TEST RESULTS FROM THE LANGLEY HIGH REYNOLDS NUMBER CRYOGENIC TRANSONIC TUNNEL

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**Abstract**

The NASA has recently developed and proof tested a Pilot Cryogenic Transonic Pressure Tunnel at the Langley Research Center, Hampton, Virginia. In addition to providing an attractive method for obtaining high Reynolds number results at moderate aerodynamic loadings and tunnel power, this unique facility enables the independent determination of the effects of: Reynolds number, Mach number, and aeroelasticity. The "proof of concept" experimental and theoretical studies are briefly reviewed. Experimental results are included which indicate pressure distributions for a two-dimensional airfoil and strain-gage balance characteristics for a three-dimensional delta wing model.

**Nomenclature**

- $a$: Local speed of sound
- atm: Atmospheres
- $c$: Chord of airfoil
- $C_n$: Normal force coefficient
- $C_p$: Pressure coefficient, $C_p = \frac{P - P_m}{\rho v^2}$
- C.R.: Center of rotation
- dB: Decibel
- $h$: Height
- F: Fahrenheit
- H.P.: Horse power
- $l$: Linear dimension
- $L_p$: Sound pressure level
- $M$: Mach number
- $M_t$: Local Mach number
- $\Delta M_t$: Change in local Mach number
- $P$: Pressure
- $P_t$: Stagnation pressure
- $q$: Dynamic pressure, $q = \frac{1}{2} \rho v^2$
- $r$: Radius
- $R$: Reynolds number
- $R_e$: Reynolds number based on chord
- $R_{ft}$: Reynolds number per foot
- $T$: Temperature
- $T_t$: Stagnation temperature
- $V$: Free-stream velocity
- $v$: Local velocity
- $x$: Linear dimension along airfoil chord line
- $z$: Vertical position
- $\alpha$: Angle of attack
- $\sigma$: Standard deviation
- $\mu$: Free-stream viscosity
- $\rho$: Free-stream density
- 2-D: Two dimensional
- 3-D: Three dimensional

**Subscripts**

- $\infty$: Free stream
- $\max$: Maximum

**Introduction**

The wind tunnel, for many years, has provided an invaluable experimental device which has enabled advanced predictions of aircraft behavior and the development of improved aerodynamic concepts. It is a well-known fact among aeronautical researchers that in order to completely simulate flight conditions in the wind tunnel, it is necessary to duplicate the two essential flow similarity parameters, Mach number and Reynolds number. The development of new wind-tunnel facilities has, to the great extent, kept pace with the advances in Mach number capability. The Reynolds number corresponding to modern high-speed aircraft, however, cannot be duplicated in current test facilities in this country or abroad. For example, a conservative estimate would reveal that the flight Reynolds numbers of advanced transonic aircraft are at least five times larger than the capabilities of our most modern facilities. To illustrate this current dilemma, where could a study be accomplished in this country or in Europe to evaluate the transonic capabilities of an advanced supersonic transport at a flight Reynolds number in excess of $300 \times 10^7$?

The urgent requirement for transonic high Reynolds number wind tunnels has been recognized by researchers for many years. However, until recently, the problems imposed by economics, power requirements, and lack of technology have obscured
the need for advanced high Reynolds facilities. The method to be taken to achieve a high Reynolds number test capability has been a subject of intense study for the past several years. During the latter part of 1971, personnel at the NASA Langley Research Center undertook a series of studies to determine the feasibility of applying the cryogenic concept to a high Reynolds number transonic wind tunnel.

Operating at cryogenic temperatures was first proposed by Smelt1 in the mid 1940's as offering an attractive approach for achieving a high Reynolds number test capability. The initial Langley Research Center study was undertaken to extend the analyses of Smelt and to determine the feasibility of applying the cryogenic concept to a high Reynolds number transonic wind tunnel. Some of the theoretical and low-speed experimental results derived from these studies have been published by Goodyer and Kilgore.2 The encouraging results obtained from the initial studies and subsequent real-gas analyses stimulated the design and construction of the Langley Pilot Transonic Cryogenic Tunnel. The tunnel was placed in operation in September of 1973. Some of the initial results obtained in the pilot tunnel were presented during the AIAA 12th Aerospace Sciences Meeting.3

The purpose of this paper is to provide a brief review of the cryogenic concept and to present some of the recent findings determined in the Langley Cryogenic Pilot Tunnel.

