness. Thus, for example, it is highly desirable to have the capability of measuring surface finish over the wing's leading edge. Towards this end, the NBS has designed a light-scattering system to measure surface finish accurately on surfaces with high curvature or flat surfaces; this system is currently being fabricated for laboratory tests.

High Reynolds Number Testing

A number of tests have been conducted over the past few years in the 0.3-m TCT to obtain high Reynolds number data. The data of figure 1 illustrate the results of one such test. Figures 6 and 7 show data from a similar test but in a somewhat different format. At the lower value of C_n (fig. 6) the section drag coefficients continually decrease from the lowest Reynolds number to the highest, 47.54×10^6 based on chord. Also, it can be seen that the drag generally increases with Mach number up to the drag rise Mach number which is about 0.8 for most Reynolds numbers. When the section lift coefficient is increased to 0.55, a somewhat different variation with Mach number is obtained with a dramatic drag rise occurring at $M \sim 0.50$. Change with Reynolds number is also slightly altered from that at the lower C_n and is clearly different from that evidenced in figure 1.

Most high Reynolds number tests to date in the 0.3-m TCT have yielded some surprises. This points up the need for a much higher concentration on diagnostic measurements and flow visualization to help understand these surprises and provide better flow (turbulence) models for the thcoretician.

Real-Gas Effects

Prior to the first high Reynolds number research workshop (ref. 1) the real-gas work at Langley had concentrated on the adequacy of cryogenic nitrogen to simulate inviscid type flows (ref. 3) with the general conclusion that for anticipated cryogenic tunnel envelopes such flows are insignificantly different from those for an ideal diatomic gas (i.e. flight case). Since that time real-gas analyses have been completed in a variety of areas. Probably the most important of these is the demonstration of the simulation adequacy of cryogenic nitrogen for viscous flows (refs. 4 and 5). Results indicated that important boundary layer parameters were not significantly different from those for an ideal diatomic gas with a Sutherland viscosity description. One real-gas effect that did appear to be measurable was a difference in level of adiabatic wall temperature. While this was not considered to be significant, a subsequent experiment (ref. 6) verified the theoretical real-gas adiabatic wall temperatures and produced a high level of confidence in this viscous work. Another area of analysis dealt with a tunnel operational consideration, that being the fan power required to compress cryogenic nitrogen gas (ref. 7) for equal stagnation conditions; it was determined to be less than that required for an ideal diatomic gas. The last area of real-gas work was an analysis of the mathematical description (equations of state) for cryogenic nitrogen gas. Even after attempts to improve this description (ref. 8) and after comparing thermodynamic properties and flow solutions (ref. 9) the simple Beattie-Bridgeman equation of state appears to be adequate and as good as the more complex descriptions. It is felt at this time that the real gas effects of nitrogen at cryogenic temperatures are well understood and have little or no effect on the ability of this gas to simulate ambient air flows.

Condensation

Work concerning the onset of condensation effects has progressed both experimentally and analytically since the 1976 workshop. Experiments involving airfoils and total pressure probes have been performed, in addition to refining of data evaluation procedures for both the newer data and the original NACA 0012-64 airfoil data. As an example of condensation onset for an airfoil with a moderate maximum local Mach number of $M_{L,max} = 1.2$, the circles in figure 8 represent the total pressures and temperatures at which condensation effects were detected by pressure orifices during an experiment with the NACA 0012-64 airfoil (ref. 10). Effects occurred at total temperatures not only below those which produced supercooled flow locally over the airfoil (line labeled SATURATION, $M_{L,\ max} = 1.2$), but below those total temperatures corresponding to supercooled flow in the test-section itself (line labeled SATURATION, $M_\infty = 0.85$). Also shown is a theoretical prediction by Sivier (ref. 11) for the beginning of self-nucleation of the nitrogen test gas. The lack of agreement between theory and data suggests that the onset of effects is not due to self-nucleation of the test gas. Instead, the probable cause of effects is the presence of seed particles on which condensation can take place at temperatures higher than those corresponding to self-nucleation.

As an example of condensation onset for an airfoil test with a higher maximum local Mach number, $M_{L,max} = 1.7$, the circles in figure 9 represent the total pressures and temperatures at which condensation effects were detected locally over the airfoil by visually looking into the test section of the 0.3-m TCT during a test with a British NPL-9510 airfoil (ref. 12). The circles correlate with the theoretical prediction of Sivier for self-nucleation of the gas and are above total temperatures corresponding to saturation in the test-section flow. To summarize the experiments, the onset of condensation effects is model dependent and can occur at temperatures above free-stream saturation. On the analytical front, the question of sound speed in two-phase mixtures has been addressed, as well as various aspects of saturated equilibrium flow (refs. 13, 14, and 15). A review of available droplet growth equations has been conducted (ref. 16) and is presently being extended. In addition, computer programming of the effects of both self-nucleation and of seed particles on the flow is in progress.

Future work in the area of condensation includes studies of self-nucleation of the nitrogen test gas as well as work involving condensation on seed particles. Langley Research Center is cooperating informally with DFVLR-Göttingen to verify empirical constants necessary for the self-nucleation theory. DFVLR has just completed a one-dimensional nozzle apparatus that is well-suited for the nucleation study. Their results can be compared to the data being generated in the two-dimensional 0.3-m TCT. There are two important reasons for this comparison. First, the environment in the 0.3-m TCT, while more representative of the NTF than that of the nozzle, makes it difficult to get the empirical constants necessary for self-nucleation theory because of the presence of seed particles. Second, it should be possible to verify the belief that there should be little difference between one-, two-, or three-dimensional flows as long as the Mach number distributions are similar. Concerning studies involving the seed particle, an instrument is being procured which will allow sizing and counting of the seed particles in the 0.3-m TCT. This information is vital for proper mathematical modeling of the condensation process.

Buffet

Two semispan buffet wing models have been tested in the 0.3-m TCT using strain gages to measure the unsteady wing root bending moment. One model is a slender, sharp-leading-edge delta wing with 65° leading edge sweep known to be relatively insensitive to Reynolds number. This configuration was chosen to provide a baseline model to demonstrate the test technique over the temperature range in a cryogenic wind tunnel. Results were obtained for the delta-wing model as shown in figure 10 at the same free-stream velocity, which gave almost the equivalent reduced frequency parameter $\frac{f\bar{c}}{V_\infty}$ and the same dynamic pressure by adjusting the Mach number and the stagnation pressure. At these low Mach numbers any Mach number effect should be small. Good agreement for the nondimensionalized unsteady wing root bending moment C_B was obtained over the range of angle of attack using this procedure. This good agreement for the wing root bending moment is considered to be verification that the root bending moment strain-gage technique for buffet measurements works satisfactorily at cryogenic temperatures.

Another example of some recent buffet measurements from the 0.3-m TCT is shown in figure 11. This wing is a zero sweep wing of aspect ratio 1.5 with an RAE(NPL) 9510 airfoil section which was expected to be very sensitive to differences in Reynolds number. Figure 11 illustrates a large shift in the buffet onset angle of attack from about 8° to 14° as a result of a variation in the Reynolds number.

Plans are being made to make buffeting measurements on a carbon-fiber-reinforced epoxy wing to extend the range of the reduced frequency parameter. Also it is planned to instrument the wing of the NTF Pathfinder I model with a wing root bending moment gage for buffet tests at high Reynolds number on a transport-type configuration.

Flow Quality

Activities aimed at measuring and improving the flow quality in the Langley 8-Foot Transonic Pressure Tunnel to enable it to carry out valid experiments on airfoils designed to have significant runs of laminar flow have been under way for several years. Test programs using a variety of screens with a honeycomb and a test section choke have been expedited. Data obtained in experiments in the 8' TPT to investigate the effect of a choke located at the rear of the test section on disturbance levels in the test section are plotted in figure 12. They show measured fluctuating pressure levels normalized to the mean value as a function of free-stream Mach number for both the Mach 0.8 choke and for the "natural" choke (test section slots were closed). There is obviously a large decrease in level, about an order of magnitude, which would indicate that at high subsonic speeds the noise from the diffuser and drive system is a significant source of test section noise.

A sample of some of the results from tests on honeycomb and screen combinations is given in figure 13. The lateral component of vorticity v' is seen to decrease with increasing number of screens up to the maximum number tested (five). On the other hand, the longitudinal component u' bottoms out beyond three screens. This is thought to be due to an organ pipe effect of the pilot facility used for these tests. It is expected, on the basis of these results, that the four screens used in the NTF will reduce the turbulence level by approximately a factor of three. Further flow quality improvements can be expected from the screening effect of the cooling coil. If a choke similar to that tested in the 8' TPT and slot covers are employed in the NTF then outstanding flow quality should be obtained throughout its speed range. Even without the choke the NTF's flow quality should be superior to any large production transonic tunnel in use today.

Flow Visualization

There are a number of flow visualization techniques available from which the researcher can choose, depending on his needs and the type of model to be tested. Unfortunately most of these techniques deteriorate in utility and may even become totally worthless at cryogenic temperatures. Research is under way both at Langley and under grant to find other means of aiding flow visualization. One approach which has found some success is that attempted at the University of Southampton (ref. 17). Liquid propane with pigmentation added was arranged to issue from 0.08-cm (1/32-in) holes upstream of the area of interest. In figure 14 the movement of the flow around a solid obstruction is defined by the use of this technique, including the formation of the vortices aft of the model. This particular experiment was carried out at a free-stream temperature of 145K. It is thought that the "propane" technique could be used to ten atmospheres and 85K, which is beyond the maximum operating conditions of the NTF. Several improvements to this approach are under investigation.

Perhaps the most common technique for flow visualization is that of tufts bonded to the model surface. The use of tufts made of wool and nylon at cryogenic temperatures has been demonstrated at the University of Southampton. They were attached using an epoxy adhesive in a region of vortical flow on a delta wing. They were exercised for about 45 minutes at a Mach number of 0.3 and were found to have suffered no degradation, i.e. they did as well in the cold environment as one would have expected at normal temperatures.

Diagnostic Measurements

Both laser and hot wire measurements have been carried out in the 0.3-m TCT to determine if ambient-temperature techniques can still be used. In the case of the former the main interest was on the scattering ability of liquid nitrogen droplets which normally exist in the tunnel. The test, which is depicted in figure 15, determined that LN_2 droplets were larger than those normally found in kerosene-smoke-seeded flows and yielded a much stronger signal than required. The method of LN_2 injection has been improved since the laser test was carried out (see ref. 18) so that the average droplet size can be maintained at a smaller value. In any case it would appear that the natural seeding afforded by LN_2 droplets under very cold conditions will be useful for laser flow measurements in the NTF.

Several hot wire measurements have been carried out in the 0.3-m TCT with variable success. Some difficulty was encountered with wire resistance and breakage as well as with large LN_2 droplets. Solutions to these problems are discussed in reference 18. Generally it would seem that hot wires can be used successfully in a cryogenic environment with the proper precautions.

PANEL RECOMMENDATIONS

The Fluid Dynamics Panel made recommendations regarding specific experiments, most of which relate to basic fluid flows, although some are more configuration oriented. Many of the experiments are already planned and the panel simply endorsed their expedition. Where the panel had a particular concern regarding the way an experiment was to be conducted this is documented to insure that it will not be overlooked. Test techniques and tunnel calibration requirements were also discussed by the panel and recommendations made for improvements or emphasis. As noted earlier, experiment suggestions were encouraged for the 0.3-m TCT as well as the NTF.

Theory Validation

Over the past four years hundreds of papers have been published which present solutions to all manner of viscous flow problems using various forms of the Navier-Stokes equations, especially the boundary layer approximations. There have also been a number of experimental programs reported on which were aimed at validating some aspect of theory. At this workshop several illustrations were given of theory developments requiring high Reynolds number validation. These related to shock/boundary-layer interactions and massive separation. Any number of examples could have been cited of both the two- and three-dimensional variety to make the point that one of the primary responsibilities of the new high Reynolds number facilities is to provide "certified" data for theory validation and improvement. The panel, of course, supported this idea with the added proviso that separated flows deserve special consideration. They also observed that a close connection or alliance between the 0.3-m TCT and the NTF would benefit this type of research.

High Priority Experiments

The panel's recommendations regarding experiments were arbitrarily broken into two categories, "high" and "second" priority. The high priority experiments are not necessarily better from a scientific standpoint; the panel simply thought that these should be done before the others.

Ten-Degree Cone. - One of the most highly rated experiments for early testing in the NTF is a ten-degree cone such as that tested already in a large number of wind tunnels and in flight (ref. 19). This cone, which will be equipped with microphones and surface thin-film gages, will provide a measure of the tunnel's flow quality and test-section environment through transition Reynolds number determinations. Comparisons of these data with those previously obtained will also enable a relative assessment of the tunnel's flow quality. Tests should be conducted with and without the tunnel wall slots closed and with the addition of a diffuser choke as well. Changing transition Reynolds numbers by varying either the tunnel pressure or temperature is also proposed.

As in all careful transition measurements, the roughness of the surface must be measured. Furthermore, the statistical and spectral characteristics of the free-stream flow disturbances must be determined over the range of test conditions. This latter information will, of course, be available since flow quality measurements are an integral part of the tunnel calibration program.

Flat Plate. - This test is similar in some aspects to the previous one in that it can be used to make transition measurements and hence obtain some measure of the tunnel's flow quality. Unfortunately, there is no transonic flat-plate data base similar to that for the ten-degree cone. The flow quality of the NTF relative to that of other tunnels would consequently be difficult to judge based on this experiment alone. Nevertheless, it is important to obtain this type of data over as wide a range of unit Reynolds number and wall temperatures as the tunnel can provide. Perhaps the most important feature of this test is the diagnostic data that would be obtained at high Reynolds number in a relatively quiet environment, i.e. with slots closed and diffuser choke in place. Both time-averaged and fluctuating flow quantities should be obtained. In addition, the direct measurement of skin friction was strongly urged by the panel, perhaps using several candidate gages. Finally, the panel wanted to stress the need to obtain a truly two-dimensional flow, a zero pressure gradient, and a measure of the model motion, if any.

Airfoils. - The panel discussed at length experiments that could be conducted to determine the details of flows about airfoils at high Reynolds number, including leading-edge, shock/boundary-layer, and trailing-edge interactions and unsteady separated flows. It was finally determined that most of these phenomena could be examined using a series of standard or baseline models in the 0.3-m TCT. An example set of airfoils suggested was a 12-percent supercritical, an NACA 0012, and a fighter type. Considerable care would be exercised in fabrication to maintain close tolerances and a high-quality finish. The test articles would have well-defined geometries and an accurate characterization of their surface roughness. As in the case of the NTF, flow environment should be well defined. The types of data and calculations proposed for these airfoil experiments are as follows:

- Transition studies
- Wake surveys
- Shock and trailing-edge interactions
- Massive separation, stall, and buffet
- Drag creep and drag rise Mach number
- Lift and moment
- Pressures and shock locations
- Examine unit Reynolds number effects
- Roughness and waviness effects
- Companion theoretical flow-field and stability analyses
- Reynolds number scaling

It is recognized that many of the same kinds of measurements proposed for the airfoil studies should also be made for wings, although no specific experiment was outlined. One such experiment was proposed by the Theoretical Aerodynamics Panel at the 1976 workshop (ref. 1) and should be considered for inclusion in the test program for the Pathfinder I model.

Wall Temperature Effects. - The effect of wall temperatures on boundary layer flows was the topic of much discussion, indicative of the panel's concern that this effect should be well understood prior to production testing in the NTF. The features of several tests soon to be carried out in the 0.3-m TCT that will provide some of the needed information were described to the panel. In each of these tests the values of T_w/T_t will be varied from 0.3 to 3.0.

The first experiment consists of an airfoil model (33-cm (13-in)) chord with a chordwise strip of thin-skin thermocouples which will be used to measure heat transfer. This strip will be protected in a special housing in the plenum prior to reaching test conditions. An injection mechanism is used to insert that part of the airfoil containing the thermocouple strip into the test section (see fig. 16). From the heat transfer data the beginning and end of transition will be determined (i.e. transition Reynolds number as a function of T_w/T_t). The surface roughness can be varied in this test, and boundary-layer trips can be added, to determine their effects on the beginning and end of transition. The second model will be a pressure model in which the region of the pressure orifices (the same as with the thermocouple model) will be heated or cooled. The effect of T_w/T_t will be determined on the shock location, the drag, and possibly the separation characteristics. In addition, the pressure model will have several chordwise thermocouples to determine the transient response as the model approaches its adiabatic wall condition.

It should be noted that one of the few real-gas effects at cryogenic and high total-pressure conditions is the ratio of T_{aw}/T_t (ΔT as much as 7° C). Table I substantiates this point and supports the conclusions given in the real gas discussion earlier. It lists the important parameters for nitrogen ratioed to those of an ideal gas at $M_\infty = 0.85$, a local Reynolds number of 140×10^6, a total pressure of nine atmospheres, and a total temperature of 120 K.

TABLE I

Parameter	$\frac{\text{Nitrogen Value}}{\text{Ideal Value}}$
C_f	0.996
θ	0.996
δ^*	1.001
δ	0.999
T_{aw}/T_t	0.967

Clearly all the ratios are essentially unity except that for T_{aw}/T_t.

The panel urged that any transition study including wall temperature variations should be done concurrently with a stability analysis (over the range of R, M_∞ and T_w/T_t) and flow quality assessment of the 0.3-m TCT. It also supported the idea that time histories of wall temperatures at various locations on several three-dimensional models (such as the Pathfinder models) should be measured in addition to such quantities as lift, drag, moment, pressure, etc.

Second Priority Experiments

A number of other basic fluids experiments were discussed by the panel but they were thought to be slightly less desirable for early testing than those already described. Several of these tests have already received considerable attention and substantial documentation exists, except perhaps for the effects of wall temperatures and very high Reynolds numbers. The second priority experiments are:

- Protuberances and depressions
- Steps and gaps
- Junctures
- Tips
- Turbulence modeling experiments
- Re-laminarization experiment

The turbulence modeling experiments were not further defined except that the emphasis here would be on strong interactions. Similarly, the details of the re-laminarization experiment were not spelled out; however, it was thought to be feasible due to the stabilizing effect of the very cold wall temperatures achievable.

Second Priority Configuration Tests

It was recognized by the panel that there are a number of both old and new configuration problems that require additional testing at high Reynolds numbers for confident application to full-scale designs. There was also recognition of the fact that while configurations research was the primary concern of another panel, certain configurations were of special importance to the fluid mechanician. Among those suggested were an airfoil with a flap control, a high lift system, and a spoiler, and a variable-camber airfoil for "optimum" ascent and descent. For wings an interest was expressed in determining the Reynolds number sensitivity of a leading edge notch designed to inhibit spin. Leading edge devices to enhance maneuverability of highly swept wings was another configuration that attracted considerable interest. Cross-flows on boosters and fuselages at high angle of attack and buildings in a cross wind were two other types of tests proposed. It should be noted that experiments are planned in the NTF and 0.3-m TCT relating to high lift systems, leading edge devices, and cross flows.

Another panel recommendation related to configurations concerns tunnel-to-tunnel comparisons for one or more three-dimensional configurations. As in the case of the ten-degree cone, which was tested in a number of facilities to compare their flow quality (through transition measurements), it is of interest to know how force and moment measurements compare as well. This assumes, of course, that all the appropriate corrections for wall effects will be applied to arrive, in each case, at the best "free air" estimate. Several models that have been tested in other facilities have already been put on the NTF schedule; new models of the Shuttle orbiter and ascent configuration are also being fabricated for test.

Unit Reynolds Number Effects

It should be noted that cryogenic tunnels afford a unique opportunity to contribute to the understanding of unit Reynolds number effects. Normally when Reynolds number is increased in a state-of-the-art tunnel this is done by increasing the pressure level, which in turn requires more drive power. This increase in power will normally increase the acoustic levels throughout the tunnel circuit and may increase vorticity as well. Both of these phenomena can have an adverse effect on transition if they provide disturbances in the frequency range that produces flow instabilites. It is thought that the so-called unit Reynolds number effect results basically from this process.

The NTF with its added degree of freedom will have the capability to increase Reynolds number while maintaining the sound pressure level constant. To demonstrate this point the 0.3-m TCT was run at approximately 4.9 atmospheres total pressure over a range of total temperatures from 108 K to 322 K. This variation in total conditions yields chord Reynolds numbers ranging from 8×10^6 to 38×10^6 with virtually no change in sound pressure level (see fig. 17). With the chord fixed, the unit Reynolds number is changed then by a factor of four. For contrast, on this same figure sound pressure levels are plotted for two other conditions with total temperatures near 100 K. These two points plus the one in the upper right hand corner (running diagonally across the plot) show that an "acoustic" penalty of approximately 10 dB is incurred when Reynolds number is increased from 8×10^6 to 38×10^6 by increasing pressure. It should be noted that experimenters utilizing temperature for Reynolds number control in cryogenic tunnels will have to pay close attention to wall temperatures relative to the total temperature.

Flow Visualization and Diagnostic Measurements

The panel gave its most vocal support to the continued development of flow visualization and diagnostic techniques. Laser and schlieren systems were thought to be indispensable to the conduct of high quality fluid-flow research. A novel skin friction gage concept with good potential for cryogenic environments was described by one member. Still another noted a probe for measuring ρu, $\rho u'$, and flow angle that may have some advantages over a hot wire in very cold and/or high-pressure flows. Finally, the panel suggested that a mini-workshop be held to expose candidate methods, techniques, and instruments for flow visualization and diagnostic measurements.

Tunnel Calibration

A strong desire by the NTF tunnel designers has been to provide the lowest turbulence and sound pressure levels possible. This goal has received strong support from both in- and out-of-house scientists. With the design long finished and the construction nearly complete, the attention of the potential users and the panels has turned more and more to the measurements which will define what was actually achieved. The panel wanted to be assured that a thorough calibration of the tunnel would be carried out, including measurements of vorticity and acoustic levels. The calibration plan was discussed with the panel and presented to the workshop as well.

The main features of the flow quality measurements to be made during the calibration of the NTF are listed in figure 18. In summary it indicates that data should be obtained at all the important points using the best instrumentation available. Both statistical and spectral data are required for pressure, temperature, and velocity. Further details are given in reference 18.

The quantification of wind tunnel wall interference effects was not discussed at length by the panel, due to the charter given in this area to the Theoretical Aerodynamics Panel. However, the panel wanted to go on record as favoring a strong program. In addition to the usual effects of buoyancy, blockage, lift, and plenum venting (non-uniform flows near wall), Reynolds number sensitivities of these factors should be considered as well. The panel also urged that theoretical corrections based on homogeneous wall boundary conditions have those boundary conditions based on flow field surveys near the wall if at all possible.

CONCLUDING REMARKS

The Fluid Dynamics Panel engaged in discussions on almost every aspect of fluid flows, but spent the most time trying to formulate the best approaches to carrying out basic "classical" experiments at high Reynolds numbers on cones, flat plates, and airfoils. Tests of this sort have the great attraction of complementing a large number of investigations already accomplished in other facilities at lower Reynolds numbers. They are also configurations which have been subjected to theoretical treatment for a number of years. When carried out properly they should yield data on the facility's flow quality and on boundary-layer transition, and should increase understanding of the utility of the various prediction methods. The need was stressed to have a thorough understanding (measurement) of the flow quality in the NTF and 0.3-m TCT as an adjunct to these tests, along with the requirement for models with smooth and fully characterized surfaces.

The difficulty of carrying out diagnostic measurements at low temperatures, high pressure, and high Reynolds numbers was fully recognized. Panel members suggested instruments that potentially could be useful in this environment and stressed the need for increased research in the techniques and instrumentation area. A further suggestion was made that this area be considered as the subject of a mini-workshop. This workshop would endeavor to get the best minds and practitioners in the diagnostic field to establish the state of the art and to "brainstorm" new approaches for extending it.

A large number of configurations were suggested for which high Reynolds number data would be most welcome. Some were similar to those proposed at the 1976 workshop, while others were new. Time did not permit detailing of the flow physics involved or the preferred test conditions and instrumentation requirements (refer to the report of the Configurations Aerodynamics Panel, ref. 20, for a more thorough discussion of configuration research needs).

Finally, the panel strongly supported the idea of carrying out a careful and complete calibration of the NTF, including flow quality and uniformity in the test section, diffuser, and contraction section as well as at other critical locations around the circuit. Increased effort in experiments and analysis aimed at evaluating wall interference was urged.

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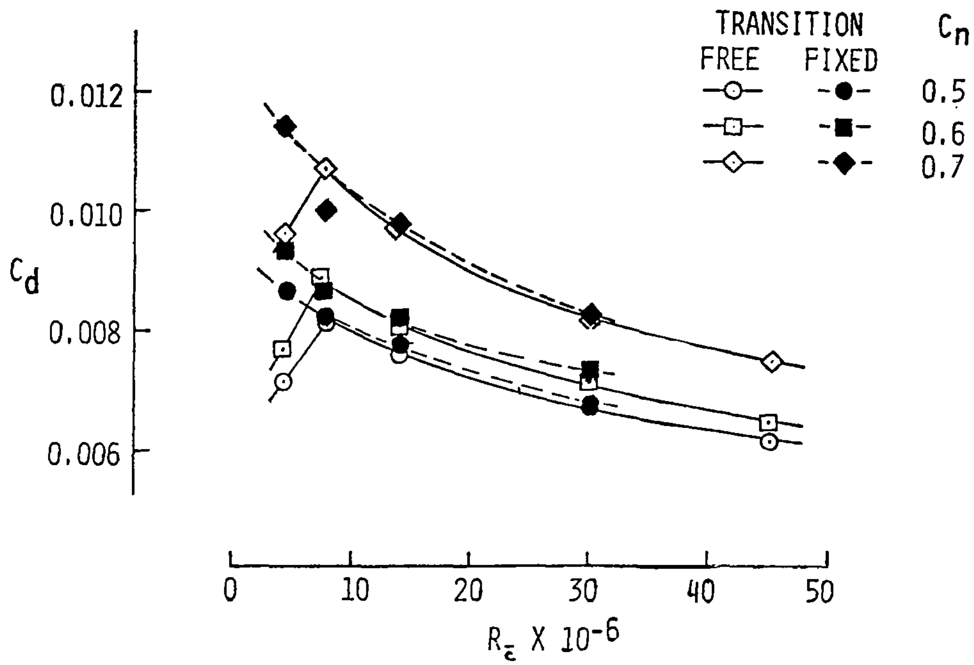

Figure 1.- Drag coefficient variations with Reynolds number for both fixed and free transition. $M_\infty = 0.76$.

- 12% SUPERCRITICAL AIRFOIL
- VARIOUS GRIT SIZES
- RANGE OF REYNOLDS NUMBER TO 50 X 10^6
- VARIOUS TRIP LOCATIONS TO SIMULATE REYNOLDS NUMBER TRANSITION
- HIGH REYNOLDS NUMBER TESTS TO CHECK VALIDITY OF SIMULATIONS
- TRANSITION MEASUREMENTS BY TWO METHODS

Figure 2.- Features of airfoil transition-trip program.

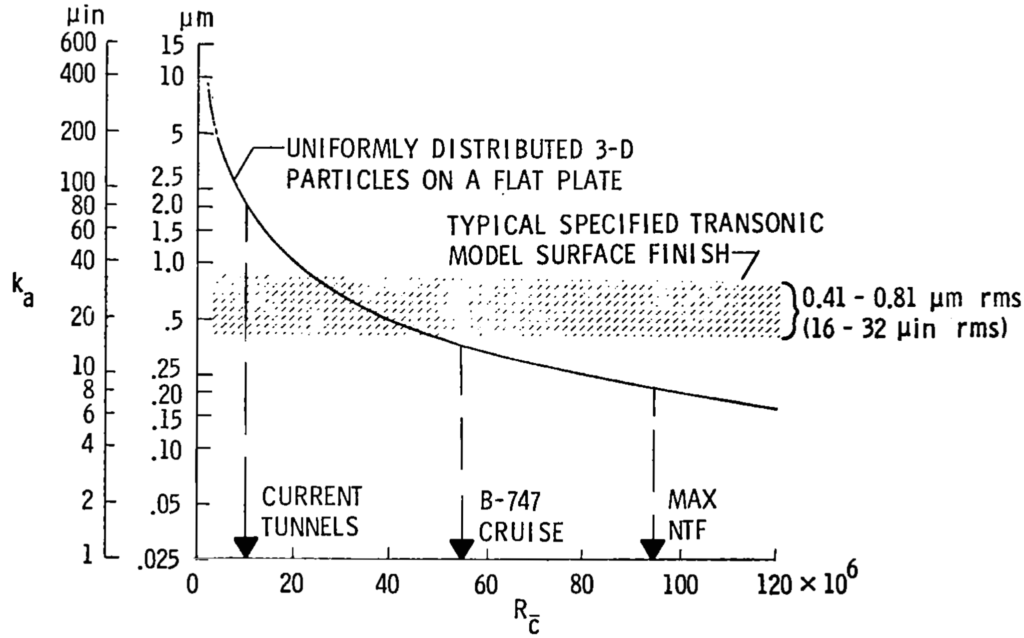

Figure 3.- Admissible roughness (k_a) for typical NTF-size models. $\bar{c}$ = 0.2 m (0.65 ft).

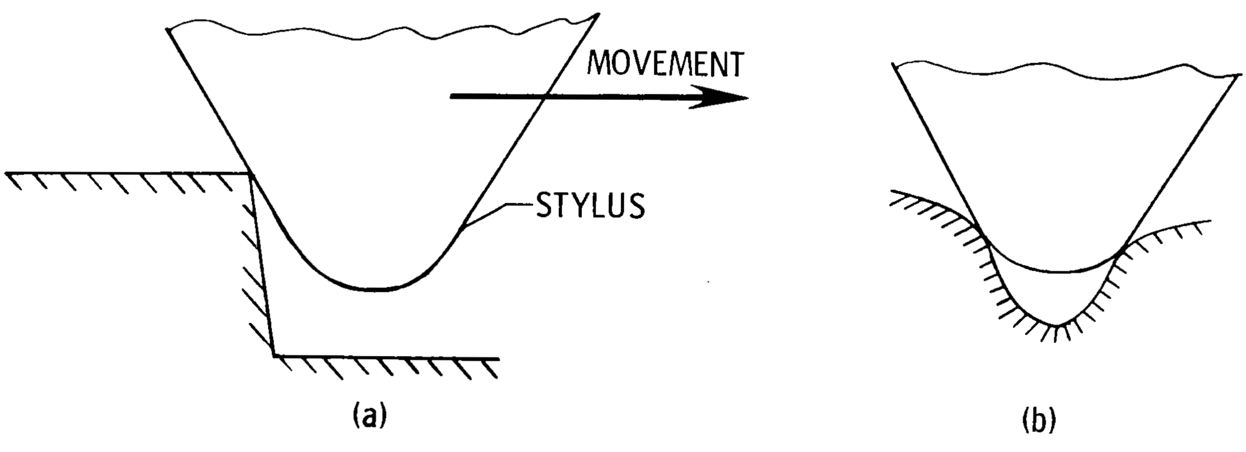

Figure 4.- Potential stylus profilometer problem areas.

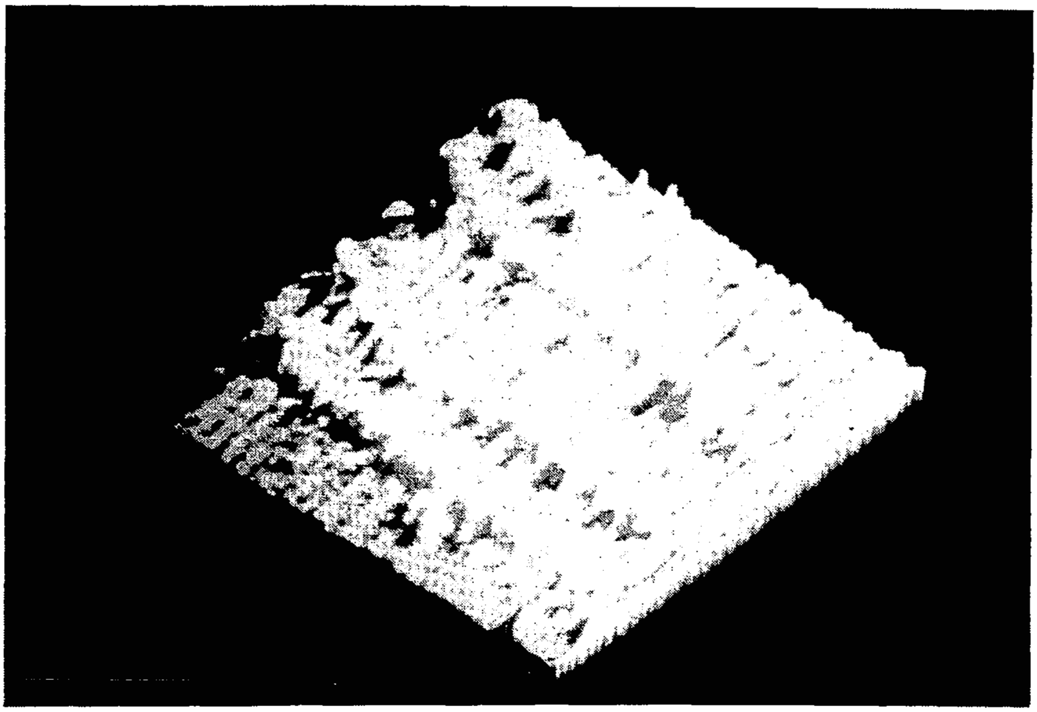

Figure 5.- Topography of typical NTF model surface determined by stylus profilometer - National Bureau of Standards (200-μm- (0.008-in-) square section).

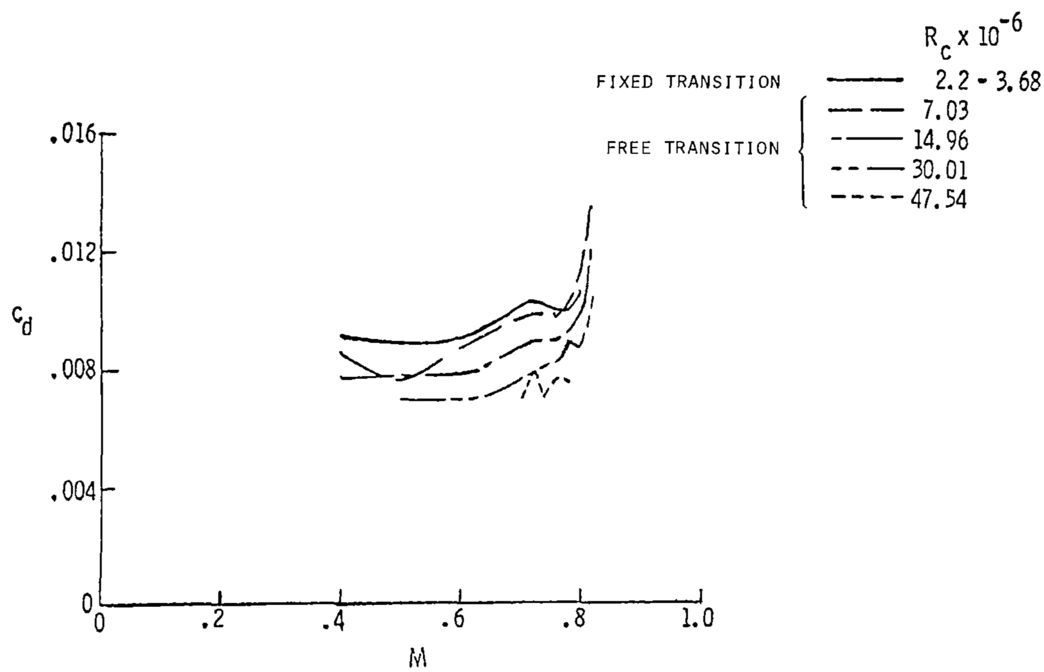

Figure 6.- Effect of Reynolds number on drag rise for a C_n of 0.35.

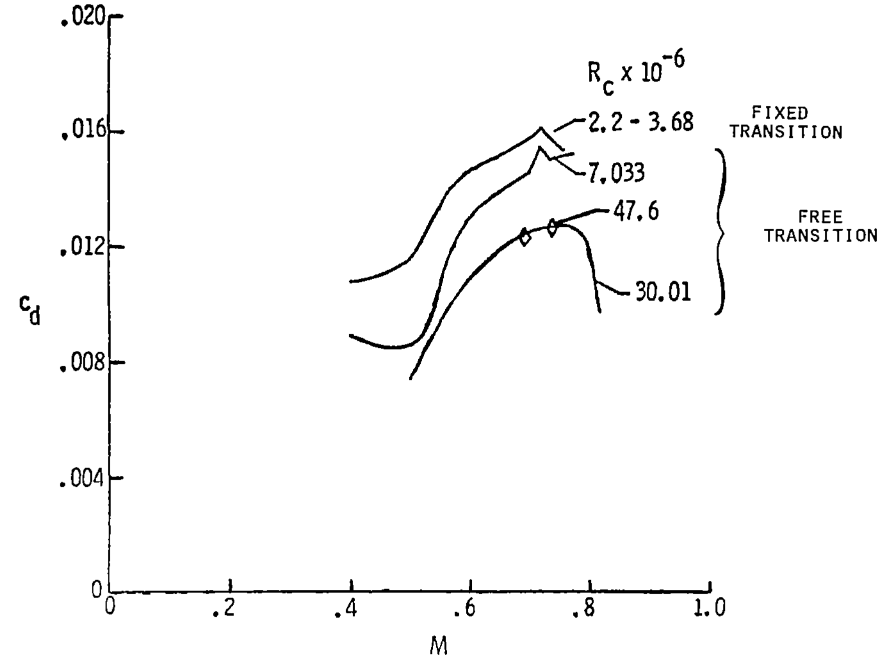

Figure 7.- Effect of Reynolds number on drag rise for a C_n of 0.55.

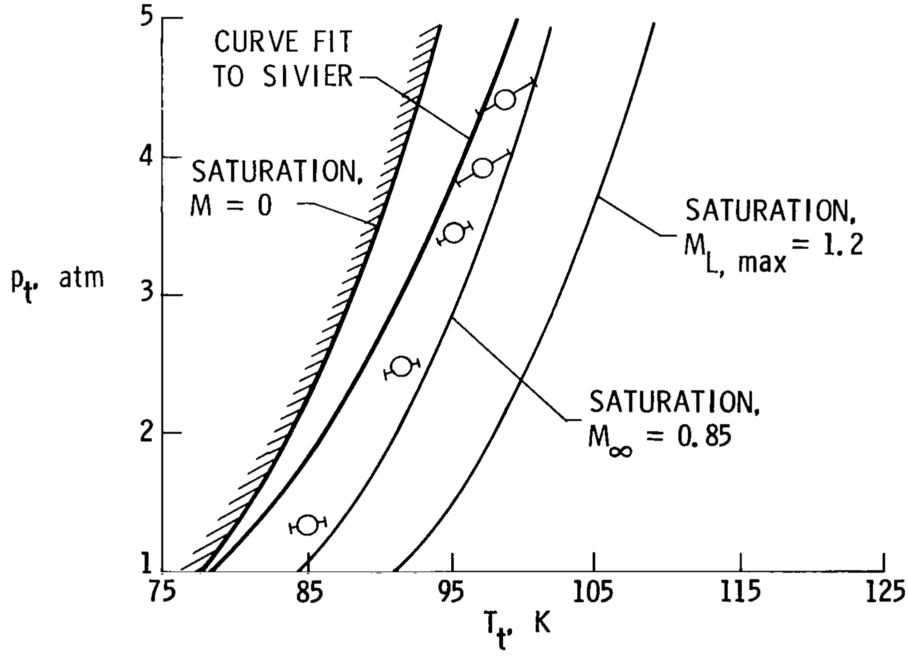

Figure 8.- Onset of effects with the NACA 0012-64 airfoil compared to predicted onset of homogeneous nucleation.

