AENG 411: Aerospace Laboratory

Wind Tunnel Testing of a Complete Aircraft

by

Tom Moline

Member of Group NO. 2

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Submitted to: Mr. Larry Boyer

Department of Aerospace and Mechanical Engineering

Parks College of Engineering, Aviation, and Technology

Saint Louis University

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# Summary

The objective of this experiment was to utilize the wind tunnel in Oliver Hall to analyze the flow characteristics associated with a model aircraft. This was accomplished by attaching the model to a tunnel balance that is capable of measuring forces and moments in 6 separate directions. Before the data collected from this balance could be used, however, it was necessary to account for disruptions in the flow around the model that developed as a result of flow interactions between the walls of the tunnel and the freestream flow, as well as the presence of the aircraft itself. Once these errors were accounted for, the data that resulted was analyzed, as to determine the stability characteristics associated with the aircraft. Ultimately, it was found that the data collected was highly impacted by error associated with the use of the tunnel balance, even when the error corrections were taken into account. Even with these errors, it was still possible to determine the lift curve slope, minimum drag, maximum lift, and static margin associated with the aircraft. All of these results can be found in Sections (7-8), with the raw data being stored in Section (9). Thus, the lab as a whole was a success, in that the flow characteristics associated with a model aircraft were determined through the use of a wind tunnel.

# Nomenclature

*B* Test Section Width

*(Value)w* Wing Coefficient, Value, or Parameter

*(Value)tail* Tail Coefficient, Value, or Parameter

*(Coefficient)u* Uncorrected Coefficient

*(Coefficient)c* Corrected Coefficient

*b* Geometric Span

*be* Effective Span

*C* Test Section Cross Sectional Area

*d* Maximum Diameter of Fuselage

*H* Test Section Height

*k* Ratio of Effective Span to Tunnel Width

*K1* Body Shape Factor for Blockage

*K2* Fuselage Shape Factor for Blockage

*l* Length of Body

*lt* Distance from CG to ¼ MAC of Tail

*q* Freestream Dynamic Pressure

*qc* Corrected Freestream Dynamic Pressure

*Re* Reynolds Number

*S*Area

*αg* Geometric Angle of Attack

*αc* Corrected Angle of Attack

*δ* Boundary Correction Factor

*εT* Total Solid Blockage Correction Factor

*εsbB* Body Solid Blockage Correction Factor

*εsbW* Wing Solid Blockage Correction Factor

*εstuts,windshields* Strut and Windshields Solid Blockage Correction Factor

*τ1* Tunnel Correction Factor for Blockage

*τ2* Downwash Correction Factor

*c­t* Tip Chord

*cr* Root Chord

*λ* Taper Ratio

*MAC* Mean Aerodynamic Chord

*CM* Moment Coefficient about ¼ Chord of MACwing

*CD* Coefficient of Drag

*CL* Coefficient of Lift

*L* Lift

*D* Drag

*M* Moment about ¼ Chord of MACwing

*AR* Aspect Ratio

Mean Aerodynamic Chord

*V* Freestream Velocity

*CG* Center of Gravity

*CLW* Wing-Only Lift Coefficient

Variation in Pitching Moment Coefficient with Horizontal Tail Incidence Angle

*a* 2-D Lift Curve Slope

Horizontal Tail Velocity Coefficient

*FA* Strut Frontal Area

*tstrut* Strut Thickness

*hstrut* Strut Height

*P* Ambient Pressure

*T* Ambient Temperature

*ρ* Ambient Density

Volume

# Introduction

## Vibration of a Cantilever Beam

A cantilever beam is a defined as any long, constant cross-sectional area structure that is firmly fixed at one of its ends and free at the other. When a structure such as this is subjected to vibration, it behaves in very specific ways that have been well documented by engineers and scientists over the years. For example, if the thickness (t), width (w), and length (L) of the beam are known (And thus the moment of inertia of the beam, I), as well as its modulus of elasticity (E) and density (ρ), the natural frequencies of said structure can be found from the simple relationship listed below:

(1)

The i subscript listed on each side of the equation refer to the mode that said natural frequency is associated with. A mode is a state of oscillation in which each part of the beam oscillates with fixed phasing, in that, if one portion of the beam deflects a certain amount at a particular time and another beam deflects a different amount at that same time, there will exist a steady time over which said configuration repeats itself over and over. The number of a mode is associated with the number of nodal points that can be observed while it is taking place. A node point is a point at which there is no displacement of the beam relative to its neutral axis over the course of oscillation. If none of these nodal points exist, the mode number associated with the beam’s vibration is one. If one exists, the mode number is two, and so on and so forth.