Why Cryogenics?

There are several approaches which would provide the desired increase in Reynolds number capability. Three of these methods are depicted in Figure 1. From the Reynolds number equation, \( R = pVl/\mu \), an obvious solution would be to increase the model component lengths, \( l \). This would require excessively large, costly facilities with tremendous power requirements. An alternate solution would be to restrict the tunnel and model sizes and increase the operating pressure and, consequently, the density, \( p \). This method appears to be feasible but the required drive power and aerodynamic forces on the model and support system are drastically increased. The third method for increasing \( R \) is to decrease the temperature of the test media. As temperature is reduced, the density, \( p \), increases and the viscosity, \( \mu \), decreases resulting in the desired increase in Reynolds number. At lower temperature, the speed of sound, \( a \), is reduced which provides advantages with regard to dynamic pressure, drive power, and energy consumption offsetting to some extent the Reynolds number decrease due to the reduction in velocity, \( V \). The illustration shown in Figure 2 provides some insight regarding the magnitude of the advantages of testing at cryogenic temperatures. The ratio of several key test parameters to the parameter at a typical ambient temperature condition are plotted against stagnation temperature in degrees Fahrenheit. It can be readily seen that at cryogenic temperatures, an increase in Reynolds number by a factor of over 6 is obtained without an increase in dynamic pressure and a reduction of about one-half in the required drive power. Due to the many inherent advantages associated with testing at low temperatures, the cryogenic approach appears to be the most feasible method for acquiring transonic high Reynolds number test results.

The Pilot Transonic Cryogenic Tunnel

General Description

The Langley Pilot Transonic Cryogenic Tunnel is a continuous flow, fan-driven facility with a slotted octagonal test section, 13.5 inches across the flats. A photograph of the cryogenic tunnel during the initial assembly stage is shown in Figure 3. In this photograph, the test section, plenum, and contraction cone are being positioned for installation. From the vantage shown in Figure 5, the motor and fan are positioned in the lower-left turn in the tunnel circuit and the flow would be in the counterclockwise direction.

The tunnel is constructed of aluminum alloy and is encased in a thermal insulation about 5 inches, consisting of urethane foam covered with an epoxy-fiberglass vapor barrier. The fan is driven by a 3000-horsepower variable frequency motor. The Mach number range of the pilot tunnel can be varied from about 0.1 to 1.2 at stagnation pressures varying from about 0 to 1.2 atmospheres. The tunnel temperatures can be reduced to about 100°F by spraying liquid nitrogen directly into the flow circuit at two points, located just before the upper-left and just after the lower-left turns shown in the photograph of Figure 3. Viewing ports of 1.5-inch diameter are provided for monitoring the test section and nitrogen injection zones. With the unique ability to separately control three tunnel parameters, that is, temperature, pressure, and Mach number, it is possible to determine independently the effects of aerelasticity, Reynolds number, and Mach number.

Since the pilot tunnel was to be the first cryogenic transonic tunnel in existence and was in itself a proof of concept vehicle, a rather extensive amount of instrumentation was incorporated in the design. For instance, the installed instrumentation includes about 100 static pressure orifices, four stagnation pressure devices, 30 temperature sensors, cryogenic microphones, dew-point devices, oxygen analyzers, 15g-flow sensors, exhaust mass flow probe, torque, proximity, and vibration devices. In addition to the "permanently" installed devices, the instrumentation included multipressure probes and rakes to evaluate vertical, lateral, horizontal pressure, and temperature distributions in the test section.