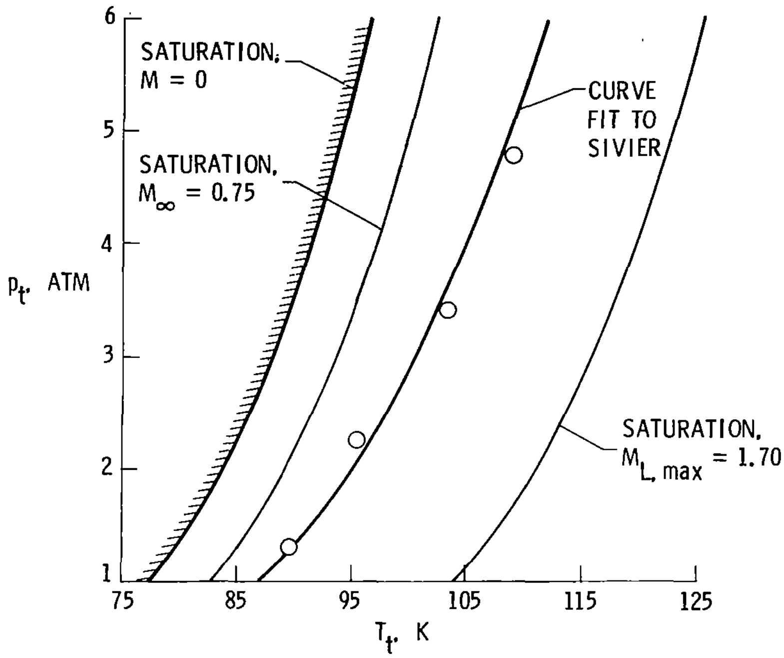

Figure 9.- Onset of condensation effects for NPL-9510 airfoil. $M_\infty = 0.75$; $\alpha = 6^o$; $M_{L,max} = 1.70$.

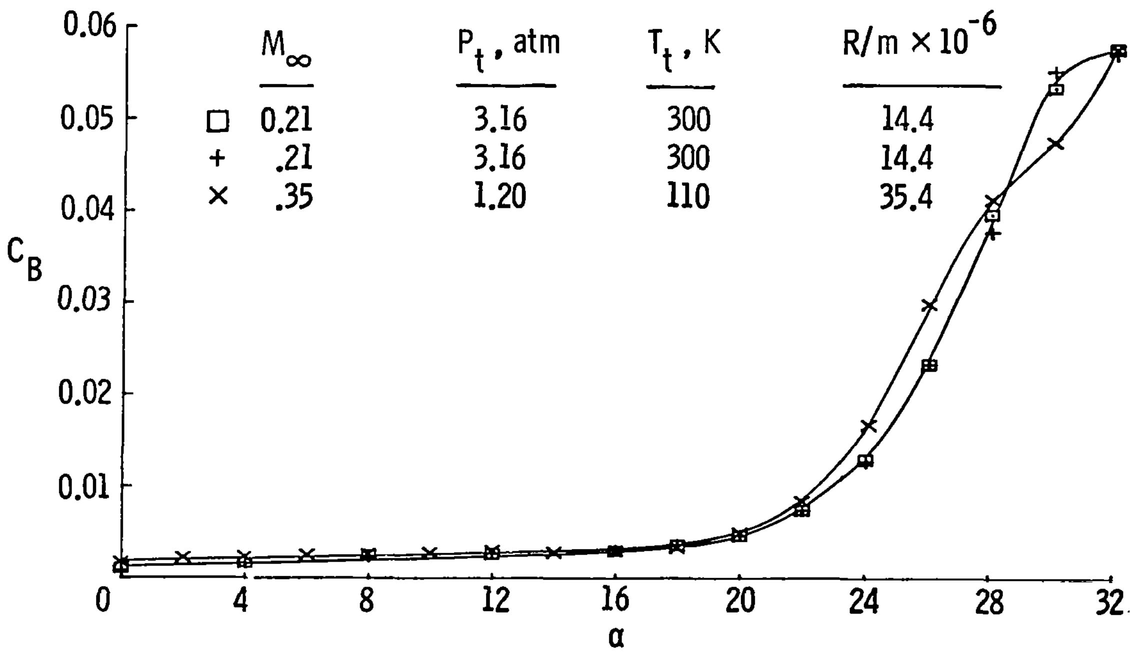

Figure 10.- Buffet data for delta-wing model as characterized by root bending moment. $q_\infty = 9578$ N/m^2; $V_\infty = 72.6$ m/sec.

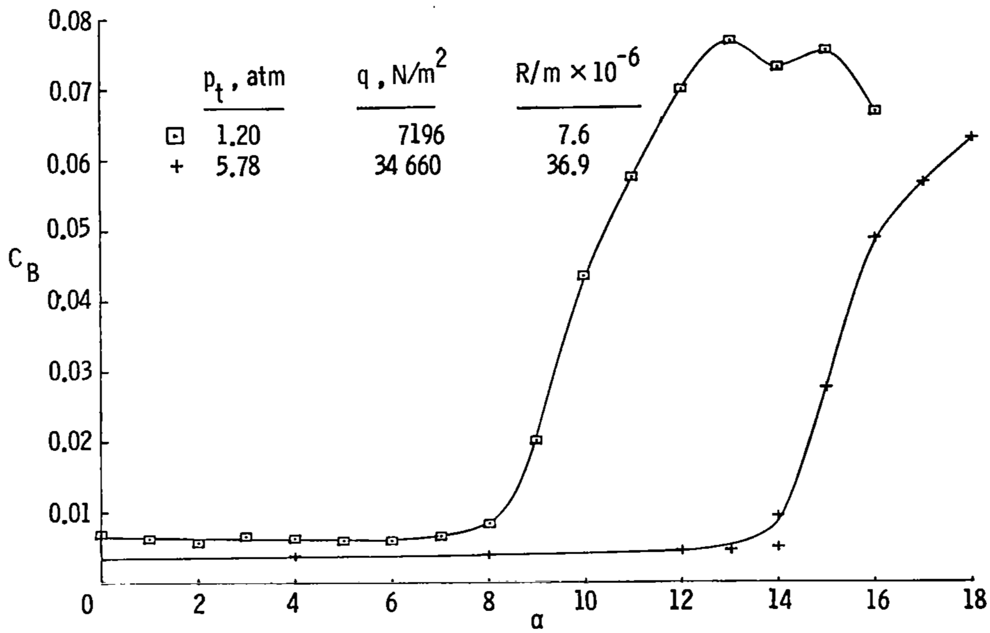

Figure 11.- Buffet data for NPL 9510 airfoil as characterized by root bending moment. M_∞ = 0.3; T_t = 300 K, V_∞ = 103 m/sec.

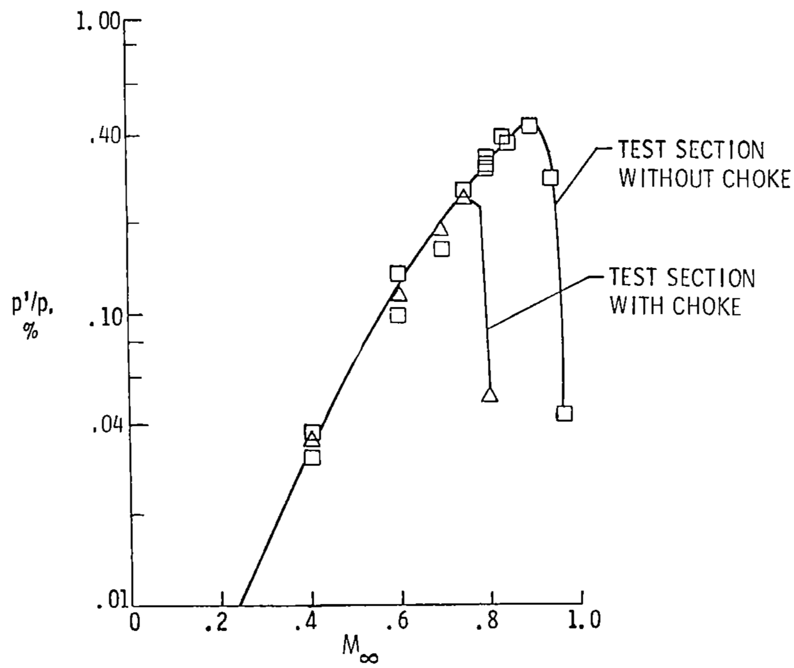

Figure 12.- Effect of choke on static pressure perturbations in test section of 8' TPT.

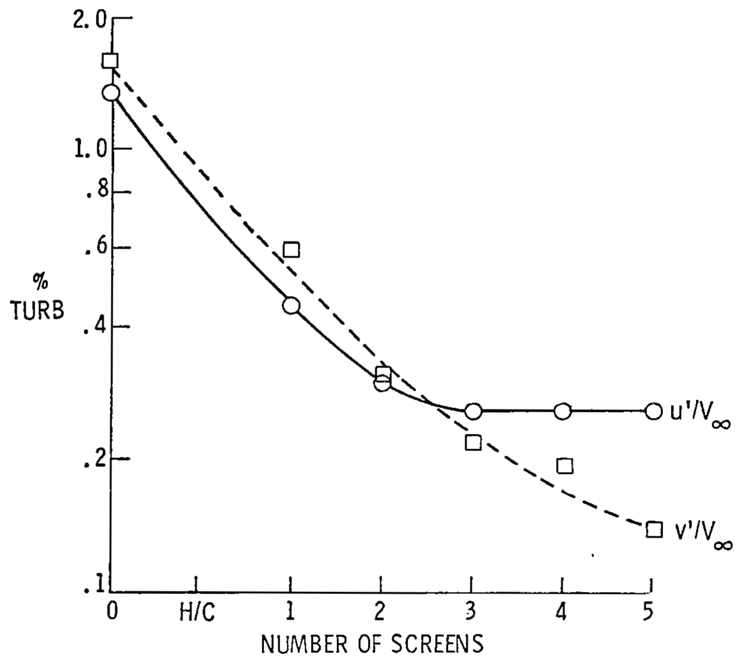

Figure 13.- Reduction of turbulence with various honeycomb-screen combinations.

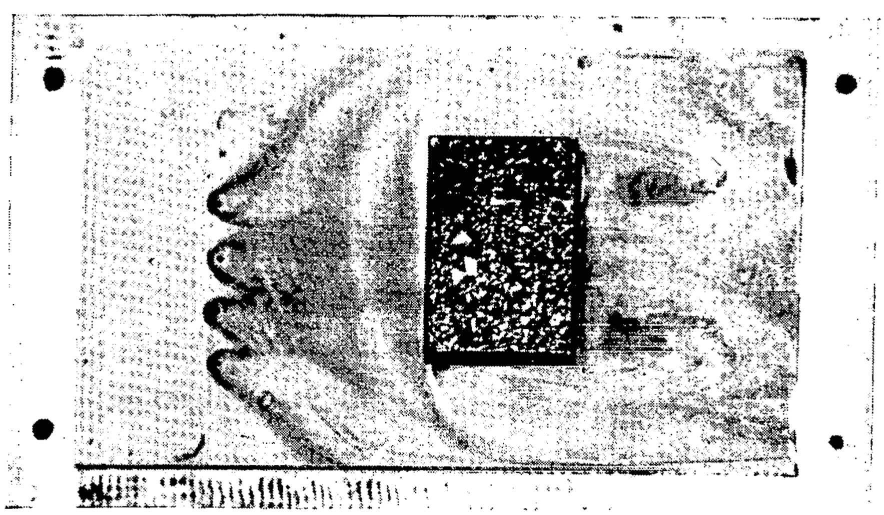

Figure 14.- Visualization of flow around an obstacle using liquid propane.

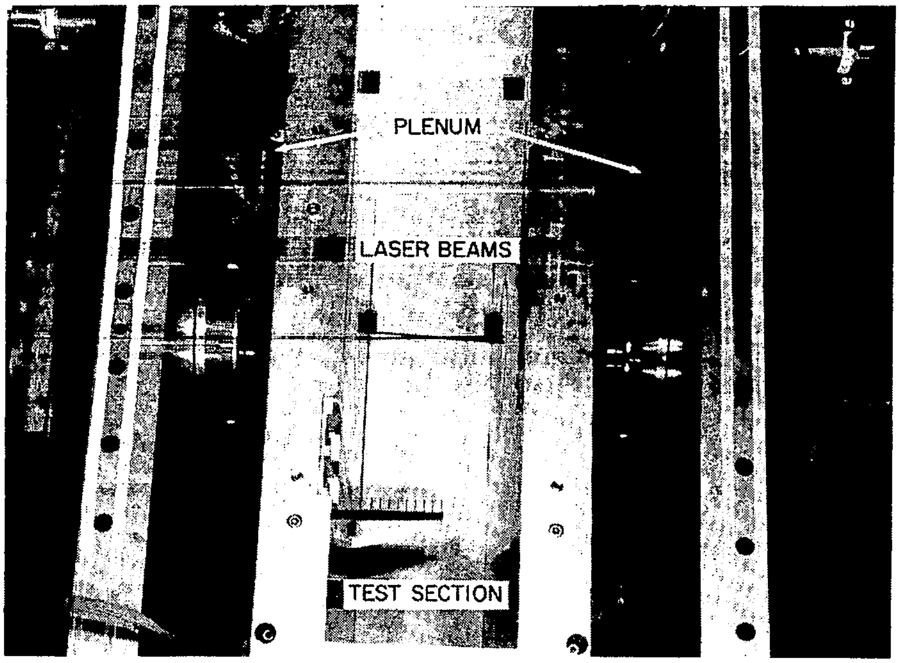

Figure 15.- Laser velocimetry in 0.3-m TCT.

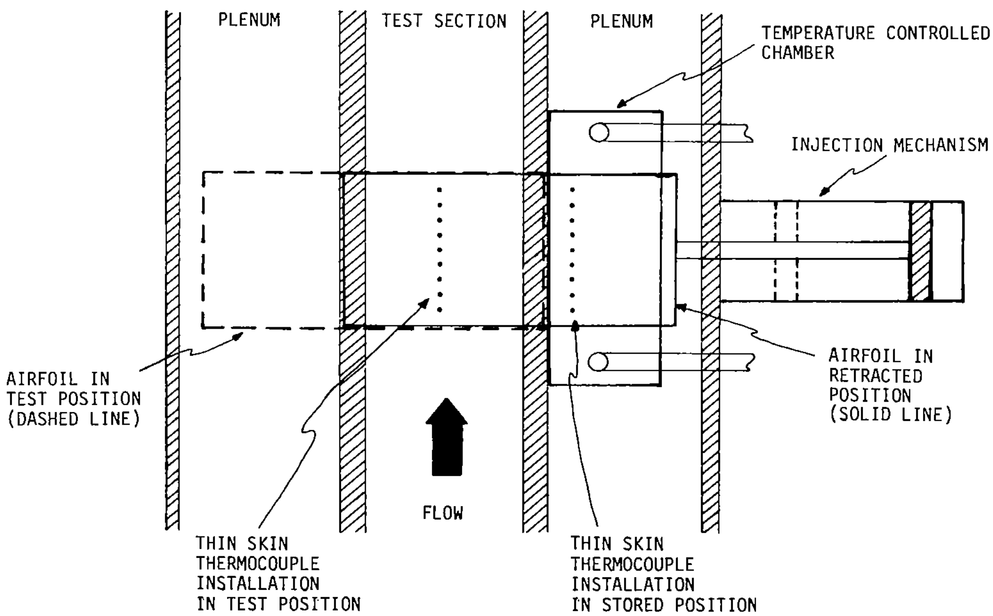

Figure 16.- Test set-up for determining effect of wall temperature on transition location in 0.3-m TCT.

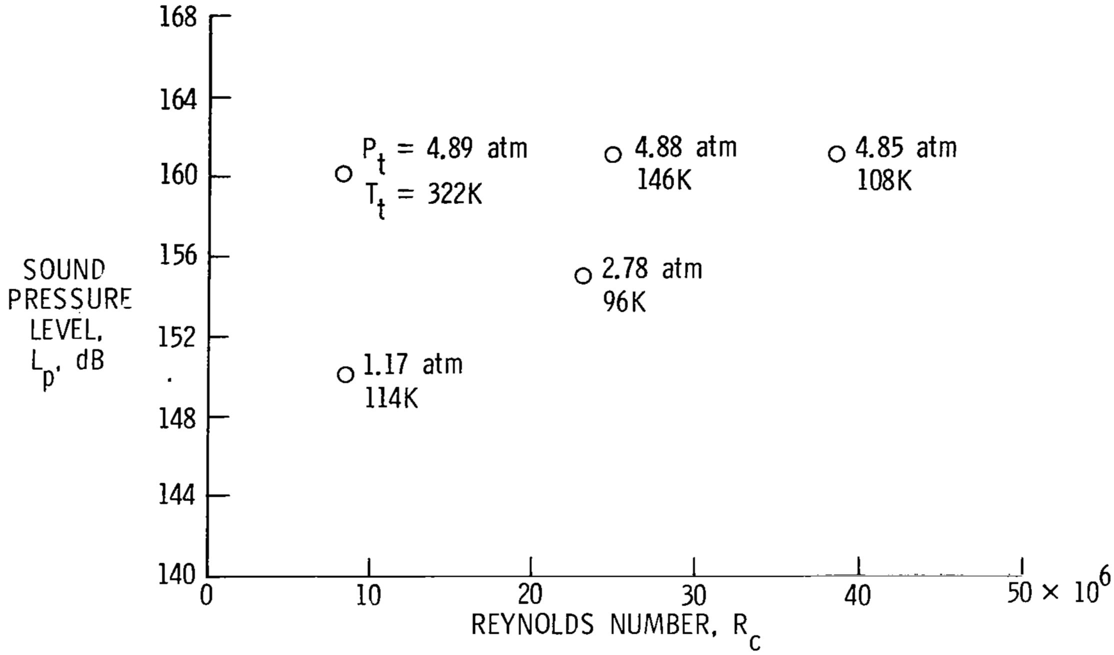

Figure 17.- Broadband sound pressure level in the test section of 0.3-m TCT. $M_\infty = 0.80$.

MEASUREMENT LOCATIONS

- TEST SECTION - PLENUM
- SETTLING CHAMBER
- RAPID DIFFUSER
- HIGH-SPEED DIFFUSER
- FAN STATION
- LN_2 INJECTION STATION

MEASUREMENT INSTRUMENTATION

- WALL-MOUNTED PRESSURE TRANSDUCERS
- HOT WIRES, HOT FILM PROBES, AND LV

MEASUREMENTS

- RMS
- SPECTRA
- CORRELATIONS

QUANTITIES

- PRESSURE
- VELOCITY
- TEMPERATURE

Figure 18.- Features of flow quality measurements to be made in the NTF.

HIGH-LIFT PANEL

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Technical Secretary	Paul L. Coe, Jr. NASA LaRC

Panel Members

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John P. Campbell	The George Washington University
Heinz Gerhardt	Northrop Corporation
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Michael D. Mack	The Boeing Company
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Frank W. Steinle, Jr.	NASA Ames Research Center

HIGH-LIFT TECHNOLOGY

Paul L. Coe, Jr. and Richard J. Margason

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SUMMARY

The NASA Langley high-lift technology program is reviewed, and elements of the program which are considered Reynolds number sensitive are discussed. The Energy Efficient Transport (EET) and Supersonic Cruise Research (SCR) models proposed for high-lift studies in the National Transonic Facility (NTF) are described. Recommendations regarding the NTF facility and test techniques are presented.

INTRODUCTION

Aircraft engine/airframe sizing studies must, obviously, consider the entire operational envelope. Previous experience has shown that low-speed, take-off, climb, and landing performance requirements impose constraints on aircraft concepts which result in less than optimum efficiency. Studies have demonstrated that the development of improved high-lift systems will relax these low-speed operational constraints, thereby resulting in the following potential benefits:

(1) Engine/airframe optimized for cruise efficiency
(2) Improved climb performance
(3) Reduced field length requirements
(4) Reduced approach velocities (improves safety and airport capacity)
(5) Reduced community noise

The distribution of block fuel consumption for the various operating phases of a typical wide-body jet is shown in figure 1 as a function of stage length. Inasmuch as the typical commercial flight is in the 500 to 1000 nautical mile range, figure 1 demonstrates that the fuel consumed during climb and cruise is approximately the same. The potential savings achieved by permitting the engine/airframe to be optimizated for cruise efficiency and improved climb performance is substantial.

As an illustration of the potential magnitude of the fuel savings which could be derived from the development of improved high-lift systems, figure 2 presents fuel burned as a function of maximum landing lift coefficient ($C_{L,max}$). Results are presented for a contemporary wide-body transport and for the same transport utilizing an aspect-ratio-12 supercritical wing developed under the NASA Energy Efficient Transport (EET) program. For the conditions specified, the contemporary transport consumes about 39 900 kg (88 000 lb) of fuel and has a required landing $C_{L,max}$ of about 2.2. Simply replacing the current wing with the aspect-ratio-12 supercritical EET wing of the same area would result in an increased structural weight (required to provide the necessary wing stiffness) which in turn would result in a somewhat higher wing loading and a corresponding increase in required landing $C_{L,max}$.

The improved performance resulting from this modification would apparently result in about a 10-percent reduction in fuel burned. If the wing were re-sized to optimize cruise efficiency, analysis shows that an 18-percent reduction in fuel burned could be achieved. However, the wing loading would be significantly increased and, as a result, the landing $C_{L,max}$ would be required to increase from approximately 2.2 to about 3.1.

A high-lift system consistent with the EET cruise geometry (see fig. 3) has been designed and tested at low Reynolds numbers ($RN_{\bar{c}} = 1.6 \times 10^6$) in the 4- by 7-Meter Tunnel at the NASA Langley Research Center (see ref. 1). The analysis presented in reference 1 shows that the maximum trimmed lift coefficient for such a configuration under these conditions would be limited to values on the order of only 2.5. Experience obtained by industry shows a substantial effect of Reynolds number on $C_{L,max}$. Unfortunately, the Reynolds number effects appear to be contradictory in nature and are configuration dependent. Hence, there does not exist, at present, a proven method to extrapolate $C_{L,max}$ from the low test Reynolds numbers to those of flight.

SYMBOLS

AR	aspect ratio
$\bar{c}$	mean aerodynamic chord
C_D	drag coefficient, drag/qS
C_L	lift coefficient, lift/qS
$C_{L,max}$	maximum lift coefficient
C_m	pitching-moment coefficient, pitching moment/$qS\bar{c}$
C_p	pressure coefficient
M	Mach number
q	free-stream dynamic pressure, Pa (lbf/ft^2)
$RN_{\bar{c}}$	Reynolds number based on mean aerodynamic chord
S	wing area, m^2 (ft^2)
V_{ref}	reference flight speed, knots
W/S_{TO}	take-off wing loading, kg/m^2 (lb/ft^2)
x/c	nondimensional chord length
α	angle of attack, deg
β	angle of sideslip, deg

Γ dihedral, deg

Λ wing sweep, deg

CURRENT STATUS AND PLANS

The Langley High-Lift Technology Program is summarized in figure 4. The program is comprised of fundamental investigations, computational methods, and conceptual applications. The various elements comprising the program are briefly described in the following sections.

Fundamentals

High-lift airfoil research is being conducted in the Langley Low Turbulence Pressure Tunnel (LTPT) rather than the 0.3-Meter Transonic Cryogenic Tunnel (0.3-m TCT). This decision was predicated on the short chord lengths required by the 0.3-m TCT. The short chord lengths would not permit the chordwise pressure distribution to be measured in sufficient detail, and therefore it appears that the LTPT is a more suitable facility for this type of research. At present, a high-lift balance and model support system for the LTPT is under development. Additionally, sidewall boundary-layer separation at high-lift conditions is being studied. A series of investigations of high-lift airfoils at representative flight Reynolds and Mach numbers is planned. These studies will include two-dimensional parametric studies of flap and slat geometries and will be used to validate analytical methods for multi-element airfoils. Based on the capabilities of the LTPT, it would appear that two-dimensional airfoil and parametric studies can be adequately conducted in this facility and will not require the NTF.

Additional fundamental investigations are being conducted. These include a grant to Wichita State University to experimentally measure the characteristics of separated wakes (fig. 5 presents a sample of the measured velocity profile for a spoiler-induced separated wake) and three-dimensional parametric wing studies conducted in the Langley 4- by 7-Meter Tunnel (see fig. 6).

Computational Methods

A substantial effort in the development of relatively fast, reliable computational methods is in progress. Such methods will be verified or validated by the fundamental investigations discussed in the preceding section and will then be used in conjunction with conceptual applications as discussed in a subsequent section.

An example of surface panel methods is shown in figure 7, which represents a mathematical model of the NASA EET concept. Results of calculations of overall forces and moments, as well as chordwise pressure distributions, are presented and compared with experimental results in figure 8. Results are presented for the inviscid calculation and for the inviscid calculation modified to include a two-dimensional or strip boundary-layer theory. As can be seen, inclusion of the strip boundary layer improves the theory/experiment agreement for attached flow conditions. Inasmuch as the present calculations are for an aspect-ratio-12 wing with only moderate sweep, the two-dimensional boundary-layer simulation apparently is sufficient. However, for lower aspect ratio or high wing sweeps, the viscous crossflow becomes increasingly important and use of three-dimensional boundary-layer theory in conjunction with the inviscid calculation would be more appropriate. Accordingly, this effort is in progress. Of course, when flow separation occurs, the above methods are no longer valid. A two-dimensional method, which includes the separated wake effects, has been developed under contract by McDonnell Aircraft Company. Figure 9 shows results of calculations for a two-dimensional airfoil at an angle of attack of 15° and compares the calculated results with experimental data. As expected, calculations which omit the separated wake yield pressure distributions which are significantly in excess of experimental values. Furthermore, the calculated section lift is about 1.5 times the experimental values. However, inclusion of the separated wake results in a calculated pressure distribution and section-lift coefficient which are remarkably close to the experimental results.

It should be noted that the methods discussed have been limited to single-element airfoils. There is at present a two-dimensional design/analysis method applicable for inviscid multi-element airfoils. This method was also developed under contract with McDonnell Aircraft Company (MCAIR) and may be extended to three-dimensional design methods. Simultaneously, a somewhat different approach which is applicable to two-dimensional multi-element airfoils and which includes separation effects (enabling the calculation of $C_{L,max}$) is under development at the NASA Langley Research Center.

Additional elements of the computational method effort include both in-house studies and studies conducted under grants to Mississippi State University for the development of fast, efficient solutions to the Navier-Stokes equations. These studies include two-dimensional multi-element airfoils with separation and are to be extended to include the three-dimensional solution.

Conceptual Applications

The understanding of and insights into the fluid mechanical phenomenon achieved through fundamental investigations, coupled with the design and analysis capability provided by the development of advanced computational methods, are used in an attempt to produce new approaches in high-lift technology by their application to aerodynamic designs. These applications include detailed studies of propulsive-lift concepts, configuration effects, and Reynolds number effects.

Propulsive lift.- The Langley Research Center has studied propulsive-lift concepts for a wide variety of applications including the low-speed flight regime of conventional subsonic transport aircraft and advanced supersonic transport configurations. However, more recently the military's concern regarding damaged runways and denied landing fields has directed the propulsive-lift effort towards advanced fighter aircraft concepts. One such in-house conceptual design is shown in figure 10.

Configuration effects.- A challenging aspect of the high-lift technology effort is to effectively integrate or develop high-lift systems compatible with particular high subsonic or supersonic cruise designs. An example of one such effort is the NASA Langley Laminar Flow Control (LFC) airfoil. Figure 11(a) shows the Mach = 0.8 cruise design, while figure 11(b) depicts the airfoil with a deployed leading-edge slat and a single-slotted trailing-edge flap for the low-speed high-lift condition. Other configurations currently under study include the previously discussed Energy Efficient Transport (EET) concept and the Supersonic Cruise Research (SCR) concept, both of which will be discussed in detail in the section concerning the high-lift/ high Reynolds number research.

High-lift/high Reynolds number research.- Experience has shown that conventional aerodynamic performance characteristics and the stall/post-stall aerodynamic characteristics are sensitive to Reynolds number. The particular Reynolds number sensitive areas will be discussed by the high-lift panel members and reported in a subsequent section. However, as an illustration of configuration-dependence and need for high Reynolds number testing of high-lift configurations, figure 12 presents $C_{L,max}$ as a function of Reynolds number. For this particular configuration, flight Reynolds number is 20×10^6. The Reynolds number range over which data are presented is representative of the Reynolds number range provided by existing subsonic atmospheric and pressure tunnels. Data are presented for the configuration with a clean wing and for the configuration having wing leading-edge devices and an extended two-segmented trailing-edge flap system. The extended trailing-edge flap is either undeflected ($\delta_f = 0^\circ$) or deflected as indicated. Of particular interest is the marked increase in $C_{L,max}$ with increasing Reynolds number for both the clean-wing configuration and the configuration employing the slatted leading edge. By contrast, the configuration employing the Variable Camber Kreuger (VCK) leading edge generally exhibits a less pronounced increase in $C_{L,max}$ with increasing Reynolds number. The point to be drawn from figure 12 is the inherent uncertainty in the prediction of $C_{L,max}$ from the extrapolation of wind-tunnel data.

The Langley Research Center, after extensive interaction with industry, is proposing in-depth NTF studies of the low-speed high-lift characteristics of two widely varying aircraft concepts. These concepts are the EET and SCR configurations mentioned previously. Figures 13 and 14 show photographs of models of these respective configurations which were used in prior Langley wind-tunnel studies. These studies were conducted at relatively low Reynolds numbers ranging from 1.5×10^6 to 2.5×10^6. Present plans are to conduct additional tests in the Ames Research Center 12-Foot Tunnel with similar smaller models at somewhat higher Reynolds number followed by tests in the NTF at flight Reynolds numbers. These proposed NTF low-speed high-lift studies are summarized in figures 15 and 16. Figure 15 illustrates what is referred to as the EET/Pathfinder 1 model. The model will be comprised of the EET wing as shown in figures 3 and 13 and will use the existing Pathfinder 1 fuselage which is virtually the same as the EET fuselage shown in figure 13. The model is intended to study the effectiveness of various leading- and trailing-edge systems and to explore tailoring of leading-edge devices in order to control stall progression and the effect of stall on wing control surfaces. The tests will produce conventional force and moment data and wing surface pressure data (including flap and slat elements) over a representative α-β range and will permit the effects of Reynolds number to be determined from values up to 40×10^6. Because of the lengthy time intervals involved in bringing the tunnel down for model changes, it is highly desirable to remotely actuate as many movable surfaces as possible.

The proposed low-speed high-lift SCR model is illustrated in figure 16. This particular configuration differs slightly from previous SCR geometries (see ref. 2 for details) and is the subject of a cooperative NASA/industry research effort. It is proposed that initial studies concentrate on the effects of leading-edge radius and deflection with follow-up studies of alternate leading-edge devices. The leading-edge radius study is intended not only to determine the effect of radius, but also to determine if it is possible to simulate higher Reynolds number conditions by using a suitably modified leading-edge radius. This would permit substantial development work to be conducted in relatively inexpensive tunnels with limited validation testing in high Reynolds number facilities. As such, the successful conclusion of this undertaking would represent a major breakthrough in aerodynamics. For this configuration, leading-edge deflection schemes range from simple constant deflection of the entire leading edge to a more complex scheduled deflection for which the deflection angles are held to a minimum inboard and increased along the span to a maximum at the wing tip. Such schedules have demonstrated marked improvement in flow attachments, and hence aerodynamic performance, for the low-speed high-lift condition. Unfortunately, tests of these concepts have all been conducted at relatively low Reynolds numbers. In addition to conventional force and moment data and wing surface pressures, experience has shown that flow visualization is particularly useful in defining the type of flow separation on this class of wings. It is, therefore, considered important that some effort be directed towards development of effective flow visualization schemes for the NTF. The conventional lift/drag polars shown in the lower right-hand corner of figure 16 illustrate the theoretical bounding polars of 100-percent and 0-percent leading-edge suction. It is anticipated that increasing Reynolds number will shift the polar further away from the 0-percent boundary and closer to 100-percent boundary, but the degree to which this will occur, while of extreme importance, is unknown.

PANEL DISCUSSION AND RECOMMENDATIONS

Reynolds-Number-Sensitive Areas

The high-lift panel considered Reynolds-number-sensitive areas for subsonic and supersonic designs in the takeoff, landing, and transonic high-lift configuration. These areas are summarized and discussed individually as follows:

(1) Performance characteristics
    (a) Leading-edge flow separation
    (b) Leading-edge suction
    (c) Vortical flows
    (d) Leading- and trailing-edge flap effectiveness
    (e) Control surface effectiveness

(2) Stall/post-stall characteristics
    (a) Maximum lift
    (b) Longitudinal stability and control
    (c) Lateral stability and control

In the area of performance, the most obvious Reynolds-number-sensitive effect is leading-edge flow separation. This is apparently most pronounced for airfoils having small leading-edge radii and high wing sweeps. This effect is universally recognized and raises significant uncertainty in data from conventional low-Reynolds-number wind tunnel tests. Furthermore, leading-edge flow separation is related to most of the areas summarized, particularly leading-edge suction.

Bodies at high angles of attack experience substantial crossflow and subsequent vortex separation. The mechanism of body vortex formation and vortex shedding appears to be Reynolds number sensitive, and consequently may have a substantial effect on aerodynamic performance.

Leading- and trailing-edge flap effectiveness is, of course, directly related to separation effects and is sensitive to Reynolds number. Furthermore, based on the data presented in figure 12, it would appear that the nature of the Reynolds number effect is dependent on the leading-edge configuration. In a similar fashion, control surface effectiveness, while a major factor in sizing aircraft components, is considered Reynolds number sensitive. For example, horizontal-tail effectiveness is of paramount concern with regard to pitchup. However, in typical high-lift studies, the horizontal tail operates at exceptionally low values of Reynolds number and, hence, is particularly deserving of attention.

Turning to the area of the stall and post-stall aerodynamic characteristics, the most obvious area, as discussed previously (see fig. 12), is the area of maximum lift, which is closely related to leading-edge flow separation and leading- and trailing-edge flap effectiveness. Longitudinal and lateral stability and control characteristics are considered to be largely dependent on the stall pattern and spanwise progression of the stall. Consequently, these characteristics will also exhibit Reynolds number dependence and warrant further attention.

Facility and Test Technique

In order to insure maximum cost effectiveness and full utilization of the unique capabilities afforded by NTF, considerable attention was given to facility and test technique development. Specific recommendations are:

(1) Develop flow visualization techniques
(2) Develop means to verify model component position under test condition
(3) Alternate model support system
(4) Maximize automation of model component parts
(5) Closed slots for subsonic high-lift development studies

Development of flow visualization techniques is considered to be essential. One concept, in particular, appears promising for application in the NTF. This concept, which was developed at the Boeing Company, employs rapidly traversing, pressure-sensitive diode lights. It is recommended that preliminary verification tests of this concept be conducted in the 0.3-m TCT. Laser velocimeter (LV) techniques were generally considered inappropriate for NTF flow visualization application. This was due to the LV requirement for the introduction of seeding particles which would create condensation problems. Holographic techniques, on the other hand, were considered promising, and it was recommended that the work at Ames and Langley (in particular the planned holographic study in the 4- by 7-Meter Tunnel) be strongly supported.

Verification of model component position under test conditions was considered to be of particular importance for high-lift investigations. This is predicated on the known sensitivity of results to parameters such as slat and/or slot gap and overlap. It is specifically recommended that photographic techniques be tested with a representative high-lift system in the 0.3-m TCT.

Substantial concern was expressed about large sting tare effects particularly on horizontal tail characteristics and on the limited angle-of-attack capability provided by the proposed sting arrangement. It is specifically recommended that alternate support systems - for example, a sidewall-mounted strut support system - be developed. The development of such a strut support system should enable sting tare effects to be determined and should also permit data to be obtained at high angles of attack (i.e., up to 90°) for stall/spin research.

Recognizing that pressurization and thermal equilibrium considerations will result in extensive time delays associated with model changes, it is recommended that substantial effort be directed towards developing means for remote actuation of model components.

Lastly, it is recognized that slotted wind tunnels present a problem in determining wall corrections for high-lift configuration studies. It is, therefore, suggested that the high-lift research be conducted with the closed slot test configuration and that existing wall pressure taps be used in correlating wind tunnel correction techniques to insure proper corrections for wall effects.

High-Lift Configuration Studies

The panel strongly supported Langley's proposed EET and SCR high-lift research in NTF (see figs. 15 and 16). The panel noted that an integral part of these studies would center on the ability to develop techniques and methods for design and construction of flap brackets that will produce the required geometry (for example, gap and overlap) throughout the NTF operational envelope.

Applications Adequately Covered in Other Facilities

In addition to applications for NTF, the members of the high-lift panel thought it appropriate to list areas where testing is adequately performed in other facilities. These include:

(1) Two-dimensional high-lift
 (a) Parametric investigations
 (b) Code verification
(2) Rotorcraft
(3) V/STOL aircraft
(4) General aviation

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1. Morgan, Harry L., Jr.; and Paulson, John W., Jr.: Low-Speed Aerodynamic Performance of a High-Aspect-Ratio Supercritical Wing Transport Model Equipped With Full-Span Slat and Part-Span Double-Slotted Flaps. NASA TP-1580, 1979.

2. Coe, Paul L., Jr.; Thomas, James L.; Huffman, Jarrett K.; Weston, Robert P.; Schoonover, Ward E., Jr.; and Gentry, Garl L., Jr.: Overview of the Langley Research Effort on SCR Configurations. NASA CP-2108, 1979, pp. 13-33.

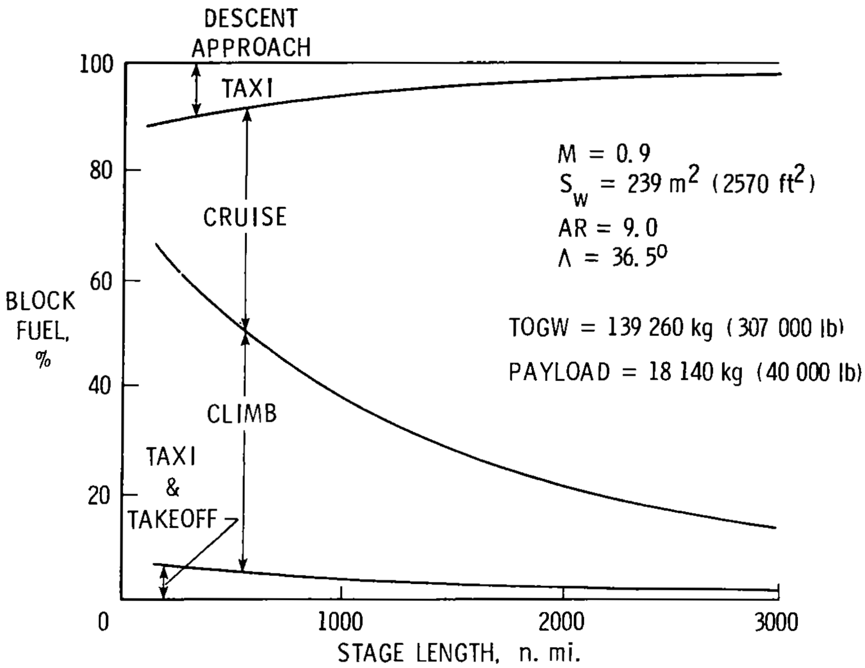


Figure 1.- Block fuel consumption as a function of stage length.

