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Abstract

High levels of disturbances in the test section of wind tunnels can have adverse effects on aerodynamic data and make the data on models difficult to interpret. Because of this, some wind tunnels are built to have low disturbance levels in their test sections. A tunnel with this characteristic is the Langley Research Center’s Low Turbulence Pressure Tunnel. This tunnel became operational in the spring of 1941 and remained in operation or in a semiactive status for 40 years without a major overhaul. From 1979 to 1982, this tunnel was rehabilitated and its cooler and screens in the settling chamber were replaced. Tests were conducted to measure the disturbance levels across the cooler, screens, and nozzle and in the test section.

Symbols

\( f \)  
frequency

\( M_\infty \)  
free-stream Mach number in test section

\( p_0 \)  
total pressure

\( \rho \)  
rms of pressure fluctuation

\( \rho_\infty \)  
mean free-stream static pressure in test section

\( q \)  
local dynamic pressure

\( R_e \)  
unit Reynolds number in test section

\( T_0 \)  
total temperature

\( t \)  
time

\( \bar{u} \)  
mean longitudinal flow velocity

\( \bar{u} \)  
rms of velocity fluctuations in longitudinal direction

\( \lambda \)  
integral length scale of turbulence

Introduction

The National Advisory Committee for Aeronautics (NACA)/National Aeronautics and Space Administration (NASA) has, from time to time, made concerted efforts to design and build wind tunnels which have test Reynolds numbers approaching those encountered in flight and in addition have low disturbance levels in the test section. One of these tunnels is the Low Turbulence Pressure Tunnel (LTP) located at the Langley Research Center (LRC). This facility combines the desirable attributes of the old Variable Density Tunnel (VDT) and the NACA Ice Tunnel (later noted as the Langley Two-Dimensional Low Turbulence Tunnel (TDLT)). The VDT was built in the 1920's and operated at 20 atmospheres total pressure in order to simulate flight Reynolds numbers of that era. However, the disturbance levels in the test section were high and these disturbances had adverse effects on aerodynamic test results. Because of this, the Ice Tunnel was built ostensibly to conduct research on icing conditions of aircraft, but it was built actually to conduct research aimed at developing a low turbulence level wind tunnel. The Ice Tunnel operated at atmospheric conditions and the low turbulence levels in the test section were obtained by the use of a large contraction ratio (19.6 to 1) for the nozzle and by placing a honeycomb and screens in the settling chamber. Once the disturbance levels were found to be low in the Ice Tunnel, a third tunnel was designed to have the high Reynolds number capability of the VDT and the low disturbance levels obtained in the Ice Tunnel. The wind tunnel built as a result of this design study was the LTP. The LTP became operational in the spring of 1941. It was built to conduct two-dimensional airfoil testing and it served as a major facility for the development of airfoil shapes during World War II.

As flight speeds increased in the 1950's and 1960's, the apparent need for the LTP decreased and the tunnel was placed in a semiactive status in 1964. It remained in this state until 1968 when the need for low-speed high Reynolds number data on hypersonic vehicles was recognized. In addition, there was an increased interest in the two-dimensional testing of advanced airfoil sections at low speeds and high Reynolds numbers and low Reynolds numbers. (In addition to being operated at total pressures above atmospheric pressure, the LTP can also operate at sub-atmospheric pressures.)

Because the LTP had been in operation since 1941 without a major overhaul and because of a renewed interest in its unique capability, the tunnel underwent a major overhaul between December 1979 and March 1982. Two of the major items replaced were the cooler and screens. After the replacement of these items, it was desired to remeasure the disturbance levels in the tunnel. This was done in 1982 and the results of this investigation are presented herein.

