The Cause and Effect of Boundary-Layer Laminar to Turbulent Transition Literature Survey

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Abstract

This paper entails a literature review of computational, flight testing, and experimental testing to determine how boundary layer transition is being viewed and determined, the key components that causes the transition of the boundary layer from laminar to turbulent to occur, as well as what parameters are affected by the transitions. Closing remarks will be made about what the author has observed through researching the boundary layer transition.

Introduction

The understanding of when and why transition from laminar to turbulent flow in the boundary layer occurs is essential to accurate flight predictions and equipment and crew safety. Where turbulent flows are described as "unpredictable" and abrasive, laminar flow is described by having a smaller amount of viscous drag but is also much more likely to separate due to its low momentum [4]. When the boundary layer transition occurs, changes in properties such as a drag, vorticity, dilatation, and heat transfer rate can be observed [4]. Understanding changes in properties such as heat transfer rate is essential to being able to design adequately surface cooled atmospheric hypersonic vehicles. The processes that cause a flow transition to take place occur due to disturbances that either happen on the body of the vehicle or in the free stream medium that the vehicle is moving through. Disturbances in the free stream consist of vortical, acoustic, and thermal perturbations while disturbances on the body consist of roughness, vibrations, periodic suction and blowing, and surface heating perturbations [6]. The freestream disturbances entering the boundary layer is defined as receptivity and provides initial disturbances conditions such as frequency, amplitude, and phase [7]. Turbulence normally begins due to the first mode instabilities (Tollmien Schlichting waves (TS)) increasing and second mode instabilities (acoustic waves) becoming damped out [4]. Some of the properties that can affect the rate and location at which the transition occurs are Mach number, Reynolds number, curvature, roughness, and bluntness [5]. Testing and observing boundary layer transition can be observed three different ways, through live flight tests, simulation/mathematical modeling, and through wind tunnel experimentation. Each method is not without limitations and the possibility of skewed results [6]. Live flight testing provides the most accurate results, but it is very hard to control your testing parameters as well as it is the most expensive of the three methods [6]. Due to the high cost of ground and flight-testing, computational fluid dynamics, specifically RANS that is coupled with structural and thermal solvers, are used to estimate the boundary layer transition at desired flight conditions including Mach number and altitude properties. Certain models such as the $Langtry - Mentery\gamma - \theta_{re}$ (LM) are used in simulation to better observe the effects of boundary layer transition. Mathematical methods such as e^N, linear stability theory (LST), and parabolized stability equations (PSE) are also used to predict transition. Though due to complexity and non-linear effects better mathematical models need to be developed so that effects such as roughness and high freestream turbulence can be predicted adequately [5]. Experimentally, wind tunnels are also used along with Rayleigh scattering simulation technique, Particle Image Velocimetry (PIV), temperature and pressure paints, pressure transducers, high performance CCD (charged coupled device) camara, and thermal couples to determine placement, intensity, and characteristics of transition from laminar to turbulent in the boundary layer [2]. Disturbances such as noise vibrations, caused by conventional wind tunnels can cause the transition flight characteristics to be higher than true inflight values making the results questionable [1]. To alleviate the questionability of wind tunnel results, quite wind tunnels are now being used because the produce similar results to true flight conditions. This is because quite wind tunnels are designed to limit disturbances that occur due to the operation of the wind tunnel. With fewer unnatural disturbances occurring, the flow transitions more naturally from laminar to turbulent as it would if it were a plane in flight. As of now, three quite hypersonic wind tunnels exist in the world, one at Purdue University, one at Texas A&M University, and one at Peking University [1]. This paper will look at examples from all three types of experimentation and determine conclusions about the effects and causes of boundary layer transition.

Literature Survey

Effects of Boundary Layer Transition on the Aerodynamic Analysis of high lift systems [1]

This article used a Mach number of .2 and a Reynolds number of a 4.3 million. The authors used RANS $Langtry - Mentery - \theta_{re}$ (LM) model which is used to predict boundary layer transition and compared it to Shear Stress Transport (SST) model with in CFD ++ simulator and experimental data for high lift wing tested in NASA Langley subsonic wind tunnel that is shown in figure 1.