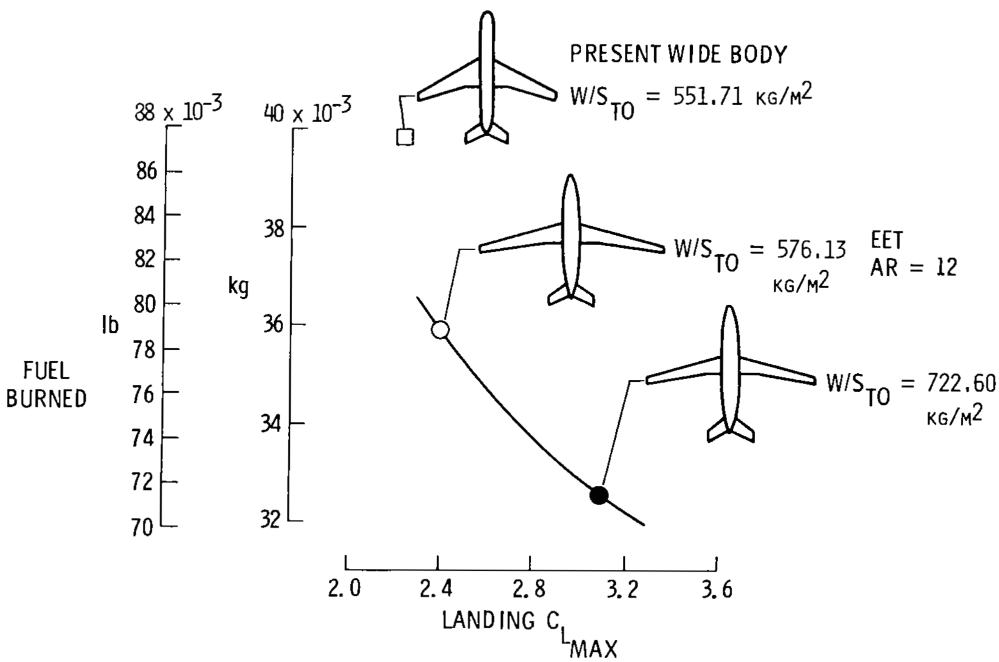


Figure 2.- Impact of high-lift technology on fuel consumption. M = 0.8; alt = 10 668 m; V_{REF} = 140 KIAS; range 2700 n.mi.

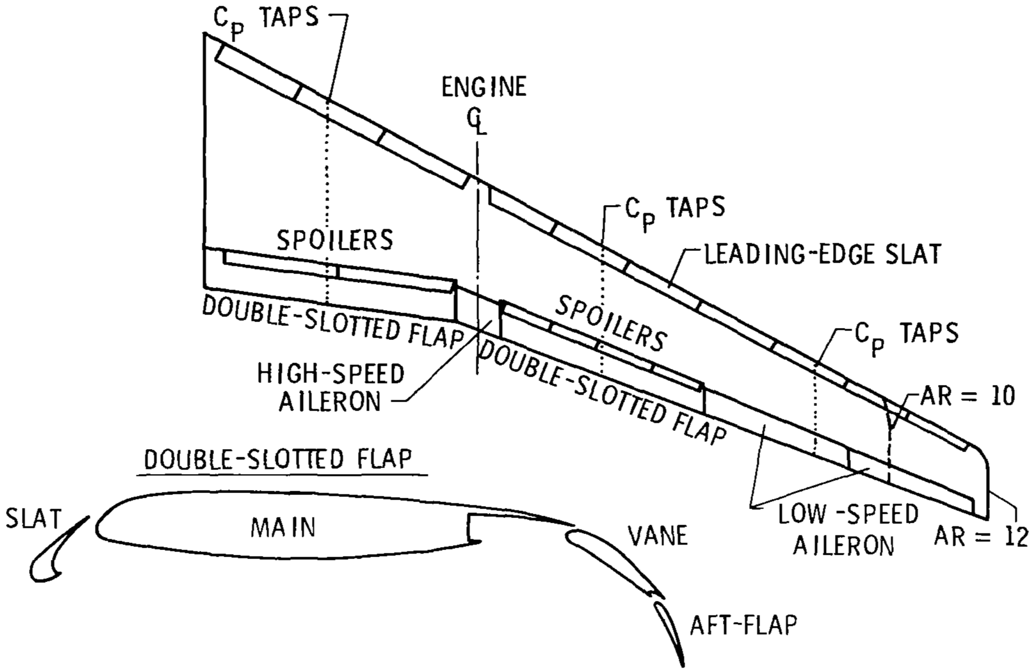


Figure 3.- High-lift supercritical wing.

HIGH-LIFT TECHNOLOGY

PURPOSE: PROVIDE PHENOMENOLOGICAL UNDERSTANDING, ANALYSIS, AND DESIGN CAPABILITY FOR THE DEVELOPMENT AND INTEGRATION OF EFFECTIVE HIGH-LIFT SYSTEMS ON CURRENT AND FUTURE AIRCRAFT

FUNDAMENTALS

- 2-D HIGH RN TEST TECHNIQUES
- 2-D PARAMETRIC STUDIES
- 2-D SEPARATED FLOW MEASUREMENTS (WICHITA STATE GRANT)
- 3-D WING INVESTIGATION

COMPUTIONAL METHODS

- SURFACE PANEL METHODS
  - 2-D BOUNDARY LAYER
  - 3-D BOUNDARY LAYER
- MCAIR CONTRACT
  - 2-D DESIGN METHOD
  - 3-D DESIGN METHOD
- 2-D MULTI-ELEMENT $C_{L_{MAX}}$ METHOD
- EFFICIENT NAVIER-STOKES EQUATIONS
- MISS STATE GRANTS
  - FAST HIGH RN SOLUTION
  - APPLIED 2-D AND 3-D N. S. SOLUTIONS

CONCEPTUAL APPLICATIONS

- PROPULSIVE LIFT CONCEPTS
- CONFIGURATION EFFECTS $\left\{ \begin{array}{l} \text{LFC} \\ \text{EET} \\ \text{SCR} \end{array} \right.$
- RN EFFECTS $\left\{ \begin{array}{l} \text{2-D EET} \\ \text{EET} \\ \text{SCR} \end{array} \right.$

Figure 4.- Elements of the NASA Langley High-Lift Technology Program.

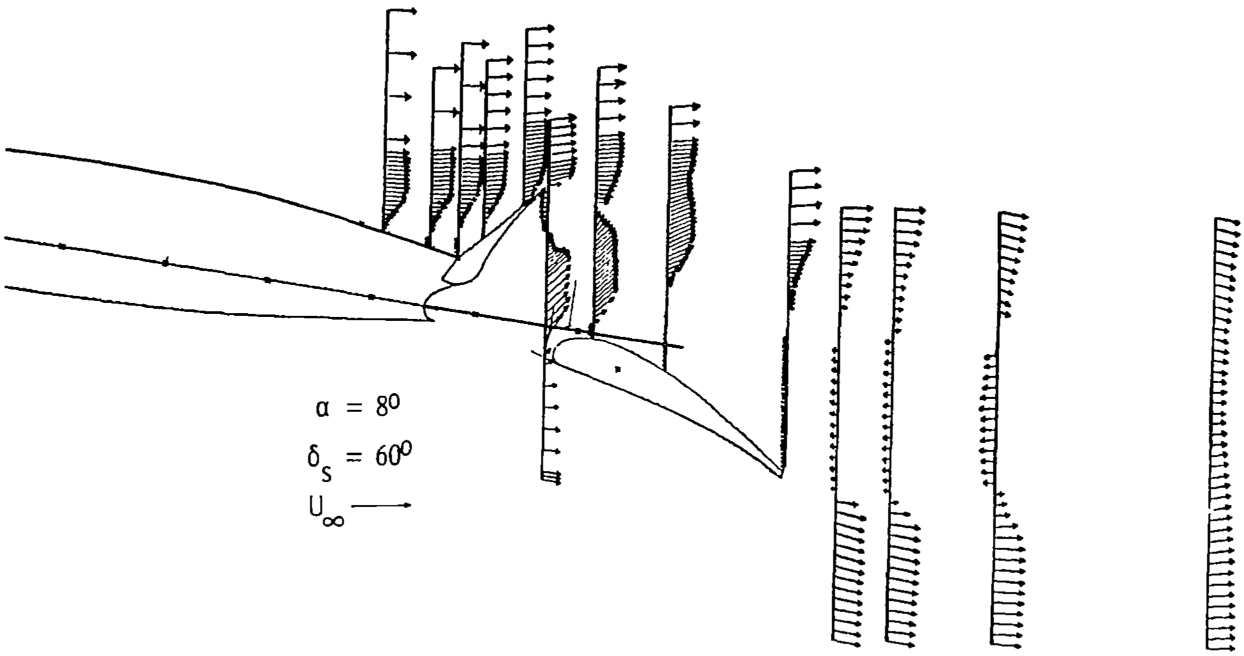


Figure 5.- Example of velocity profiles for flap and spoiler obtained using dual split film. WSU-Grant NSG-1165.

1- x 3- METER SEMISPAN WING

OBJECTIVE

PROVIDE EXPERIMENTAL BASE FOR DEVELOPMENT OF METHOD TO PREDICT THE SPANWISE VARIATION OF LIFT AND DRAG USING WAKE FLOW-FIELD MEASUREMENTS

MEASUREMENTS

FORCE AND MOMENT, WING SURFACE AND WAKE PRESSURE MEASUREMENTS, LASER VELOCIMETRY

CONFIGURATION VARIABLES

SWEEP, TRAILING-EDGE FLAPS, POWERED NACELLES

STATUS

CONFIGURATION	SWEEP 0^{0}	20^{0}	40^{0}
CRUISE	C(8/78)	C(2/80)	C(2/80)
CRUISE WITH POWERED NACELLE	C(2/80)	C(2/80)	C(2/80)
HIGH-LIFT FLAPS	C(4/79)	P(6/81)	P(6/81)
HIGH-LIFT FLAPS WITH POWERED NACELLE	P(11/81)	P(11/81)	P(11/81)

C = COMPLETE P = PLANNED

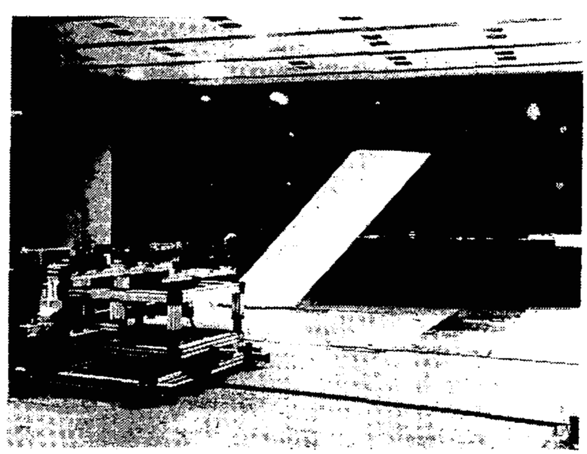


Figure 6.- Parametric study of 3-D wing conducted in Langley 4- by 7-Meter Tunnel.

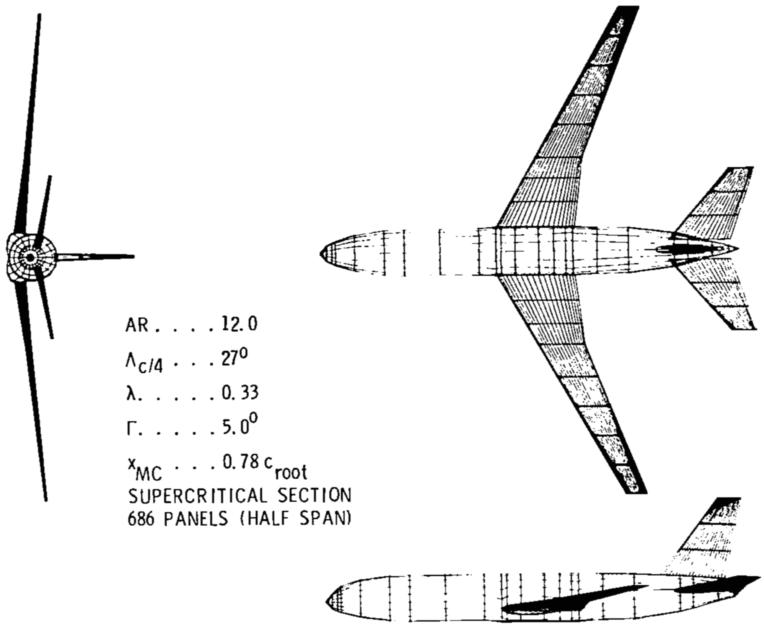


Figure 7.- Panel method representation of advanced transport.

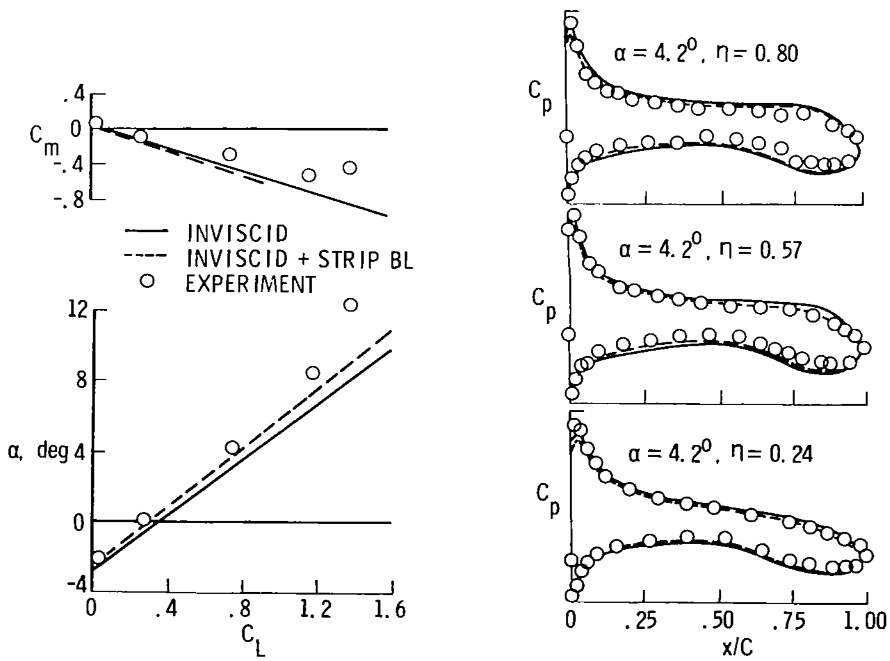


Figure 8.- Comparison of experimental results and panel method with strip boundary layer for advanced transport.

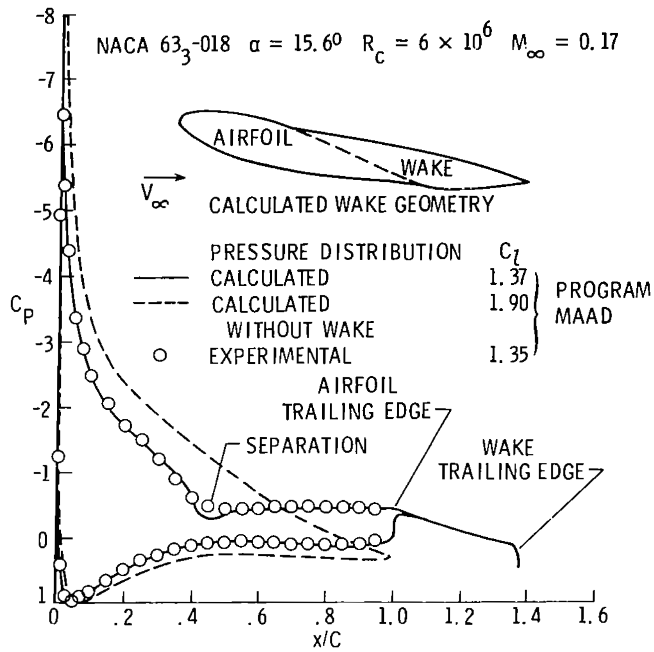


Figure 9.- Experiment/theory comparison for airfoil-wake solution method.

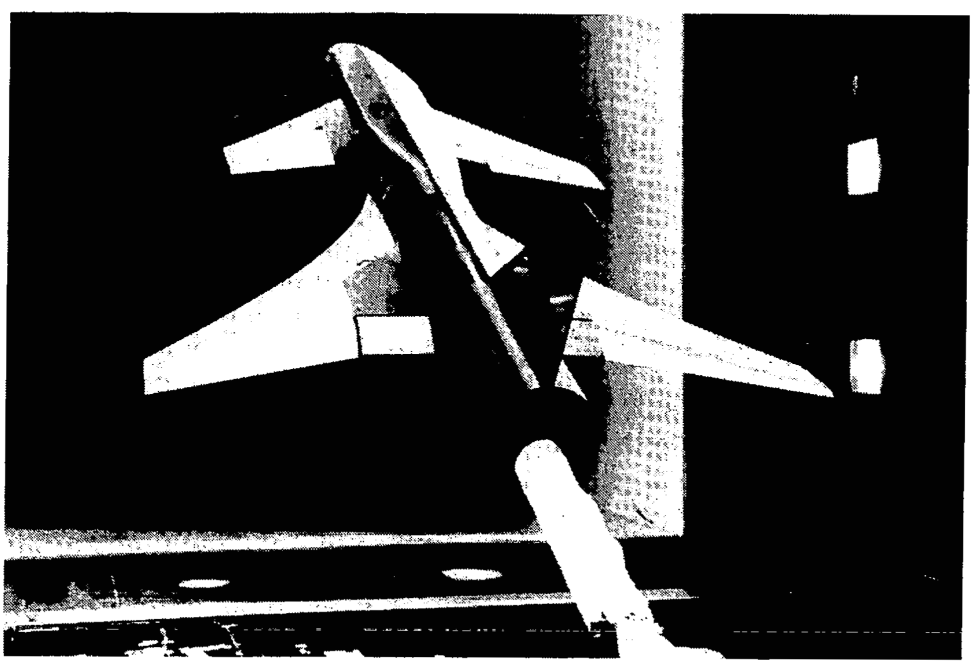
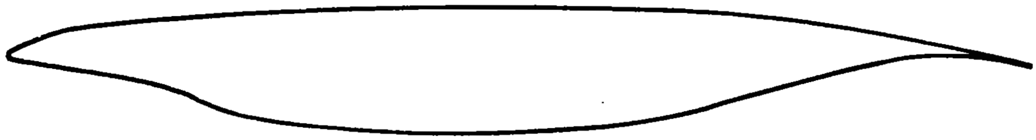
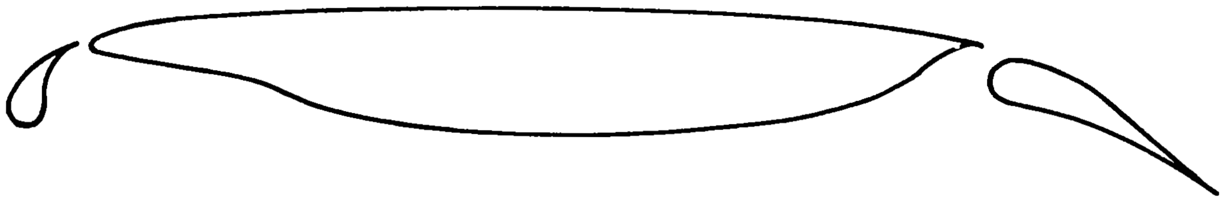


Figure 10.- Photograph of NASA-LaRC powered-lift wing-canard model mounted for tests in 4- by 7-Meter Tunnel.



(a) Cruise geometry.



(b) Low-speed high-lift geometry.

Figure 11.- Sketch of the NASA-LaRC LFC airfoil.

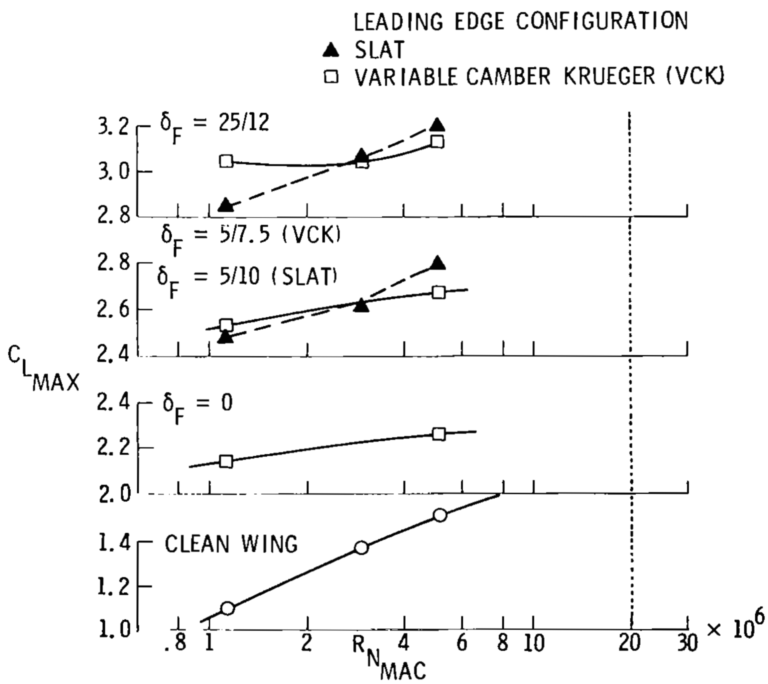


Figure 12.- Effect of R_N on $C_{L_{max}}$ for high-lift configuration (tail off). Douglas Aircraft Company contract NAS1-14744.

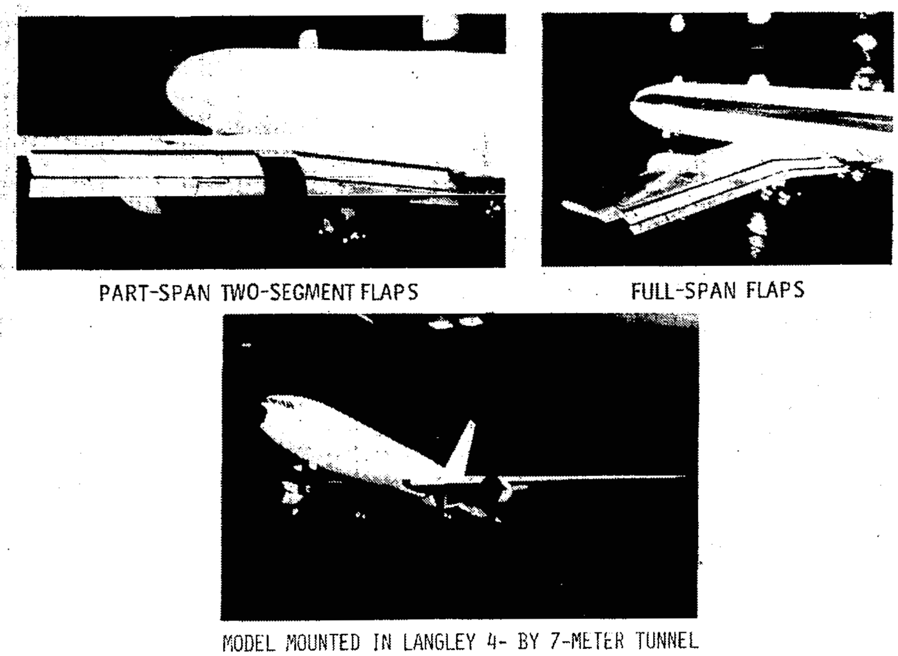


Figure 13.- Photographs of NASA-LaRC EET high-lift model.

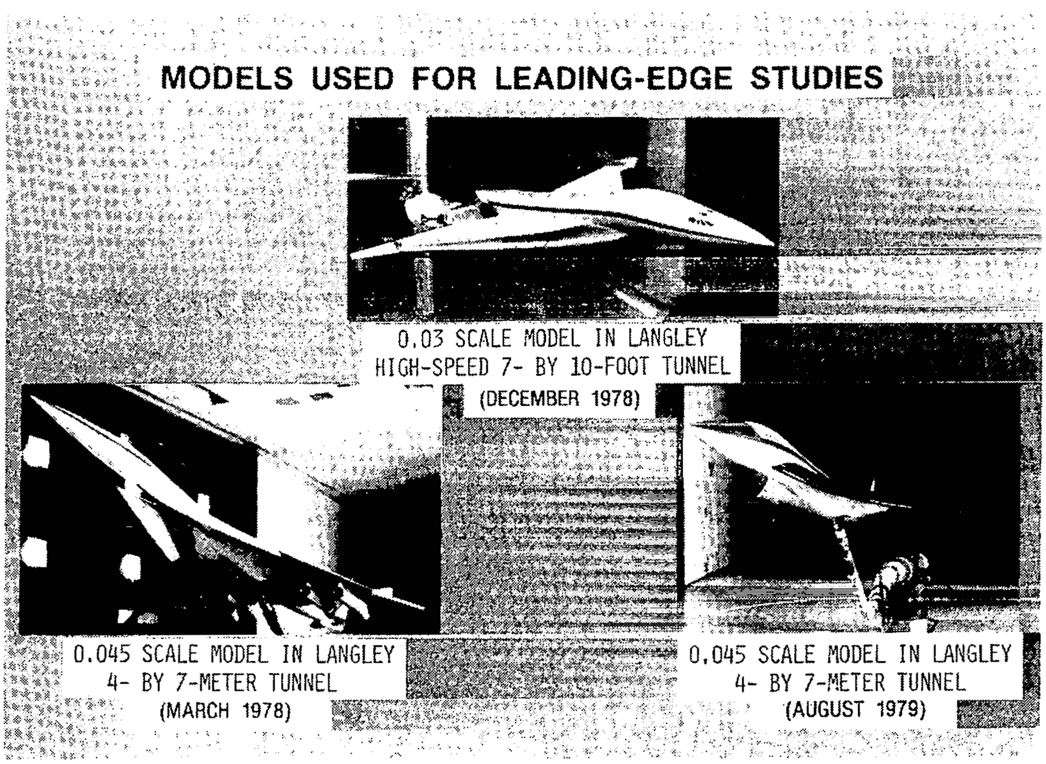


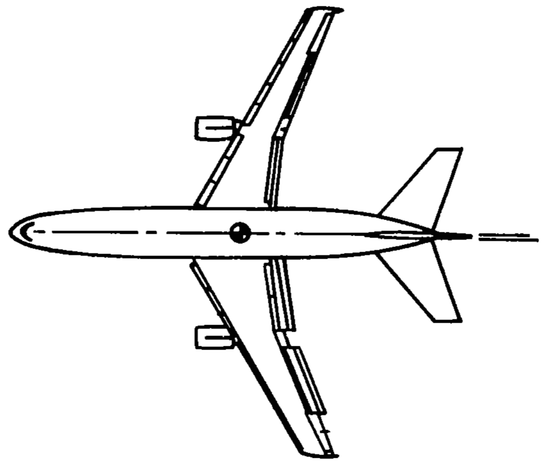
Figure 14.- Photographs of NASA-LaRC SCR high-lift models.

CONFIGURATION VARIABLES

- LEADING-EDGE FLAP
- TRAILING-EDGE FLAP
- WING CONTROL SURFACES
- REMOTELY CONTROLLED TAIL

TEST VARIABLES

- $M \approx 0.3$
- $\alpha = -6^0$ TO 26^0
- $\beta = \pm 10^0$
- $RN = 1 \times 10^6$ TO 40×10^6



MEASUREMENTS

- SIX COMPONENT FORCE AND MOMENT
- THREE ROWS OF WING PRESSURES INCLUDING FLAPS AND SLATS

MODEL DESIGN

- PATHFINDER I MODEL FUSELAGE/TAIL WITH HIGH-LIFT EET WING

Figure 15.- NASA-LaRC proposed high-lift investigation of EET/Pathfinder I configuration.

CONFIGURATION VARIABLES

- LEADING-EDGE RADIUS
- LEADING-EDGE DEFLECTION SCHEMES

TEST VARIABLES

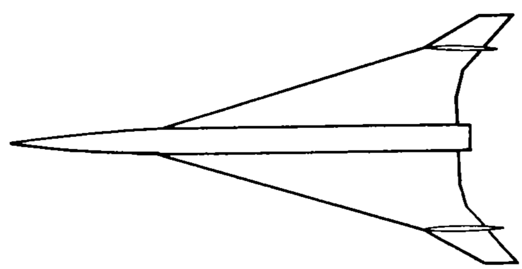
- $M = 0.3 \longrightarrow 1.2$

  $\alpha = -6^0 \longrightarrow 18^0$

  $RN_{\bar{c}} = 6 \times 10^6 \longrightarrow 80 \times 10^6$

MEASUREMENTS

- SIX COMPONENT FORCE AND MOMENT
- PRESSURES
- FLOW VISUALIZATION



EXPECTED RESULTS

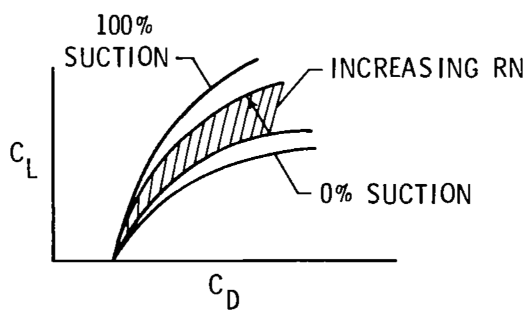


Figure 16.- NASA-LaRC proposed high-lift investigation of SCR configuration.

CONFIGURATION AERODYNAMICS PANEL

Chairman	Edward C. Polhamus NASA LaRC
Technical Secretary	Blair B. Gloss NASA LaRC

Panel Members

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Richard G. Bradley	General Dynamics
Peter J. Butkewicz	Wright-Patterson AFB
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John A. Dahlin	Douglas Aircraft Company
Bann W. Farquhar	The Boeing Company
Douglas Kirkpatrick	Naval Air Systems Command
James M. Luckring	NASA LaRC
Dal V. Maddalon	NASA LaRC
Earl A. Minter	Vought Corporation
Mark R. Nichols	GWU/JIAFS
Larry G. Niedling	McDonnell Douglas Corporation
James C. Patterson	NASA LaRC
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CONFIGURATION AERODYNAMICS

Edward C. Polhamus and Blair B. Gloss
NASA Langley Research Center

INTRODUCTION

It is anticipated that a rather sizeable portion of test time in the National Transonic Facility will be devoted to aerodynamic configuration research and design concept optimization studies applicable to the various classes of aerospace vehicles. In order to cover this broad subject adequately, the Workshop format provided for additional aircraft configuration-related panels in the technology areas of high-lift system development and aeroelasticity and flutter while the spacecraft configuration research and development requirements were covered by the Space Vehicles Panel. The Configuration Aerodynamics Panel, therefore, limited its discussions primarily to the static aerodynamic research related to aircraft-type configurations in their cruise or combat maneuver modes. The discussions were centered on three general classes of aircraft: subsonic transport aircraft, transonic tactical aircraft, and slender wing aircraft; these particular classifications tended to also group the configurations by aircraft wing planform types. The purposes of this paper are to: (1) review the status and plans of Langley's NTF configuration research program which was presented to the panel as a background for an interchange of ideas with regard to the needs and priorities in the configuration aerodynamics area, and (2) to document the main thrusts of the panel discussion and the resulting recommendations related to near-term configuration research in the NTF.

SYMBOLS

$\bar{c}$	mean geometric chord
$C_{B,D}$	dynamic wing root bending moment coefficient, $\frac{M_D}{qS\bar{c}}$
C_L	lift coefficient, $\frac{\text{lift}}{qS}$
c_p	pressure coefficient, $\frac{p_s - p_r}{q}$
f	frequency
M	free-stream Mach number
M_D	time-averaged rms value of dynamic wing root bending moment
$(n/d)_{max}$	maximum section normal force-to-drag ratio
p_r	free-stream static reference pressure

p_s local static pressure

P_t free-stream stagnation pressure

q free-stream dynamic pressure

R Reynolds number per unit length

$R_{\bar{c}}$ Reynolds number based on mean geometric chord

s distance measured along wing surface, see figure 10

S reference area

T_T free-stream stagnation temperature

V free-stream velocity

α angle of attack

Λ leading-edge sweep angle

δ flap-deflection angle

CURRENT STATUS AND PLANS

The presentation of Langley's current status and plans for utilization of the National Transonic Facility for configuration research of various classes included: (1) some of the more pertinent research areas that it was felt should be addressed in the initial phases of the NTF aircraft configuration research program, (2) a description of precursor research being carried out which is related to the NTF research program or associated with the development of cryogenic testing techniques required for NTF research, and (3) a brief description of the current plans with regard to the initial models and general testing program.

Subsonic Aircraft

Design trends related to subsonic transport aircraft, such as the application of supercritical technology and the large increases in size being considered for future aircraft, have accentuated the need for improved simulation of the aerodynamic characteristics with regard to Reynolds number. Some examples of possible future large transport aircraft types are illustrated in figure 1. Gross weights in the range of two million pounds are being considered and, for this size aircraft, cruise Reynolds numbers on the order of 100 million could be encountered. Inasmuch as it is anticipated that supercritical technology will be utilized for the wing designs, an improved simulation of shock boundary layer interactions is, of course, an important aspect that will benefit from the unique capabilities of the NTF. Related interactions include those associated with nacelles, pylons, stabilizing surfaces, controls, etc. With regard to controls, it would appear that, because of the high inertias of these large aircraft and the control sensitivity to Reynolds number, a considerable amount of research and development may be required in the aerodynamic control area. Static aeroelasticity and high speed buffet are additional areas in which the NTF will be expected to provide valuable information. In addition to the extensive research program in the Langley 0.3-Meter Transonic Cryogenic Tunnel related to cryogenic wind tunnel operational procedures and techniques, aerodynamic research related to that which will be carried out in the NTF is underway and is providing precursor type experience and data of value to the NTF program.

Precursor studies.- A high Reynolds number aerodynamic research program that is closely related to the wing design aspects of subsonic transport configurations is the airfoil program currently being carried out in the Langley 0.3-m Transonic Cryogenic Tunnel. The tunnel's maximum stagnation pressure of 6 bars and cryogenic temperature capabilities provide Reynolds number levels comparable to the full-scale chord Reynolds numbers encountered by the current large transport aircraft at their cruise conditions. A description of this facility is presented in reference 1. The major portion of the current high Reynolds number airfoil research program in this facility can be divided into three categories. First, there is a series of five NACA and NASA airfoils selected to provide both the opportunity for correlation with other facilities and to extend supercritical airfoil research data to higher Reynolds numbers than those obtainable in other facilities. The second series consists of one Langley Research Center advanced transport airfoil design and four industry-designed airfoils to be tested as part of the NASA-Industry Energy Efficient Transport program. This series is providing some of the prospective NTF users in the aircraft industry with cryogenic model construction and testing experience as well as broadening the high Reynolds number airfoil research program. The third series consists of two foreign-designed transport type airfoils that will be tested under a recent exchange agreement.

Although limited to two-dimensional flow, these tests should provide valuable information in the following areas: (1) Reynolds number effects on shock/boundary-layer interactions; (2) guidance to future subsonic transport configuration research in the NTF; and (3) cryogenic testing and model construction experience. An example of some preliminary results from a subsonic transport type airfoil recently obtained as part of the 0.3-m transonic program is presented in figure 2 where the parameter, $M(n/d)_{max}$, an approximation to the range performance parameter, is presented as a function of Mach number for a range of Reynolds numbers. These preliminary data illustrate the range of conditions that are being covered in the study.

The cryogenic pressure tunnel has the unique capability to essentially isolate static aeroelastic effects from both Reynolds number and compressibility effects, and the ability (through the control of the speed of sound) to match flight conditions over the altitude range with a single static aeroelastic model (see ref. 2). These unique testing capabilities are of particular importance for subsonic transport aircraft with their relatively flexible, high-aspect-ratio swept wings and Reynolds-number-sensitive supercritical airfoils. Although large model stresses will be generated when the high levels of stagnation pressure available in the NTF are required, the unique advantage of the NTF in providing a given Reynolds number at one-fifth of the model loads encountered in an ambient temperature pressurized air tunnel of the same size should not be overlooked.

Defining the flight conditions for which the onset of wing buffeting occurs and establishing the intensity as the buffet region is penetrated is also an important role envisioned for the NTF and one in which the cryogenic tunnel again offers unique capabilities. In addition to those capabilities cited above, advantages such as, for example, increases in the ratio of aerodynamic to structural damping and control of the reduced frequency are afforded by the ability to control temperature over a wide range. A program to develop the technology required to utilize wing root strain gage techniques for buffet-onset tests under cryogenic conditions has recently been successfully carried out and is reported in reference 3. An example, taken from reference 3, of some precursor buffet tests conducted in the Langley 0.3-m Transonic Cryogenic Tunnel on the rectangular wing having a transport-type airfoil section is presented in figure 3. In this figure $C_{B,D}$, the root mean square average of the dynamic portion of the wing root bending moment, is plotted as a function of angle of attack. A range of Reynolds numbers obtained by various combinations of tunnel stagnation pressures and temperatures is presented and illustrates the pronounced effect of Reynolds number on the angle of attack at which

<u>NTF plans</u>.- The initial NTF subsonic transport-type aircraft model, designated Pathfinder I, is a representative wide-body transport configuration incorporating an advanced high-aspect-ratio supercritical wing. The configuration has a low wing and tail and has a design lift coefficient of approximately 0.55 at a cruise Mach number of 0.82. The wing has a 9.8 aspect ratio based on the trapezoidal planform area of 0.183 m^2 (1.98 ft^2) (including fuselage intercept) and a 1.345-m (52.97-in) span. The total area of the wing including the leading- and trailing-edge extensions is 0.214 m^2 (2.30 ft^2). The wing has 5^o dihedral and the supercritical airfoil has a thickness-to-chord ratio of 0.145 at the side of the fuselage, 0.12 at the geometric break and 0.106 at the tip. The quarter chord line of the trapezoidal planform has a sweep of 30^o. The model will be instrumented with a six-component strain gage balance, static pressure orifices in the wing, thermocouples, buffet gages, and accelerometers. This model is illustrated in figure 4 and is the same configuration, with a few minor exceptions, as a recent NASA Langley supercritical transport design developed in the Langley 8-Foot Transonic Pressure Tunnel.

The Pathfinder I model can be used for several areas of investigation in the NTF. Since the wings are removable, the Reynolds number effects on several types of wings can be established, and an assessment made of the validity of aft grit location techniques used in existing tunnels to simulate high Reynolds number conditions. Also, the effect of Reynolds number on buffet onset can be established for high-aspect-ratio wings. Nacelles can be added to the wings, although no designs currently exist, to investigate the effect of Reynolds number on nacelle interference. Because of the replaceable (or removable) components of Pathfinder I (see fig. 4), the effect of Reynolds number on forebody forces and induced flow fields, tail surface interactions, and model support effects can be investigated.

The initial wing of the Pathfinder I model does not have a provision for aileron type controls. However, a semispan wing model (being fabricated for unsteady pressure studies and described in ref. 4) planned for testing in the NTF could provide some initial research on control effectiveness at high Reynolds number. This model, shown in figure 5, is representative of the subsonic transport class of wing.

Transonic Tactical Aircraft

The second general class of aircraft configurations considered is the tactical fighter aircraft designed to provide high levels of maneuverability in the high subsonic and transonic speed regime while providing some supersonic dash capability. The wings of this class of aircraft tend to have moderate sweep and aspect ratio. The highly-swept low-aspect-ratio-wing tactical aircraft of the supercruiser type will be addressed in the section on slender wing aircraft.

Some of the viscous-dependent flows that can be encountered on these transonic tactical aircraft can be envisioned with the aid of figure 6, which illustrates two configuration types of current interest. Typical of these concepts are the relatively short-coupled lifting surfaces designed for favorable interference and involving both supercritical and vortex flows. The basic shock boundary layer interactions can be complicated further by the merging wing and fuselage boundary layers for the forward-swept wing configurations. The high angle-of-attack maneuver requirements involve strong viscous cross-flows on the long forebodies resulting in large forebody forces and vortex flows which can interact with the empennage and the inlet. The NTF capability will also be utilized to establish the effectiveness of flow control and maneuver devices at, or near, full scale Reynolds numbers and to define the buffet and static aeroelastic effects as described in the previous section. The unique aeroelastic testing capability is also expected to provide for simulation of aeroelastic tailoring concepts for a wide range of flight conditions.