Facility and Test Conditions

The LTP is a pressurized, closed-circuit, continuous-flow wind tunnel (See figure 1). The test section was designed for two-dimensional testing of wing sections and the test section is 7-1/2 feet high, 7-1/2 feet long, and 3 feet wide. The nozzle is relatively short with a

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contraction ratio of 17.6 to 1. The nominal range for the test conditions are:

\[
\begin{align*}
0.1 \leq & \quad P_0, \text{ atm} \leq 10 \\
60 \leq & \quad O_F \leq 120 \\
0.05 \leq & \quad M_m \leq .4
\end{align*}
\]

A cooler/heater is available to cool or heat the working fluid to the desired temperature within the constraints given above. The cooler/heater is made of two rows of tubes. The tubes are 0.75 inch in diameter with 0.013 inch thick fins pressed over the tubes with a fin spacing of 8 fins per inch. The total thickness of the cooler in the flow direction is 5 inches. There are nine screens installed in the settling chamber 18 inches downstream of the cooler. The screens are of 39 mesh having wires of 0.005 inch in diameter. The individual screens are 3 inches apart and the most downstream screen is 15 inches upstream of the entrance to the nozzle.

**Instrumentation and Data Recording and Reduction**

Hot wire probes, operated with a constant temperature anemometer, were used to measure fluctuating velocities across the cooler, screens, contraction and in the test section. Pressure transducers were mounted in the walls of the test section flush with the walls.

Data were recorded on a tape recorder and the recorded data were later used to obtain autocorrelations and spectra using a commercially available analyzer. The rms levels in the test section were very low and the output levels from the anemometer were corrected for electronic noise using the technique of the appendix.

**Results and Discussion**

The velocity fluctuations in the flow direction measured upstream and downstream (upstream of screens) of the cooler are presented in figure 2. The levels of the velocity fluctuations do not appear to change significantly across the cooler. The levels of the fluctuations range from about 5 to 15 percent depending on the unit Reynolds number. The invariance of the level of the velocity fluctuations across the cooler agree with the results of reference 4. However, the scale of turbulence is significantly reduced across the cooler and results substantiating this will be presented subsequently.

The measurements made downstream of the screens are also shown in figure 2. These levels are very low at the lower unit Reynolds numbers, being around 0.5 percent. The level of the fluctuations increased to about 0.4 percent at the highest unit Reynolds numbers. The reduction in the levels of the velocity fluctuations across the screens are very large and range from factors of about 30 at high unit Reynolds numbers to over 100 at the lower Reynolds numbers. The pressure drops across the screens were measured by John B. Peterson, Jr., and are presented in figure 3. These pressure drops and the theory of Dryden (ref. 1) were used to predict the theoretical attenuation of the velocity fluctuations across the screens and the results are presented in figure 2. There is good agreement between the theory and data up to a test section unit Reynolds number of about $3 \times 10^6$. Above this value of Reynolds number, the data are somewhat below the levels predicted by the theory.

Velocity fluctuations measured in the test section are shown in figure 4 for various Mach numbers over the Reynolds number range of the test. The levels range from about 0.025 percent at $M_\infty = 0.05$ and $Re/ft$ 1 x $10^6$ to 0.30 percent at $M_\infty = 0.20$ and $Re/ft$ 10 x $10^6$. The variation of the fluctuations with unit Reynolds number, at each Mach number, appears to have a minimum. This minimum occurred at higher unit Reynolds numbers as the Mach number increased.

The data that are available from hot wire measurements made in 1941 (refs. 1 and 2) are also presented in figure 4. Except for the datum point at the lowest unit Reynolds number, there is very good agreement between the present and the 1941 data. The disagreement between the two sets of data at the lower unit Reynolds numbers is due to a Mach number difference. For the 1941 data, the Mach number for this point was about 0.02. This value of the Mach number is below the lowest Mach number investigated during the present test. The extrapolation of the present data to $M_\infty = 0.02$ at $Re/ft = 5 \times 10^6$ would indicate a velocity fluctuation below 0.01 percent.