Mach Number Calibrations

An extensive series of calibrations were conducted to determine the Mach number distribution in the test section. The complicated matrix of the operating envelope was covered in detail with special attention given to obtaining comparisons which would indicate any disparities that might occur between ambient and cryogenic temperature comparisons. The calibrations were made with a 3/4-inch-diameter, center line, pressure probe which surveyed about 40 inches of the test section. A photograph of the pressure probe installed in the test section is shown in Figure 4. (In this photograph and in several subsequent photographs, the insulated plenum chamber and several of the test section side plates have been removed to expose the test section.) Figure 5 indicates a sample of results which were obtained. Two different test...
conditions are included here. The results shown in the upper portion of the figure indicate the local Mach number, \( M_1 \), distribution in the test section at a stagnation pressure of 1.2 atm and a cryogenic temperature of about \(-25^\circ F\). In the lower portion of the figure, the Mach number distribution shown was recorded at a stagnation pressure of 5 atm and a temperature of 137\(^\circ\). In both cases the flow similarity parameters were almost identical. (The Mach number was 0.85 and the Reynolds number was about 10^7 Ft. \( F \)). It will be noted from Figure 5 that center of rotation, C.R., stations are indicated for two different models. (The two-dimensional and three-dimensional model studies will be discussed in a following section of this paper.). In this region of the test section, the maximum "scatter" in the local Mach number was about 0.008. It will also be noted that there is a marked similarity between the two calibrations and that in both cases there is about 20 to 30 inches of "good" flow upstream of the model. These results are fairly typical of the high subsonic, low-transonic calibration which were obtained. At the low Mach numbers the distributions were somewhat smoother and at the higher Mach number conditions the downstream sections of "good" flow. It should be noted, however, that at this time there has been no attempt made to optimize the test section flow conditions for the highest Mach number.

Temperature Distribution Calibrations

One of the recurring questions at the outset of the Langley cryogenic studies concerned the temperature distributions around the tunnel circuit. As mentioned previously, the value range of operating temperatures is obtained by spraying liquid nitrogen directly into the tunnel circuit to cool the structure and test media and to remove the heat added to the stream by the drive fan. In order to determine the extent of the mixing process and to evaluate the temperature distributions near the sharp turns in the circuit, an elaborate temperature survey ring was placed in the "screen" section of the tunnel. A photograph of the temperature sensors which were located at the intersection of the "spokes" and concentric rings. The screen section was located just downstream of the turn in the tunnel circuit shown at the upper-right portion of Figure 3.

Figure 7 indicates typical examples of the distributions which were determined with the temperature ring. These particular samples were taken at a test-section Mach number of 0.85 and a stagnation pressure of 5 atmospheres. The results shown at the left were recorded when the tunnel stagnation temperature, \( T_1 \), was 127.7\(^\circ\) F and the data at the right were taken at a cryogenic temperature of 127.4\(^\circ\) F. It should be mentioned that the tunnel stagnation temperature, \( T_1 \), used in the reduction of data is measured with a refined, individual, temperature sensor located just upstream of the temperature survey ring toward the turn vanes. (See small insert sketch.) The symbols included with the two ring sketches indicate the locations of the various measurements shown in the temperature distribution plots. The temperature plots indicate the temperature distribution across the tunnel. In both cases it would appear that it was slightly warmer near the walls of the tunnel. However, the standard deviation, \( \sigma \), (measure of dispersion around the mean), was only about 1\(^\circ\) F and at both the cryogenic and ambient temperature conditions, the multitudes of measurements were surprisingly close to the actual tunnel temperature condition.

Since the survey ring was located "upstream" of the smoothing screens and contraction section (see sketch, Fig. 7), it might be expected that the additional mixing effect would result in an even more uniform distribution in the test section. This did prove to be the case. The ultimate concern, of course, is the temperature distributions of the flow media in the region of the test section. A temperature survey rake was used in the test section which could be oriented across the test section in vertical, horizontal, or lateral positions. The photograph of Figure 8 shows the rake mounted in the vertical position. The rake consisted of seven evenly spaced temperature probes which protruded from a streamline strut support. An example of test section temperatures is shown in Figure 9. These results indicate the excellent distribution which was obtained at a cryogenic temperature of about -283.7\(^\circ\) F. (Standard deviation, \( \sigma \), was 0.5\(^\circ\) F.)

An interesting point should be mentioned here. When test section temperature distributions were taken at the very beginning of a series of tests, that is, just after "cool down," there was a moderate "bucking" shape in the distributions, indicating slightly warmer temperatures near the test section walls and plenum chamber. This trend was very similar in nature to the behavior previously discussed regarding the distributions determined with the survey ring in the screen section of the tunnel.

The temperature studies have shown remarkably good distributions. This is particularly obvious when it is considered that it is normal for wind tunnels to have temperature gradients in the test section of over 20\(^\circ\) F. These encouraging results have indicated that the method in which liquid nitrogen is injected into the tunnel can be readily solved and does not present a difficult problem area.