Precursor studies.- Experience currently being gained as part of a Langley study of the application of advanced transonic theories to the design of aircraft having high levels of transonic maneuver performance is being utilized in the selection and design of the first NTF transonic tactical aircraft research models. The two configurations of the current Langley study are shown in figure 7. These transonic maneuver research configurations were designed for a $M = 0.9$, $C_L = 0.86$ maneuver condition using three-dimensional transonic theory in a design-by-analysis mode described in reference 5. These configurations are being utilized to establish the level of maneuver performance that supercritical flow technology can provide for these extreme conditions, recognizing, of course, that some type of variable-geometry adaptive wing concept would be required to apply the benefits without penalizing other performance aspects. The aft-swept wing configuration has been constructed and tested and currently is being "tuned" with the aid of pressure measurements coupled with additional theoretical studies. The forward-swept wing is designed with the inclusion of the canard flow field effect and the model is under construction.

NTF plans.- The first NTF maneuvering aircraft model, designated Pathfinder II, is in the preliminary design stage and is similar to the aft-swept configuration of figure 7. The configuration has a midwing designed for a lift coefficient of 0.86 at a Mach number of 0.90. The wing has an aspect ratio of 3.13 based on the trapezoidal planform area of 0.1394 m^2 (1.5 ft^2) and a 0.66-m (2.17-ft) span. The wind has zero dihedral and a leading-edge sweep angle of 45°. The model will be instrumented with a six-component strain gage balance, static pressure orifices in the wing and fuselage, thermocouples, buffet gages and accelerometers. It is planned to provide for the testing of alternate fuselage forebodies to study the effects of Reynolds number on high angle-of-attack cross-flow effects.

Slender Wing Aircraft

In recent years, there has been an increased interest in slender wing aircraft, and some of the classes of aircraft expected to utilize this type of wing are illustrated in figure 8 with derivatives from basic delta, double-delta and arrow planforms. The tactical supercruisers and the supersonic cruise transports require these highly swept wings to provide the desired high levels of supersonic aerodynamic efficiency, while the so-called "stealth" aircraft may utilize relatively high sweep angles and sharp leading edges to reduce radar detection and to reduce gust response during the low altitude penetrations. Many of the research areas already discussed in relation to the other aircraft types apply to these slender wing concepts and will not be repeated. However, a type of flow which tends to be more characteristic of the slender wings is the leading-edge vortex flow. Research in this area deals with both the suppression of the vortex flow for cases where it is undesirable and its optimization when vortex lift characteristics are beneficial. An example of the application of vortex flow in design is the transonic maneuver condition for a tactical supercruiser. At the high-maneuver angles of attack required, it is unlikely that attached flow can be maintained along the highly swept leading edges and, therefore, approaches are being developed for utilizing the vortex to provide the optimum combination of vortex lift and leading-edge thrust recovery. Some examples of this type of design are presented in references 6 and 7. For the stealth type of aircraft where conventional high lift systems are relatively ineffective as well as undesirable, application of vortex lift for takeoff and landing is a possibility under investigation. Regarding the supersonic cruise transport class of aircraft, leading-edge separation and the resulting vortex flow are likely to be encountered at the 2.5-g maneuver condition and, thereby, will influence the structural design.

Several features of these vortex flows are sensitive to viscous effects which cannot, at present, be theoretically modeled, and the NTF will, of course, provide valuable information. The origin of the primary vortex feeding sheet and its feeding rate are sensitive to Reynolds number, especially for the rounded leading-edge case and for the secondary vortex regardless of the leading edge condition. Both the symmetric and asymmetric vortex breakdown characteristics, which play an important role in high angle-of-attack stability, are Reynolds number dependent.

Precursor studies.- A strong aerodynamic research program relative to leading-edge vortex flows has been underway at Langley for several years, covering the development of theoretical analysis and design methods, experimental verification studies, and applied research directed towards the design of a variety of vortex maneuver lift concepts. The major portion of the configuration design research has been associated with wings having sharp leading edges for which the vortex feeding sheet remains fixed at the edge, and for this vortex flow design condition the major features of the flow are not critically dependent on Reynolds number.

As part of the preparation for utilization of the NTF to extend the research to the more general Reynolds-number-dependent vortex flows, both experimental and theoretical studies are underway.

One such study deals with the effect of Reynolds number and Mach number on the characteristics of the leading-edge vortex developed on a rounded leading-edge wing, and utilizes the F-111 TACT aircraft. Detailed wing pressure distribution measurements were made over the forward portion of an inboard wing station, indicated in figure 9. Also shown in figure 9 are the test conditions covered in the Reynolds number - Mach number envelope, and it will be noted that Reynolds numbers as high as 50×10^6 (based on the MAC) were obtained over the transonic speed regime. A sample of the upper-surface pressure distributions is presented in figure 10 for an angle of attack of 6^o and a Mach number of 0.6, and it is of interest to note that strong Reynolds number effects are still present on this relatively sharp leading edge in the Reynolds number range above 20 million. Complete suppression of the vortex flow did not occur until a Reynolds number of 40 million was reached. In addition to illustrating the need for the NTF Reynolds number capability for configuration research and development in the area of leading-edge vortex flows, this study will assist in the definition of meaningful experiments for the NTF.

Regarding leading-edge vortex breakdown, a theoretical study has been initiated for the purpose of developing a three-dimensional theoretical model to account for the interaction between the outer flow field and the inner viscous core of the vortex. If it is successful in correlating existing vortex breakdown data, this method will be utilized to establish definitive full scale Reynolds number experiments in the NTF to improve the understanding of vortex breakdown and to develop design criteria for controlling vortex breakdown.

Wing buffet associated with vortex breakdown has also been studied as part of the cryogenic tunnel buffet measurement technique development program described earlier. An example of some preliminary results obtained in the 0.3-m Transonic Cryogenic Tunnel (ref. 3) is presented in figure 11. The buffet coefficient $C_{B,D}$ extracted from the unsteady signal of the root bending moment gage is plotted as a function of angle of attack for a 65^o delta wing having sharp leading edges. The Mach number was 0.35 and the Reynolds number, based on the mean aerodynamic chord, was 4.8×10^6. To investigate the unique capability of the cryogenic tunnel in which control of the speed of sound can be utilized to provide a range of reduced frequencies $(f\bar{c}/v)$ while maintaining constant Mach and Reynolds number, data for two values of reduced frequency are presented. Also indicated is the angle of attack at which vortex breakdown has reached the wing trailing edge as obtained from the study of reference 8. The buffet measurements indicate that the onset of buffet coincides with vortex breakdown at the trailing edge and that the intensity, which increases as the vortex breakdown moves towards the wing apex, is highly dependent upon the reduced frequency. The demonstrations of the capability of utilizing a conventional buffet technique in a cryogenic wind tunnel, and of the unique capabilities with regard to controlling various buffet parameters, along with the research results obtained, should be of value in assuring efficient use of the NTF for buffet studies utilizing simple root bending gages.

NTF plans.- Although definite plans for a transonic NTF model representative of a slender wing aircraft design have not yet been made, some basic fluid dynamic studies relating to the slender wing configurations are planned. Currently, plans are being made to design relatively simple delta wings to study Reynolds number effects on leading-edge vortex flows. These wings will have removable leading edges so that both sharp leading-edged wings and rounded leading-edged wings can be investigated. Pressure instrumentation will be installed in both the upper and lower surfaces with the vast majority of orifices being located in the upper surface. Results from advanced panel methods which model the leading-edge separation-induced vortex flows (see ref. 9) can then be evaluated with high Reynolds number test results and the importance of the secondary vortex and vortex breakdown determined for full-scale conditions. Also, the onset of buffet for this type of wing can be studied at high Reynolds numbers, using either strain gages installed near the wing root, accelerometers, or highly responsive pressure instrumentation.

PANEL DISCUSSION AND RECOMMENDATIONS

Prior to establishing recommendations relative to detailed research programs for the NTF, the configurations aerodynamic panel reached a consensus regarding the types of configuration design concepts that would be considered for the near term research. These configuration design concepts are illustrated in figure 12. In the transport aircraft area, it was recommended that a slender-wing supersonic transport configuration be considered as part of the near term research program and coordinated with the high-lift research supersonic transport model. As companions to the Pathfinder II, three additional types of tactical aircraft configurations were recommended for early inclusion in the NTF research program. The first configuration would utilize a transonic wing having a supercritical-type airfoil, designed for attached flow both at the transonic cruise and transonic maneuver conditions, and utilizing variable geometry to approximate a good transonic camber distribution for the transonic maneuver. This model would probably be of the same planform as Pathfinder II and would utilize leading and trailing edge flaps to approximate the Pathfinder II maneuver design shape.

Considerable interest was expressed in Langley's current research directed toward application of various vortex-lift concepts to maneuvering aircraft of both the moderately swept transonic wing type and the slender-wing type, and the panel recommended that a near-term extension of the NTF program include both types of wings. The transonic wing configuration would utilize wing warp designed for optimum supercritical attached flow in the high subsonic cruise and would be capable of generating vortex lift for maneuver. Various vortex-lift concepts, including both passing and augmented types such as spanwise blowing and fluid strakes, would be included in the program. The slender-wing tactical aircraft research would center on a supersonic cruise class of configuration typical of those of current interest which utilizes a derivative of a highly swept double-delta type wing. A supersonic-cruise design warp would be utilized with vortex lift flaps used to provide transonic maneuver capability.

Subsonic Transport Aircraft

During the discussion of the Pathfinder I program, the panel expressed some concern that this first NTF model would not be designed to allow research into the Reynolds number effects on wing flap and control effectiveness. The Langley representatives pointed out that this was due to the aerodynamic loads encountered at the upper test Reynolds number and suggested that after the planned testing was completed, control study capability would be added to the model. Wing flap and control effectiveness tests could then be made at stagnation pressures of the same order of magnitude as current tunnels while experiencing a factor five increase in Reynolds number over the current tunnels' Reynolds number range, due to the cryogenic capability of the NTF. In view of this, the Pathfinder II research program was divided into two phases. The first phase, with the simplified model, will be tested to the design Reynolds number of the model and the second phase, with controls installed, will be tested over a reduced, but creditable, Reynolds number range. A general outline of the research areas recommended for coverage with the Pathfinder I model is as follows:

PHASE I:

(1) Establish Reynolds number effects on configuration with clean wing

(a) Performance
(b) Stability
(c) Buffet onset

(2) Establish pure aeroelastic effects

(3) Evaluate boundary-layer trip methods used in existing facilities to simulate full-scale Reynolds number

(4) Investigate support interference effects at high Reynolds numbers

(5) Perform correlation tests in another aircraft development facility

(6) Study wall interference effects (small model at same M, $R_{\bar{c}}$ and q as large model)

PHASE II:

(1) Using the modified model at reduced dynamic pressure, study effect of Reynolds number on:

(a) Aileron and flap effectiveness
(b) Spoiler characteristics
(c) Flap support fairing effects
(d) Nacelle-pylon-wing interference

Transonic Attached Flow Wing - Tactical Aircraft

As mentioned earlier, the panel expressed an interest in the extension of the transonic (nonslender) wing tactical aircraft configuration research to include wings designed for transonic cruise which incorporate either passive or jet-augmented vortex lift for maneuvering. While the panel was in general agreement with the NTF research program plans regarding the various flow phenomena to be investigated, it was felt that the program should include a study of the effect of Reynolds number on both external store carriage characteristics and support interference in the high angle-of-attack range of interest for maneuvering aircraft. An outline of the major items to be covered in the proposed program is as follows:

(1) Basic Reynolds number studies of the wing flows

(a) Attached flow
(b) Separated and vortex flows
(c) Investigation of boundary-layer trip techniques

(2) High angle-of-attack forebody cross-flow effects

(3) Component interference

(4) External store characteristics

(5) Transonic maneuver characteristics

(a) Variable geometry for attached flow
(b) Vortex maneuver lift
- passive
- jet-augmented

(6) Basic study of leading-edge suction sensitivity

(7) High angle-of-attack model support interference

Slender Wing Tactical Aircraft

For the slender wing tactical aircraft of the supersonic cruise type, a configuration having a wing designed for the supersonic cruise and utilizing vortex-lift flaps for the transonic maneuver requirements is envisioned for the near-term NTF program. Some recommended research items are as follows:

(1) Reynolds number sensitivity of subsonic/transonic cruise performance, stability, control, and aerodynamic loads

(2) Basic Reynolds number studies of leading-edge vortex flow

  (a) Leading-edge radius and camber effects
  (b) Primary and secondary vortex characteristics
  (c) Symmetric and asymmetric vortex breakdown
  (d) Vortex-lift devices

Propulsion Integration

Although the NTF plans for the initial testing equipment do not include the procurement of propulsion simulation equipment, tunnel shell penetrations and model support strut passages are such as to permit the transport of gaseous nitrogen to either a model mounted on a sting or to a side-wall-mounted semi-span model. The passage in the model support strut was sized to accommodate a pipe or pipes that would allow 35.4 kg/sec mass flow rate at a sting-mounted model having a nozzle pressure ratio of 8 with the tunnel stagnation pressure of 8.8 bars and a Mach number of 1.00. The panel was asked to review anticipated needs in the area of high Reynolds number propulsion research to aid in the selection of future propulsion simulation equipment for the NTF. To accomplish this in the time allotted, a subpanel was formed from the membership of the Configuration Aerodynamics Panel. The comments which follow were formulated by the propulsion sub-panel.

The propulsion system problems which the subpanel highlighted as particularly sensitive to Reynolds number effects tend to be associated with subsonic transports, fighters and V/STOL-STOL aircraft. The major area of interest for the subsonic transport class of aircraft is the nacelle-strut interference on the wing performance. The effects of Reynolds number on forebody flow fields affecting inlet conditions, boattail and base pressure, boattail and base drag, and flow fields around highly integrated airborne/propulsion configuration for the fighter-type aircraft also require study. Lastly, the effects of Reynolds number variation on the stability, control, and performance of V/STOL-STOL type aircraft warrant investigations in the NTF. To accomplish these studies the subpanel suggested several NTF propulsion system requirements. Both flow-through and powered simulators for subsonic transport and logistic aircraft will be needed to properly investigate the Reynolds number effects on propulsion systems/airframes. Further, a capability to simulate the jet/afterbody flow for high-speed aircraft, V/STOL and STOL aircraft with thrust vectoring and/or reversing, and spacecraft thrust augmentors will be needed.

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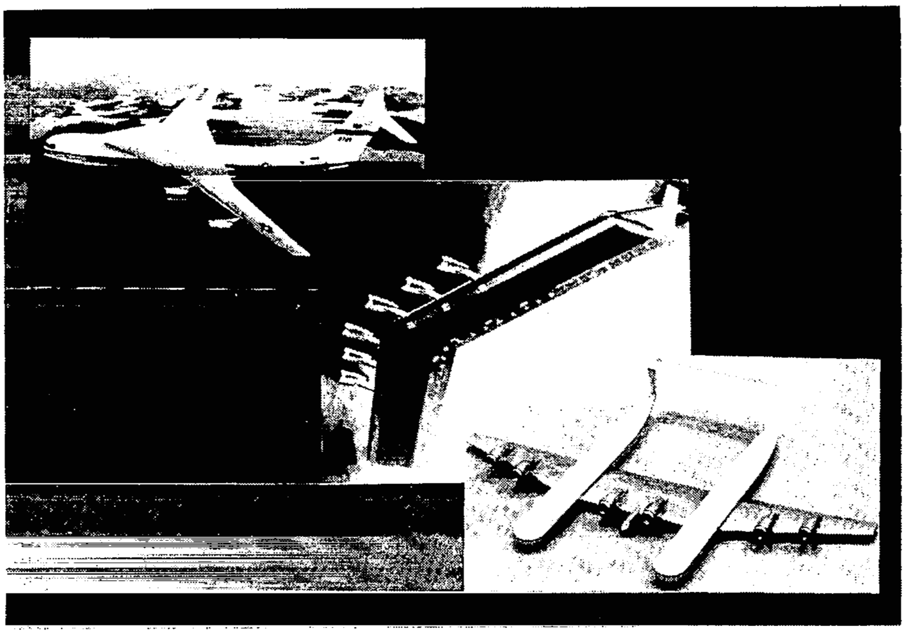

Figure 1.- Large transport aircraft.

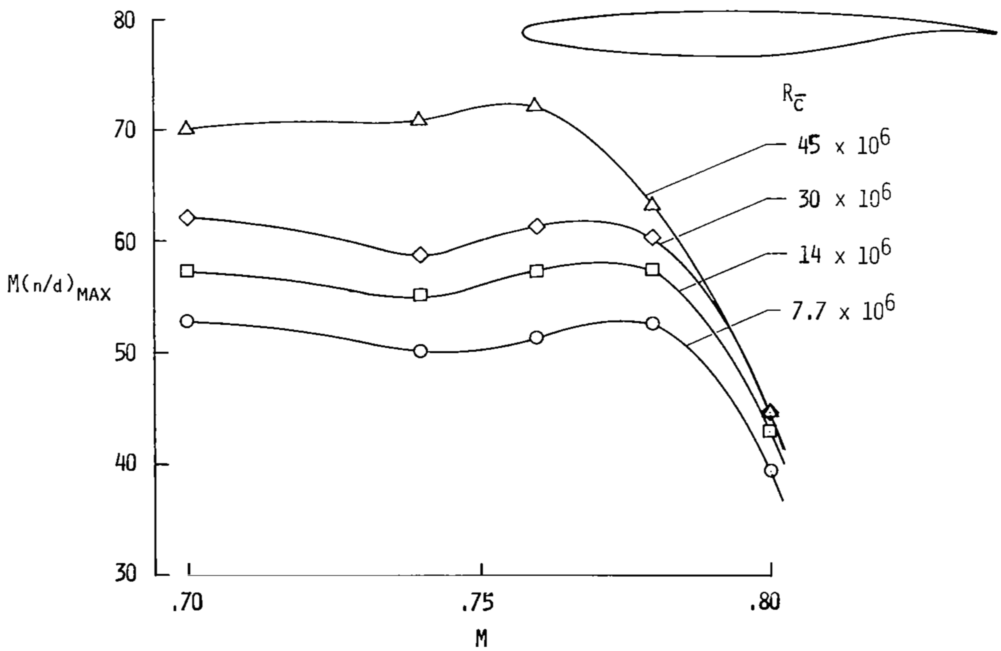

Figure 2.- Some preliminary results from the 0.3-m Transonic Cryogenic Tunnel airfoil program.
$\bar{c}$ = 15.24 cm.

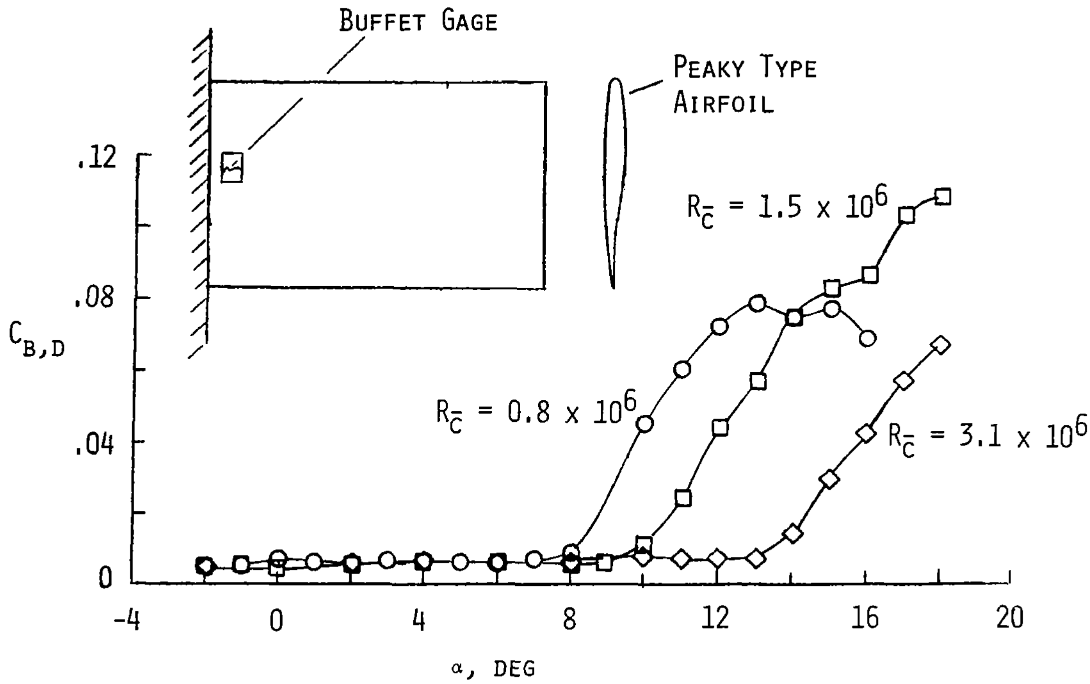

Figure 3.- Effect of Reynolds number on buffet onset for a rectangular wing with a peaky type airfoil.
M = 0.35; $\bar{c}$ = 10.16 cm; semi-span = 15.29 cm.

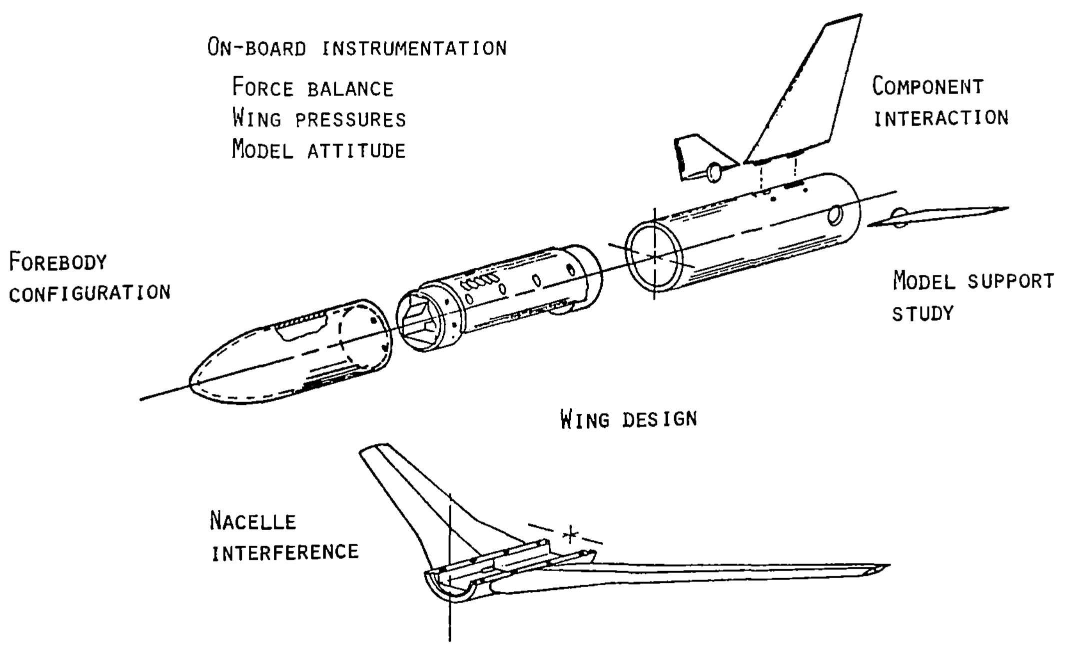

Figure 4.- Configuration research capability of the NTF Pathfinder I model.

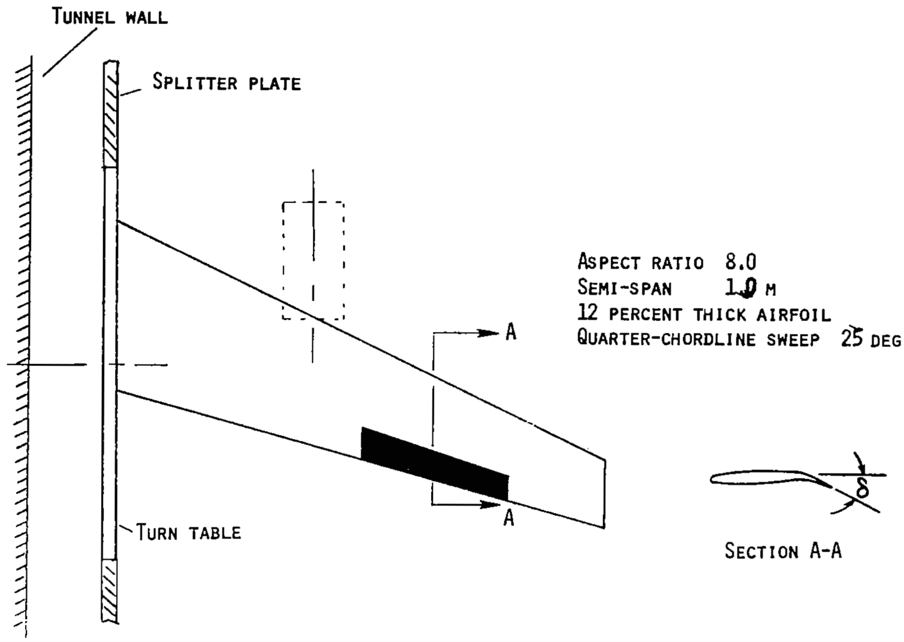

Figure 5.- NTF semi-span model.

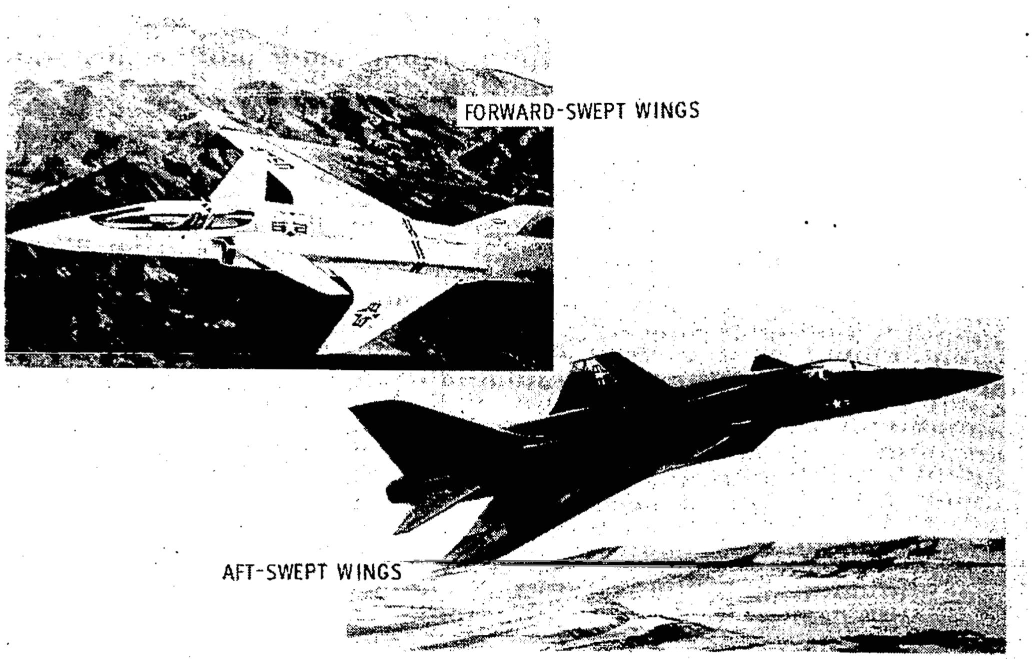

Figure 6.- Tactical aircraft configurations.

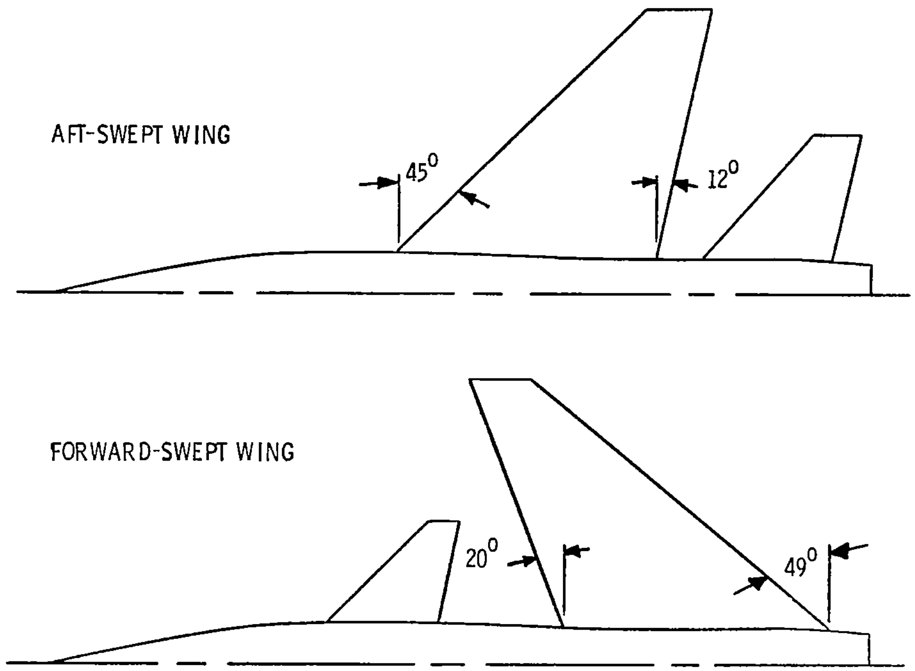

Figure 7.- Current transonic tactical research models.
$M = 0.90$; $C_L = 0.86$.

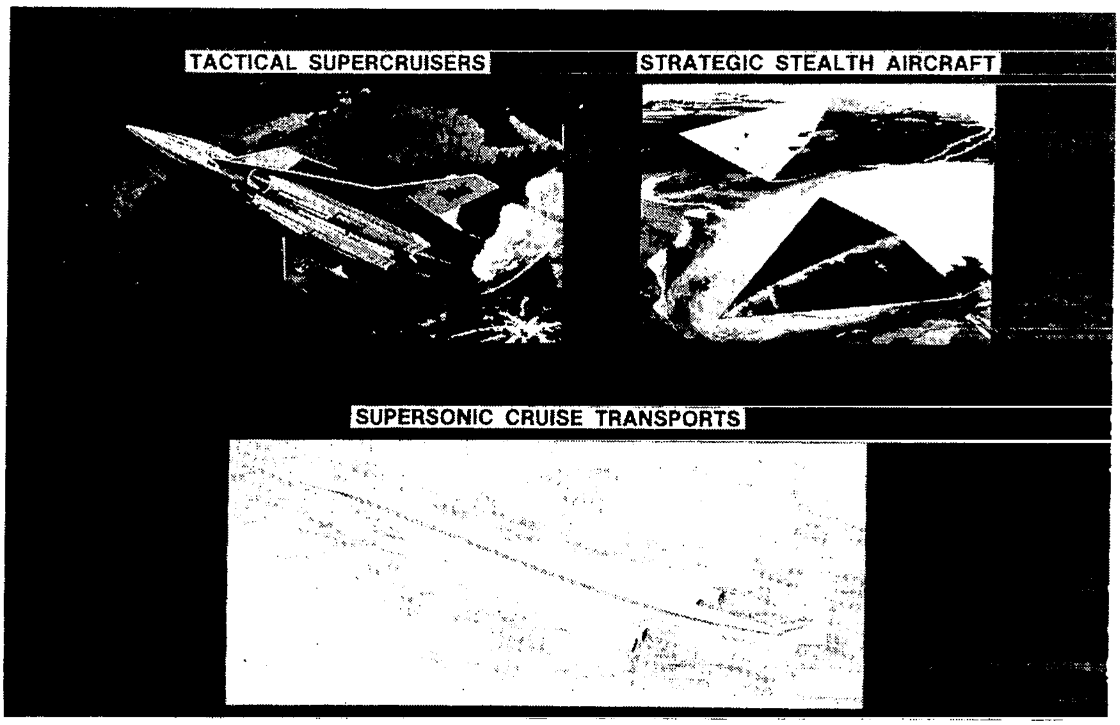

Figure 8.- Slender wing configurations.

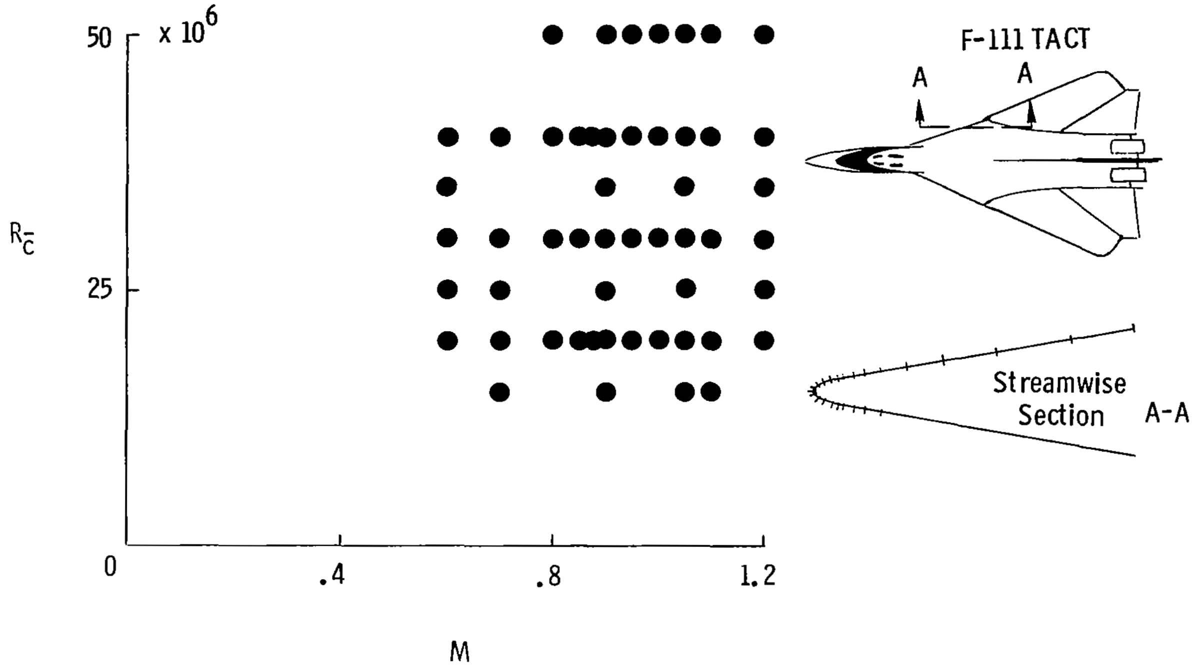

Figure 9.- Flight test envelope for a high-Reynolds-number leading-edge vortex experiment.

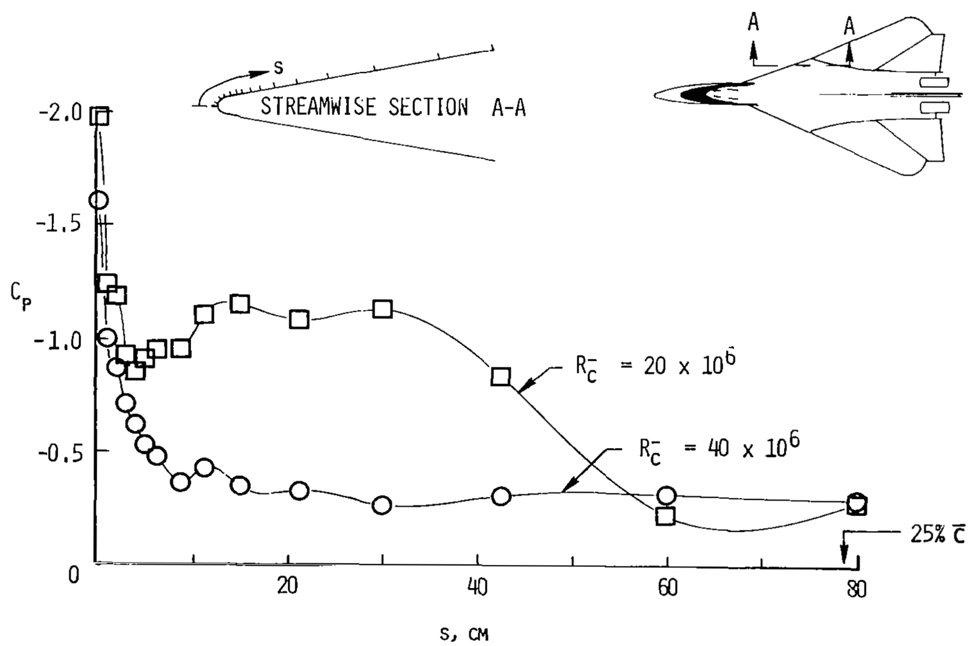

Figure 10.- Reynolds number influence on vortex flow development - TACT flight experiment.
$\alpha = 6^o$; M = 0.6.

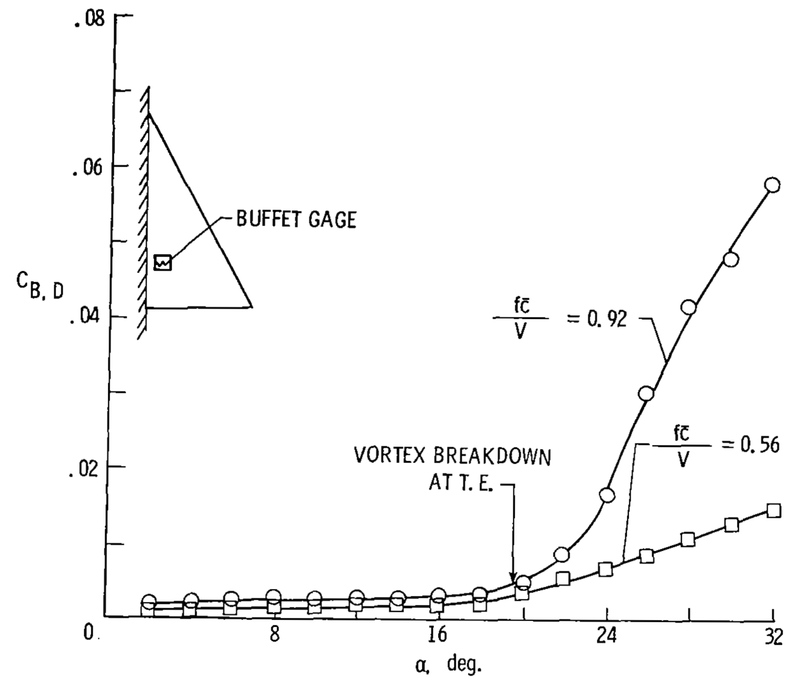

Figure 11.- Some preliminary buffet data from the 0.3-m TCT tests on a $\Lambda = 65^o$ delta wing.
$M = 0.35$; $R = 35.4 \times 10^6$ per meter; $\bar{c} = 0.1355$ m.

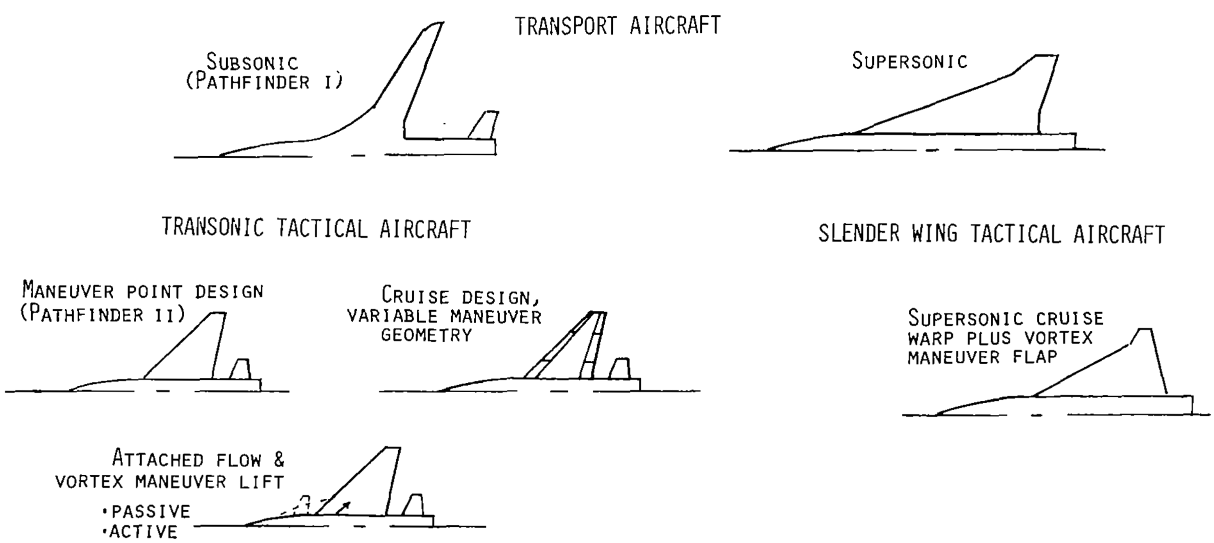

Figure 12.- Recommended configuration design concepts for NTF research programs.