The pressure fluctuations measured at the wall in the test section are presented in figure 5 for a total pressure of one atmosphere. The level of the fluctuating pressure normalized by the local static pressure (figure 5a) ranged from about $10^{-5}$ to $10^{-3}$. When the fluctuating pressures are normalized with the local dynamic pressure (figure 5b), the values range from $6.9 \times 10^{-3}$ to $13.0 \times 10^{-2}$ over the Mach number range of the test. The lower level of $\overline{P}/q$ at $M_\infty = 0.05$ agrees well with the pressure fluctuations measured under a turbulent boundary at $M_\infty = 0$ as reported in reference 5. At the higher Mach numbers, the increase in $\overline{P}/q$ can probably be attributed to the increase in the pressure fluctuations in the free-stream due to noise from the diffuser propagating upstream into the test section or to noise from the fan due to increased power.

**Integral Length Scale**

The autocorrelation functions were obtained upstream and downstream of the cooler, downstream of the screens, and in the test section. The integral time scales were obtained by integrating the autocorrelation function up to the first zero crossing. The integral length scales were obtained by multiplying the integral time scales by the local free-stream velocity.
The integral length scales measured in the test section with the hot wire mounted normal to the flow and with the pressure transducer mounted flush with the wall are presented in figure 6. At a total pressure of 15 psia, the length scales in the free-stream ranged from about 3 feet at low Mach numbers up to 12 feet at the higher Mach numbers. At 60 psia, the length scales were much smaller at the higher Mach numbers, having lengths which ranged from about 3.5 to 5 feet.

The integral length scales measured at the wall under the turbulent boundary-layer were small, and at 15 psia, ranged from about 0.15 foot at low and high Mach numbers and reached a peak of about 0.3 foot at a Mach number of about 0.22.

The length scales measured across the cooler are shown in figure 7. Upstream of the cooler, the length scales increased from about 0.14 to 0.8 foot as the test section Mach number increased from 0.05 to 0.40. There is a significant reduction in the length scale across the cooler. The reduction factors range from about 2.5 to 10 with the larger reductions occurring at the higher Mach numbers.

As shown in figure 2, there were only a slight change in the levels of turbulence across the cooler. However, the length scales were reduced significantly. The smaller length scales downstream of the cooler should promote the decay of turbulence and these reduced length scales should have a favorable influence on the efficiency of the screens in further reducing the levels of turbulence in the settling chamber. Therefore, in addition to maintaining a constant total temperature in the test section, the cooler is also an effective flow manipulator which improves the flow quality in the tunnel test section.

It was not possible to obtain a realistic integral scale of turbulence downstream of the screens because of the low disturbance levels existing there. The best estimate of the integral scale of turbulence was obtained at Mach numbers of 0.40 and 0.30 at 15 and 60 psia, respectively. At these conditions, the scales of turbulence were about 0.016 to 0.020 foot. Using these values, the reduction factor for the length scale due to the screens was about 5.

Hot Wire Spectra

Examples of the spectra obtained with the hot wire anemometer upstream of the cooler are presented in figure 8. The spectra are broadband without any significant discrete frequencies observed over the frequency range of the spectra. This absence of discrete frequencies is apparently due to the large, broadband velocity fluctuations which occur and overshadow other possible disturbances. The upper frequencies of the measurable velocity fluctuations at 15 psia range from about 300 Hz at 0.05 Mach number to 5000 Hz at 0.40 Mach number. At 60 psia, there is a slight increase in these upper frequencies and they range from about 500 to 7000 Hz at $M_w = 0.05$ and 0.29, respectively.

The spectra measured downstream of the cooler (upstream of the screens) are presented in figure 9. These spectra are also broadband with only a few low-level, discrete frequencies being measurable at the higher frequencies at the low Mach numbers and 15 psia. There is a shift in the spectra to frequencies higher than those measured upstream of the cooler and this fact agrees with the reduction in the integral scales of turbulence measured across the cooler as noted before. At 60 psia, the spectra are also broadband, without evidence of measurable discrete frequencies.