Figure 1 High Lift Wing placed in Nasa Langley wind tunnel.

The purpose of this experiment was to determine how LM model picked up flight parameters such as the coefficient of lift, drag, pressure, and skin friction when compared to SST model. These coefficients are critical because along with determining flight performance they can also be used to determine when the transition from laminar to turbulent flow is going to occur. The authors confirmed that as the skin friction or surface temperature grew the likely hood that transition would occur also increased, similarly the author also determined that as an adverse pressure gradient increased the likely hood of flow separation increased. The results of the LM model better followed the results that were obtained from the experimental data, proving that the SST model alone was not adequate for properly modeling the flow around the wing and body of the aircraft. There were many instances such as in figure 2 where the LM model observed flow separation or separation bubble that the SST model did not.

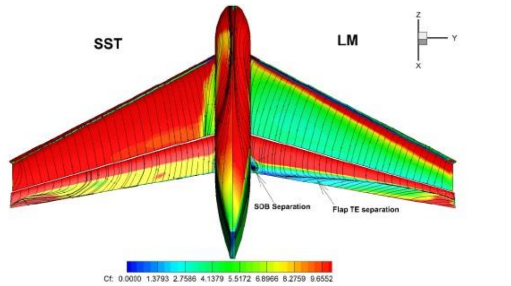


Figure 2 Measure of Skin friction using LM method (right) and SST method (left)

The e^N mathematical model was also used to verify the location of the boundary layer transition which matched well with the LM prediction. In conclusion, this article showed that a boundary layer model should be included when using CFD to simulate flow that could possibly have transition occur. The article also confirmed that as transition occurs that skin friction coefficient and surface temperature increase.

Transition in hypersonic boundary Layers [7]

The effect of high-speed flow in the boundary layer was observed in this article. In a hypersonic boundary layer flow transition is important factor in understanding when and how much aerodynamic heating, drag, and entropy production will occur. A Mach number of 6 and a Reynolds number of 9.7 x 10^6 was considered by using the hypersonic wind tunnel at the Peking University. Due to the high-speed flow, Rayleigh-Scattering visualization, fast response pressure transducers, CCD camara and particle image velocimetry (PIV) were used to observe the effects on the boundary layer. A linear stability theory was also preformed to confirm results from experimental data. A flared tip nose cone with 13 pressure transducers placed down it is center with a nose cone angle of 5 degrees and a zero angle of attack was used in the wind tunnels test section which is shown below in figure 3.

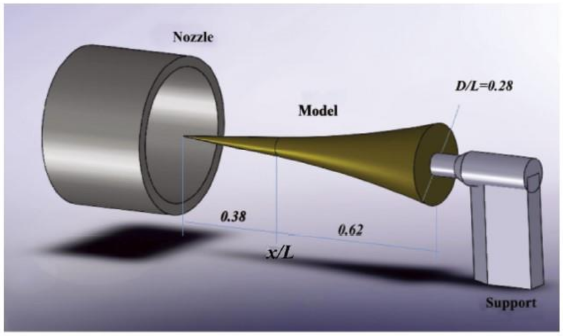


Figure 3 Model of Flared Cone in M6QT

For the PIV method TIO2 particles were seeded into the flow and were illuminated using a laser [2]. The authors focused on looking at the first and second instability modes that occur within the boundary layer to determine when the flow transitioned from laminar to turbulent. Using the pressure transducers, the authors determined that the 2nd mode waves are acoustic waves that have a much higher frequency then the firsts mode waves. Using the Rayleigh-Scattering visualization, PIV, and the pressure transducer it was concluded that as the first mode instabilities increase the flow will go turbulent, but also that as the 2nd mode instabilities go to zero when the flow goes turbulent which is shown in figure 4. Where section A is when secondary mode waves begin to form, section by B is when the second mode waves go to zero, and section C is when the flows goes completely turbulent for both figure 4 and 5.