AEROELASTICITY AND UNSTEADY AERODYNAMICS PANEL

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Joe J. Nishikawa	The Boeing Company
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Joseph R. Stevenson	Rockwell International
Paul B. Tucker	McDonnell Douglas Corporation
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REPORT OF THE PANEL ON AERELASTICITY AND UNSTEADY AERODYNAMICS

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NASA Langley Research Center

INTRODUCTION

The program title of this panel was "Aeroelasticity and Flutter." However, since flutter is a "sub-set" of aeroelasticity (as are "buffet" and "divergence") and since the scope of the panel's charter included unsteady aerodynamics, it is considered appropriate to refer to the present panel as indicated in the title of this report. In the previous workshop on high Reynolds number research (ref. 1) the report of the panel on dynamics and aeroelasticity, drawing on some of the special features of the Langley Research Center Transonic Dynamics Tunnel (TDT) which was designed specifically for dynamic aeroelasticity testing, recommended certain features that ought to be considered in the design of the National Transonic Facility (NTF) for similar testing. That panel also recommended certain studies in the areas of aeroelasticity and unsteady aerodynamics that ought to be performed, and briefly discussed the benefits and difficulties of conducting those types of studies in the NTF. More recently, some in-depth considerations have been given to the testing of dynamically scaled flutter models in the NTF (ref. 2). The status of these earlier recommendations and the additional considerations of the present panel will be reviewed in this report.

DESIRABLE SPECIAL FEATURES FOR THE NTF

The previous panel recommended that for studies of dynamic aeroelasticity and unsteady aerodynamics certain desirable features ought to be considered in the design of the NTF. These included a rapid tunnel shutdown capability, a means for directly viewing flutter models, added protection of the fan blades from model debris, a sidewall model mount with a pitch oscillation capability, a "cold room" capability for determining model vibration characteristics at cryogenic temperatures, and a data system capable of acquiring dynamic data.

The panel understands that most of these features are being incorporated. For extra fan blade protection a removable honeycomb is being designed for installation at a later date. Rapid tunnel shutdown capability will be provided by quick-response controllable fan guide vanes; however, the resulting rate of change in tunnel dynamic pressure has yet to be established. A "cold room" capability is planned although details of how model vibration characteristics (mode shapes, frequencies, and damping) will be determined have not yet been worked out. This is one of the areas of concern to the panel relative to conducting flutter tests in the NTF. Typically several months of measurements are required to determine the validity of scaled flutter models at the home lab before they are brought to a remote facility for testing. Airframe manufacturers are not likely to have cold room facilities at their lab, so that much of the model validation tests may have to be accomplished at the NTF. Some inefficiency would be anticipated in off-site remedial changes to the model.

The panel was pleased to learn that the data acquisition system is being designed with the capability for handling dynamic data, but there were some reservations about having only a 14-analog-channel permanent capability; however, the permanent capacity can be augmented by add-on equipment as required. Also, it is considered unfortunate that requirements for integrity of the pressure shell have precluded the capability for direct view of the flutter models.

RECOMMENDED STUDIES

Generally, the recommended studies of the previous panel are endorsed except for reordering the priority and adding an element to one of the study areas. The four specific topics which the panel recommends be initially programmed for tests in the NTF (or, preferably as precursor work in the 0.3-m Transonic Cryogenic Tunnel) are, in order of priority:

1. Transonic unsteady aerodynamics

2. Control surface unsteady aerodynamics

3. Buffet onset and loads

4. Flutter

Each of these topics will be addressed relative to the current status and the panel's observations.

Transonic Unsteady Aerodynamics

The objective here would be to evaluate the effects of Reynolds number on the unsteady pressures measured on oscillating airfoils and wing planforms as a function of Mach number, mean angle of attack, and oscillation frequency and amplitude. The panel considered this area first priority rather than third priority as established by the previous panel. In addition, it was recommended that the studies include both low- and high-aspect-ratio wings and external stores. The ground work has begun for the first studies in the NTF in this area. A sidewall model mount capable of oscillating a relatively high-aspect-ratio wing model in a pitching motion is being designed. This conceptual design will lead to a final system design that will be capable of oscillating the wing at amplitudes up to ±1/4 degree at frequencies up to 100 Hz. The high-aspect-ratio wing model (fig. 1) is being built by Lockheed-Marietta under contract to the Air Force Wright Aeronautical Laboratory for testing initially in the NLR tunnel in the Netherlands, and later in the NTF. After the NLR tests in April 1981 the model will be sent to the NASA Langley Research Center for reinstrumentation and refurbishment for the NTF tests. It is anticipated the model will be available for testing in the NTF in 1983. The model is being designed so that later a pylon-mounted nacelle and wing tip extensions can be added. The panel strongly endorsed this study and emphasized the need to consider configurations with external stores.

Control Surface Unsteady Aerodynamics

Reynolds number effects are important in control surface aerodynamics due to boundary-layer growth toward the trailing edge and interaction with shocks. The objective of the first studies is to quantify these effects and to obtain unsteady aerodynamic force, moment, and pressure measurements due to control surface motion at flight Reynolds numbers. No definite plans have been made for these studies other than to develop appropriate instrumentation which will be applicable to all oscillating or unsteady pressure tests and to design the oscillating wing model just described so that an oscillating control can be added later relatively easily.

Buffet Onset and Loads

The objective of these studies is to establish the significance of Reynolds number effects and aeroelastic effects separately on buffet onset and intensity change with Mach number and angle of attack. Preliminary studies have been conducted by Richmond Boyden of the Langley Research Center in the 0.3-m Transonic Cryogenic Tunnel. Some of the results have been reported in reference 3, and there are plans to extend these studies by testing similar carbon fiber composite models to be provided by the British Royal Aircraft Establishment in the 0.3-m tunnel.

Flutter

Some in-depth consideration has been given to the testing of dynamically scaled flutter models in the NTF and for studying the effects of Reynolds number on flutter using simple "trend" models (ref. 2). The results, some of which will be discussed later, highlight the benefits and limitations of the NTF for this type of testing. However, no actual "precursor" flutter studies have been made in the 0.3-m Transonic Cryogenic Tunnel. The panel was not critical of the progress in this area although the desirability of ascertaining the effects of Reynolds number on flutter was emphasized. However, there was little enthusiasm for testing dynamically scaled aeroelastic models of actual aircraft in the NTF because of anticipated complexities, high cost, and potential damage to the tunnel. As a matter of fact the testing of scaled aeroelastic models in the NTF was considered to be the type of testing that should not be done unless there was a reasonably clear indication that accurate results could be obtained only through the use of the unique capabilities of the NTF. An example of this situation might be the dynamic instability phenomenon encountered by the B-1 prototype which has been hypothesized to be a shock-induced, self-excited wing bending oscillation with significant Reynolds number dependence. The phenomenon is discussed by J. R. Stevenson (ref. 4).

Even with the unique capabilities of the NTF, determining the effects of Reynolds number on flutter is not straightforward. The problem is that alterations in tunnel state conditions (pressure and/or temperature) to change Reynolds number at a given Mach number also produce significant changes in the important flutter parameter, mass-density ratio. The dimensionless flow parameters that are important in simulation studies are:

$\frac{V}{a}$ Mach number, M (1)

$\frac{b\omega}{V}$ reduced frequency, k (2)

$\frac{m}{\rho b^2}$ mass density ratio, μ (3)

$\frac{Vb}{\nu}$ Reynolds number, RN (4)

$\frac{V^2}{bg}$ Froude number, F (5)

where

a	fluid free-stream speed of sound
V	fluid free-stream velocity
ρ	fluid free-stream density
ν	fluid free-stream coefficient of kinematic viscosity
g	acceleration due to gravity
b	characteristic length
ω	characteristic oscillation frequency
m	body mass per unit length

From these basic similarity parameters other dependent ratios relating model quantities to full-scale quantities may be derived. If these dimensionless parameters have the same values for the model and the full-scale aircraft and the mass, stiffness, and, to a lesser degree, the damping distributions are the same for the model and full-scale aircraft, then the flexible and rigid body response or behavior of the model will be similar to the aircraft providing the model is geometrically similar to the aircraft, orientation to the airflow is similar to that of the aircraft, and the model is supported in a manner that does not significantly affect the model response or behavior.

Froude number is important only for cases where deflections of the model due to gravitational forces are significant. Although recognized as important, the Reynolds number of the model and full-scale airplane cannot generally be matched in conventional tunnels because of tunnel performance and/or size limitations, and so RN effects have been approximately accounted for by various other means, or in the case of flutter tests, ignored.

By use of the relationship of the speed of sound to the gas constant, ratio of specific heats and temperatures, and Sutherland's law relating viscosity and temperature and the equation of state, the reduced frequency, mass-density ratio, and RN at a given Mach number can be expressed in terms of static pressure P and temperature T for a given model as follows:

$$k \propto \frac{1}{T^2}$$

$$\mu \propto m\left(\frac{T}{P}\right)$$

$$RN \propto P\left(\frac{T + 114}{T^2}\right)$$

That is, Reynolds number can be changed independently of reduced frequency (by holding the test temperature constant) but cannot be changed independently of the other important flutter parameter, mass-density ratio. The effect of Reynolds number variation on relative mass-density ratio (proportional to the reciprocal of the density) for a given model is shown in figure 2. The data are for Reynolds numbers of 6.1×10^6 and 30.5×10^6 per meter (20×10^6 and 100×10^6 per foot) for three selected Mach numbers. The vertical lines at the ends of the curves for each Mach number represent the approximate limits of the NTF dynamic pressure capability for those Mach numbers at each Reynolds number shown. The figure shows that significant variations in mass-density ratio occur with changes in Reynolds number and that a five-fold change in Reynolds number is far too great for an assessment of Reynolds number effects on flutter because of the minimal "overlap" in Mach number for that range of Reynolds number. Thus determining the effects of Reynolds number on flutter, even in the NTF, is not as simple as might be assumed. This problem is addressed in reference 2, where it is proposed that a fortunate characteristic of the metals magnesium, aluminum, and steel can provide a solution to this dilemma. That is, the ratio of the moduli of torsional and bending elasticity to the material density is essentially constant so that geometrically similar solid models made of these materials will have the same natural frequencies and therefore will flutter at the same mass-density ratio for a given Mach number. The densities of aluminum and steel are respectively approximately 1.56 and 4.48 times that of magnesium. Therefore, if the magnesium wing is tested at a $RN = 20 \times 10^6$ and the steel wing at $RN = 89 \times 10^6$ the mass-density ratio at the flutter dynamic pressure and Mach number will be the same for all three Reynolds numbers (except for possible Reynolds number effects). This is illustrated in figure 3, which shows the estimated flutter boundaries (solid lines) of three simple solid metal models (rectangular planform, cantilevered wings with a 25.4 cm-(10 in.) chord, 101.6 cm-(40 in.) span, and a 65A009 airfoil). The dashed lines are loci of tunnel stagnation pressure for several Mach numbers that produce the constant Reynolds numbers indicated. The flutter dynamic pressures (stagnation pressures) were estimated using preliminary design empirical methods based on a "flutter index," $F.I. = \frac{V_F}{b\omega\sqrt{\mu}}$, which experience has shown accounts for variations in mass, stiffness, size, and flutter velocity V_F of wings with geometrically similar shapes, centers of gravity, and aerodynamic centers. Possible effects of cryogenic temperatures on the stiffness and damping characteristics of the models have not been considered. To minimize loads problems the models should be tested at zero lift conditions. In conducting the tests, as the stagnation pressure is increased at a constant Mach number to increase the dynamic pressure until flutter is reached, the stagnation temperature is adjusted to maintain the RN constant.

These same models may be used to evaluate Reynolds number effects on buffet loads by measuring model response and damping at constant values of the mass-density ratio parameter at the same Reynolds numbers used for the flutter study, but at dynamic pressures at each Mach number that are below those that would produce flutter or overload the model for the maximum angle of attack to be used. The procedure would entail setting the tunnel to the desired RN, M, and q and measuring the model response and damping as angle of attack is changed from low to high values.

This kind of flutter/buffet study should be among the first studies to be scheduled in the NTF (or preferably as precursor studies in the 0.3-m Transonic Cryogenic Tunnel) since the results will bear directly on the question of the necessity or desirability of conducting more sophisticated "flutter clearance" and buffet loads tests in the NTF.

CONCERNS

The panel had several concerns which may be grouped under two general categories: (1) model verification procedures in a cryogenic temperature environment and (2) test methods. For flutter studies, the panel expressed concern over the ability to verify scaled aeroelastic model stiffness and vibration characteristics (frequencies, damping, mode shapes) under cryo temperature conditions. For tests of scaled flutter/buffet models in conventional tunnels these characteristics are verified at the contractor's shop or lab and the model is altered if necessary to obtain the desired frequencies, etc. This process can take several weeks or even months. At the tunnel site the models usually are checked again only to verify that no significant changes have occurred because of the disassembly, transportation, and assembly processes. The contractor or model builder will not have the capability to make the necessary scaling verification measurements under cryo conditions at the home lab and so they must be made on site at the tunnel. This is inconvenient for model alterations and expensive for the contractor. Indeed, one of the concerns is that the techniques for making the required verification measurement at cryo temperatures have not been defined. The panel felt that information relative to the "cold-room" and the capabilities for model calibration there (stiffness, strength, vibration frequencies, damping, etc.) should be generated and disseminated. It was observed that flutter models will not be able to meet strength factors of safety greater than 1.5 for all model components and obviously there must be an exception to the stated requirement that all models must be shown to be free from flutter up to two times the maximum anticipated tunnel dynamic pressure. That is, facility requirements for flutter model integrity should be established and disseminated.

Even for "non-flutter" models the stiffness properties will need to be established for model deformation considerations and to verify predicted flutter conditions. For tests involving high dynamic pressures all models should be considered to be potential flutter models. A suggestion was made that it would be helpful to users if guidelines were developed that indicated estimated flutter dynamic pressures for a variety of solid metal models as a function of wing thickness ratio, aspect ratio, and sweep angle. This could be a formidable task.

Relative to test methodology, the panel felt unsure of how the NTF would actually be operated for flutter tests. The usual method of testing flutter models is to change tunnel flow conditions very slowly to approach the flutter boundary cautiously, whereas the control of the NTF is oriented to getting "on-point" as rapidly as possible. Also, since it has been indicated that special screens are being designed to be installed for flutter testing it would seem likely that an attempt would be made to group flutter tests "back-to-back" to minimize screen installation and removal time.

The panel supports vigorous efforts to develop pressure and acceleration instrumentation for dynamic measurements and an appropriate dynamic data acquisition system. Several panel members were disappointed with the stated intent to have only a 14-analog-channel permanent capability for dynamic tests. It was noted, however, that these permanent channels can be augmented by auxiliary equipment when required.

Finally, the panel concurs with the observations of several other panels that a means of flow visualization and of measuring model deformations is needed, and would only add that the capability to handle <u>unsteady</u> flow conditions and <u>dynamic</u> deformations should be included as design goals.

CONCLUDING REMARKS

Most of the recommendations of the Panel on Dynamics and Aeroelasticity of the first workshop (ref. 1) relative to special features that should be included in the design of the NTF for unsteady aerodynamics and dynamic aeroelasticity studies have been incorporated.

The recommended studies of the previous workshop panel are endorsed, but with the priority changed to give first priority to transonic unsteady aerodynamics. Also the unsteady aerodynamics studies should be expanded to include a low-aspect-ratio wing and to include wing external stores. Flutter studies in the NTF are of low priority. Initial studies should determine significance of RN on flutter using simple models.

The separation of RN effects from the recognized important flutter parameter mass-density ratio will be difficult even in a cryo tunnel because RN cannot be changed independently of mass-density ratio. A potential solution to this problem makes use of similar models having different mass densities and correspondingly different stiffnesses.

Concerns of the panel include ill-defined model verification and calibration procedures (vibration, strength, and fatigue characteristics) in a cryo temperature environment, flutter test methodology, timely development of instrumentation for unsteady pressure and acceleration measurements, and timely development of flow visualization and model deformation measurement capabilities (static and dynamic).

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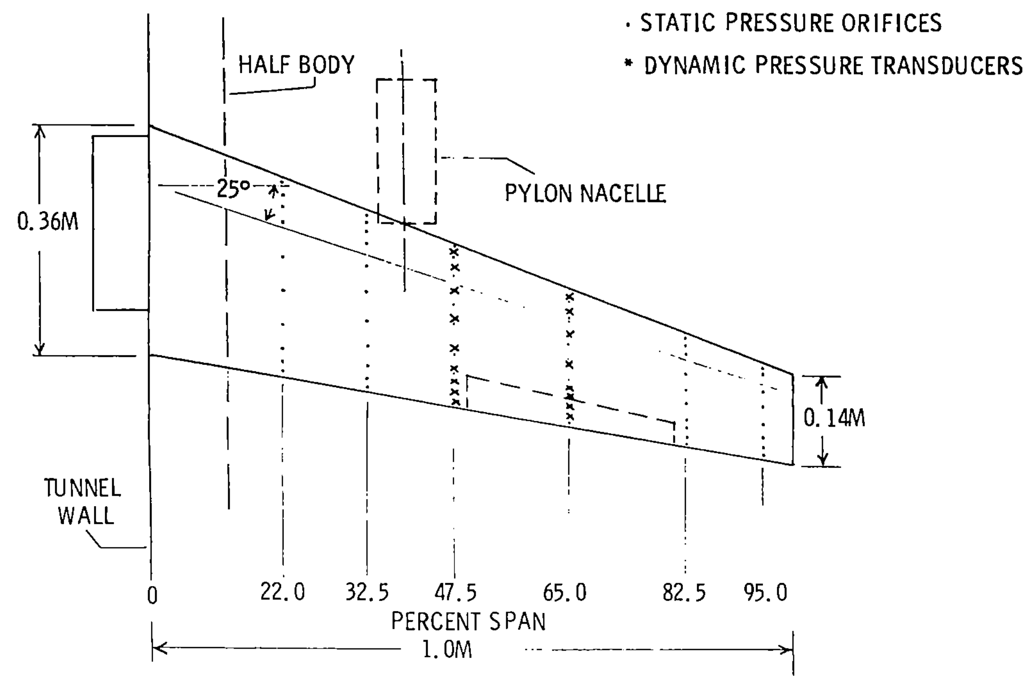

Figure 1.- "LANN" wing configuration.

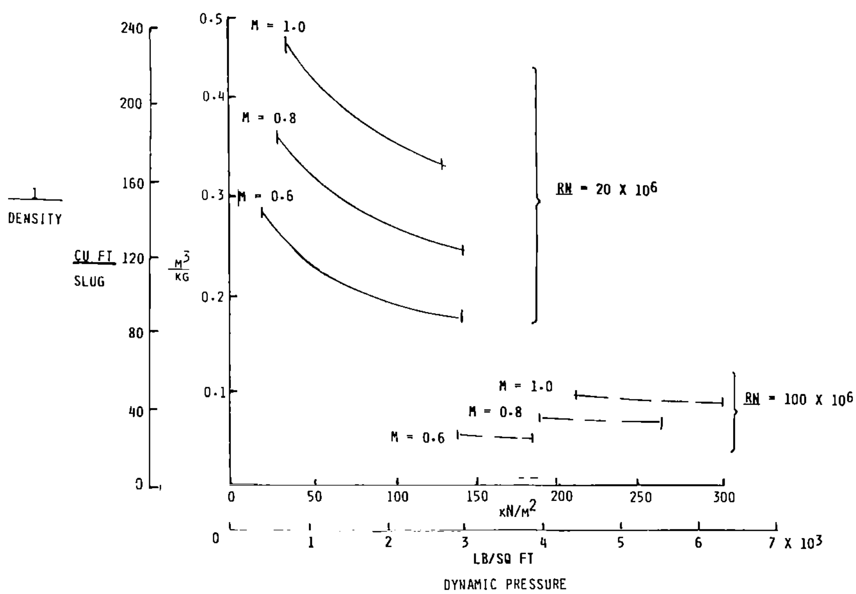

Figure 2.- Effect of Reynolds number variation on mass-density ratio.

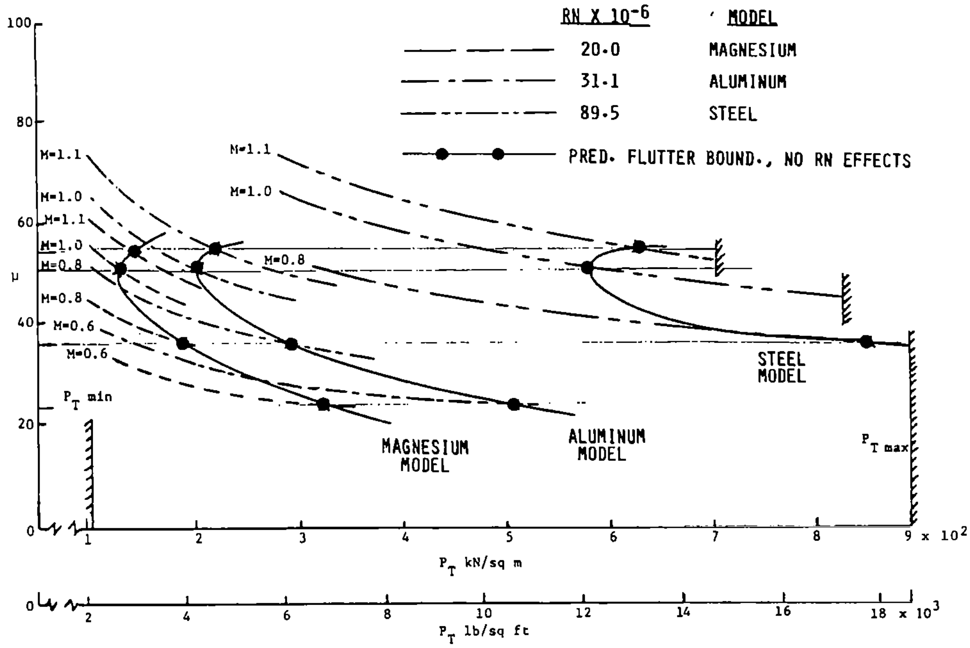

Figure 3.- Effect of change in Reynolds number on mass-density ratio.

WIND TUNNEL/FLIGHT CORRELATION PANEL

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REPORT OF THE WIND TUNNEL/FLIGHT CORRELATION PANEL

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INTRODUCTION

The 1976 Workshop on High Reynolds Number Research (ref. 1) recognized the fact that a significant portion of the National Transonic Facility test time will be devoted to providing data on configurations intended to operate at Reynolds numbers beyond those attainable by present production wind tunnels. It is therefore appropriate that a separate panel addressing wind tunnel/flight correlation be included in the present workshop as well as in future deliberations pertaining to the National Transonic Facility. It is anticipated that this panel will provide a focus for establishing user confidence in the ability of the NTF to adequately simulate the flight environment.

In an attempt to define what needs to be accomplished to establish user confidence, it is important to look back and reflect on what has already been done and what experience has been gained. It can generally be assumed that there are as many interpretations of the word correlation as there are instances of correlatable data. It is therefore important that a definition or definitions of the word correlate be identified at the present time in order to provide a basis from which the Wind Tunnel/Flight Correlation Panel can address needs. These definitions reflect discussions with several industry colleagues, a cursory search of the literature, and my own experience from having been involved extensively in both wind tunnel and flight research.

For the purpose of this discussion, then, correlation has three definitions. The first is simply an attempt to compare wind tunnel and/or theory with flight results. These data are generally obtained with little or no prior consideration of correlation and no follow-on tests to explain anomalies. The results are simply noted and reported. This is the most common method of attempting correlation. Much of the data obtained from early flight tests fall into this category. An example, shown in figure 1, is the X-15 fuselage base pressure coefficients. It is obvious from this figure that significant differences exist at transonic speeds between wind tunnel and flight data even for a relatively simple configuration with no power effects. More recent attempts at this type of correlation can be identified in the HiMAT and AD-1 research vehicles. It is also interesting to note that in many instances this type of correlation is attempted by individuals who were not involved in either the wind tunnel or flight tests.

The second definition of correlation is one encompassing a detailed study of the total drag of a vehicle as determined from wind tunnel and flight tests. This implies that one has very carefully controlled and defined both the wind tunnel model and full-scale airplane geometry. It also implies an understanding of the propulsion system, loads and aeroelasticity, trim effects, and all other pieces which influence the full-scale vehicle characteristics. This is an extremely difficult task and our experience to date at transonic speeds has seen only limited success as evidenced by the XB-70 data shown in figure 2.

Finally, correlation can be defined as attempting to understand the fundamental mechanisms of fluid flows associated with aircraft components in specific areas of the flight environment. The C-141 wing shock problem and the F-102 and F-106 base drag are examples of this type of correlation. Some other examples which come to mind are the previously completed pressure distribution measurements obtained for the F-8, TACT, and T-2C Supercritical Wing. A more recent example would be the results obtained for the KC-135 Winglets Project (fig. 3). In this instance the necessary data were obtained to understand the anomalies which occurred at the winglet root section ($\eta = 1.01$). Boundary-layer measurements, spillage, nozzle and afterbody drag, and local skin friction also fit this category. An example of one of these more fundamental experiments, shown in figure 4, is the recently completed wind tunnel/flight correlation of transition Reynolds number. The flight results for this experiment undoubtedly represent some if not the most accurate and critical measurements yet obtained. Special consideration was given to angles of attack and sideslip measurements as well as Mach number, altitude, and turbulence. As a result, these data represent the standard from which wind tunnel measurements can be judged. This same hardware is now being used to provide wind tunnel and flight data for the purpose of correlating and validating theoretical methods for predicting leeside separation at angle of attack. It is a view shared by the writer and many others, as evidenced by the 1976 workshop, that this third category of wind tunnel/flight correlation should be the one concentrated on in the near future.

Operating costs and performance requirements associated with high development and energy costs as well as intense foreign competition are forcing designers to extract the utmost in efficiency from new and/or derivative aircraft. These requirements are manifested in complex systems to provide relaxed static stability and active controls, both flight and propulsion, which continue to push aircraft designs near the limit. As a result, stability and control characteristics are becoming just as important as drag. This means that one must predict with reasonable certainty the stability levels and control requirements for future aircraft. The data shown in figures 5 and 6 were assembled and correlated in an attempt to establish criteria for estimating the possible error band for the Space Shuttle Orbiter vehicle. It should be obvious from these results that the ability to predict characteristics of vehicles with low or negative stability levels is not well in hand.

There will probably continue to be aircraft configurations which under certain flight conditions will exceed even the R capability of NTF, as shown in figure 7. However, we soon will have the capability with NTF of addressing problems at or very near the full scale R of most current aircraft. The Configuration Aerodynamics Panel of the 1976 Workshop clearly discouraged attempts at absolute drag correlation (fig. 8). That panel also made some specific recommendations, one of which was the transition R experiment. The other recommendations were to establish experiments for correlating pressure distribution and section drag as determined by wake meausrements. I believe that it should be the task of the Wind Tunnel/Flight Correlation Panel to start from the 1976 Workshop recommendations and define the kinds of experiments required to provide the necessary confidence to the user that NTF can simulate the flight environment.

PANEL CONSIDERATIONS AND RECOMMENDATIONS

The Wind Tunnel/Flight Correlation Panel discussed the validation of the National Transonic Facility (NTF) in the following order:

(1) Basic tunnel calibration

(2) Establishing confidence in the tunnel

(3) Areas of concentration

(4) Recommendations

The first three items relate to experimental studies which the panel believed to be of primary importance to the aerospace community in determining industry utilization of the NTF. The fourth item addresses an approach to validating the facility.

Basic Tunnel Calibration

The panel was unanimous in its concern for providing a complete calibration of the facility prior to conducting any R & D tests. This calibration should include a complete mapping of the test section including total and static pressure measurements in the longitudinal, lateral, and vertical planes. Dynamic measurements should be made to define the frequency and spectra for turbulence definition and scaling and acoustic environment. The influence of cryogenic operation on these measurements as well as flow angularities needs to be established. Early consideration should be given to conducting tests with the existing 10-degree cone hardware. This hardware has been used to obtain transition Reynolds number data from some 23 wind tunnels throughout the United States and Europe. In addition, flight tests were conducted with the identical hardware to establish the free-air data base for assessing wind tunnel turbulence effects. As was the case with the Fluid Dynamics Panel, the Wind Tunnel/Flight Correlation Panel identified the desirability of providing a longitudinal heat-transfer measurement capability in the cone experiment for NTF. While the 10-degree cone experiment is important in the tunnel calibration, it is equally important to establish unit Reynolds number and heat-transfer effects on the transition location for lifting surfaces. For this reason a two-dimensional airfoil experiment should be considered as part of the NTF calibration.

Establish Confidence in Tunnel

Because of the large investment by the aerospace community in wind tunnel facilities and the amount of developmental testing required, it must be recognized that the NTF will provide an added capability and not represent a replacement for existing facilities. It can be expected that a majority of the wind tunnel tests will be conducted in these existing facilities. Therefore, it will be extremely important to establish a tunnel-to-tunnel correlation. An early priority for the NTF should be the definition and conducting of an experiment or experiments to provide user confidence in tunnel-to-tunnel measurements. It is recognized that perfect correlation will probably not exist. However, providing sufficient data to allow an acceptable correlation is important to assure that intelligent use of the NTF can occur.

Areas of Concentration

The major portion of the deliberations addressed those areas which the panel viewed as requiring wind tunnel/flight correlation for validating the NTF. These areas were separated into seven categories.

Wing cruise drag and drag rise.- During the opening session of this workshop, data were presented which showed differing Reynolds number effects on the drag characteristics for various classes of airfoils. It was pointed out that drag alone (C_D vs M) was insufficient for determining the causes of these variations. It was the consensus of the Wind Tunnel/Flight Correlation Panel that both wind tunnel and flight data are required to provide detailed pressure distributions, accurate definition of shock wave location, and boundary layer and wake surveys for determining airfoil section characteristics. While the wake survey data are important, it was recognized that model wakes will be extremely small for the higher Reynolds numbers attainable in NTF. Therefore, it may not be practical or possible in many instances to provide accurate measurements from the wind tunnel. Even so, wake data should be obtained in flight for the wind tunnel/flight data analysis. Such data would also be extremely beneficial in the validation/modification of analytical methods. As was discussed previously in the basic tunnel calibration relative to wind tunnel-heat transfer measurements, consideration should be given to obtaining such information from the flight articles.

Wing separation and stall.- Discussions in this area addressed leading-edge and shock-induced separation for both thick and thin wings. The $C_{L_{max}}$ dependency (with and without flaps) on Reynolds number is a major concern in aircraft design (particularly transport class), as are stability characteristics (longitudinal and lateral/directional) and Mach buffet. Correlation studies should be conducted to address these areas as well as those of buffet, buffet-intensity rise, dynamic lift, and hysteresis effects. Another area which requires attention is that of the effect of Reynolds number and turbulence on leading-edge suction. The question of whether or not there exists a cutoff Reynolds number for the various classes of airfoils and wings which can be used in calibrating existing facilities against NTF needs to be addressed.

Afterbody and base drag.- The accurate prediction of full-scale aircraft drag has historically been hampered by the inability to adequately determine afterbody and base effects. This has been true for fighter aircraft and to a lesser extent for transport aircraft, including both commercial and military logistic vehicles. Although model support system interference effects can be a major contributor to afterbody and base drag measurements, the predominant effect is believed to be that of Reynolds number simulation. Wind tunnel and flight data are therefore required to provide a correlation for those classes of airplanes having afterbody and base configurations sensitive to Reynolds number.

Propulsion effects.- In many instances propulsion effects can be considered a major contributor to afterbody and base drag. Classical fighter aircraft configurations, for example, generally have aft fuselage-mounted engines. In these instances, as much as forty percent (40%) of the total vehicle drag can be associated with afterbody effects, including boattail, base, and propulsion system drag. Engine inlet-spillage effects have also been shown to be major contributors to aircraft drag. Future fighter aircraft incorporating an advanced nonaxisymmetric nozzle with thrust vectoring and reversing could be susceptible to significant Reynolds number effects. These effects could be manifested in plume geometry, flow turning, and cooling and supercirculation.

The propulsion-system installation effects for transport category aircraft, both civil and military, can be difficult to assess. The advent of high bypass ratio turbofan engines has resulted in a large engine-to-wing size relationship for transport aircraft. In these instances nearly the entire wing span can be influenced by propulsion system effects. A similar situation exists for configurations incorporating upper-surface blowing. It is extremely important that the sensitivity of the propulsion system installation be understood early in the development cycle of aircraft to minimize performance penalties and/or costly modifications. It is recognized that initial configuration testing in the NTF will not address propulsion system effects. However, the panel strongly recommended that early planning be initiated to provide the capability for propulsion testing in the NTF. The need for wind tunnel and flight data correlation was recognized by all panel members.

Vortex flows.- The broad basic research needs relating to vortex flow were addressed by the Fluid Dynamics and Configuration Aerodynamics Panels. The discussions of the Wind Tunnel/Flight Correlation Panel focused on those areas where correlation is desired or required for validating the NTF. The three areas identified included forebody vortex shedding, vortex bursting, and structural loads. Ample evidence exists to suggest that the impact of forebody vortex shedding on the high angle-of-attack stability characteristics of fighter aircraft is susceptible to scale effects. Vortex bursting can have a significant effect on the stability characteristics of high performance aircraft. The burst location and subsequent stability, control, and structural implications are susceptible to Reynolds number effects and require full-scale flight data for validating wind tunnel results. It is generally assumed, based on wind tunnel studies, that vortex flow generated by separation from the sharp leading edges of highly swept surfaces may be Reynolds number independent. However, there is presently no conclusive demonstration of this assumption, particularly under full-scale conditions. Even for the extremely sharp leading-edge case, Reynolds number effects have been observed with regard to the secondary vortex effects on total lift and surface load distribution. Also, the effect of compressibility on vortex flow has not been investigated, in particular the effect of shock waves above the wing on the vortex location and stability. In view of the complexity and uncertainty of vortex flow interference and nonlinear aerodynamic effects, and the probability that many future aircraft will employ some versions of the vortex lift concept, it is important to provide wind tunnel/flight correlation for validating the full-scale simulation capability of such flows in the NTF.

Cavity flows.- It is an extremely difficult and many times impossible task to predict the unsteady aerodynamics associated with the flow relating to exposed landing gear wells and/or open bomb bay areas of full-scale aircraft. The buffet loads associated with landing gear wells can have significant structural implications. In the case of bomb bay cavities, the unsteady flow not only creates structural implications, it is also a major factor affecting weapons separation. The small size of wind tunnel models combined with Reynolds number capabilities of existing facilities precludes obtaining aerodynamic data at flight conditions. The NTF will provide such a capability and full-scale validation is strongly recommended.

Excrescences.- It was pointed out by some panel members that there exists a continuing need for a data base from which to predict excrescence drag. However, it was also pointed out that, in general, the NTF Reynolds number capability exceeds that required for such testing. Also, the size of most models to be tested in the NTF (as in most wind tunnels) would be so small as to preclude actual scaling of excrescence drag contributors such as gaps, steps, antennas, etc. While there may be instances where the NTF can and will be used to determine excrescence drag, such tests should generally be done in other facilities.

Recommendations

The recommendations for achieving validation of the National Transonic Facility will be addressed individually.

Open-ended flight/wind tunnel program.- The experience gained from previous attempts at correlating wind tunnel and flight data clearly suggests a need for retaining the ability to retest configurations, both in the wind tunnel and in flight. This is extremely important if an understanding of discrepancies is to be provided. It is imperative, in the use of NTF, that one utilize its unique capabilities to isolate the effects of Reynolds number, aeroelasticity, etc., if such an understanding is to be provided. Implicit in this correlation program is the inclusion of the appropriate wind tunnel, wind tunnel model, and full-scale airplane instrumentation. The implication of this is that both wind tunnel models and full-scale airplane be retained in the correlation configuration until the final analysis is complete and all questions have been satisfactorily addressed.

Fighter and transport category aircraft required.- Because of the diversity of the organizations represented by the panel members, it was not possible to achieve consensus for one representative configuration for conducting a wind tunnel/flight correlation to validate the NTF. The concerns and requirements for tactical and air superiority fighter aircraft, civil and military transport, and spacecraft are substantially different. Therefore, it was decided that a recommendation be made for pursuing wind tunnel/flight correlation in several categories. The fighter category should include configurations which address both attached-flow and separated-vortex-flow wing designs. The transport category should include configurations having low-wing arrangements with gentle afterbody slopes (B-747, DC-10, L-1011 type) as well as high-wing arrangements with steep afterbodies (C-5A/AMST type). Finally, a configuration such as the Space Shuttle Orbiter should be included, if possible, in the overall correlation of wind tunnel and flight data. However, it was left to the Space Vehicles Panel to address the specifics of the Orbiter correlation requirements.

Total drag correlation not advisable.- The Configuration Aerodynamics Panel of the High Reynolds Number Research Workshop held at the Langley Research Center in 1976 strongly recommended that attempts at total drag correlation be discouraged. This position was also taken by the present Wind Tunnel/Flight Correlation Panel. Even so, it was recognized by all panel members that total drag measurements are an ultimate goal of correlation. However, such a task is extremely difficult even for simple airplane configurations. Considering the complexity of modern aircraft and state of the art of flight measurement techniques, it is believed that developing an understanding of component effects is the area in which wind tunnel/flight correlation can best be accomplished in the near future. Such an approach will also provide an acceptable validation of the full-scale simulation capability of the NTF.

Wind tunnel flight correlation team essential.- The question of how best to accomplish the required correlations for validating the NTF was discussed in considerable detail. The use of previously obtained data from correlations such as the C-5A, C-141, and Transonic Aircraft Technology (TACT) efforts was pursued at length. An attitude of pessimism about the usefulness of these data generally prevailed among the panel members for the following reasons:

(a) The individuals responsible for those efforts have long since moved on to other tasks. In some instances they have retired.

(b) Resurrecting and understanding the data would be a monumental or more likely an impossible task.

(c) Substantial improvements in wind tunnel and flight test techniques have occurred since the completion of those efforts.

(d) The ability to address the "why" areas of correlation by retesting no longer exists for these configurations.

(e) The configurations are not representative of modern technology.