Noise Level Measurements

In wind-tunnel testing, background noise in the test section is a concern since excessive noise levels can obstruct the proper simulation of unsteady aerodynamic parameters usually of interest in dynamic tests. In addition, test section noise can affect angle-of-attack measurements and certain static or steady-state parameters. Due to the fact that as temperature is reduced, there are large reductions in drive power, it was expected that background noise would be reduced when a given Reynolds number was obtained at cryogenic temperatures rather than at ambient temperature. A preliminary study has been made to evaluate the effects of cryogenic operation on wind-tunnel "noise." The photograph shown in Figure 10 shows the noise measuring equipment which consisted of two cryogenic microphones, one mounted in the plenum area and the other flush mounted in the test section. The results shown in Figure 10 indicate some
trends that were determined with the test section microphone. Sound pressure level, \( P_{\text{ref}} \), in decibels is plotted against Reynolds number per foot at a Mach number of 0.80 and various pressure and temperature conditions. The noise levels are presented in terms of the broadband (10 hertz to 20 kilohertz) sound pressure level with the reference pressure taken to be \( 2.9 \times 10^{-9} \) pounds per square inch. These measurements were made during the testing of two-dimensional airfoil at 3° angle of attack and the absolute level of the noise measurements may appear to be high. The results, therefore, do not represent a pure indication of the minimum background noise and should only be used as comparative levels to indicate the general effects of changes in pressure and temperature. In Figure 11, the tunnel conditions (stagnation pressure and temperature) at which the noise measurements were taken are shown adjacent to each of the plotted points. It will be noted that at a constant Reynolds number of about 15 million, when the desired condition was obtained at the lower pressure, 1.17 atmospheres by reducing the temperature to -254.7°F, the sound pressure level was reduced by about 10 dB. The decrease in the upper portion of the figure indicates that at an almost constant pressure (about 4.9 atm), the Reynolds number is increased from 19 to 47 million per foot by reducing the tunnel temperature to -698.3°F with only a 1.5% increase in noise level. It might have been expected that a reduction in noise level would occur at the higher pressure condition when the tunnel temperature was reduced to the cryogenic condition. It is not completely understood at this time, but the absence of a reduction is believed to be associated with the increased noise resulting from the increased nitrogen injection and exhaust requirements. If this is the case, special attention to this subject would very possibly result in additional noise reductions in large cryogenic tunnels. An extensive analysis of test-section noise has not been made to date; however, based on this limited amount of information, the results have been promising.

Experimental and Theoretical Drive Power and Fan-Speed Studies

During the initial calibrations and aerodynamic testing, measurements were made of both the drive-shaft torque and fan speed to enable comparisons with theoretical predictions at various temperature, pressure, and Mach number conditions. At this time, the drive power results have not been completely evaluated. However, based on power supplied to the drive motor, it appears that drive power varies roughly as predicted, that is, for constant pressure and Mach number, power varies directly with the speed of sound \( \sqrt{T} \).

A sufficient amount of fan-speed measurements have been made to evaluate the agreement between theory and experimental results. Figure 11 shows the theoretical and experimental variation of fan speed with temperature at a Mach number of 0.85 and a stagnation pressure of about 4.9 atm. The Reynolds number variation for this series of studies varied from about 19 \( \times 10^5 \) per foot at the highest temperature to 100 \( \times 10^5 \) per foot at the lowest cryogenic condition. The experimental results, shown by the circular symbols, indicate that the fan speed actually decreases somewhat faster than predicted by simple theory (speed \( \propto \sqrt{T} \)). The lower values of fan speed at the cryogenic conditions are believed to be associated with the beneficial effect of the greatly increased Reynolds number around the tunnel circuit and/or the increased fan efficiencies at the lower operating temperatures.

Two-Dimensional Airfoil Study

As mentioned earlier, the real-gas studies which were conducted by Langley Research personnel had substantiated that flow characteristics are insignificantly affected by nitrogen "imperfections" at cryogenic temperatures. Shortly after the initial tunnel calibrations, a series of two-dimensional airfoil tests were conducted to provide experimental confirmation of the cryogenic concept. The configuration selected for these studies was a 12% thick, NACA 664 airfoil equipped with pressure orifices. A photograph of the model is shown in Figure 12. The 5.4-inch chord airfoil completely spanned the 13.5-inch test section. The sketch shown at the lower left of the figure indicates that at subcritical speeds, the 664 series airfoils have a " Riotop" vortex in the flow direction, which appears to be more promising. A photograph of the model installed in the test section of the cryogenic pilot tunnel is shown in Figure 13.