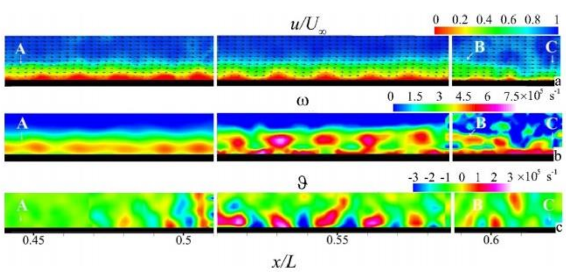


Figure 4 PIV visualization of velocity (top), Vorticity (middle), and Dilatation (bottom)

The 2nd mode waves cause vortical waves to occur causing the flow to go turbulent which can be seen by using the Rayleigh-Scattering visualization in figure 5 below.

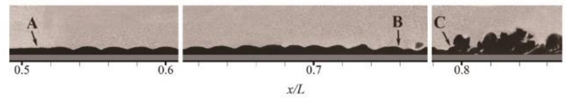


Figure 5 Rayleigh-Scattering Visualization M6QT

The author describes Rayleigh-Scattering visualization as, “a cooling and expansion process. The diameter of the cooled particle is much smaller than wavelength of the laser thus allowing for the visualization to be seen as the particle is heated in the near wall region.”. In conclusion, the authors of this article determined that as the first mode instabilities enter the boundary layer a second higher mode instability is also created and while the second mode instability initially increases it then begins to decrease, while the first mode instability continually increases, then when the 2nd mode goes to zero and the first mode grows large the transition to turbulence occurs.

<u>Recent progress in the Study of transition in the hypersonic boundary layer [2].</u>

This article like the one above also investigates the Peking hypersonic wind tunnel, but unlike the article above it investigates the difference of running the wind tunnel “noisy” vs running the wind tunnel “quiet” and documents the difference in the observed flight characteristics which can be seen in the figure below.

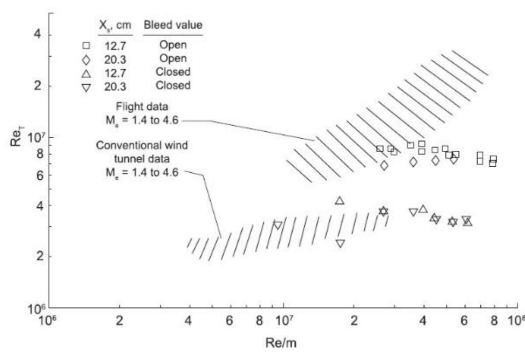


Figure 6 Difference between true flight transition Reynolds number and conventional wind tunnel transition Reynolds number

The authors stat that, “the fluctuations dominated by acoustic noise that is radiated from the turbulent boundary layer on the nozzle wall has a dramatic effect on the boundary layer transition.”. For this article, the Mach number was set at 6 and the Reynolds number was $2.5*10^6$. It was determined that quite wind tunnels due to their lack of disturbances better predicted the in-flight characteristics. They observed that the second and first mode instability waves where related through the nonlinear parabolized stability equations and that as the secondary waves went to zero that the transition from laminar to a turbulent boundary layer occurred. Through PSE evaluation it was also determined that the second mode waves were the cause that led to aerothermal heating which happen due to the compression and expansion of the flow which can be seen in figure 7.

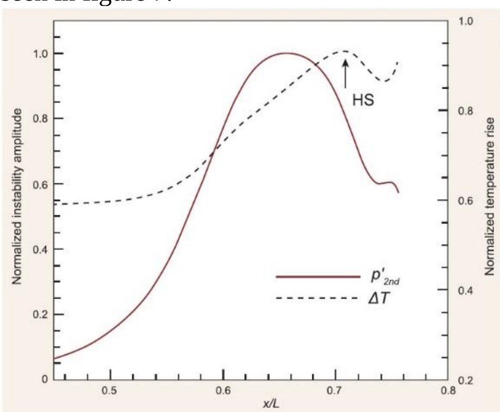


Figure 7 Second Mode instability wave compared to heat transfer rate.