The consensus of the panel was to recommend the initiation of new correlation efforts, addressing the areas of concentration, to be accomplished by:

(a) Establishing a dedicated team of government and aerospace community investigators to carry out the correlation efforts

(b) Defining an open-ended nonproprietary wind tunnel/flight test program utilizing advanced technology configurations

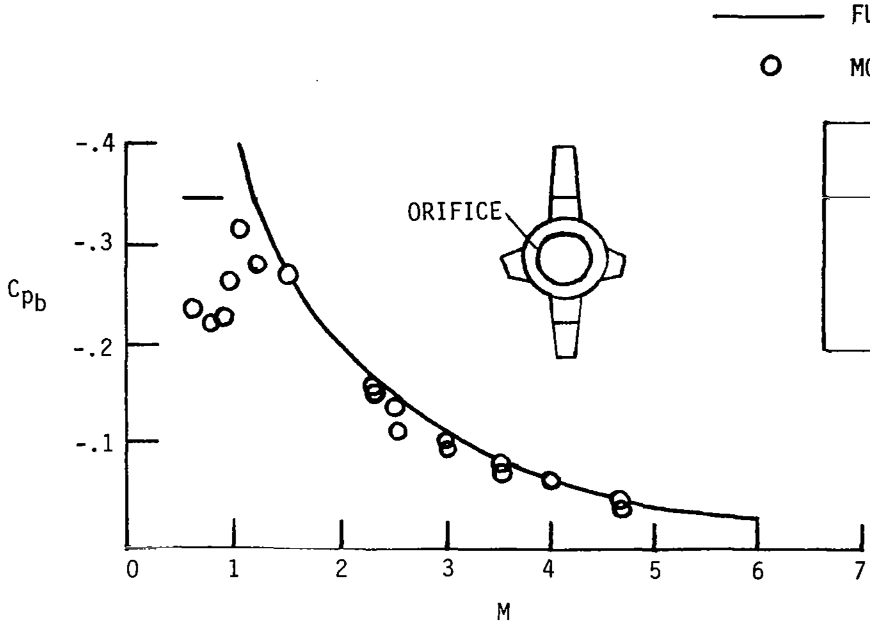
(c) Establishing accountability to assure that the correlation and validation of the NTF are completed in a satisfactory manner

The dedicated team of investigators should be encouraged and, if possible, required to interact in all aspects of the correlation effort. They should all have a thorough understanding of the data requirements, wind tunnel calibration, wind tunnel, and flight tests. Such an approach would assure an effective correlation effort.

Accountability.- The subject was discussed because of a general belief that very little feedback had been provided from the 1976 High Reynolds Number Research Workshop. There was unanimous agreement among the panel members that some method of accountability should be established to insure that action is taken by NASA to consider and/or carry out the recommendations of all the individual panels. There was also agreement that meeting once every 4 years is insufficient to adequately address the NTF and validation requirements. As the operational date of the NTF gets nearer, the panel should reconvene at an appropriate time (perhaps annually) to provide the necessary and desirable interaction with the NTF staff.

REFERENCE

1. Baals, Donald D., ed.: High Reynolds Number Research. NASA CP-2009, 1976.



C_{D_b}/C_{D_0}, FLIGHT	M
0.88	0.8
.55	1.5
.35	3.0

Figure 1.- X-15 fuselage base pressure coefficients (power off).

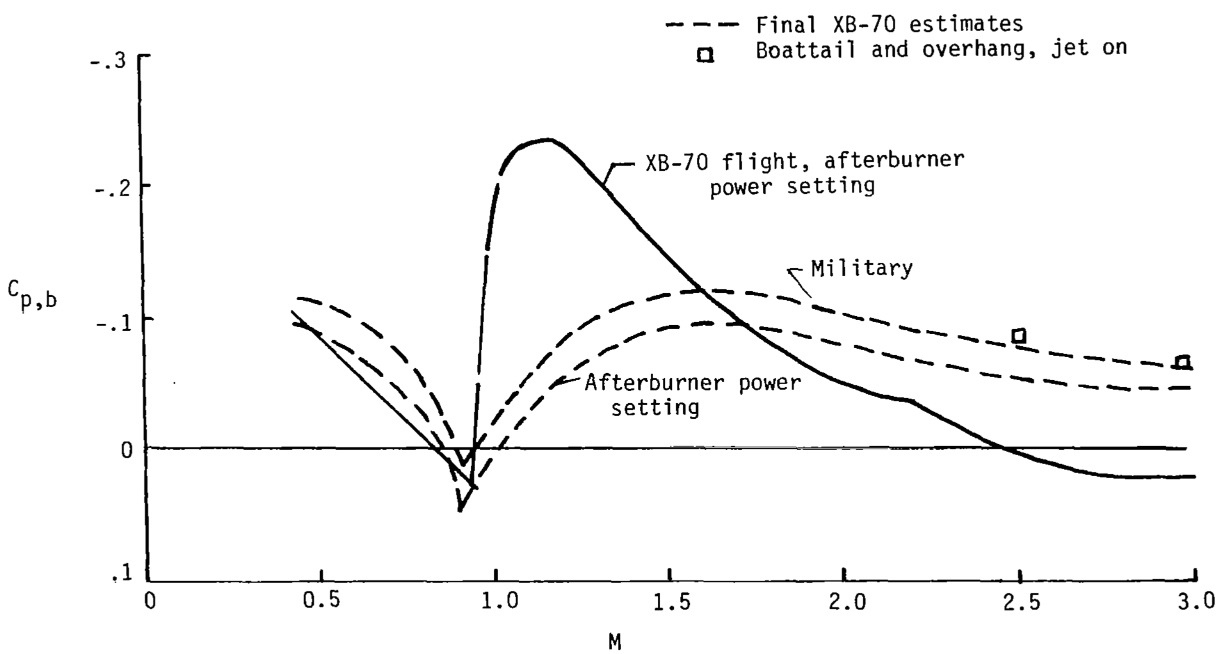


Figure 2.- Comparison of flight-measured XB-70 average base pressure coefficient with predicted values.

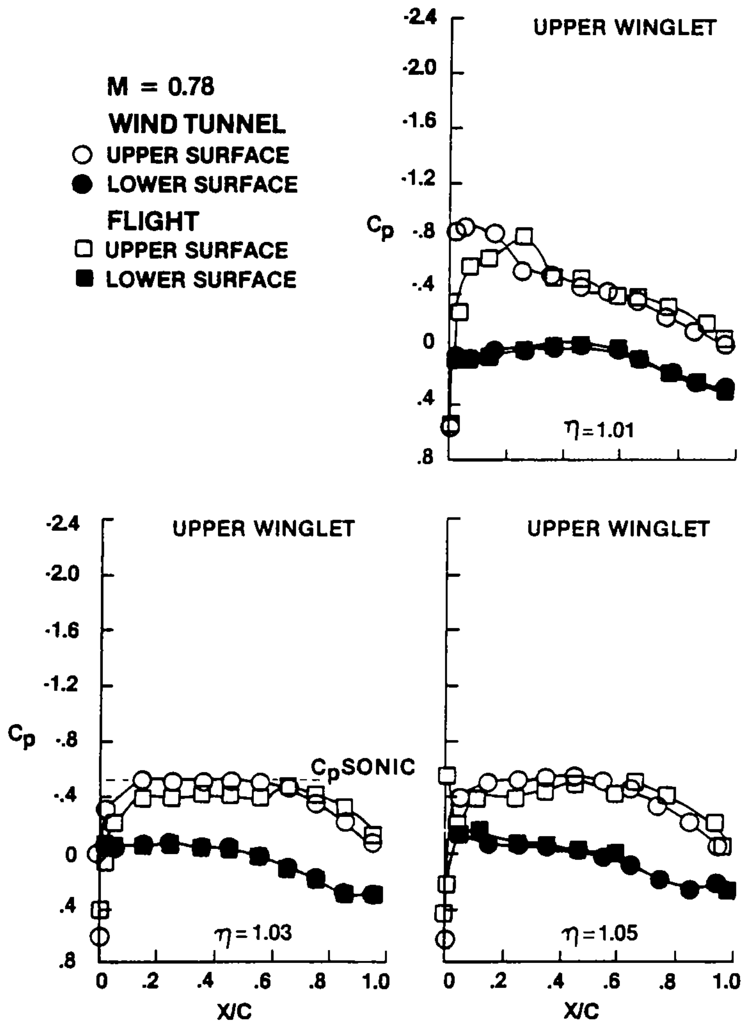
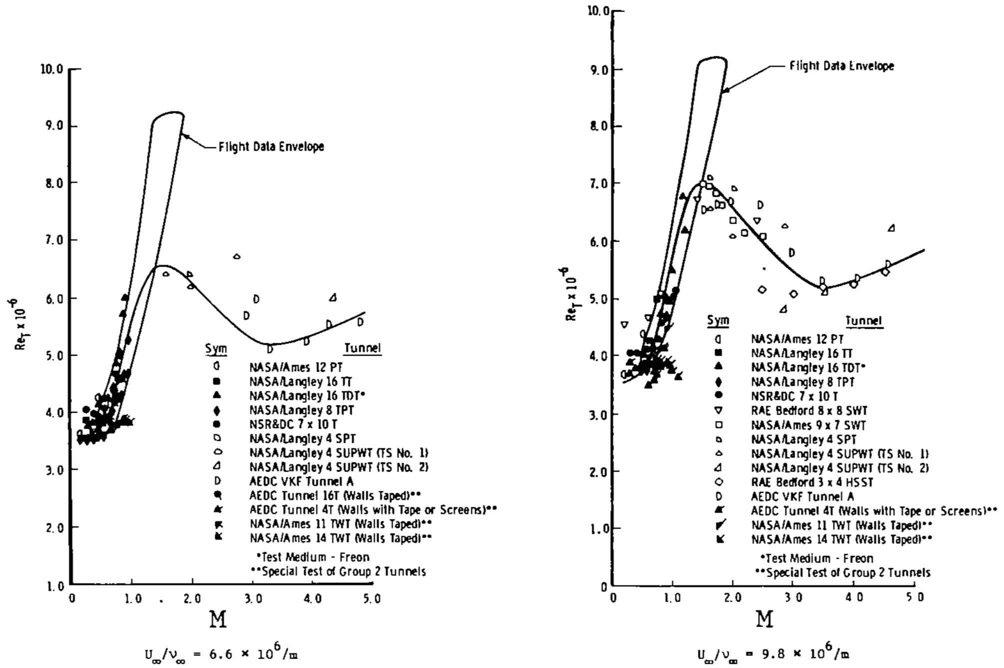
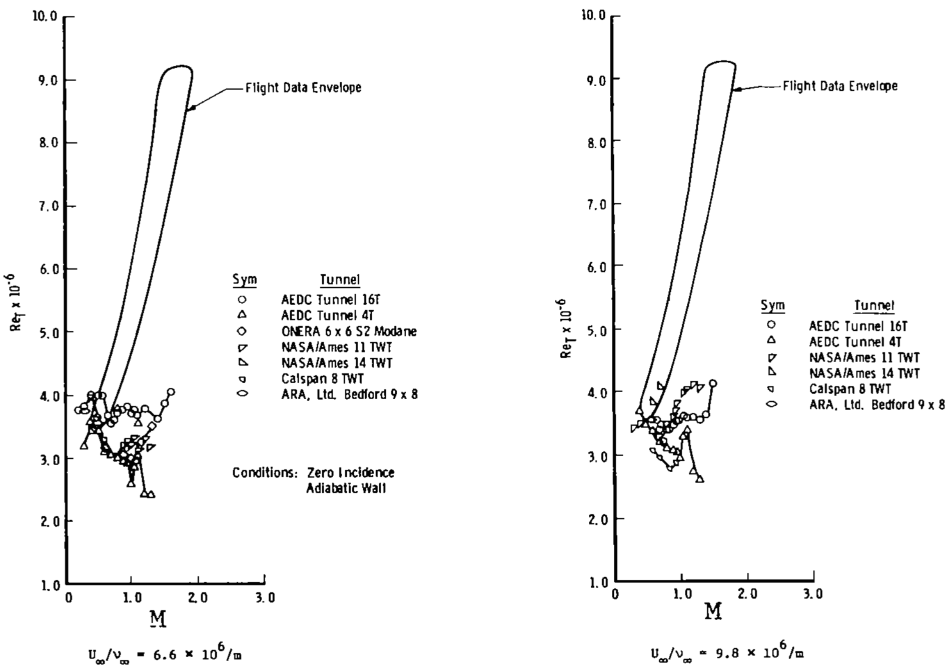


Figure 3.- Comparison of wing pressure distributions measured in flight and in the wind tunnel (KC-135 Winglets Project).



(a) Transition Reynolds numbers in lower-disturbance-level wind tunnels.



(b) Transition Reynolds numbers in higher-disturbance-level wind tunnels.

Figure 4.- Transition experiments on a 10-degree cone in wind tunnels and in flight.

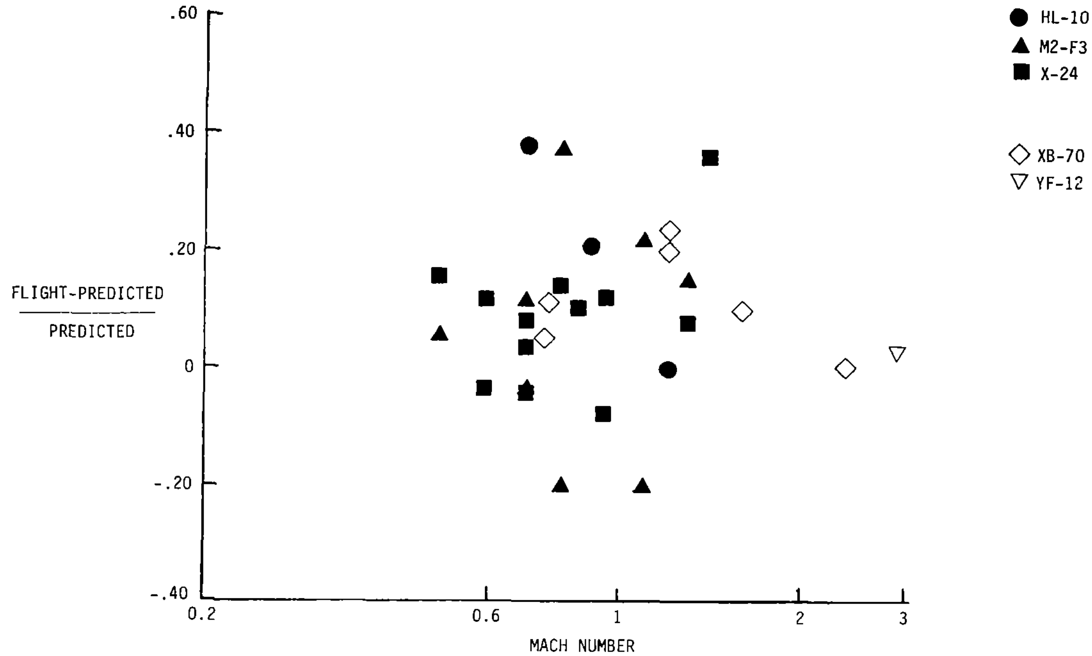


Figure 5.- Correlation of flight and predicted C_{m_δ} for lifting bodies.

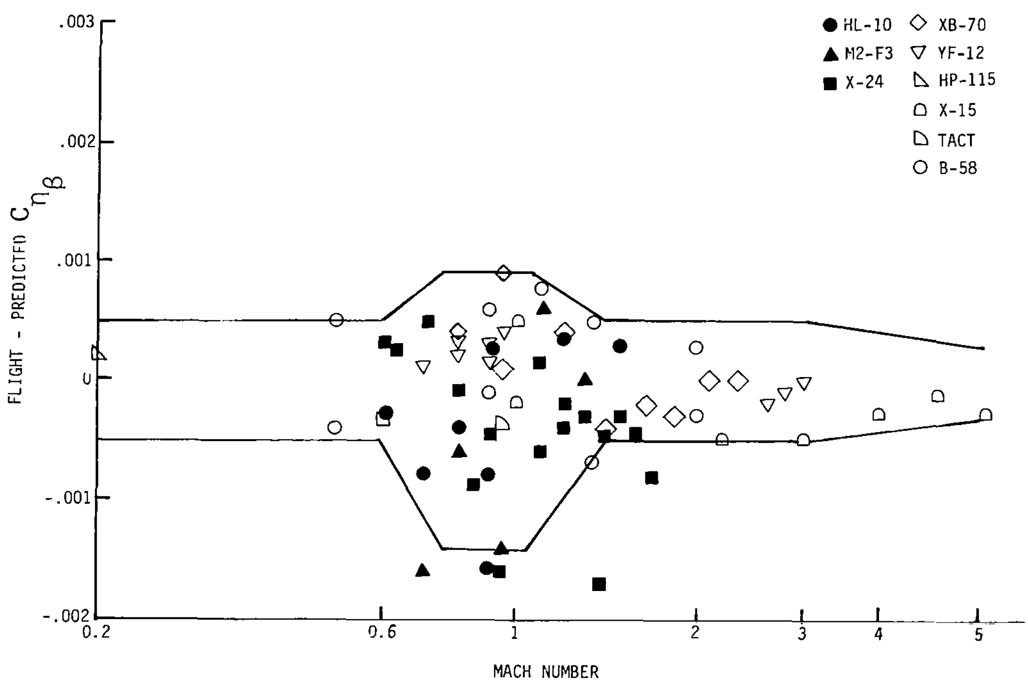
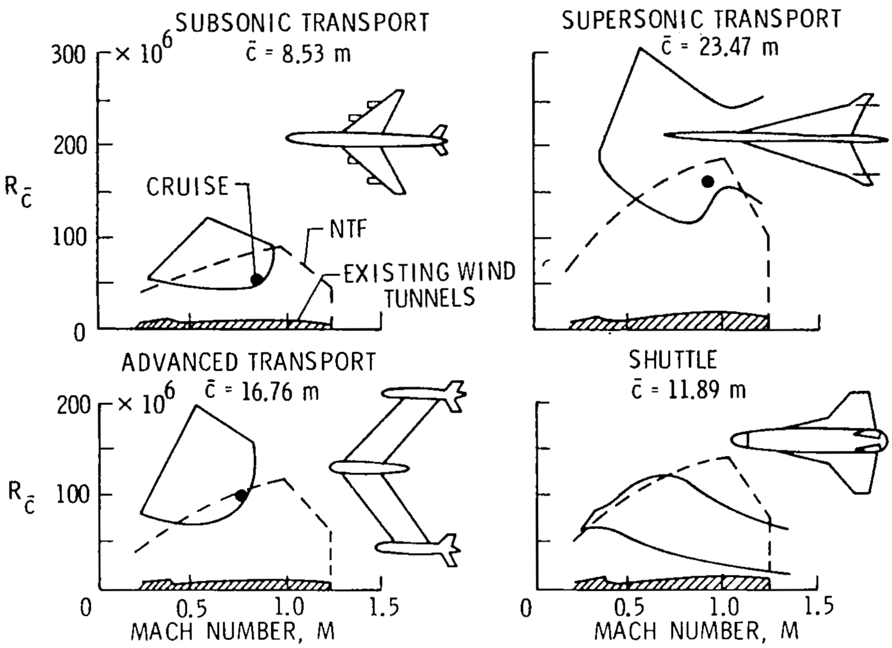
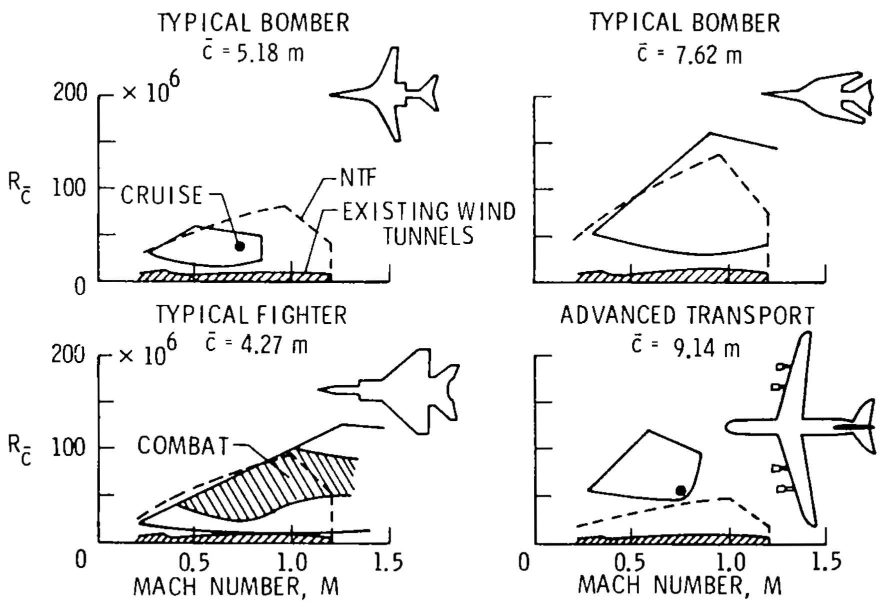


Figure 6.- Correlation of flight and predicted C_{n_β} for conventional aircraft.



(a) Typical commercial aircraft.



(b) Typical military aircraft.

Figure 7.- Comparisons of Reynolds number and Mach number envelopes for full-scale flight vehicles, the NTF, and existing wind tunnels.

NTF MODEL-TO-FLIGHT COMPARISONS:

- AVOID ABSOLUTE DRAG COMPARISON BECAUSE OF THRUST MEASURING UNCERTAINTIES
- THEREFORE BASE THE COMPARISON ON:
  - (A) PRESSURE DISTRIBUTION MEASUREMENTS
  - (B) SECTION DRAG FROM WAKE PROFILE MEASUREMENTS
- THE AIRCRAFT CHOSEN FOR THIS TASK MUST HAVE:
  - (A) A PRECISE, CALIBRATED, "AIR-DATA" SYSTEM FOR DEFINITION OF M, q, AND P_∞
  - (B) PRECISE α AND β DATA
- STRUCTURAL DEFORMATION MEASUREMENTS ARE MANDATORY FOR BOTH MODEL AND AIRCRAFT

Figure 8.- Recommendations of the Configuration Aerodynamics Panel of the 1976 Workshop on High Reynolds Number Research (ref. 1).

SPACE VEHICLES PANEL

Chairman	Delma C. Freeman, Jr. NASA LaRC
Technical Secretary	George M. Ware NASA LaRC

Panel Members

Bernard Spencer, Jr.	NASA LaRC
True Surber	Rockwell International
Gerald Walberg	NASA LaRC
Howard Zeck	The Boeing Company

HIGH REYNOLDS NUMBER RESEARCH REQUIREMENTS
FOR SPACE VEHICLE DESIGN

Delma C. Freeman, Jr.
NASA Langley Research Center

INTRODUCTION

There are currently three areas of entry vehicle design that require high Reynolds number wind-tunnel testing: space shuttle development, development of future space transportation systems (1995-2000 time frame), and planetary entry data analysis. Determination of the aerodynamic characteristics at flight Reynolds numbers would greatly simplify the design process for these vehicles by reducing uncertainties in the predicted flight stability and control parameters. Because entry vehicles operate in the transonic and subsonic flight regimes at higher angles of attack than conventional aircraft, understanding the Reynolds number effects on separated flow is important.

CURRENT STATUS AND PLANS

Space Shuttle Orbiter

A look at the space shuttle development will give some insight into the space vehicle design requirements for high Reynolds number testing. Figure 1 shows an artist's conception of the first launch of the space shuttle scheduled for 1981. A summary of specific design problems where high Reynolds number subsonic and transonic test results would have been beneficial follows. Numerous people associated with the shuttle development have indicated that an accurate simulation of the flow in the interstage area between the tank and the orbiter during ascent would have resulted in a better estimate of the aerodynamic loads on the lower surface of the orbiter. High Reynolds number transonic data would help to understand the interaction of the flow from the orbital maneuvering system pods and vertical tail. Understanding this complex flow would result in less uncertainty in the level of directional stability and would also give insight into the aeroelastic characteristics of the vertical tail. Calibration of the air-data sensors used in the approach and landing tests at high Reynolds numbers would have been beneficial. A more accurate calibration of these sensors would have increased the accuracy of the aerodynamic data extracted from the flight tests. NASA has a large effort underway to extract aerodynamic data from the shuttle flights. Using these data and results from tests in the NTF would offer an excellent opportunity to assess the ability of the wind tunnel to predict flight aerodynamics.

As discussed earlier, determining the flow properties in the interstage between the Space Shuttle Orbiter and tank has been a difficult problem. Understanding these flow properties is necessary to adequately estimate the loads on the thermal protection system during ascent. In order to accurately estimate the aerodynamic loads, the strength of any shocks generated and the boundary-layer characteristics must be known. Figures 2 and 3 give an indication of the problem. In figure 2, the estimated boundary-layer thickness upstream of the crossbeam at the aft attachment point for the tank for a 0.03-scale model is presented for a Mach number of 1.4. These estimated thicknesses show that the boundary layer from the tank and orbiter overlap, creating a region of subsonic flow ahead of the rear strut crossbeam. Figure 3 shows the estimated boundary-layer thickness for the full-scale vehicle. This estimated flight case shows that the boundary layers from the tank and orbiter do not intersect and the crossbeam is in a region of supersonic flow which would generate a shock. The strength of this shock must be known accurately to determine the loads on the tiles where it impinges on the lower surface of the orbiter. Tests of a large panel at flight Reynolds numbers would help to answer these questions.

A comparison of the orbiter flight Reynolds numbers with the NTF capability is presented in figure 4. During the shuttle development wind-tunnel testing, experience showed that a blockage ratio no larger than 0.002 was required to obtain accurate test results at Mach numbers near 1. Based upon this experience, the largest orbiter model that could be tested at transonic Mach numbers in the NTF would be a 0.015-scale model. The comparison of flight Reynolds numbers and NTF capability presented in figure 4 is based on tests with this size model. The figure shows that at Mach numbers less than 0.75 the NTF cannot duplicate flight Reynolds numbers for the orbiter. However, since blockage is not a problem at the lower Mach numbers, a larger model can be tested to obtain flight Reynolds numbers at Mach numbers less than 0.75.

There are several areas where there are requirements for high Reynolds number subsonic and transonic testing. One of these areas is shuttle post-flight testing. Undoubtedly, there will be problems that will require wind-tunnel data to resolve. Also, within NASA there is a program underway to improve the performance of the shuttle, and this work will require additional subsonic and transonic aerodynamic data.

Development of Future Space Transportation Systems

There is an ongoing effort within NASA to identify the technology requirements for future space transportation systems. Projections indicate that these new vehicle systems will be required in the 1995 time frame. An example of these vehicles is shown in figure 5. The single-stage-to-orbit vehicle shown is 66.8 meters long and is designed to deliver a 29 500 kg payload to a 100 nautical mile Earth orbit. The vehicle has an entry weight of approximately 193 000 kg. Utilizing a control configured vehicle (CCV) design philosophy, the vehicle will be flying statically unstable during most of its atmospheric entry, requiring an accurate prediction of flight values of stability and control for control system design. High Reynolds number wind-tunnel testing is required to reduce the uncertainties in the predicted aerodynamics and, therefore, optimize the vehicle design.

A comparison of the entry flight Reynolds number of the CCV vehicle, the entry flight Reynolds number for the shuttle orbiter, and the maximum Reynolds number capability of the NTF is presented in figure 6. The comparison shows that the CCV configuration has about the same high Reynolds number requirements as the shuttle orbiter. The comparison is made for a 0.01-scale model of the CCV configuration which has a blockage ratio of about 0.002.

There are currently two classes of future space transportation systems projected for the 1995 time frame. A priority vehicle with a payload capability of 4450 to 29 500 kg is designed to deliver personnel and payloads to Earth orbit. Some requirements for this vehicle dictate a quick response capability to a wide variety of orbits. The second class of vehicle being studied is the heavy-lift system. This vehicle will be designed to deliver very large payloads (up to 225 000 kg) to orbit. The requirements for this system are still not well defined, but these vehicles will be very large with high flight Reynolds numbers during both ascent and entry.

Planetary Probes

There is a requirement for high Reynolds number testing to support the planetary probe programs. These probes are used to determine the atmospheric characteristics of the various planets. Determining these atmospheric characteristics requires very good definition of the probe aerodynamics. The results of work with the Pioneer Venus Probe have shown that for very blunt bodies accurate transonic wind-tunnel results can only be obtained with very small blockage ratios, requiring high Reynolds number facilities to duplicate flight values. A photograph of a wind-tunnel model of the Pioneer Venus Probe is shown in figure 7. A comparison of the flight Reynolds number of this probe with the NTF capability is presented in figure 8. As can be seen, the NTF easily provides flight values of Reynolds numbers with the recommended blockage ratio for this probe.

PANEL DISCUSSION AND RECOMMENDATION

The purpose of the panel discussion was to determine the space vehicle design requirements for transonic and subsonic high Reynolds number wind-tunnel tests. The space vehicle designs considered were Space Shuttle Orbiter, space shuttle launch vehicle, advanced space transportation systems, planetary probes, and cruise missiles.

The panel recommended that pathfinder models of a 0.02-scale Space Shuttle Orbiter and a 0.01-scale launch vehicle be tested in the NTF. These models should be designed to obtain force and moment, pressure, and spectral data and should also be utilized for flow-visualization studies. The objectives of developing the shuttle pathfinder series would be: postflight shuttle development, wind tunnel to wind tunnel and wind tunnel to flight data correlation, and shuttle enhancement studies.

There are several preflight concerns for the shuttle system that could result in postflight test requirements. These concerns, which are presented in figure 9, should not affect minimum safety, but better definition of these problems would result in the removal of design boundaries and mission constraints.

The requirements identified by the panel for high Reynolds number testing for advanced space vehicles and planetary probe development are presented in figure 10. One requirement is to determine the Reynolds number effects on very thick wings with large leading-edge radii. For some of the proposed configurations where structural efficiency dictates circular bodies, Reynolds number effects on body crossflow are important. Reduction in the uncertainties for predicted flight aerodynamics by use of high Reynolds number wind-tunnel tests will simplify the vehicle design process. For the design of planetary probes, the only real requirement is that high Reynolds number capability be available with an accurate understanding of blockage and sting effects for bluff-body testing.

In order to accomplish the testing required for space vehicle design, the space vehicle panel has identified requirements for the following capabilities.

High Angle of Attack

To analyze the abort flight regime of the shuttle orbiter, there is a requirement to test at angles of attack up to 40° at both subsonic and transonic Mach numbers. The ability to obtain aerodynamic data at flight values of Reynolds numbers for these tests would eliminate uncertainties in the stability and control parameters for control system design. Experience in the development of the cruise missile has also demonstrated a need for high Reynolds number high α data for these vehicle designs. In order to accomplish this high angle-of-attack testing, there are requirements for a high α support system, high load balances, an understanding of the blockage effects at high α, and a flow-visualization capability.

Dynamic Stability

Even though there are estimation techniques that do accurately predict the damping derivatives for aircraft-type configurations, they are not sufficient for space vehicle design. During entry and abort, these vehicles fly at high angles of attack (12° to 40°), and regions of separated flow exist. Present estimation techniques cannot predict the damping with separated flow on the vehicle. Capability should be developed to measure the pitch, yaw, and roll damping and the roll and yaw cross derivatives in the NTF.

Blade or Strut Support System

Base flow studies and the determination of propulsion and reaction control system plume effects require alternatives to the standard sting support system. A blade or strut support system would eliminate sting effects in the base region, allowing base flow studies at the high Reynolds numbers of the NTF.

Surface Roughness Studies

Space vehicles have some unique surfaces because of the requirements for TPS, nozzles, vents, hatches, etc. In order to assess the effect of these rough surfaces, there is a requirement to obtain both flow visualization and spectral data at high Reynolds numbers.

Boundary-Layer Transition Measurements

In the design of space vehicles, there is a requirement to understand the boundary-layer transition on very thick wings (10 to 20 percent). These transition studies will require the use of thin-film gauges or laser techniques to determine the boundary-layer characteristics.

Bluff-Body Studies

In order to test planetary probes, there will be requirements for special balances capable of measuring the high axial forces and low normal forces of bluff bodies.

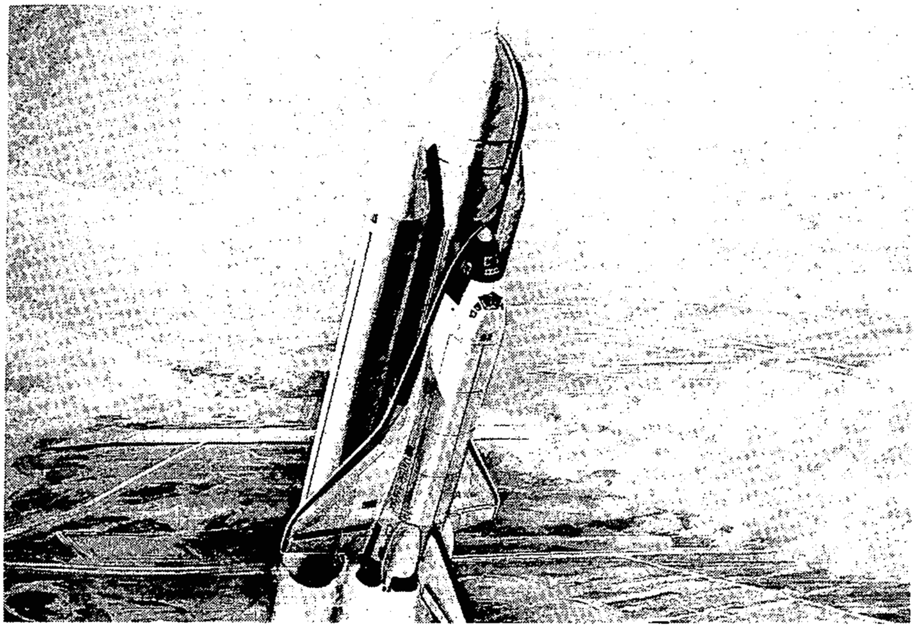


Figure 1.- Artist's conception of the first Space Shuttle Launch.

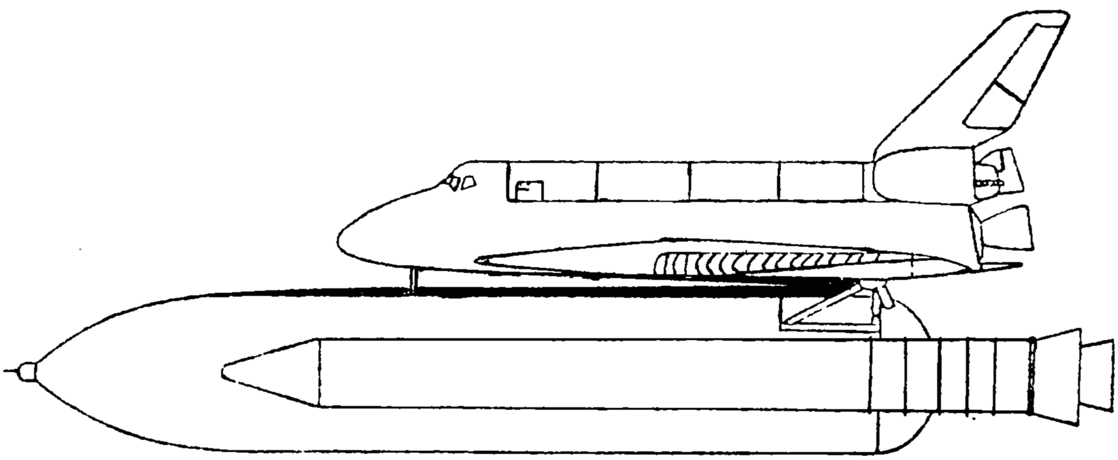


Figure 2.- Estimated boundary-layer thickness upstream of the cross beam. M = 1.4; 0.03 scale.

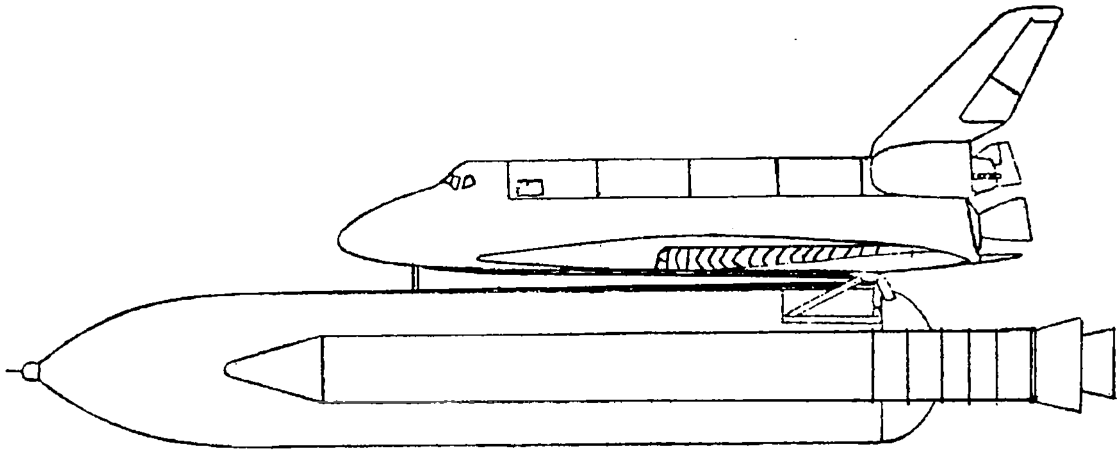


Figure 3.- Estimated boundary-layer thickness upstream of the cross beam. M = 1.4; full scale.

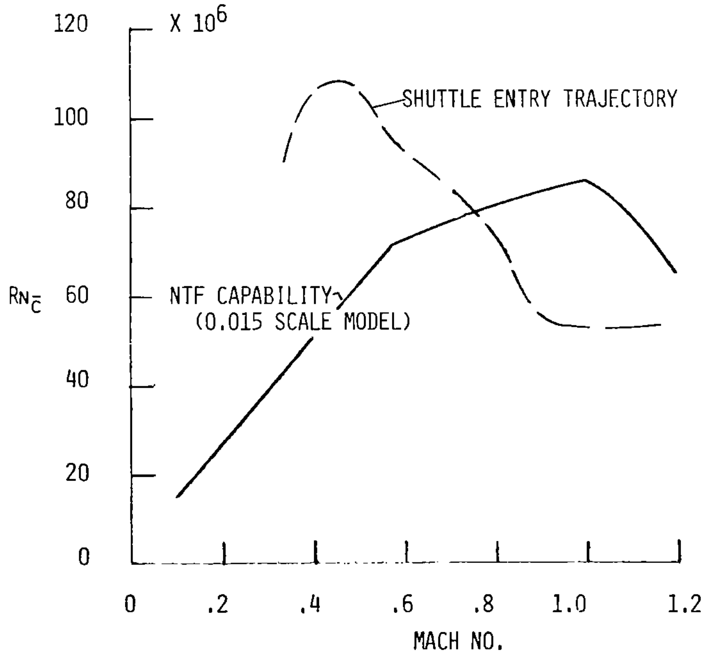


Figure 4.- Comparison of the Space Shuttle Orbiter entry flight Reynolds number and NTF capacity. 0.015-scale orbiter model.

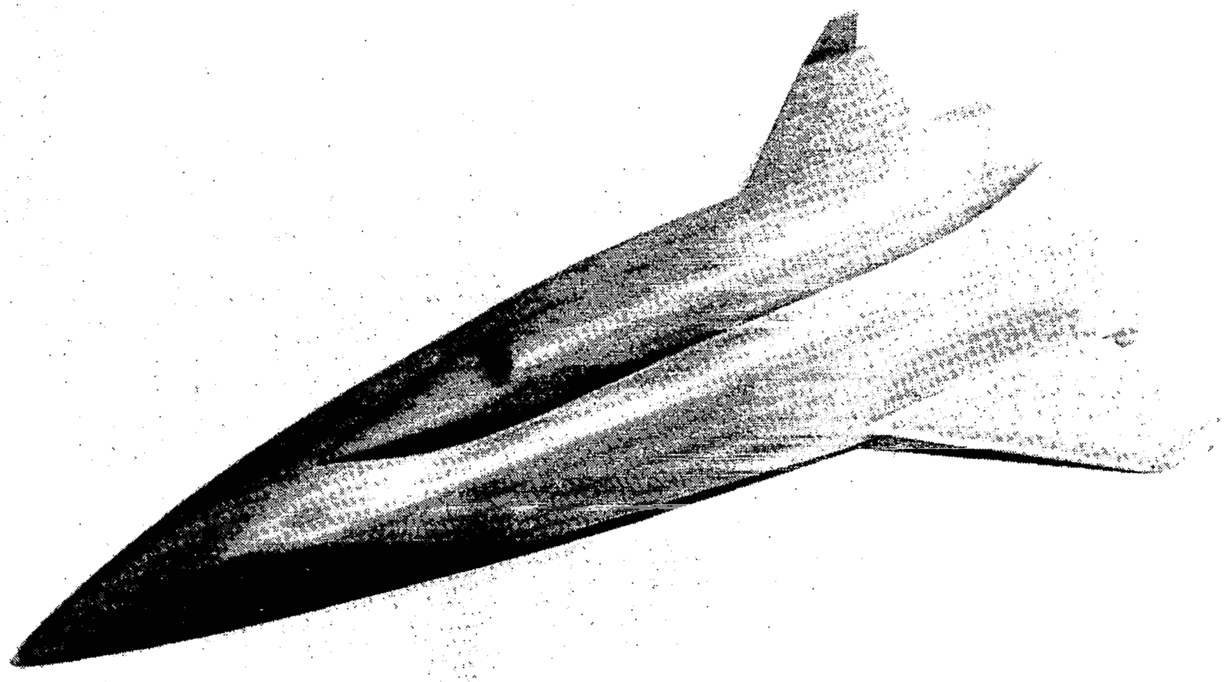


Figure 5.- Photograph of a wind-tunnel model of SSTO CCV configuration.