There were several conditions which were selected to assess a pure cryogenic evaluation: (1) ambient and cryogenic temperature tests were to be made in the same tunnel, on the same model, at identical Mach and Reynolds numbers; (2) the airfoil was to be tested with free transition to allow any possible temperature effect on boundary-layer development; (3) the symmetrical airfoil was to be tested at a lift coefficient of 0 to eliminate any shape or angle-of-attack change due to dynamic pressure difference; and (4) test conditions would exceed the leading-edge Mach number of typical sonic transports.

Figure 14 illustrates the operating envelope considered in the 2-D airfoil studies. Stagnation pressure, \( P_{\text{st}} \), is plotted against Reynolds number per foot, and based on model chord. The boundaries of the envelope are dictated by the stagnation pressure (horizontal line at temperature (diagonal curves). The temperature curves indicate the operating envelope at 120°F, 250°F, and at the theoretical local and free-stream saturation limits. The circular symbols illustrate the \( M = 0.85 \) test configurations performed at Reynolds numbers ranging from about 4.9 \( \times 10^5 \) to 100 million per foot. Results are included herein which will indicate the two-dimensional pressure distributions obtained at identical Reynolds numbers at ambient temperature and at a cryogenic temperature of -250°F. (Note the darkened symbols, Fig. 14.) In addition, pressure results will be discussed which indicate the feasibility of obtaining increased Reynolds number capability by testing beyond the saturation boundaries.

Figure 15 indicates comparisons of pressure results which were obtained at cryogenic and ambient temperature conditions. These pressure coefficients were obtained with the two-dimensional airfoil at an angle of attack of 0°. The circular symbols indicate upper surface pressure results at a stagnation pressure of about 4.9 atmospheres and ambient temperatures. The square symbols indicate
results obtained at about 1.2 atmospheres and -250° F. Pressure coefficients are plotted against model chord station for two different Mach number conditions. (Mach 0.85 and Mach 0.75.)

Several types of flow phenomena are illustrated here. The Mach 0.75 results are unmistakably subcritical. The Mach 0.85 results reflect that the local Mach number reached the speed of sound at a Cp of about -0.3 and continued to increase until a shock was produced. In both cases, there was an excellent agreement between the cryogenic and ambient temperature results. This agreement was achieved despite the sensitivity of the airfoil to changes in Mach number, and the large variation in the speed of sound at the two temperatures. These results clearly demonstrate the ability to set the tunnel conditions accurately and substantiate the theory that cryogenic gaseous nitrogen is a valid test medium.

Figure 16 indicates an example of the studies which were made to examine the effects of the theoretical saturation boundaries. The variation of pressure coefficients are shown for two different pressure and temperature conditions at a Reynolds number of 27 million. The circular symbols indicate results obtained at -210.2° F, well above either the local or free-stream saturation limit. The square symbols reflect pressure coefficients obtained at a temperature equivalent to the theoretical free-stream saturation boundary. The excellent agreement between the results suggests that additional Reynolds number capabilities can be expected from cryogenic wind tunnels by testing at temperatures approaching the theoretical free-stream saturation boundary.

The two-dimensional airfoil tests substantiated that cryogenic test conditions can be set accurately and that gaseous nitrogen is a valid transonic test environment allowing the achievement of high Reynolds numbers without increases in aerodynamic loading. In addition, this particular series of studies indicated that substantial increases in Reynolds number capability may be realized by testing beyond the theoretical saturation boundary.

Three-Dimensional Model Tests

Three-dimensional model tests have been made in the cryogenic pilot tunnel to evaluate the acquisition of strain-gage balance results for conventional, sting-mounted, models at cryogenic temperatures. The configuration selected for this study was a sharp leading-edge, delta model having an aspect ratio of 1.07 and a sweep of 75°. The overall length of the model was 7.07 inches and the maximum span was 4.22 inches. A photograph of the model installed in the pilot tunnel is shown in Figure 17.