For a simple cantilever beam, the location of the nodal points for a particular node can be found from the relationship listed below, where λ and σ are empirical values associated with finding the natural frequencies of constant area cantilever beams, and χ’ is the non-dimensional length said beam (x/L).

(2)

Based on these predictions, the actual natural frequencies and nodal points associated with a particular cantilever beam can be found through vibration testing. This is accomplished by setting the vibration frequency of the beam to those around the predicted natural frequencies, at which point specific values for the natural frequencies for each node can be found through the use of a strobe light. This is done by only altering the frequency of the strobe light until the beam appears to be stationary. The stationary wave that appears when this value is found is called a standing wave and represents the maximum deflection of the beam above and below its neutral axis for a given mode. With those results in mind, one can draw comparisons between the natural frequencies that were predicted by theory and those that were actually observed, as to assess the material properties associated with a given beam.

## Vibration of a Cantilever Wing

The definition of a cantilever wing is essentially the same as that for a cantilever beam, with the only difference between the two being the shape of the structure that is being secured. Though the definition of these two structures is similar, the way they behave when subjected to vibration is not. In the case of a cantilever beam, its width is small enough to discount any two-dimensional effects that may result from its vibration, effectively reducing it to a one-dimensional object that spans the length of the beam. For a wing, however, its area and width are large enough to allow for these two-dimensional effects to come into play, thus making it very difficult to develop standardized equations for its behavior when subjected to vibration.

With this in mind, there are two ways to go about finding the modal characteristics of a particular wing shape: Finite Element Modeling (FEM) Analysis and Experimentation. FEM Analysis is quick and simple, consisting of modeling the wing shape in a Computer Aided Design (CAD) program, setting up the fixtures associated with the wing (in this case, fixed at the root and free everywhere else), and then simulating vibrations in the vertical direction as to model the behavior of the wing at its various natural frequencies. This type of analysis works will for simplified shapes and fixtures (such as that of a thin, trapezoidal, cantilever beam), but becomes more time intensive and difficult as the complexity of the shape to be tested increases. However, the results that are obtained from this method are very precise and clear, making them a good reference source for physical testing.

Experimentation, on the other hand, allows for exact answers as to the natural frequencies associated with a given mode of vibration, as well as offers a general idea of the nodal lines associated with said modes. This is accomplished by coating the top of the wing with a fine, granular material (such as sand), and then adjusting the vibration frequency of the wing until the material begins to form specific shapes on the surface of the wing. The nodal points then are found based on the areas of the wing at which the material begins to congregate, since it is these areas where the least amount of displacement occurs in the vertical direction. Though it is clear where these nodal lines generally lie, they are by no means an exact measurement of where they occur in the wing itself. In order to have a more confident understanding of these nodal locations, it is absolutely necessary to compare them to those predicted by FEM analyses.

# Design of Test

## Cantilever Beam Test

The Cantilever Beam Test Apparatus consisted of a long, constant-area beam that was secured at its middle to a pneumatic, sinusoidal vibration apparatus. The frequency of the vibration apparatus was controlled through the use of a signal generator, while the frequency itself was measured through the use of a strobe light.

## Cantilever Wing Test

The Cantilever Wing Test Apparatus consisted of a trapezoidal, swept wing that was fixed at its root to a vibration apparatus. The frequency of the vibration apparatus was controlled by a function generator, while the nodal lines on the wing were found through the use of sand particles.