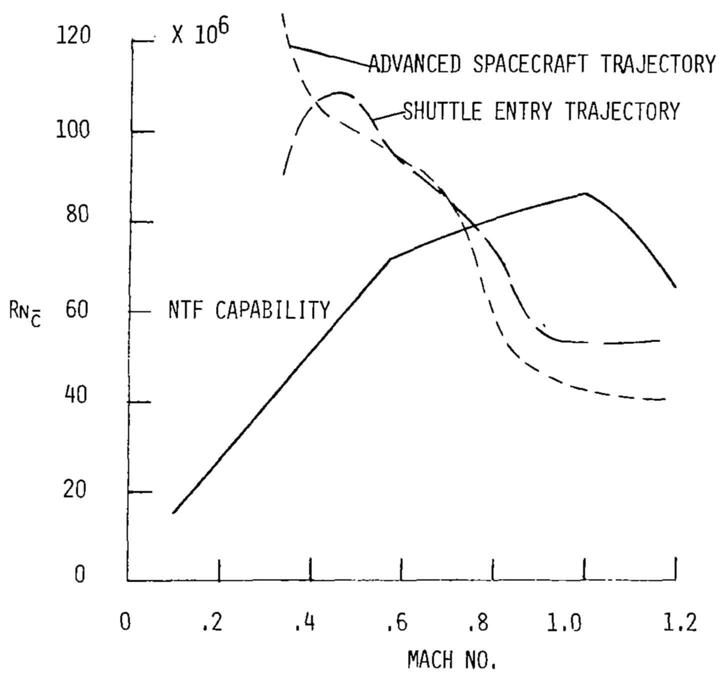


Figure 6.- Comparison of entry flight Reynolds numbers of CCV configuration (0.01-scale model), Space Shuttle Orbiter (0.015-scale model), and NTF capability.

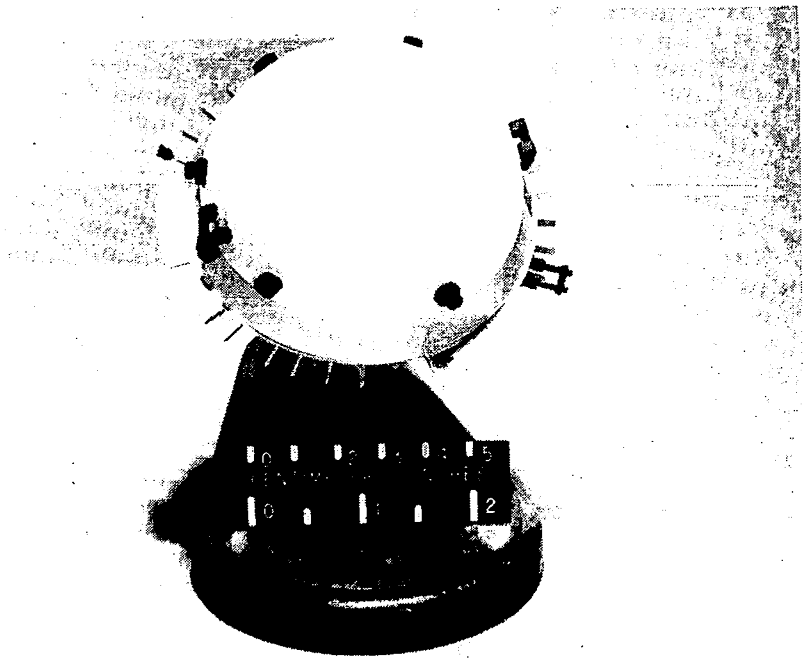


Figure 7.- Photograph of the 0.092-scale Pioneer Venus Sounder probe model.

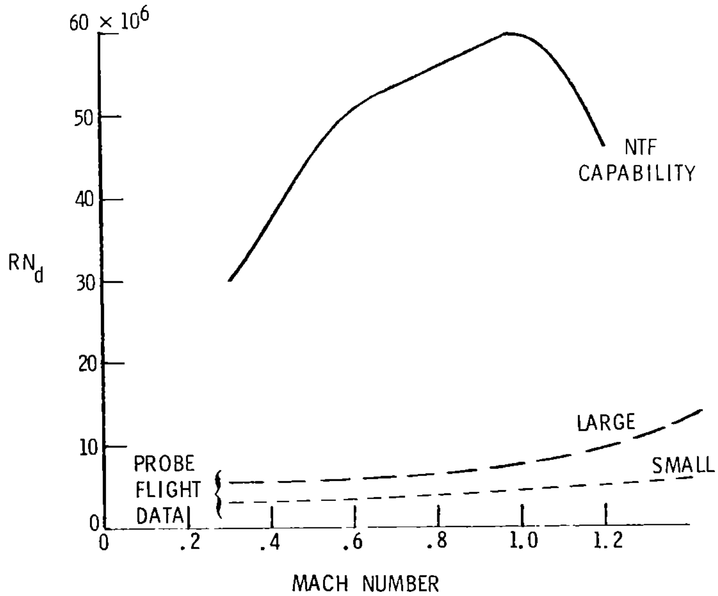


Figure 8.- Comparison of Pioneer Venus Probe flight data Reynolds number requirements and NTF capability.

PREFLIGHT CONCERNS - ORBITER

- CONTROL EFFECTIVENESS ($C_{n_{\delta_a}}$)
- ELEVEN HINGE MOMENTS
- β HYSTERESIS
- PLUME INDUCED EFFECTS (RCS, AND SSME FOR ABORT)

PREFLIGHT CONCERNS - LAUNCH VEHICLE

- INTERSTAGE BOUNDARY-LAYER CHARACTERISTICS AND SHOCK IMPINGEMENT (STATIC AND DYNAMIC)
- HINGE MOMENTS (LOAD RELIEF)
- BASE EFFECTS (POWER ON AND POWER OFF)

Figure 9.- Projected shuttle post-flight test requirements.

ADVANCED SPACE VEHICLES

- SHOCK-INDUCED SEPARATION AND R_N EFFECTS DUE TO THICK WINGS AND LARGE LEADING-EDGE RADII
- R_N EFFECTS ON BODY CROSS-FLOW
- ACCURATE PREDICTION OF FULL-SCALE STABILITY AND CONTROL FOR CCV DESIGN

PLANETARY PROBES

- ACCURATE DETERMINATION OF BLOCKAGE AND STING EFFECTS FOR BLUFF BODIES

Figure 10.- Advanced space vehicle requirement for high Reynolds number data.

THEORETICAL AERODYNAMICS PANEL

Chairman	Jerry C. South, Jr. NASA LaRC
Technical Secretary	Frank C. Thames NASA LaRC

Panel Members

E. Clay Anderson	NASA LaRC
Elwood Bonner	Rockwell International
Clinton E. Brown	NASA Headquarters
Morton Cooper	Flow Research
Douglas L. Dwoyer	NASA LaRC
Mohamed M. Hafez	The George Washington University
Renaldo V. Jenkins	NASA LaRC
William B. Kemp, Jr.	GWU/JIAFS
Ching F. Lo	Arnold Air Force Station
Robert E. Melnik	Grumman Aerospace Corporation
Perry A. Newman	NASA LaRC
J. Scott Reasor	Lockheed California Company

REPORT OF THE PANEL ON THEORETICAL AERODYNAMICS

Jerry C. South, Jr., and Frank C. Thames
NASA Langley Research Center

INTRODUCTION

This panel covered both aspects of the interaction of NTF and the theoretical aerodynamics discipline, namely, ways in which the NTF can be used to benefit theoretical aerodynamics and ways in which this discipline can aid the NTF. The NTF can significantly impact theoretical aerodynamics in three areas:

- Development and validation of computational fluid dynamics computer codes

- Determination/validation of Reynolds number (Re) scaling laws which allow data from conventional, moderate Reynolds number tunnels to be "scaled up"

- Extension of the data bases of entrainment-type turbulence models to include high Reynolds number data

Of these, the panel's principal focus was on the first area. Further, the aerodynamic programs considered were those used primarily for flight vehicle design and analysis purposes. The more fundamental aspects of fluid mechanics - transition, detailed turbulence modeling, shock/boundary-layer interactions, etc. - were left to the Fluid Dynamics Panel. The second area listed - determination of Reynolds number scaling laws - would serve to broaden the data base for theoretical aerodynamics development work in a number of areas. Finally, the last item addresses the need to obtain high Reynolds number base data to update some of the more widely used turbulent boundary-layer methods.

The theoretical aerodynamics discipline can benefit the NTF in one important (and usually troublesome) area - the quantitative description of wind-tunnel wall interference effects. Without an accurate wall interference assessment or correction procedure, validation of aerodynamic codes designed for simulation of free-air conditions is virtually impossible. This is particularly true for transonic flows which can be inordinately sensitive to small perturbations in geometry and/or flow parameters.

The next section of this paper presents a more detailed look at the four areas of theoretical aerodynamics/NTF interaction outlined above. The final section covers the recommended experiments to achieve the common goals of NTF and the theoretical aerodynamics community.

BACKGROUND

Aerodynamic Code Development/Validation

To access the usefulness of the NTF in aiding the development/validation of aerodynamic codes, it is reasonable first to review the status of current and anticipated near-term capabilities.

The current state of the art in transonic computational aerodynamic analysis codes (geometry-specified) can be summarized as follows:

- Airfoils (2-D): 2-D conservative full-potential transonic inviscid analysis plus Green's integral boundary method plus trailing-edge singularity and viscous wake thickness and curvature accounted for (ref. 1)

- Wing/bodies (3-D): 3-D conservative full-potential analysis plus 3-D integral boundary-layer method plus wake simulation (refs. 2-4)

- Wing/body/pod/pylon/winglet (3-D): 3-D nonconservative, small-disturbance, potential analysis plus 2-D strip boundary layer on wing only (ref. 5)

It is interesting to note that all of these methods are of the viscous-inviscid interaction type and that the viscous (boundary-layer) solutions are all integral procedures. Sample results from these three methods are shown in figures 1 to 3. In addition to these analysis routines, there are two transonic airfoil design codes (pressure-distribution-specified) that are generally available. These are the shockless design code of Bauer et al. (ref. 6), and the Carlson program (ref. 7) which can handle more general cases with shocks. The design codes are also based on potential/boundary-layer interaction theory. Future developments, however, should see the advent of methods based on a still more powerful theoretical foundation.

The expected computational aerodynamic capabilities for the near future are listed in table 1. The initial five items are in the final stages of development and are sure to be generally available by the end of calendar year 1982. The latter three efforts are in more formative stages of development and their release may well extend into the mid-1980's. Note that four of the items in table 1 are based on the numerical solution of either the 2-D or parabolized 3-D Reynolds-averaged Navier-Stokes equations with varying degrees of sophistication associated with turbulence closure (NTF inputs to turbulence modeling developments are covered by the Fluid Dynamics Panel). Design-type solutions to the fully 3-D Reynolds-averaged Navier-Stokes equations require computer resources beyond the capabilities of current or projected super-computers.

All of the current and expected capabilities outlined above are developed and "validated" using data taken at Reynolds numbers significantly less than those encountered in flight for full-scale vehicles. If we couple this fact with the fact that all of these methods are approximate in some sense, it is readily apparent that data obtained at near-flight Reynolds numbers constitute a pressing need of the theoretical aerodynamics discipline. Several of the recommended NTF experiments are proposed to fill this data-base requirement.

Reynolds Number Scaling

The purpose of this effort is to try to establish a set of rules (scaling laws) which can be used to rescale data acquired in a low-Reynolds-number installation (say, $Re = O(10^6)$) so that, in effect, the data appear to have been taken at a higher Reynolds number ($Re = O(50 \times 10^6)$). The existence of such a set of laws would allow Reynolds-number-sensitive testing in existing low-Reynolds-number tunnels. The development of a comprehensive set of scaling rules (if, indeed, they exist) would, of course, entail an extensive, carefully thought-out program. As a first step, this panel recommends that an attempt be made in the Pathfinder I test series to validate the Reynolds number scaling rule proposed in reference 8. Basically, this rule states that the transition strip should be positioned on the wing such that the trailing-edge displacement thickness in the low-Reynolds-number experiment matches the thickness that would occur at full-scale conditions. The recommended experiment to test this rule is covered in a later section.

High Reynolds Number Data Base Generation for Entrainment Boundary-Layer Methods

The point was made in a previous section that current and near-term computational aerodynamics methods will continue to rely heavily on approximate formulations of fluid flow problems. It is anticipated that potential flow/boundary-layer interaction methods will carry the burden of design and analysis computations for quite some time (particularly for 3-D calculations). Of the available boundary-layer methods, the integral entrainment type of method is in wide use in both two- and three-dimensional calculation (e.g., ref. 1 and references cited therein). All of these methods contain a number of correlation constants which are determined from test data of limited Reynolds number range. A broader data base, extended to higher Reynolds numbers, is needed to extend the validity of the entrainment methods. The required test is a rather simple one (flat plate) and can be carried out in the smaller 0.3-meter Transonic Cryogenic Tunnel (TCT) at Langley. Details of the proposed test are given in a later section.

Wall Interference Assessment Studies

The study of wall interference is one area in which the theoretical aerodynamics discipline can aid the NTF. Accurate wall interference assessment procedures, or the accounting for wall effects, are mandatory for successful aerodynamic code validation. For this reason, this panel strongly recommends that measurements necessary to establish wall interference corrections or to simulate wall effects be made from the very inception of NTF operations. This will guarantee the buildup of a data base which can be utilized for future developments of correction methods. Current wall interference assessment methods (ref. 9) require special measurements. The principal ones are:

- Tunnel sidewall, floor, and ceiling static pressure distributions

- One or more flow angle measurements upstream of the working section where the flow is essentially potential in nature

This panel recommends that both NTF and the 0.3-m TCT be instrumented to acquire these measurements on a routine basis. These data will aid in the establishment of the interference data bank alluded to above.

All of the recommended experiments covered in the next section address the problem of wall interference to some degree. Two of the experiments - the axisymmetric body drag rise test and the isolated wing test - have wall interference as their primary focus. In each of these experiments, two model sizes will be tested.

RECOMMENDED EXPERIMENTS

A brief description is given in this section of a set of five experiments designed to implement the requirements set forth in the previous section. In addition to fulfilling the stated requirements, the panel established two other guidelines for the tests:

- It should be possible to conduct the proposed tests with currently available measurement and data acquisition systems.

- The experiments should have a generic nature - that is, the data acquired should be useful for a number of aerodynamic applications.

The recommended experiments have not been placed in any priority ordering - all have essentially the same level of priority. However, since two of the tests are merely additions to experiments definitely scheduled for early work in the NTF, it is presumed that they will be first in the recommended series to be accomplished. Details of the individual tests are given below (for the two tests mentioned immediately above, only the additions are described).

Experiment Number 1 -

Additions to the Axisymmetric Body Drag Rise Experiment

Purpose.- Develop data base to establish an axisymmetric wall interference assessment and correction procedure.

Test plans.- Two separate tests are to be conducted, one in NTF and another in the 0.3-m TCT, using two geometrically similar bodies of revolution (see fig. 4). The bodies are sized such that the body diameter to tunnel width ratio for the small body in the 0.3-m TCT equals the same ratio for the large body in the NTF. In addition, the small body should be small enough to incur negligible interference effects to NTF. Data are to be acquired at various Reynolds numbers for Mach numbers near one. The effects of small angles of attack/sideslip are also to be obtained.

Measurements.-

- Six-component force and moment data
- Body/sting pressure distributions
- Wall pressure distributions
- "Axisymmetric" wall pressure distribution (see fig. 4)

Experiment Number 2 -

Additions to the Pathfinder I Experiment

Purpose.- Obtain configuration data for code validation at high Reynolds numbers and attempt to verify transition-strip-placement Reynolds number scaling rule.

Test plans.- Obtain data at Reynolds numbers from 2×10^6 to 40×10^6 at constant dynamic pressure. (This latter requirement minimizes model geometrical anomalies due to differential deflections.) The data are to be taken at three Mach numbers (Pathfinder I design Mach number and ±0.1) and three lift coefficients (design and ±0.5). In addition, the low Reynolds number data are to be acquired for several transition strip locations (say, 10, 20, and 30% chord).

Measurements.-

- Transition location (high Reynolds number only)
- Wake rake pressures
- Configuration forces, moments, and surface pressures
- Tunnel wall pressures
- Model deformation
- LDV measurement of stream velocity in the potential flow region and near model

Experiment Number 3 -

High Reynolds Number Flat-Plate Test

Purpose.- Obtain a data base to support the extension of entrainment-type boundary-layer methods to high Reynolds number.

Test plans.- Test flat plate in 0.3-m TCT. Obtain data at several Mach numbers and a sequence of prescribed Reynolds number. Include capability to vary boundary-layer shape factor and to obtain small separation zones.

Measurements.-

- Detailed boundary-layer profile data (hot wire or LDV)
- Plate surface pressures
- Tunnel wall pressures

Experiment Number 4 -

Isolated Wing Test

Purpose.- Computer code verification at high Reynolds number and assessment of 3-D wall interference effects.

Test plans.- Model should be identical to one tested extensively at other facilities. Two geometrically similar model sizes are to be tested, one "large", another "small". Data are to be acquired for various Mach numbers, lift coefficients, and Reynolds numbers.

Measurements.-

- Wing surface pressures
- Configuration forces and moments
- Wake rake
- Boundary-layer profiles
- Wall pressures

Experiment Number 5 -

Wing Design Method Evaluation Experiment

Purpose.- Verify three-dimensional transonic wing-design code.

Test plans.- Modify a given wing model surface shape)one which has been extensively tested) using predictions of a 3-D wing-design code. Test model at conditions in the neighborhood of the wing design point.

Measurements.-

- Wing surface pressures
- Wing forces and moments
- Wake rake
- Tunnel wall pressures

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TABLE 1.- EXPECTED COMPUTATIONAL AERODYNAMIC CAPABILITIES

Anticipated availability date	Capability
1982	Grumman 3-D Pod-Pylon-Wing Code + 3-D Integral Boundary Layer + Wake
1982	3-D Transonic Wing Design Methods
1982	2-D Laminar/Turbulent Navier-Stokes Transonic Single Element Airfoil
1982	3-D Wing in Wind Tunnel (Inviscid)
1982	3-D Parabolic Navier-Stokes Subsonic Wing-Fuselage Turbulent Juncture Flow
1983	2-D Laminar/Turbulent Navier-Stokes Low-Speed Two-Element Airfoil
1984	3-D Parabolic Navier-Stokes Transonic Wing Tip Flow
1984	3-D Transonic Potential Flow/Free Vortex Sheet Interaction for Wing-Strake Flow

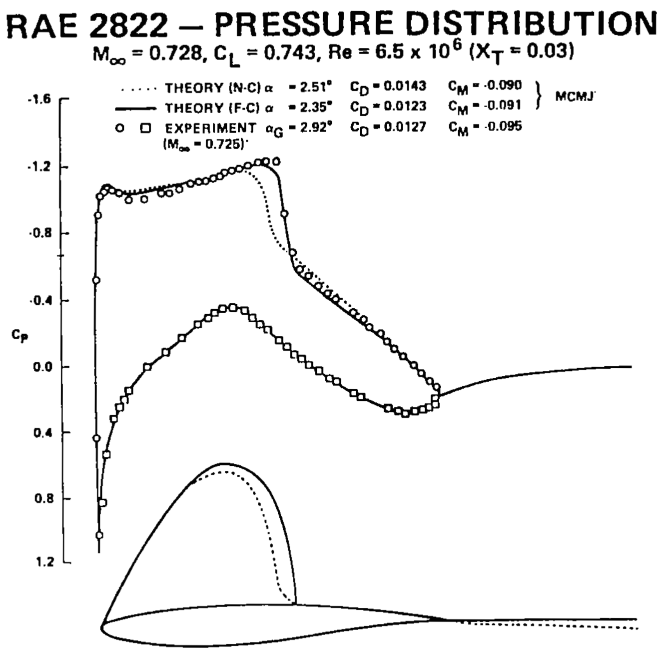
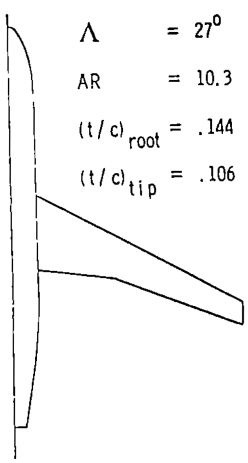
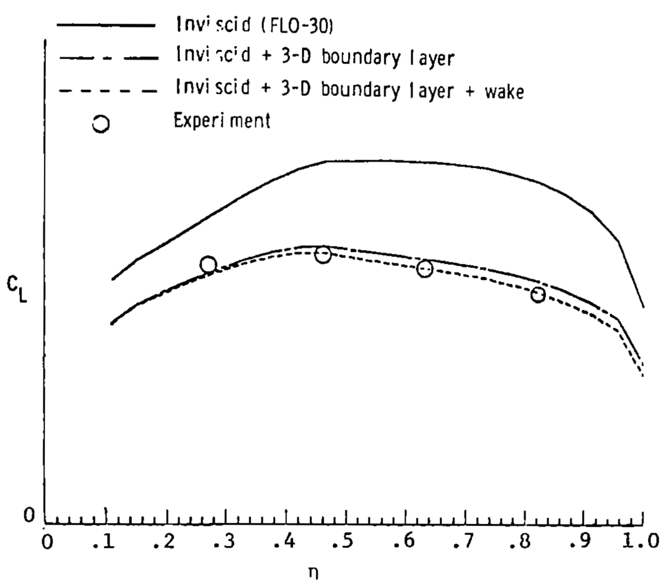


Figure 1.- Comparison of experimental data with predictions of the GRUMFOIL code (ref. 1).



Test Case Configuration



Spanwise Lift Distribution Comparison

Figure 2.- Comparison of experimental data with predictions of Streett (ref. 4).

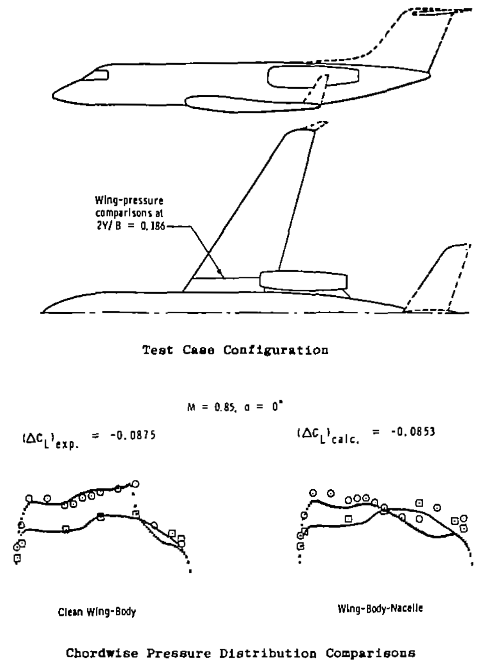


Figure 3.- Prediction of nacelle interference effects on a Gulfstream wing using Boppe's method (ref. 5).

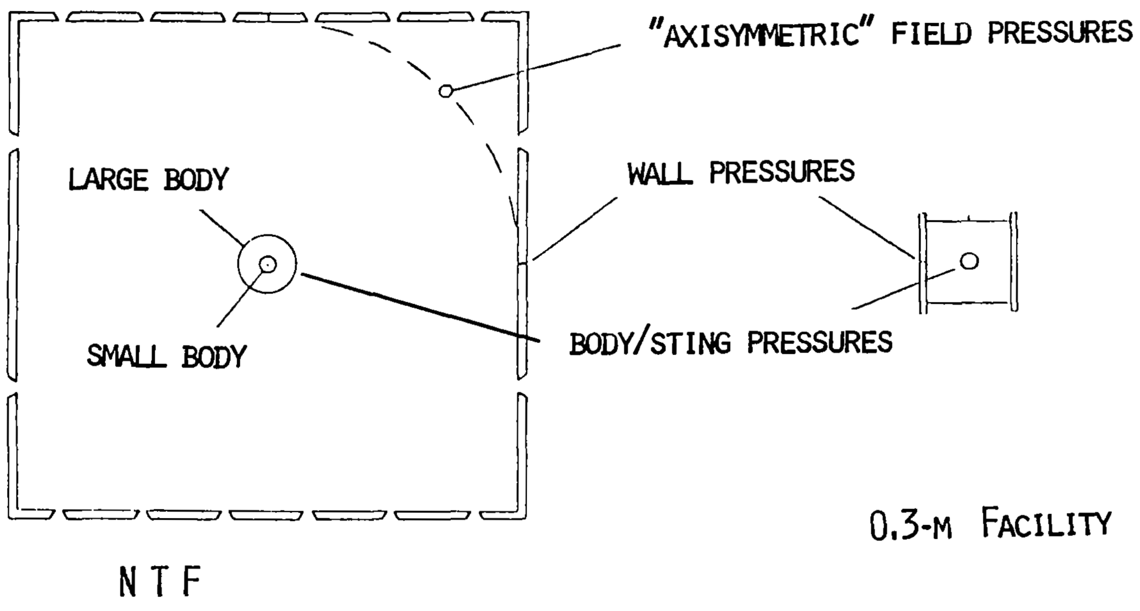


Figure 4.- Schematic of models and pressure measurement stations for experiment number 1.

QUESTION AND ANSWER PERIODS

AND

ROUND TABLE DISCUSSION

QUESTION AND ANSWER PERIODS

Donald D. Baals, editor
George Washington University

INTRODUCTORY STATEMENT

Following the presentation of each paper by the various authors and panel chairmen, the Workshop attendees were invited to raise questions or introduce comments appropriate to the discussion. The following summation of these extemporaneous questions, answers, and comments has been drawn from detailed personal notes, conversations, and other source material with only minimal post-conference review by the participants. As a result, the following commentary may err in detail, but the general content of the commentary should provide a valuable adjunct to the formal papers and panel presentations contained in this conference publication.

NTF PROJECT STATUS

(ROBERT R. HOWELL)

Comparison of NTF Operating Costs

The NTF is projected to have higher data production rates than for existing facilities, but a need was expressed for a comparison of operating costs with other facilities. The author noted that NTF is designed to generate 8000 equivalent polars per year, which is about four times the productivity of the Langley 8-Foot Transonic Pressure Tunnel, and about twice that of the Unitary Plan Wind Tunnel. NTF cost per polar in air is less than that for existing ambient-temperature tunnels. For operation with cryogenic nitrogen, however, the NTF operating cost increases by a factor of about four. The data cost for the 8' TPT is about $600 per polar, for NTF about $2000 per polar (but at greatly increased test Reynolds number).

Precision Measurement of Model Attitude

Precise measurement of the attitude of the model under test was recognized as a problem requiring further effort. Initial NTF models will incorporate an advanced internal accelerometer system, but the resultant accuracy may not be satisfactory. An industry-developed optical measurement system of high accuracy will also be utilized. Further, a dedicated optical system for measurement of model attitude is in the process of development at Langley.

Determination of Model Shape Under Load

Measurement of surface deflection (shape) under load represents a more difficult problem than that of model attitude only. Since the NTF has the unique capability for holding the aerodynamic load constant while varying Reynolds number (or varying the load at constant Re), measurement of the wing deflection under load becomes a basic requirement. Several industry-developed optical systems for measuring wing deformation are now under study for application to NTF.

NTF MANAGEMENT CONSIDERATIONS

(ROBERT BOWER)

Relationship Between NTF and 0.3-m TCT and Their Accessibility For Basic Research

An observation was made by a representative of the university community that Langley has had in operation for some time a small 0.3-m Transonic Cryogenic Tunnel. It was questioned whether this facility is to be considered a part of NTF in the broader research view, or whether it is to be a separate facility with a separate organization and operating staff. This is a key question relative to university-type research activities, for a large tunnel (NTF) may not be the best way to attack certain research problems.

It was further stated that the original concept of NTF being acceptable for sponsored research never had a constraint that all such research would be carried out in this large-scale facility, which in many ways and for many things is unrealistic to use. Much of the early discussion about the 0.3-m TCT had a lot of input with regard to it being accessible to the researcher. Now, the 0.3-m TCT is being used practically as a production tool for the development of 2-dimensional airfoils. The idea of doing basic research in that facility has been blocked by the fact that it is a very important and unique tool in a very practical sense. One should consider the 0.3-m TCT as a part of the cryogenic facility which was the original concept for NTF. There is good reason why the 0.3-m TCT should <u>not</u> be off to the side; in some ways it should be considered as an adjunct to NTF.

In response, the author noted that because of the nature of NTF as a national facility, there is need for a dedicated group to manage the NTF. The 0.3-m TCT, along with NTF, is in the Transonic Aerodynamics Division (formerly the Subsonic-Transonic Aerodynamics Division) and is tied both physically and organizationally to the same managing group, so there will be a strong interplay between them. The 0.3-m TCT is being used for many pilot experiments, data base, etc. and can be used for planning and documentation of NTF.

Right now the 0.3-m TCT has installed a 2-dimensional test section and is getting transonic airfoil data at a chord Reynolds number of 48 million, which is believed to be the highest yet achieved for this type of test. Early next year there will be installed a 2-D, self-streamlining wall test section to further enhance experimental capability. The facility will remain in a 2-D mode probably for the next couple of years as part of an extensive research program with industry for developing airfoils at high Reynolds number.

Throughout this time period, however, the 0.3-m TCT will still support basic research--at least that research which can be carried out in a 2-D test section. To enhance the productivity, the tunnel will be operated on two shifts and incorporate time savings items such as Mach number control and automation of the LN_2 supply system. It is not clear at this time when the 0.3-m TCT might receive a new test section beyond the 2-D, self-streamlining wall. It is clear, however, that the tunnel will continuously be used in a complementary mode with NTF, as will other facilities at Langley, to perform research most effectively. Potential users of the NTF and the 0.3-m TCT were strongly urged to start their planning immediately and propose to Langley as soon as possible experiments in both of these facilities.

Operational Funding for NTF

A member of the NTF Liepman Committee cautioned that the operational funding plan for NTF should not be structured so that it is dependent upon the users coming in with money. NTF funding must be provided up front to operate the facility and make access as easy as possible for the user. One must not have to go through a great procurement cycle and get on a list 4 years in advance with funding, etc., before one can get on the NTF schedule. How the user has to pay facility operating costs will be very important, for the user may be turned off because the project has become too expensive.

PATHFINDER MODEL STATUS

(CLARENCE YOUNG)

Relative Costs of Cryogenic Model Construction

For Pathfinder I model, no real cost drivers have been found, but there will be more work required for model construction than for a regular transonic model. For a Pathfinder I type model, the cost ratio should be less than 2 to 1; a fighter type model would be a tougher job. Materials costs for Pathfinder I are estimated at $12,000. No accounting has yet been made of engineering man hours. Machining costs will be comparable to that for "340 stainless" steel. All high-strength materials (cryo or non-cryo) are difficult to machine, thus machining costs are accordingly high.

Why the Complex Shapes for Pathfinder Models Instead of a Simple "Standard" Design

The question was raised as to why a simpler shape, for which there is a data bank of existing wind tunnel and flight-test data for comparison, was not selected for one of the initial Pathfinder models. The concern is that the relatively complex model selected precludes a program of wind-tunnel/flight correlation. Further, since there exists no data base of applicable aerodynamic data, the Pathfinder model results will have only limited use in a continuing aerodynamic program.

An NTF staff member responded by pointing out that the intent of the Pathfinder model program was to start with a model that would be representative of an advanced transport configuration for which a meaningful data base can be established. In the concept of Pathfinder I, the major emphasis was on structural and engineering-type problems--design, materials, fabrication--that sort of thing. But in the process of design and fabrication of this typical model for cryogenic testing, we did not want to put all this effort into a model that might be quickly discarded because of an inability to be utilized in a continuing aerodynamic research program downstream.

Pathfinder I is <u>not</u> a key model in the NTF qualification program, where one compares the results from NTF with other tunnels around the world. Consideration will be given to testing one of the so-called "standard" models (e.g. AGARD) in the NTF calibration plan, which comes up in our wind tunnel/flight correlation part of the activity. Perhaps there is a question of timing here, but it is not considered to be a case of one instead of the other. They represent two different problems. Pathfinder I is a more difficult model to design and build, so it is of most immediate concern to get it going.

Model Structural Divergence Considerations

In a comment from the audience, it was pointed out that in the past, some very high-strength models used for high Reynolds number tests have run into model/sting divergence limits which were well below the stress limits.

This structural divergence problem for NTF has been a design consideration from the beginning. It turns out that the "q" limit is really the limiting factor if one is trying to match the wind-tunnel model to the airplane flight conditions. One can limit the lift load on the model to control the stress problem, but the divergence problem is a "q" problem independent of lift. So structural divergence of the model/sting combination becomes the ultimate limiting factor on model test conditions.

Model Tolerances and Surface Finish

The problem of model tolerances and surface finish was noted to become more severe as the unit Reynolds number increases, because the boundary-layer displacement thickness becomes smaller. Experience has shown that as one controls the model dimensions and surface finish more tightly, then the model costs can suddenly increase out of hand. Will this be a problem with NTF models?

The author response pointed out that papers on this subject had been presented at the Langley Cryogenic Technology Conference of November 27-29, 1979 (NASA CP-2122). Surface finishes of 0.20 to 0.25 μm (8 to 10 microinches) (rms) have been projected, but it is currently believed this may be too restrictive. Dimensional tolerances of ± 0.025 mm (0.001 in) are being specified. Langley has an on-going program to establish model surface finishes for high Reynolds number testing. Fabrication costs undoubtedly will increase relative to more conventional model construction practices, but there are not yet enough data to determine how much of an increase.

MODEL EXPERIENCE IN THE LANGLEY 0.3-METER TRANSONIC CRYOGENIC TUNNEL

(PIERCE L. LAWING AND ROBERT A. KILGORE)

Free-Flight Correlation

In a high Reynolds number wind tunnel, the most important consideration is a check with free-flight tests. One area of concern is pressure orifice diameter. Because the displacement thickness of the boundary layer is so small, the effects of orifice diameter have to be correlated. Whether the wind-tunnel acquired data are good or bad depends on how these results compare with the free-flight, high Reynolds number case. A question was raised as to whether such checks have been made.

The response revealed no model for which there were equivalent flight data; however, validating tunnel parameters to assure accuracy in predicting flight results is an on-going program. Current studies in assessing wind-tunnel wall effects as well as plans for early installation of a variable "streamline" wall in the 0.3-m TCT are steps toward improved free-flight correlation. The problem of orifice size under test conditions of very thin boundary layers will be specifically addressed in the Panel discussions.

Changes in Model Surface Temperature

Changes in surface temperature and the attendant influence on skin friction, heat transfer, and shock boundary-layer interaction were expressed as major areas of concern. Time histories of model surface temperature will be influenced by model materials, construction details, the time allowed for equilibrium conditions, etc. These effects impact the validity of tunnel data.

This problem area will be addressed more thoroughly in subsequent Panel discussions. It was pointed out, however, that both NTF and the 0.3-m TCT have continuous-flow capability. The basic data taken to date have been run with everything in equilibrium, and this same mode of operation is planned for NTF.

Incorporation Of Control Surfaces (Ailerons, Flaps, etc.)

Relative to cryogenic model construction and testing, a concern was expressed as to how to incorporate and deflect control devices on 2-D and 3-D models. A response from the NTF staff pointed out that the new model fabrication techniques addressed in the present paper will permit the instrumentation of thin, highly-loaded surface elements. Mounting of the control elements will be by a bracket arrangement much like we do in tunnels today. The problem is whether one can fair the bracket with a filler agent that mates with the wing surface material and does not chip out. Although aerodynamic tests of control surfaces have yet to be made, development of filler agents that might be used in a cryogenic environment to fair the brackets has met with success.

RESEARCH ON INSTRUMENTATION PROGRAMS FOR NTF

(JOSEPH F. GUARINO)

From the fluid dynamics area, there was expressed a shared disappointment that NTF will not have some means of flow visualization. Instrumentation programs presented are aimed at measurements on the model surface and not into the flow stream. The avowed purpose of NTF was to study scale effects where the succession of turbulent and viscous phenomena is changing as the Reynolds number increases. Since there is no way at present to observe these phenomena, it becomes very important that something be done either with respect to flow visualization or with respect to flow measurements in the stream. One cannot rely simply on instrumentation within or on the surface of the model. This is considered a severe deficiency in NTF research instrumentation capability as now proposed.

The Langley staff reviewed briefly the technical problems associated with various flow visualization systems. The schlieren system has been the main concern for the past several years. The current NTF test section wall design represents a compromise between some desires to have all walls fully slotted with no provision for windows, and the other extreme in which there would be few slots and many windows. As noted below, substantial viewing area has been retained. Thermal stresses due to operation at cryogenic temperatures and mechanical stresses of a 9-atm test environment dictate windows of thick-wall quartz of high optical quality--a most expensive material. A schlieren system designed for use in the 0.3-m TCT unfortunately suffered one of the problems under investigation for NTF--a cracked window. This experience, as well as the high cost of large, quartz windows, has led to the decision to operate the NTF initially with the large, side wall windows blanked off with steel inserts.

The development of advanced instrumentation for operating in the high density, cryogenic environment of NTF will be a continuing process for years to come. A program is being developed to apply a laser-velocimeter system to NTF. Experiments are under way to develop a fluid to replace the old oil-flow approach for cryogenic test conditions. New interferometry techniques are under study using laser holography. Possibly a limited-view shadowgraph system can be incorporated. Hopefully, advanced instrumentation techniques will provide effective flow-visualization and flow-diagnostic systems within the aforementioned NTF constraints.

Editor's note: The test section of NTF has been designed to incorporate three 70-cm by 85-cm (27.5-in by 33.5-in) windows on the center line of each of the side walls. In addition, there are ten 15-cm (6-in) diameter viewing ports in each of the side walls as well as an array of 15 similar viewing ports in the top and bottom walls for a total of 50.

EXPERIENCE IN THE 0.3-METER TRANSONIC CRYOGENIC TUNNEL

(CHARLES L. LADSON)

The 0.3-m TCT operating in the 2-D airfoil testing mode was noted to have performance characteristics similar in some respects to the Ames High Reynolds Number Channel. A question was raised as to whether these facilities have duplicate roles or whether they have a different purpose.

In response, it was stated that the Ames facility has the same test section height, 61 cm (24 in.), but only about 40 percent of the Reynolds number capability of the 0.3-m TCT. Further, the Ames facility is non-cryogenic and attains its Reynolds number capability by high stagnation pressure (15 atm). The research emphasis of the Ames facility is in the area of turbulence modeling in direct support of their work in computational fluid dynamics, as opposed to the cryogenic technology and advanced supercritical airfoil configuration research being carried out in the Langley 0.3-m TCT.

FLOW CONDENSATION STUDIES IN THE 0.3-METER TCT

(ROBERT M. HALL)

Nucleation Sites On the Surface Of the Model

Nucleation sites on the surface of the model are not an area of concern, for the flow within the boundary layer is heated up and approaches stagnation temperatures near the model surface. Consequently, condensation will be more likely to occur in the regions of lower temperature and lower pressure outside the boundary layer.