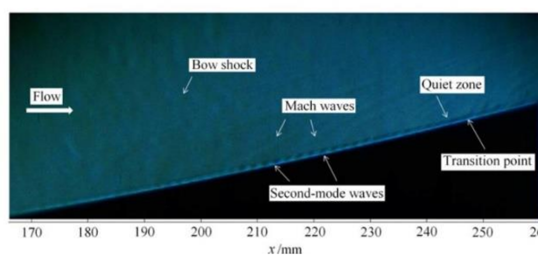
Similarly, the first mode wave was related to the increase in shear stress and is determined as the main cause of why a flow transitions from laminar to turbulent. In fact, the author believes that the second mode wave transfers energy to the first mode wave causing the wave to grow in amplitude and is the eventual cause for the flow to become turbulent. It was also noticed through experimentation that the second mode waves did not occur in slower supersonic flows, but only begin to become more prevalent when the Mach number is equal to or greater then four. In closing the author breaks down the transition to turbulent in three steps, "(i) receptivity, (ii) linear eigenmode growth or transient growth, and (iii) nonlinear breakdown to turbulence.".

Hypersonic Boundary Layer Transition on a flared cone [6]

This article also used the M6QT wind tunnel at Peking University in Beijing China which is shown in figure 8.

Figure 8 M6QT Hypersonic Wind tunnel

The Mach number was set to 6 and the Reynolds number was 5.8 X 10^6. The authors looked at the effect of placing a flared tip nosecone into the hypersonic quite wind tunnel. The purpose of the flared tip is because of the geometry it causes natural transition disturbances to occur in the boundary layer. The experiments consisted of using pressure transducers and pulsed schlieren imaging to determine results. The primary focus of this article was placed on the effects and causes of both the first and second mode waves and how they relate to boundary layer transition. They defined the life of the second mode wave in three stages, an initial stage, a growth stage, and a region they called the quiet zone. The quiet zone was described as a region where the second mode waves begin to weaken and eventually merge as shown in the figure below.



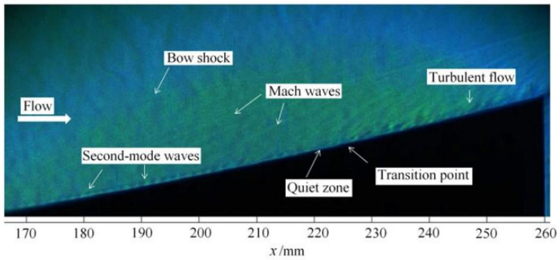


*Figure 9 comparison between (top) Reynolds number of 5.8 X10^6 and (bottom) 9.1*10^6*

From their research the authors also determined that "the wavelength of the second mode was twice that of the boundary layer thickness and that the laminar boundary layer thickness is inversely proportional to the square root of the Reynolds number", meaning that as the Reynolds number increases the wavelength of the second mode decreases. From their investigation of boundary layer transition, it was also noticed that as the Reynolds number was increased the boundary transition happened further upstream, meaning at a higher Mach number the transition will happen sooner this observation can also be seen from figure 9 above.

Flight Data for Boundary-Layer Transition at Hypersonic and Supersonic Speeds [3]

Like this literature review, this article was written as an overview to boundary layer transition and provided a review of over twenty articles and what was determined from many different types and kinds of experiments. The author states that the transition period is key to understanding flight parameters such as heat transfer, shock-shock interactions, and skin friction rates, specifically in hypersonic flights. The article gives example to why all methods of experimentation have their own disadvantages when trying to predict boundary layer transition. A summary of these issues is that true flight data has a very high cost, and it is very hard to control flight parameters such as angle of attack or maintained speed, wind tunnels provide noise that can cause unnatural disturbances in the flow such as vibrations or roughness from the experimental apparatus, while mathematical models do not provide any definite insight when the disturbances become nonlinear. The Mach number in these experiments ranged from 1.7 to 14.3, and consisted of flight test data, wind tunnel experimentation, and numerical analysis. From these articles similar observations could be noticed and that is what will be discussed about this article. The research determined in many cases that initial flow disturbances were caused by roughness, waviness, bluntness, curvature, or Mach number. As the initial disturbances occurred, they began to grow, the rate at which the grew was determined by Mach number, pressure gradient, and wall temperature. Modes of instabilities that were considered were Gortler, first and second TS, and three-dimensional cross flow. It was generally agreed upon that the turbulence first occurs when the instabilities of the second TS waves break down and that turbulence begins to grow of the wall as vortical waves become more prevalent. The author concludes that there is much more work to be done in the field of boundary layer transition and that as of now we only have a very generalized understanding of causes and location prediction.