# Test Procedure

## Cantilever Beam Test Procedure

The procedure for conducting vibration testing of a cantilever beam is listed below:

1. The length, width, and thickness of the beam were measured through the use of a tape measure and caliper.
2. The area moment inertia of the beam was calculated by multiplying the width by the height cubed and then dividing the result by 12.
3. The theoretical natural frequencies of the beam were then found through the use of Equation (1) in Section (3).
4. The vibration frequency of the testing apparatus was then set to a value near that associated with the first natural frequency of the beam.
5. The lights of the testing area were then turned off and a strobe light was used to calculate the actual natural frequency associated with the first mode of the beam.
6. The location of the nodal points relative to the fixed end of the beam were then found through measurement
7. Steps 4-6 were then repeated for the second and third vibration mode of the beam.

## Cantilever Wing Test Procedure

The procedure for conducting vibration testing of a cantilever wing is listed below:

1. The dimensions of the wing were measured though the use of a caliper.
2. Sand was sprinkled over the top of the wing.
3. The frequency of the vibration apparatus was set to a value near that of the first mode of the wing.
4. The frequency of the vibration apparatus was adjusted until the sand particles located on the top of the wing began to form specific patterns on the top of the wing.
5. The frequency at which this occurred, as well as an image of the top of the wing, were recorded.
6. This same process was then repeated for three more modes of vibration.

# Test Results

## Cantilever Beam Test Results

Before any data was collected from experimentation, the geometry of the beam being tested was measured, its area moment of inertia was found, and its material properties were recorded, as listed in Table 6-1.

**Table 6-1. Beam Dimensions and Material Properties**

|  |  |
| --- | --- |
| **Strut Characteristics** | |
| Length (in) | 20.5625 |
| Width (in) | 1.02 |
| Thickness (in) | 0.055 |
| Moment of Inertia (in4) | 1.414x10-5 |
| Area (in2) | 0.0561 |
| Young’s Modulus (psi) | 10x106 |
| Density (lb-sec2/in4) | 25.55x10-6 |

Once these values were recorded, the theoretical natural frequencies of the beam were found through the use of Equation (1) in Section (3). These values were then compared to the natural frequencies obtained from the actual experiment. These values, along with the location of each nodal point, are listed in Table 6-1.

**Table 6-2. Theoretical and Actual Natural Frequencies/Nodal Points**

|  |  |  |  |  |
| --- | --- | --- | --- | --- |
| **Mode** | **Frequencies (Hz)** | | **Nodal Point Locations (in)** | |
| **Theoretical** | **Actual** | **First** | **Second** |
| 1 | 3.95 | 4.16 | N/A | N/A |
| 2 | 23.35 | 26.05 | 16.25 |  |
| 3 | 66.5 | 72.95 | 17.75 | 10.0 |

**Table 6-2. Airplane Geometry**

|  |  |
| --- | --- |
| **Airplane Characteristics** | |
| Tip Chord (in) | 1.289 |
| Root Chord (in) | 2.579 |
| Wingspan (in) | 24.375 |
| Wing Thickness (in) | 0.35 |
| Tail Root Chord (in) | 1.75 |
| Tail Tip Chord (in) | 1 |
| Tail Thickness (in) | 0.17 |
| Fuselage Diameter (in) | 2 |
| Fuselage Length (in) | 12 |

**Table 6-3. Wind Tunnel Test Section Geometry**

|  |  |
| --- | --- |
| **Wind Tunnel Characteristics** | |
| Width (in) | 40 |
| Height (in) | 28 |
| Length (in) | 54 |

Based on the values recorded in each of these tables, various correction factors and flight characteristics associated with the model can be found, as discussed in the next section.

Finally, for each case discussed in the procedure section, angle of attack, force, and moment measurements were taken. Due to the number of these values, and thus the relative size of the tables associated with them, the data has been stored in the appendix.

# Discussion of Results

Based on the airplane characteristics listed in Table 6-2, the wing aspect ratio, taper ratios, volumes, and areas were calculated and stored in Table 7-1 below.