Model Erosion in Cryogenic Tunnels

A Boeing airfoil model under test in the 0.3-m TCT came out eroded, which raises the possibility that solid ice particles could erode the model finish.

The author observed that not all models have been subject to erosion. The initial NACA 0012-64 calibration airfoil (circa 1975) was tested for about 100 hours without apparent erosion. As in the Boeing tests, however, there was a small amount of erosion on the most recent airfoil tested. At present, there appears to be a population of metallic particles going through the tunnel. Samples of gas which have been vented from the tunnel have been found to contain a very low density of metallic impurities in the range of 0.5 to 2 μm. This may have contributed to the erosion problem.

Condensation Experience At Hypersonic Speeds

An observation from the floor pointed out that there has been a fair amount of experience on condensation effects in the hypersonic regime. Most of these experimenters decided that the thing to do was to stay completely away from any degree of supersaturation. The 0.3-m TCT tests were conducted on a simple airfoil, but the fact is that future experiments will be on quite different configurations. This raises the question of how much supersaturation can be tolerated, and this depends very much on specific configurations. If the configuration has sharp corners, strong gradients, weak gradients, or shocks, the results might be quite different. There remains a question as to how these results might be generalized. There is a lot of work to be done in this area.

The author noted that the computer studies of Sivier showed that the length effects relative to supersaturation are not very critical. If one increases the temperature perhaps a half degree (K), one can essentially compensate for having a model twice as long. There may be some surprises later in the program, but for the present there is confidence that the length effects are not going to bother us as much as had been expected.

Liquid Nitrogen Impurities

A concern was expressed as to the specification for allowable impurities in the LN_2 and whether the LN_2 will be filtered.

The author stated this is an area of concern that has received more attention this past year. Specifications are very restrictive but it is recognized that residual impurities may still be sufficient to cause some problem. In the previously noted analyses of vented gases, extremely few solid particles have been found. Their size distribution appears to be very similar to aerosol distributions found in the background air. However, it may not be only the solid particles in the LN_2 which are causing all problems; there may be some liquid contaminants as well. The LN_2 specification for contaminants is very restrictive, but possibly it does not take very much contamination to form the necessary nuclei to initiate condensation. Determining the background of nuclei is under active investigation at the moment.

Contaminancy of O_2

A question was raised as to the general awareness of earlier data taken on the effects of contaminancy of O_2 and other gases which condense at much higher temperatures than N_2 and therefore may act as nuclei. The author was familiar with the subject work conducted at the California Institute of Technology in the early 1950's. A quantative analysis of the gas circulating around the 0.3-m TCT shows the gaseous contaminants to be far below the threshold where condensation effects would be found.

FLOW QUALITY MEASUREMENTS IN TRANSONIC WIND TUNNELS

AND PLANNED CALIBRATION OF NTF

(P. CALVIN STAINBACK)

Fluctuating Wall Static Pressure Measurements

The author was congratulated on making fluctuating pressure measurements and was requested to provide further information on the experimental setup. Pressure measurements in the flow field were taken on an ogive-cylinder with a static tap leading to a transducer mounted inside the probe. Probe diameter is approximately 0.5 cm (0.2 in); the pressure transducer diameter is about 0.23 cm (0.09 in). Thus there is a very short length from the cavity to the surface where the pressure is sensed. In the case of wall measurements, the pressure tap is flush with the wall.

In response to a question, it was admitted that a preliminary look at the data indicated an unexplained effect of phase change noted in correlating data taken with a probe and data taken on a wall.

Hot-Wire Calibration

In the area of turbulence measurement, it was observed that hot wires at transonic speeds are notoriously difficult to calibrate. The author noted that hot-wire data taken in the 0.3-m TCT do not exceed M = 0.8 (approximately). Data at higher speeds, however, have been taken in the 8-Foot Transonic Pressure Tunnel at Mach numbers up to 1.2, but not with the 3-wire probe.

Proposed NTF Tests With a Standard Transition Cone

A suggestion was made from the floor that the standard 10-degree transition cone that has been tested in wind tunnels around the world and in flight be tested in the NTF as part of its calibration. A response from the NTF staff revealed that tests of the transition cone were already scheduled.

Fluctuating Pitot-Pressure Measurements

An observation was made that fluctuating pressure measurements made to date provide pressure levels relative to other facilities, but that these are not absolute measurements, for they are made under a turbulent boundary layer. Is there any thought being given to the suggestion of Mabe of the RAE of making fluctuating pitot measurements towards derivation of absolute criteria of flow unsteadiness?

The author responded by noting that fluctuating pressure measurements have been made in the 8' TPT using a pitot probe, and that these results were believed to agree with measurements from the static pressure probes in the free stream. In fact, the latter measurements were found to agree with the wall pressures measured under the turbulent boundary layer. This is a somewhat surprising result but it might be only a special case.

Instrumentation Accuracy Relative To Flow Quality Measurement

The NTF is quoted and expected to have excellent flow qualities. A question was raised as to whether the NTF has instrumentation or pressure transducers accurate enough to measure the level of flow quality anticipated (e.g. total pressure gradients across the test section being within 0.1%).

It was noted by the author that his concern was with pressure fluctuations about a mean, rather than variations in the mean itself. Fluctuating pressures can be measured well within 0.1% of the mean. Accuracy in measurement of the variation in static properties across the test section will be dependent on the ultimate accuracy of the electronic static pressure measuring system.

STATUS REVIEW AND PLANNED EXPERIMENTS IN NTF

(PERRY A. NEWMAN)

Wall Corrections Where the Shock Wave Extends To the Wall*

In the subject presentation, confidence was expressed that one can correct for situations where the sonic line and shock waves extend to the wall. What are the expectations for correcting in that extreme situation where the Mach number is really one and the flow is choked?

For the aforementioned conditions, the author expressed little hope for extrapolating to free air. For less extreme situations, however, surprisingly good results have been obtained under conditions with significant supercritical flow over the model when wall corrections, using some version of linearized theory, have been based on measured pressures as boundary conditions in the numerical simulation. It remains to be seen how well the full nonlinear method will do when the flow at the wall is supercritical. There is a fundamental concern, however, whenever the wall interferes very heavily with the shock. Detail wave reflections probably will not be correct without wall adaptation.

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*Author's note: Both the question and the reply above apparently relate to flow choking as it occurs in a solid wall test section in which a sonic throat is formed at the model location. It should be noted, however, that the primary function of a transonic test section is to eliminate choking by providing a signal propagation path from the flow downstream of the model to that upstream through the ventilated walls and the plenum chamber. The severity of the wall interference which remains after the appropriate corrections to Mach number and angle of attack are applied depends on how closely the flow constraint imposed by the ventilated walls approximates that imposed by the surrounding flow in the free-air case. The assessment/correction procedures described in the subject presentation should provide a heretofore unavailable capability to quantify not only the appropriate Mach number and angle-of-attack corrections, but also the remaining uncorrectable wall effects. Only time will tell whether an intelligent exercise of this capability will lead to the identification of a wall porosity or slot geometry such that the residual uncorrectable effects are reduced to an acceptably low level even at near sonic test conditions.

Flow Measurements Near the Wall

The wall-interference assessment/correction procedure presented herein is heavily dependent upon measurements in the flow field. How is one going to measure the pressure or flow angularity near the wall?

It appears that most of the correction procedures can function from measurement of the pressures at the wall rather than the more difficult flow-angularity measurements in the flow field. Flow-angularity measurements are required, however, if one wants to extrapolate to an equivalent angle of attack in free air. But, in the wall-interference assessment/correction procedure, measurement of flow angle at probably one spot well upstream of the model should be sufficient rather than a whole distribution of flow-angle measurements. In the development of an adaptive wall, many of the aforementioned problems must be addressed; thus, much of the technology under development in the flow-measurement aspects of adaptive walls should be directly applicable to the wall-interference assessment/correction procedure under consideration for NTF.

An Adaptive Wall For NTF?

In a possible application of an adaptive wall to NTF, a question was raised as to whether the slot area would be changed or a solid wall would be contoured.

The author noted that the initial wall configuration going into NTF is essentially a more conventional slotted wall, not an adaptive wall. In the development of adaptive walls, most of the current research emphasis is on either a porous wall behind which pressures are adjusted or a solid wall where the contour is mechanically changed. A likely approach might be a combination of the two. The productivity of a fully adaptive wall will probably be poor because of the run time consumed in an iterative adjustment process. A partially adapted wall operated in the correctable condition range may be the best compromise in practice.

Langley Summary Assessment of Wall Corrections

A Langley staff member provided a wrap-up of the wall interference assessment procedure by noting that results from 2-D airfoil tests indicate there are three aspects of the wall-induced perturbations which must be considered. One aspect is the things one can correct for - the change in general level of Mach number and angle of attack. A second aspect is the type of disturbance that we are used to seeing from the old incompressible theory - that is, gradual, long-wave distortions that are apparent over the model as general slopes in the wall-induced velocities. The third aspect is seen only in the transonic test regime and this is the existence of localized irregularities in regions of supercritical flow. One interpretation is that the localized irregularities are closely related to the supersonic wave-reflection characteristics of the tunnel wall. Now that there is the hope of capability for looking at what is going on in the tunnel in sufficient detail to be able to separate these three different aspects, we might be able to focus our solutions in three different ways.

By first making corrections for Mach number and angle of attack, we unload any adaptive-wall requirements from this aspect of the problem. Next, it should be possible to employ a proper selection of slot width, porosity, or similar wall treatment to at least minimize the amount of wave reflection at very high speeds for the flow situation where the supersonic bubble extends to the wall. Finally, one might have to go to some sort of adaptive-wall capability to get rid of the long-wave slopes. But this can be accomplished with a very coarse resolution in the wall control; a fine resolution is not required.

SESSION ON NTF USER PLANNING

(RICHARD H. PETERSON, CHAIRMAN)

Need For Controlling and Measuring Disturbances In Wind Tunnels

In the Workshop presentations, a very definite need has been seen for controlling disturbances in wind tunnels and in measuring them. Mr. Stevenson's presentation reported on the effect of a honeycomb in smoothing flow in a number of wind tunnels to reduce an excessive fluttering of the model. This demonstrates the need for good management of the flow.

One thing that is missing is that what was left was not measured. The flow spectra are important in looking at the aspect of model flutter, so one must take spectral measurements of the flow in a wind tunnel in order to know exactly what the quality of the flow is. This is borne out by the results of flight experiments on the standard 10-degree transition cone where spectra were measured. There was an ability to differentiate between good tunnels and bad tunnels based on their spectra. This does not mean that good tunnels necessarily had exactly the same spectra as in flight, but it was determined that even though turbulence levels in good tunnels were two or three times larger than those encountered in flight, the content was in frequency ranges not important to transition. It is believed that we must go to some spectral information in order really to determine the carryover of certain wind-tunnel information with respect to flight information.

It was urged that not only should all tunnels be well conditioned but their spectra should be measured as well. This is a comment not only on NTF but on any tunnel that is used for advanced R&D. It is an endorsement of the efforts to be made in calibrating NTF.

Langley Response To the Problem Of Flow Quality

There is a program at Langley to look at flow quality in all our tunnels, fortified with a high desire to get the best flow quality we can in NTF with the funds available. The Langley 8' TPT tests demonstrated that cooling coils act a great deal like honeycombs. If there is a very large-scale turbulance in a tunnel (also dependent on whether the turbulence is homogenous or not), the honeycombs can provide a fantastic improvement in flow quality. Cooling coils located ahead of screens can similarly improve flow quality.

The 8' TPT incorporates both a honeycomb and cooling coils because of the interest in obtaining a very high flow quality to be able to perform laminar-flow experiments. The cell size was predicted on the scale of turbulence coming out of the cooling coil itself. A provision was made for the NTF to have a honeycomb as well, but it was subsequently eliminated as a cost-saving measure. Perhaps the honeycomb may someday be incorporated, but the presence of cooling coils just upstream of the screens is considered adequate. The combination of specially designed low-turbulence cooling coils and the four screens should provide flow quality in the NTF far superior to transonic tunnels currently in operation.

Flow Visualization and Flow-Field Measurements

A comment from the floor observed that it has become very obvious in the course of the discussions that there must be flow visualization and flow-field measurement capability in NTF. If one must infer the fluid mechanics from surface measurement only, then we are going to be short a lot of information. There should be a very definite effort to develop the necessary flow-field diagnostic capability for NTF.

Model Surface Finish Relative To Boundary Layer Transition

A member of the university research community observed that there are many ways of increasing the transition Reynolds number even for large-scale configurations. There is a major effort underway in boundary-layer control through suction. There is also a good possibility for increasing transition Reynolds number by surface cooling, with the prospect of application to cryo-fueled aircraft. There are preliminary results from an Ohio State University tunnel operating at high subsonic speeds and 80% adiabatic recovery temperatures in which transition Reynolds numbers on the order of 40 million have been reported. This gives promise of the transition level which can be attained with cryo-fueled aircraft if one were to use this means of boundary layer control.

In conjunction with these high levels of transition Reynolds number (10 to 30 million) a new round of questions can be raised relative to surface waviness, roughness, or curvature. What is the tolerance of the boundary layer with high transition Reynolds number to these various vehicle surface conditions? Thus far we are relying on very old information for ascertaining the influence of these factors. The time has come to perform a more scientific study of these manufacturing irregularities with respect to high performance aircraft. It was urged that the NTF or equivalent tunnels (perhaps the 0.3-m TCT would be adequate) study the effects of surface conditions under the situation where the transition Reynolds number would be relatively high.

Surface Finish Requirements for NTF Models

The subject of model surface finish, presumably for the all-turbulent case, drew lively comments from the floor. There was noted an apparent lack of a body of information on surface finish requirements for NTF models. The unique high Reynolds number capabilities of NTF result in extraordinarily thin boundary layers, and the associated phenomena are important and demand investigation.

Experiments with different degrees of roughness on a zero-gradient flat plate, however, are not considered adequate. One should test the same airfoil with different degrees of roughness and compare these results with an aerodynamically smooth airfoil to determine what surface conditions are critical. Whether the shock is going to be in the same location is very critically determined by the detailed skin friction. Meaninfgul results can be obtained only with realistic pressure gradients. Such experiments are very important also from an economic standpoint, for current NTF model requirements of surface finish and accuracy (typically 0.20 μm (8 to 10 microinches) and 0.025 mm (0.001 in) accuracy) can be costly.

The Langley response noted that there is a comprehensive experimental program underway to deal with transition strips and surface roughness. Tests are currently planned in the 0.3-m TCT on a 2-dimensional airfoil to study the effects of surface conditions. The Bureau of Standards has a current contract to investigate the accuracy of instrumentation available for measuring model surface finishes.

Impact Of Small Heat Transfer On Model Results

Concern was also expressed with the effects of small heat transfer in interpreting model results. It is important whether the models under test are really adiabatic or isothermal, for small differences in heat transfer may have the same effect as changes in roughness. (Small changes in heat transfer do change the local skin friction.) Temperature ratios need not be large, for values of 1.02 to 1.04 can produce significant effects.

It is very difficult to make (and test) a model which does not have some heat transfer, and what there is will not be uniform nor correspond to the real airplane. Thus, the NTF with its great capability may not correlate closely with flight because the small effects of model surface roughness and heat transfer may be significant in the overall evaluation of the data out of NTF. It was hoped that these two factors would receive detailed research attention in the 0.3-m TCT and the NTF when it comes on line.

In response a Langley spokesman stated that heat transfer tests are underway in the 0.3-m TCT. A response from industry (Douglas Aircraft) revealed that there are on-going 2-D airfoil tests having as the main objective the establishment of heat transfer tolerances under conditions of small heat transfer. Immediate concern is with 1- and 2-percent deviations and not the 10, 20, and 30% range commonly reported in the literature. Industry-sponsored tests in the Langley 0.3-m TCT will address as a secondary concern this problem of heat transfer.

Tests With External Stores

A representative from industry observed that the carriage of external stores on aircraft had not been previously addressed in the Workshop. Noting that the range of an aircraft with external weapons in a war environment is probably the critical range, it was recommended that some of the early Pathfinder models address the problem of external carriage of stores.

FLUID DYNAMICS PANEL

(PERCY BOBBITT)

Model Transition Determination

A concern was expressed that there was no capability in NTF for observing transition on the model in a visual sense. Since one cannot incorporate sufficient orifices and instrumentation on a complicated model to determine transition location, there should be emphasis on some visual form of transition.

The Langley response pointed out that heat-transfer gages provide limited information on boundary layer transition. The development of techniques for transition location was identified as a problem area for NTF that has to be worked out to a successful conclusion.

Temperature-Gradient Effects On the Boundary-Layer Profile

The gradient or slope of viscosity as a function of temperature is known to be very steep at the extremely low temperatures at which NTF can operate. Thus it would be possible to get a different boundary-layer profile at one temperature level than at another, even though the free-stream Reynolds number and Mach number were unchanged. A question was raised as to whether an estimate had been made of the magnitude of this effect and whether the panel had discussed any need for an experimental verification of data validity in the extremely low temperature range.

The panel leader commented that boundary layer calculations incorporating the real gas thermodynamic and transport properties have been made, and these should provide the kind of results discussed. Whether further sensitivity studies would indicate a need for experimental validation is not known at this time.

Comments from an NTF staff member pointed out that theoretical investigations of real gas effects on the boundary layer indicate that the primary effect will be found on the recovery temperature. For adiabatic conditions, such integral properties of the boundary layer as momentum and displacement thickness, as well as the skin friction, have been found to be little influenced by real gas effects in the operating temperature range of NTF.

WIND TUNNEL/FLIGHT CORRELATION PANEL

(THEODORE AYERS)

Absolute Data--Wind Tunnel vs. Flight

A comment from the floor expressed a long time concern that in the area of wind tunnel/flight correlations, the tendency is to accept any flight data as absolute and any other data as wrong. Although much discussion has been given to calibrating a wind tunnel, little or nothing has been siad about calibrating flight test data. Initial effort should concentrate on those areas where the changes of wind tunnel/flight correlation are good, rather than concentrating on areas where there are gross disagreements which show how bad the correlation might be. There are many things one can measure directly in flight which are not directly relatable to wind tunnel data. Emphasis should be placed on showing what the real relationships are and how the correlation studies may be carried out.

The chairman observed that the aforementioned comments are a valid concern. There is an implied response in the panel comments about concentrating on pressure measurements, wake surveys, boundary-layer measurements, etc., for we all agree it is difficult to get some measurements in flight. The ability to acquire critical flight data is steadily improving.

F-15 Research Demonstration

A member of the Fluid Dynamics Panel pointed out that the NASA Flight Center had already demonstrated with the F-15 aircraft that one can run experiments that maintain wind-tunnel-like conditions in flight. This is a technique which should be further developed at the Flight Center, perhaps with other aircraft, and should be very helpful for a wide range of testing regimes. The feasibility of conducting similar programs with another one or two aircraft that cover other ranges of speed and altitude should be evaluated.

The chairman responded that the Flight Center is currently incorporating the F-15 type research capability in one of the NASA F-104 aircraft. There is danger that the F-15 may be lost due to funding constraints, and every effort is being made to find other ways to retain that capability and further develop it.

Research Adequacy Of the Pathfinder Models

In the area of wind tunnel/flight correlation a panel member emphasized the fact that the initial research models should be configurations for which we have, or can get, some flight data to assess how well the NTF can really simulate the high Reynolds number full-scale type of data. One should not develop a lot of model data at high Reynolds number for which there is no flight counterpart. In this respect, the Pathfinder models do not address the area of concern of this panel.

SPACE VEHICLES PANEL

(DELMA FREEMAN)

A member of this panel observed that no mention of missiles had been included in the panel presentation, whereas this was an area of discussion. The panel leader, in noting that this area inadvertently had been omitted, pointed out that research on space vehicles at high angle of attack applies to cruise missiles as well.

THEORETICAL AERODYNAMICS PANEL

(FRANK THAMES, PRESENTER)

Laser Velocimeter Capability In NTF

Although a recommendation for LV capability for NTF had not been included in all panel charts, there was general agreement within the panel as to the desirability of LV measurements, for this capability will be useful in a wide variety of programs. The installation of a laser velocimeter is a high priority item, specifically in the context of wall interference effects.

The Question of Model Size

Relative to wall interference effects, a question was raised concerning the definition of "small" and "large" models used in the panel presentation, and what levels of blockage are being considered.

Subsequent discussion revealed that "large" models are of the Pathfinder model size, which is characteristic of existing models (8' TPT, etc.). Relative to "small" models, these are considered to be about half the size of Pathfinder I and II, but there needs to be a data base to establish the exact size.

Axisymmetric Model Program

A Langley staff member discussed proposed axisymmetric model tests in which a small model designed for the 0.3-m TCT would have the same blockage as a larger model installed in the NTF. Studies are currently underway to establish the best longitudinal area distribution for the axisymmetric model. Suggestions from the workshop attendees were invited.

Wall Correction Codes

Relative to the problem of trying to calibrate the wall correction codes, a suggestion was received from a Boeing aerodynamicist that consideration be given to making tests in NTF with the slots sealed. This will not provide a free-air comparison, of course, but it will provide a well-posed boundary condition and eliminate one of the unknowns in the theoretical comparison.

Storage Of Model Coordinates In a Data Bank

A general suggestion was made that coordinates of the models to be tested in NTF (as well as in the 0.3-m TCT) be measured with great precision and stored in an easily accessible data bank. As the boundary layer gets progressively thinner at the higher test Reynolds numbers, small variations in model coordinates might show up as unexplained variations in the resultant data. It would be worthwhile to be able to resolve that question quickly by ready access to the detailed model geometry before launching into an extended analysis to explain the data discrepancies.

ROUND TABLE DISCUSSION

Clinton Brown, moderator
NASA Headquarters

INTRODUCTORY REMARKS

In the course of the Workshop proceedings, good progress has been made in determining the kinds of experiments and the motivation behind these experiments now being proposed for NTF. Success in this undertaking should enable the existing investment in our facilities to be used wisely, at minimum cost, and in the national interest.

FLOW VISUALIZATION CAPABILITY IN NTF

A comment from the floor noted that flow visualization is considered a very important adjunct to NTF. Recognizing the fact that the present tunnel configuration provides for only very small viewing apertures, information was requested as to what modifications to NTF would be necessary to provide for flow visualization capability.

Although detailed sketches of the test section layout were not immediately available, a Langley staff member pointed out that there is more visibility in the test section than one might believe. For example, in the side walls there are three 76- by 91-cm (2½- by 3-ft) (nominal) windows, three along each side wall plus ten 15-cm (6-in) diameter viewing ports in each side wall. And in the vicinity of the model (in the ceiling and the floor of the test section) there are an additional 30 viewing ports.

Any significant change in window arrangement encounters structural limitations imposed by the design tunnel pressure of nine atmospheres. With NTF operating experience in hand, it might be feasible to de-rate the upper pressure capability, but even so one would be constrained by the structural design problem which dictates "x" amount of strength as well as by the attendant space available for viewing.

The NTF instrumentation package incorporating laser-velocimeter capability has been addressed in previous discussions. There is not as yet a dedicated system with a firm date for installation, but activities leading to such a system are underway in two areas: (1) research to assure that an L-V system works in a cryogenic environment such as the 0.3-m TCT, and (2) general engineering on the system. Once it has been demonstrated that an L-V system can function in a cryogenic environment, then incorporating such a system into the NTF should be relatively straightforward.

RESEARCH PRIORITIZATION AND SCHEDULE

A member of the Fluid Dynamics Panel commented that the various panels had submitted a long list of very worthwhile experiments, but there is a certain amount of overlap, as one would expect, among the various panel recommendations. The Panel on Aeroelasticity and Unsteady Aerodynamics made the only attempt to prioritize their recommended experiments. The question was raised as to whether there will be prepared: (1) an overall research priority list, and (2) an overall schedule (at least for the early years).

The moderator observed that the first research items introduced by the panels tended to represent the higher priority experiments. The NTF has always had a flat-plate boundary-layer experiment in its initial program, along with the 10-degree transition cone. Generally, the prioritization to date has been for configurations as opposed to the more basic research. The problem before us now is how to integrate the configuration-oriented experiments with the basic studies-- how to interleave these to get a more balanced program. This has not been done as yet.

ACCOUNTABILITY FOR RESEARCH RECOMMENDATIONS

A question was raised from the floor relative to research accountability; that is, who is going to do something about getting these recommended programs under way. From the flight research standpoint, these programs must be initiated promptly because of the long lead time to procure the experimental aircraft and install flight instrumentation.

WALL EFFECTS AND CORRECTIONS

An industry representative pointed out that a major area of concern expressed by several panels is that of wall effects and wall corrections. Progress in wall effects need not wait for NTF to come on line; many of the recommendations can be implemented with existing tunnels. Perhaps this effort should receive higher priority to insure that the wall-correction procedures will be available and verified when NTF comes on line.

The moderator noted that there are several programs under way at Langley that relate to wall effects. Theoretical studies are under way aimed at the wall-interference assessment and correction procedure. Also, there is a substantial effort on the instrumentation itself-- where to make measurements and how these measurements feed into the theory.

Some basic work is planned on the use of pressure pipes (e.g. Calspan) to make pressure measurements out in the flow field to define boundary conditions. There is only one tunnel appropriate for doing that kind of work (0.3-m TCT), and access to this facility is somewhat limited. There is a substantial amount of preliminary work going on that feeds into the total. But realistically most of this work will probably be completed by the time NTF comes on line.

GENERAL COMMENT ON WALL INTERFERENCE

A staff member from Langley commented that not only should there be some pre-NTF operational examination and development of these wall measurement techniques, but there is an extension of the whole area of computational fluid dynamics which has direct application to NTF and which warrants early utilization by all facilities.

Concern was expressed in a proposal of the High-Lift Panel that the slots in NTF be closed for high-lift work in NTF. This represents a situation in which, with the slots covered, one knows the wall interference picture and boundary conditions a little bit better, but in the process one must accept a larger magnitude of wall interference, including larger distortion of the flow, to a degree for which corrections cannot be made.

The wall interference assessment and correction procedures proposed for NTF using experimental measurement of conditions on an outer boundary represent an approach to the whole problem of examining what is going on inside the wind tunnel and thereby providing greater insight into the flow phenomena. This procedure can be addressed for subsonic speeds by panel methods. For example, in a high-lift problem, the model does not have to be represented in great detail. The element that must be represented with most accuracy is the outer boundary condition, and that is the one measured experimentally.

A similar concept can be addressed in the area of model support interference. Little of a specific nature has been accomplished here, but it is believed we are on the threshold of some new approaches to this problem. Further effort must be expended in applying the computational techniques that already exist to this specific problem area. It is not up to Langley alone to carry out this effort. Contributions to this effort can be made by any member of the aerodynamics community.

ROLE OF A SOLID WALL IN HIGH-LIFT TESTING

The chairman of the High-Lift Panel commented on the applicability of a solid wall in testing at high lift. The previously noted concerns that lead one to a slotted-wall configuration are certainly pertinent at transonic speeds; however, at low Mach numbers of 0.2 to 0.4-- which is where the high-lift concern exists-- solid walls have been found to be more appropriate. Considerable experience has been obtained with open walls, solid walls, and slotted walls, and for the lift-coefficient range typical of commercial transports, wall-correction techniques employing either panel methods or the methods of Heyson are found to be quite adequate.

ARNOLD ENGINEERING AND DEVELOPMENT CENTER (AEDC)

ACTIVITIES IN WALL INTERFERENCE

A commentary was made on progress in wall interference work at the AEDC where primary emphasis is on development of an adaptive wall. In the approach being followed, the wall is not called upon to adapt fully to free-air simulation. Instead, wall adaptation is limited to a certain point where the residual interference becomes correctable. This program is supported by experimental work in the AEDC 16-foot transonic tunnel and by analytical flow research to develop the supporting analytical techniques.

NASA activities in these areas have already been discussed; the intent of this comment is to inform Workshop attendees that AEDC is also active in these areas and contributing to NTF progress.

WIND TUNNEL/FLIGHT CORRELATION WITH SPACE SHUTTLE

The possibility was introduced of an active wind-tunnel/flight correlation program with respect to the Space Shuttle. It was asked whether any plans are under way now to make the necessary flight measurements that could be used for the purpose of correlation. The chairman of the Space Vehicles Panel stated that there is a fairly large effort in the Orbiter Experiments (OEX) program in which members of Langley, Johnson, and Dryden Centers will be involved in extracting the flight data. Data from the Approach and Landing Tests (ALT) program at Mach numbers of about 0.4 and 0.6 are also available and have already been published (ref. 1).

REYNOLDS NUMBER EFFECTS ON EXTERNAL STORES

A member of the Configuration Aerodynamics Panel observed that one of the major problems in the aircraft industry is the carrying of weapons and stores. There are very significant Reynolds number effects associated with carrying such ordnance. It is very important that such programs have access to NTF for detailed studies, because this is a strong factor for those aircraft designers who are trying to deliver ordnance. The panel chairman agreed that such an item should be added to the panel list of recommendations.

CONSIDERATIONS OF AN NTF OVERSIGHT COMMITTEE

A large number of diverse recommendations were noted concerning what to include in the early program of NTF. It was asked whether, if the Langley staff takes these recommendations and generates an initial program for NTF, the "oversight" committee mentioned in the paper on NTF Management Considerations (ref. 2) could be structured so that it would review the recommended programs.

The Langley response by Robert Bower questioned whether the noted "oversight" committee is the proper one for NTF program review. Because of past associations, a more likely candidate for program review and prioritization might be the existing NASA Aerodynamics Advisory Committee.

Follow-through on the research recommendations is very important. Within NASA Langley, a Steering Committee made up of technical representatives from Center directorates and chaired by Wayne McKinney has been established to review and recommend experiments to be performed in the early stages of NTF. It is expected that this committee will continue to function with the technical responsibility for prioritizing various programs insofar as Langley is concerned. This committee will seek advice from the Aerodynamics Advisory Committee and, where appropriate, from whichever organization results from that oversight-- perhaps a higher level group.

Langley is now in the early stages of implementing the NTF organization presented in the paper on NTF Management Considerations. There will be a dedicated branch to look into the initial experiments in the NTF. These will involve, of course, experiments from a broad range of research areas at Langley as well as outside the Center. This group will have the direct responsibility for taking the recommendations from this Workshop and proposing a program. Participation of NASA Headquarters will be required to provide funding, but there will be a definite setup, a line organization, to do this.

How we follow through on these recommendations will be very important. It is equally important for the potential user-- largely represented by the attendees here at this Workshop-- to get involved early in the use of NTF. Cooperative types of effort are encouraged very strongly. Proposals and ideas for experiments and tests in NTF, as well precursor experiments in the 0.3-m TCT, are invited. It is extremely important that we implement what we are talking about today, not just in-house here at Langley but with the attendees at this Workshop. One needs to gain confidence in the facility and not wait until 1985 to see what success it has and then start thinking about models and becoming involved in the program.

HOW EXPERIMENTS GET PRIORITIZED

A panel leader commented that the deliberations of these Workshops contribute heavily to the way in which various experiments are prioritized and will be conducted in NTF. The way research has been prioritized over the past 50 years will probably remain unchanged. In the areas of different disciplines, those priorities expressed by the user community will be adhered to. How they are going to be interleaved will be the responsibility of those at Langley.

The moderator pointed out that with all the panel research proposals, a multi-million dollar program of significant proportions is evident. It is clear that final prioritization will take place when dollars are laid side by side with the various programs.

EXPLORING THE UNIQUE CAPABILITIES OF NTF

In setting priorities, a panel member who had been actively involved in the early history of NTF recalled that in preparing the advocacy package for a high Reynolds number facility, many examples of high Reynolds number effects were gathered. One could look at the examples and find very few hard core high Reynolds number cases. In many of the remaining examples there were other kinds of effects-- aeroelastic effects and concurrent M/RN effects, etc. A point in setting priorities is to keep in mind the unique capabilities of NTF. It is not so much the high Reynolds number capability alone, but the capability to independently vary the RN, M, and q. The kind of testing that is to be carried out in NTF should take advantage of that unique capability.

USES FOR NTF AS YET UNSEEN

The chairman of the Configuration Aerodynamics Panel agreed wholeheartedly with the previous comments. Many uses for NTF will be found that are not envisioned at present. There is a tendency, because of the importance of finding out what happens at full-scale Reynolds number, to think of NTF as a high dynamic pressure tunnel. The Configuration Aerodynamics Panel did not consider static aeroelastic effects, for it was not clear in which group that area really belonged. Because of limited time, the panel restricted its efforts primarily to the constant "q" modes available in the tunnel, which of course are unique.

The static aeroelastic effects are very important in configuration development, and the NTF is not necessarily a high "q" tunnel for the same RN. In fact, it has a lower "q" than any other tunnel of that size. It also has the unique capability of making aeroelastic studies through the altitude range with a single model. This is possible because one can control the speed of sound and can match the altitude effects on RN, M, and q, which cannot be done in a conventional pressure tunnel (constant temperature). One should not think of the NTF as just a high "q" tunnel, because the cryogenic concept is fundamentally a low "q" concept for attaining a given Reynolds number and Mach number.

AIRFOIL PROGRAM CONFORMITY WITH PANEL RECOMMENDATIONS

In the airfoil program presented by the Fluid Dynamics Panel, a recommendation was made to choose two or three airfoil shapes as standard or baseline, followed by a long list of research items to be investigated for each airfoil. A question was raised as to whether the Langley airfoil program is consistent with the Panel recommendations regarding well-defined geometry and surface characteristics, transition studies, wake surveys, etc.

The panel leader commented that the panel recommendations include additional items such as wake surveys, transition using surface thin-film gages, precision definition of the environment in terms of spectra, etc. Most of these items have been investigated at Langley using several models, not just one model for which the geometry, surface conditions, and environment are well defined. These elements have not been put together in the same way as recommended by the panel. The panel recommendations are believed to represent an extension in concept of the current airfoil program to provide a greater research return for the effort expended.

Relative to the comment about classes of airfoils, an industry member pointed out that there is a difference in the effect of surface finish; for example, a supercritical airfoil has a fairly flat static pressure distribution on the upper surface compared to a natural, laminar-flow type airfoil which may have a favorable pressure gradient. The Fluid Dynamics Panel in its recommendations was trying to address general classes of airfoils and the effect of surface roughness on those classes of airfoils.

THE PROBLEM OF INTERFERENCE-FREE AIRFOIL DATA

One of the possible problems with airfoil tests is associated with wall interference. A member of the Theoretical Aerodynamics Panel commented that in the past it has been very difficult in the course of changing model sizes to separate Reynolds number effects from changes in wall effects. A careful airfoil test in the NTF, where airfoils of different sizes are tested at the same Reynolds number, would provide another way to define wall interference corrections. One would suspect that even in the small 0.3-m TCT, right now we do not have a satisfactory understanding of Mach number, blockage, or angle-of-attack corrections. Hopefully, larger tunnels such as NTF will provide a better grasp of the problem.

The leader of the Fluid Dynamics Panel observed that probably we do not have what would be called "interference-free" values of anything tested in the 0.3-m TCT. Based on pressure distributions at several span-wise locations, the flow about the airfoil looked very two-dimensional, but there is no pretense that the data are free of wall effects. The 0.3-m TCT will be provided shortly with a passive side-wall boundary-layer control system. In a few years this will be modified to an active boundary-layer removal system. Later still, an active streamline wall will be installed to further study the wall interference problem in that facility.

The 0.3-m TCT at present has a fixed slotted wall. Airfoil data are corrected for blockage, lift, and buoyancy. In some instances attempts have been made to utilize the wall-correction procedures outlined in the Workshop papers. To approach an interference-free condition, however, one may have to go to the larger NTF with longer span/chord ratios and reduced size relative to the test section. It remains a most difficult problem, as the Workshop attendees recognize, to obtain interference-free data.

DESIGN APPLICATION OF 2-D AIRFOIL DATA IN THE PRESENT WORLD

The moderator introduced a question of personal concern by noting that the idea of airfoil development arose essentially out of the old lifting-line theory. The idea that we can test 2-D airfoils and then apply them to the real world of a 3-dimensional, finite-aspect-ratio wing involves an evolutionary thought of employing some kind of lifting-line analysis. We have come a long way with airfoils in the cruise range, but the emphasis within NASA and throughout the country seems to be tending toward high-lift devices. Thus we are continuing 2-D development of airfoils for the high-lift condition.

Flow fields about an aircraft are highly 3-dimensional; accordingly, comments were solicited with regard to how much priority should be placed on the concept of further and extensive 2-D airfoil testing at high angles of attack. It was asked whether those from industry who actually design the airplane find that 2-dimensional airfoil data have applicability in the 3-dimensional design problems.

Transport Type Aircraft

The initial response from industry noted that insofar as high lift was concerned, there appears to be a need for obtaining 2-D high-lift data showing general trends up to 40 million Reynolds number. The Energy Efficient Transport (EET) effort has shown that if one designs a high-lift system for high Reynolds numbers, the shape of the flaps, gaps, and overlaps would be different (cf. lower RN), and perhaps it might be necessary to design for different deflections. So there is a need for 2-D work to look at what happens to the high-lift design at the higher Reynolds number. This is in comparison to a high-lift system designed at one million Reynolds number, which is typical of most of our transport development in atmospheric pressure, low-speed tunnels.

But there remains a problem peculiar to the transonic regime. The designer will always have to integrate a 2-D airfoil into a 3-D wing. The 3-D problem, however, is perhaps more difficult where one must consider handling qualities, Mach tuck, and similar constraints. Insofar as the airframe designers are concerned, they would tend to regard the 3-D integration as a tougher problem. The 2-D aspect relative to 3-D, at least transonically, is of equal importance.

Fighter Type Aircraft

For the low-aspect-ratio wings characteristic of fighter-type aircraft, serious concern exists as to whether the 2-D airfoil work is of much value to those in the fighter business. There are many questions raised even in developing codes for 2-D airfoils; thus, some doubt is cast on the 2-D methods themselves. When a designer tries to incorporate an airfoil into a low-aspect-ratio wing designed to operate in regions of separation, the problem becomes so 3-dimensional that there is serious question as to whether 2-D data, either computational or experimental, are of a great deal of value.

An opposing view from a company having extensive fighter design experience was simple and direct; two-dimensional airfoil work seems to work well for them. It was pointed out that most of their company wing design methodology is based on a class of airfoils developed by the company many years ago, and the designer is quite happy with 2-D airfoil shapes to work with. The company has spent a lot of time learning how to apply 2-D airfoils contained in the data base to the design of a 3-D wing that recovers the 2-D design pressure distributions. The results have been successful for fighter aircraft, so much so that those in the research department who have been trying to initiate 3-D wing calculations have received very little support.

A SUMMATION OF 2-D VS 3-D AIRFOIL WORK

A summary commentary from the university community pointed out that the question is not one of "either/or"; perhaps it would be better to ask whether 2-dimensional airfoil data have validity. If one has to start with 2-D data to calculate or determine 3-D effects, then the 2-D airfoil program is highly necessary. A consensus that 2-D airfoil data do have validity served as the basis for the 2-D airfoil program that was standard some years ago and had as its highlight the NASA Airfoil Conference (ref. 3) held at Langley Research Center in March of 1978. Although industry attendees at this conference strongly supported 3-D work, there was general recognition that 2-D airfoil development was highly necessary. It was clear, however, that the next step would be 3-dimensional work. But that phase is so highly configuration-dependent that it is difficult to establish a concentrated program similar to the systematic effort that has characterized 2-D airfoil development.

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16. Abstract

This report is a compilation of papers presented at the Workshop on High Reynolds Number Research held December 9-11, 1980, at the Langley Research Center. It also includes panel recommendations for research programs for the National Transonic Facility in the following areas:

Fluid dynamics
High lift
Configuration aerodynamics
Aeroelasticity and unsteady aerodynamics
Wind-tunnel/flight correlation
Space vehicles
Theoretical aerodynamics

A Workshop on High Reynolds Number Research was also held in 1976 and is reported in NASA CP-2009.

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