The spectra measured downstream of the screens are shown in figure 10. The velocity fluctuations in this region are very low and because of this, many low-level, discrete frequencies are measurable in the spectra over the Mach number and total pressure ranges of the present test. Discrete frequencies occur at 60 Hz and its harmonics. The higher harmonics are particularly evident at $M_w = 0.05$ and 0.10. These higher harmonics become less evident as the Mach number is increased since, at the higher Mach number the broadband energy from velocity fluctuations dominated these discrete frequencies. The fundamental frequency of the fan blades is below 233 Hz and some of the discrete frequencies could be due to the fan. At 60 psia, the spectra also contain discrete frequencies associated with 60 Hz, but these frequencies decrease in relative values as the Mach number is increased.

The spectra measured in the test section at 15 psia are shown in figure 11. Because of the low levels of the velocity fluctuations, discrete frequencies were measurable at 60 Hz and its harmonics. The relative levels of these frequencies decrease significantly with increasing Mach number. Note that attempts were made to eliminate the effects of the 60 Hz noise from the rms levels presented in figure 4 by using the technique presented in the appendix. There are significant discrete frequencies occurring at about 2500 Hz at the higher Mach numbers. These frequencies are higher than the fan blade fundamental frequencies and there appears to be no evidence of higher harmonics; therefore, the discrete frequency at 2500 Hz has not been explained at this time.

Conclusions

The Langley Low Turbulence Pressure Tunnel (LPTT) was completed in the spring of 1941 and was designed to have a very low turbulence level in its test section. Over the years, there was a deterioration of the cooler and screens. There was a general rehabilitation of the tunnel from 1979 to 1982 and after the replacement of the cooler and screens, flow quality measurements were made in several locations to determine the disturbance levels. From the measurements made across the various components and in the test section, the following conclusions can be made.

1. The velocity fluctuations measured upstream and downstream of the cooler were approximately equal and ranged from 5 to 15 percent.
2. The velocity fluctuations downstream of the screen were low. The levels ranged from 0.05 to 0.40 percent from low to high unit Reynolds numbers.

3. The attenuation of the measured velocity fluctuations across the screens agreed with the predictions from a theory by Dryden.

4. The velocity fluctuations in the test section ranged from 0.05 percent at 0.05 Mach number to 0.3 percent at 0.2 Mach number at the highest unit Reynolds number. The present hot wire data agreed well with the data reported for the tunnel in 1941.

5. The pressure fluctuations measured at the walls of the test section at \( M = 0.05 \) agreed with previous measurements made under a turbulent boundary at \( M \sim 0 \).

At higher Mach numbers, there was an increase in the measured wall pressure fluctuations, probably due to noise from the diffuser or from the fan.

6. The autocorrelation functions indicated that there were large reductions in the integral scale of turbulence across the cooler and screens.

7. The spectra measured upstream and downstream of the cooler were mostly broadband. There were several measurable discrete frequencies in the spectra downstream of the screens and in the test section. Many of these frequencies were due to 60 Hz and its harmonics.

References


Appendix

Correcting Hot Wire Data for Electronic Noise

Symbols

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>A and B</td>
<td>constants in equation A - 6</td>
</tr>
<tr>
<td>E</td>
<td>mean voltage</td>
</tr>
<tr>
<td>( E_n )</td>
<td>mean voltage across hot wire when electronic noise is measured</td>
</tr>
<tr>
<td>( e' )</td>
<td>instantaneous voltage across hot wire due to velocity fluctuations</td>
</tr>
<tr>
<td>( e'_n )</td>
<td>instantaneous voltage across hot wire due to electronic noise</td>
</tr>
<tr>
<td>( \tilde{e}_T )</td>
<td>total rms voltage measured by rms meter</td>
</tr>
<tr>
<td>( \tilde{e}_n )</td>
<td>rms voltage due to electronic noise measured by rms meter</td>
</tr>
<tr>
<td>G</td>
<td>amplifier gain</td>
</tr>
<tr>
<td>( G_n )</td>
<td>amplifier gain for electronic noise measurements</td>
</tr>
</tbody>
</table>

\[
S_u = \frac{3 \log E}{\log u} \text{ velocity sensitivity} \\
\tilde{u} \text{ mean velocity} \\
\tilde{u}_n \text{ apparent mean velocity indicated during measurement of electronic noise} \\
u' \text{ instantaneous velocity fluctuations} \\
u'_n \text{ apparent instantaneous velocity fluctuations due to electronic noise}
\]