Longitudinal forces and moments were measured with an electrically heated, three-component balance. Pressures at the base of the model were measured and considered in the reduction of the data. Angle of attack was determined with an "off-the-shelf" electrically heated accelerometer enclosed in a thin, styrofoam shell. A sample of the results which were obtained are shown in Figure 18 which indicates the variation of normal force coefficients with angle of attack at a Mach number of 0.95. The theoretical prediction of normal force determined by the method of Polhamus is shown by the solid line. The circular symbols indicate experimental results obtained at a pressure of 1.2 atm and a cryogenic temperature of about -250° F. The square symbol results were taken at test conditions of 5 atmospheres and a warm temperature of 125.8° F. The Reynolds number, based on model chord was the same, 8.4 x 10^6, for both of the experimental cases. It will be noted from these results that there was an excellent agreement between theory, and the experimental results obtained at both the ambient and cryogenic temperature.

The three-dimensional model results have provided additional evidence that gaseous nitrogen is a valid test medium even under conditions of separated and reattached (vortex) flow. In addition, there has been no indication of any major problem areas associated with obtaining angle-of-attack or strain-gage balance results at cryogenic temperatures.

Conclusions

1. The cryogenic approach offers a feasible method for acquiring transonic high Reynolds number test results at acceptable levels of dynamic pressure and tunnel drive power.

2. With the unique ability to separately control three tunnel parameters, that is, temperature, pressure, and Mach number, it is feasible to determine independently the effects of aerelasticity, Reynolds number, and Mach number.

3. Temperature studies conducted in the Langley cryogenic pilot study have shown remarkably good distributions indicating that the method in which liquid nitrogen is injected into the tunnel can be readily solved.

4. Noise level studies conducted in the cryogenic pilot tunnel have indicated that there are very promising noise reduction trends associated with cryogenic operation.

5. Two-dimensional and three-dimensional model tests have substantiated that cryogenic test conditions can be set accurately and that gaseous nitrogen is a valid test environment.

References


\[ R = \rho V \mu \]

- **INCREASE SIZE**  \[ \text{(INCREASED I)} \]
- **INCREASE PRESSURE**  \[ \text{(INCREASED \( \rho \))} \]
- **DECREASE TEMPERATURE**
  - INCREASED \( \rho \)
  - DECREASED \( \mu \)
  - DECREASED \( \alpha \)

**CRYOGENIC TUNNEL**

**Figure 1.** Three methods to increase wind-tunnel Reynolds number.

**Figure 2.** Some effects of cryogenic operation.

**Figure 3.** Photograph of pilot transonic cryogenic tunnel during initial assembly.

**Figure 4.** Photograph of pressure calibration probe installed in test section.

**Figure 5.** Mach number calibration results at \( M = 0.90 \).

\[ P_T = 1.2 \text{ ATM, } T_I = -290.9, \ R \eta = 19.8 \times 10^6 \]

\[ P_T = 5.0 \text{ ATM, } T_I = 137°, \ R \eta = 19.5 \times 10^6 \]

**Figure 6.** Photograph of temperature calibration ring installed in screen section of cryo tunnel.
Figure 7. Temperature distributions in screen section. $M = 0.85, P_T = 5 \text{ atm}$.

Figure 8. Photograph of temperature calibration rake installed in test section.

Figure 9. Temperature distribution in test section. $M = 0.85, P_T = 5 \text{ atm}$.

Figure 10. Test section sound pressure levels adjacent to a model at angle of attack. $M = 0.80$.

Figure 11. Comparison of theoretical with experimental fan speeds at $M = 0.85$.

Figure 12. Proof of concept, two-dimensional airfoil model.
Figure 13. Photograph of two-dimensional airfoil installed in test section.

Figure 14. Test conditions for two-dimensional airfoil at $M = 0.85$.

Figure 15. Comparison of two-dimensional airfoil pressure distributions at ambient and cryogenic conditions. $\alpha = 0^\circ$.

Figure 16. Comparison of two-dimensional pressure results indicating the effects of operating at the theoretical saturation limit. $M = 0.85$, $\alpha = 0^\circ$, $R_e = 27 \times 10^6$.

Figure 17. Photograph of three-dimensional delta model installed in test section.

Figure 18. Experimental and theoretical normal force results determined for the three-dimensional model at $M = 0.85$. 