**Table 7-1. Extended Aircraft Characteristics of Wing, Tail, and Fuselage Sections**

|  |  |
| --- | --- |
| **Aircraft Characteristics** | |
| Wing Taper Ratio | 0.50 |
| Center Chord (in) | 2.76 |
| Wing Area (in^2) | 49 |
| Wing Volume (in^3) | 17.3 |
| Tail Taper Ratio | 0.571 |
| Tail Area (in^2) | 7.56 |
| Tail Volume (in^3) | 1.29 |
| Wing MAC (in) | 2.01 |
| Tail MAC (in) | 1.41 |
| Wing CG Relative to MACw (in) | 0.50 |
| (in) | 9.5 |
| Volume (in^3) | 21.60 |

Once these values were found, the lift, drag, and moment coefficients associated with each test run were calculated. This was accomplished through the use of the standard CD, CL, and CM equations, based on the drag, lift, and pitching moments found in Tables 9-1 through 9-4, SW, MACw, and the temperature and pressure found within Oliver Hall during testing, which was found to be 70 ⁰F and 101 kPa respectively, which translated to a density of 0.002377 slugs/ft3.

After this, the process of correcting these raw coefficient values began, which involved running through the process described in detail in the introduction. The first step in this process involved plotting CL2 vs. CD in the wing on, full speed test run. The results of this process are plotted in Figure 7-1 below. This plot, which theoretically should show a linear increase in CD over CL2, clearly did not. This is likely due to the fact that the corrections that are to be put into place later in this section have not yet been accounted for, meaning that the raw data is not a good representation of what the aircraft would actually experience in a real world scenario. However, it was necessary to find a CDo value in order to begin the correction process, so a linear curve was fit to the data, and a value of 0.00001551 was found.

**Figure 7-1. CL2 vs CD Plot for an Airspeed of 65 mph and with the Tail Attached**

Based on this value, the total blockage correction factor was found through the use of Equations (3-7) in the Introduction. From this blockage factor, a corrected dynamic pressure was calculated through the use of Equation (8). The results and intermediated steps of this process are recorded in table 7-2 on the next page.

**Table 7-2. Blockage Correction Factor Buildup and Results**

|  |  |
| --- | --- |
| **Blockage Correction Factor Calculations** | |
| K1 | 1.04 |
| K3 | 0.93 |
| εsbwing | 0.000412 |
| τw | 0.86 |
| τ1 | 0.855 |
| b/B | 0.61 |
| εSBF | 0.000458 |
| εstruts | 0.00247 |
| CD0 | 0.00001549 |
| εwbTail | 1.71E-07 |
| εtotal | 0.00334 |
| Dyanmic Pressure Correction (lbf/in^2) | 727.9 |

With this corrected dynamic pressure value in mind, each flight characteristic coefficient was recalculated, resulting in uncorrected versions of each coefficient. The next step in refining these values then involved accounting for the effect of the tunnel walls on the angle of attack that the aircraft experienced. This was accomplished by finding the lift curve slope of the aircraft from the test run where the wing was detached. This slope, calculated from the uncorrected values shown in Figure 7-2, was found to be 0.00048316 /rad. Based on this value, and through the use of Equation (9) in the Introduction, the change in angle of attack imparted by the tunnel on the aircraft was 0.108 degrees. The results and steps involved with this process are shown in Table 7-3 below.

**Table 7-3. Angle of Attack Correction Factor Buildup and Results**

|  |  |
| --- | --- |
| **Angle of Attack Corrections** | |
| Clw (/rad) | 0.00048316 |
| bv/b | 0.78 |
| be | 12.58 |
| k | 0.314 |
| δ | 0.115 |
| τ2 | 0.49 |
| Δα | 0.108 |

The final correction factor to be calculated was that of the pitching moment coefficient. This was accomplished through the use of Equations (11-13) in the Introduction. The results of this process are listed in Table 7-4 on the following page.

**Table 7-4. Pitching Moment Correction Factor Buildup and Results**

|  |  |
| --- | --- |
| **Pitching Moment Corrections** | |
| aη | 0.0533 |
|  | 0.72529583 |
| dCMcg/ddeltas | -0.038658268 |
| btail/B | 0.075 |
| tau2,tail | 0.15 |
|  | -8.14018E-07 |

These values, combined with the corrections made through the use of Equations (1-2) in the Introduction, allowed for the calculation of the corrected coefficient values for each test run. The results of this work are shown in Tables 9-1 through 9-4 in the Appendix.