When electronic noise is great enough to influence measurements made with a hot wire anemometer, the noise must be subtracted from the gross output from the anemometer to obtain the correct net result. To make this correction, the hot wire equation can be expressed as:

\[
\frac{(e' + e'_n)}{GE} = S_u \frac{(u' + u'_n)}{\tilde{u}} 
\]

The level of the electronic noise can be approximately obtained by covering the hot wire probe and measuring \( \tilde{e}_n \) and \( E_n \). Under this condition, equation A-1 can be expressed as:

\[
\frac{e'_n}{G_n E_n} = S_u \left( \frac{u'_n}{u'_n} \right)
\]
Forming the mean square of equation A-1 and assuming that there is no correlation between the electronic noise and the velocity fluctuations gives:

\[
\frac{(e' + e_n')^2}{G^2 E^2} = S_u^2 \left[ \left( \frac{u'_u}{u} \right)^2 + \left( \frac{u'_n}{u} \right)^2 \right]
\]

A-3

Solving for the velocity fluctuation of the flow gives:

\[
\left( \frac{u'_u}{u} \right)^2 = \frac{(e' + e_n')^2}{S_u^2 G^2 E^2} - \left( \frac{u'_n}{u} \right)^2
\]

A-4

Substituting equation A-2 into A-4 results in:

\[
\left( \frac{u'_u}{u} \right)^2 = \frac{(e' + e_n')^2}{S_u^2 G^2 E^2} - \left( \frac{u'_n}{u} \right)^2 \times
\]

\[
\frac{e'_n^2}{S_u^2 G_n^2 E_n^2}
\]

A-5

A value is required for the quantity \( \frac{\bar{u}_n}{\bar{u}} \) in equation A-5. From the calibration of the hot wire, we have:

\[
\log E = A \log \bar{u} + \log B
\]

A-6

which can be written as follows by noting that

\[
\bar{u} = \left( \frac{E}{B} \right) S_u
\]

A-7

Then using A-7 \( \frac{\bar{u}_n}{\bar{u}} \) becomes

\[
\frac{\bar{u}_n}{\bar{u}} = \left( \frac{E_n}{E} \right) S_u
\]

A-8

Substituting equation A-8 into A-5 gives the final result:

\[
\left( \frac{u'_u}{u} \right)^2 = \frac{(e' + e_n')^2}{S_u^2 G^2 E^2} - \left( \frac{E_n}{E} \right)^2 S_u
\]

A-9

or in terms of the rms voltages measured with a

\[
\left( \frac{u'_u}{u} \right)^2 = \frac{\tilde{e}^2}{S_u^2 G^2 E^2} - \left( \frac{E_n}{E} \right)^2 S_u
\]

A-10

Fig. 1.- Schematic of the Low-Turbulence Pressure Tunnel and Probe locations.
Fig. 2.- Velocity fluctuations measured upstream and downstream of the cooler and across the screens.

Fig. 4.- Velocity fluctuations measured in the test section. ○ Data from reference 2.

Fig. 3.- Pressure drop measured across a single screen.

Fig. 5.- Pressure fluctuations measured at the wall in the test section. \( P_0 = 1 \text{ atm.} \)
(b) $\bar{p}/q$

Fig. 5.- concluded.

Fig. 7.- Integral length scale of turbulence measured across the cooler.

Fig. 6.- Integral length scale of turbulence measured in the test section.

Fig. 8.- Spectra measured upstream of the cooler.
(b) $P_0 = 60$ psia

Fig. 8.- concluded.

(a) $P_0 = 15$ psia

Fig. 9.- Spectra measured upstream of the screens.

(b) $P_0 = 60$ psia

Fig. 9.- concluded.

(a) $P_0 = 15$ psia

Fig. 10.- Spectra measured downstream of the screens.
(b) $P_o = 60$ psia

Fig. 10.- concluded.

Fig. 11.- Spectra measured in the test section. $P_o = 15$ psia