Based on the values listed in each of those tables, the first plot that was created was that of CL vs alpha for the uncorrected and corrected cases, both with and without a tail. The results of this are shown in Figure 7-2 below.

**Figure 7-2. CL vs alpha With Corrected and Non-Corrected Coefficients**

As can be seen from the graph, there isn’t a large difference between the non-corrected and corrected CL values in both the tail and no tail runs. This suggests that the effect of the aircraft and struts on the total airflow through the tunnel was minimal, which makes logical sense, since the total cross section of the plane is much smaller than that of the test section. Another interesting conclusion that can be made from this graph is that the maximum CL in the tail run is higher than that with no tail. This suggests that the tail is positively deflected, adding a small amount of lift to the overall lift of the aircraft, specifically, a change of 0.00002 between each configuration. **Thus, the maximum CL of the total aircraft was found to be 0.00021. Also from this plot, the zero lift angle of attack was found to be -10 degrees and the lift curve slop was calculated as 0.00001551 /rad, as previously discussed.**

The next graph that was plotted based on the collected data was that of CD vs alpha, as shown in Figure 7-3 below. From this graph, one notes that the correction factors incorporated into the calculation process actually had an impact on the final results. This suggests that in terms of pressure drag build up, the struts and tunnel walls actually had a damaging impact on the overall drag that the aircraft experienced, thus making it worthwhile to factor their effects out of the results. It is also important to note that the drag associated with the aircraft with its tail on is significantly less than that with its tail off. This is likely due to the extra buildup of pressure drag on the aircraft, since its stability was solely reliant on the forces of the wing as opposed to those with the tail and the wing. **Thus, the CDmin  of the aircraft was 0.0000005.**

**Figure 7-3. CD vs alpha With Corrected and Non-Corrected Coefficients**

With this in mind, the next two graphs to be plotted were those of Cm vs CL and alpha, as shown in Figures 7-4 and 7-5 below. As can be seen from each of these graphs, the uncorrected data appeared to be much more accurate than the corrected values. This may suggest that the corrections already in place in the LabView control software already accounted for the variations in Cm without any further user input, thus leading to the discrepancies seen when incorporating the error corrections.

Regardless, it can be seen that the aircraft achieves greater stability without a tail than with a tail, though both exhibit a negative slope, thus meaning that stability can be achieved. **The slope in the case with a tail can be found to be -0.00127 /rad, meaning that the static margin of the aircraft is 0.00127, which is fairly close to the CG of the aircraft.**

With all of this correction in place, there was still a large amount of error with the lab. This suggests that the balance used to collect data is still not particularly calibrated. Another potential source of error was air leakage between the balance and the bottom of the test section, which could lead to the development of extraneous data and error between the raw and calculated error corrections.

**Figure 7-4. CM vs Alpha with Corrected and Non-Corrected Coefficients**

**Figure 7-5. CM vs CL with Corrected and Non-Corrected Coefficients**

# Conclusion

The objective of this experiment was to analyze the flow characteristics associated with a model aircraft through the use of wind tunnel testing. This was accomplished by placing an aircraft model in the test section of the wind tunnel in Oliver Hall for various angles of attack in two separate configurations: with without a tail. Various correction factors were calculated as to account for the drag and wake disruption associated with the presence of the model and the walls of the chamber, as discussed in the Introduction. Based on these correction factors, and the geometry of the model, the lift, drag, and moment coefficients associated with the model at each angle of attack was found. The results were then plotted in Figures 7-2 through 7-5, from which several aircraft stability characteristics were found, as discussed in Section (7). Ultimately, it was found that there existed a large amount of error in the data that was collected. As discussed in Section (7), this was likely due to mis-calibration of the tunnel balance, as well as leakage airflow at the bottom of the test section. However, this error didn’t change the fact that the data collected indicated that the model used could achieve static longitudinal stability. Thus, wind tunnel testing was able to provide relevant data about a model size aircraft, thus indicating the success of the lab as a whole.