Discussion and Conclusion

From the articles detailed above many similarities can be drawn between many different researchers' conclusions. The most general of these similarities being that understanding when, what, and why the boundary layer transition happens is key to sustained and accurately predicted flight parameters and is even more critical when talking about hypersonic speeds due to the fact of aerothermal heating. All papers stated that testing using conventional wind tunnel would produce unnatural disturbances that would affect the experimental results. Disturbances in the flow occur due to roughness, waviness, bluntness, or curvature of the vehicle or model. Disturbances rates were compared by considering the geometry, Reynolds number, and Mach number. A relationship between the boundary layer and Reynolds number was observed, as the Reynolds number was increased the transition in the boundary layer occurred further upstream on the body of the aircraft. Different methods to obtain boundary layer characteristics were used such as CFD, PIV, Rayleigh Scattering Visualization, Pressure Transducers, in flight data, and Schlieren imaging but all results returned similar results regarding disturbance waves (TS, Gortler, and Cross flow). It was observed that first mode TS waves would enter boundary layer through the interaction between the free stream and flight body, these waves would amplify until turbulence occurs. A secondary high frequency TS wave (acoustic) was also produced through freestream disturbances, at Mach numbers greater than four, that originally grew but then begins to decrease, as the second mode wave merged and went to zero turbulence also occurred.

The decrease in second mode waves led to vortical waves appearing. A relationship was also derived stating that as the Reynolds number increased the length of the second mode wave decreased. In fact, in one paper it was suggested that the secondary mode wave transferred energy to the first mode wave explaining why the first wave grew and the second wave dissipated. Secondary mode waves were related to an increase aerothermal heating which occurs due to the compression and expansion of the flow, while first mode waves were related to an increase shear stress between the freestream and the body of the aircraft or model. The effect, observation, and transition of these waves allowed for a visualization of the transition from a laminar to turbulent boundary layer.

The effects and causes of boundary layer transition is only just now being understood, further research and mathematical models need to be determined to allow for nonlinear relationships to be better understood, as well as ways to remove unnatural disturbances that due not truly occur in flight. Future work that should be considered using computational tools to confirm results that were determined through flight test or conventional wind tunnel experimentation.

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References

[1] Halila, G. L. O., Antunes, A. P., Silva, R. G. D., and Azevedo, J. L. F., "Effects of boundary layer transition on the aerodynamic analysis of high-lift systems," *Aerospace Science and Technology*, vol. 90, 2019, pp. 233–245.

[2] Lee, C., and Chen, S., "Recent progress in the study of transition in the hypersonic boundary layer," *National Science Review*, vol. 6, 2018, pp. 155–170.

[3] Schneider, S. P., "Flight Data for Boundary-Layer Transition at Hypersonic and Supersonic Speeds," *Journal of Spacecraft and Rockets*, vol. 36, 1999, pp. 8–20.

[4] Schneider, S. P., "Hypersonic laminar–turbulent transition on circular cones and scramjet forebodies," *Progress in Aerospace Sciences*, vol. 40, 2004, pp. 1–50.

[5] Schneider, S., "Hypersonic Boundary-Layer Transition on Blunt Bodies with Roughness," *46th AIAA Aerospace Sciences Meeting and Exhibit*, 2008.

[6] Zhang, C.-H., Tang, Q., and Lee, C.-B., "Hypersonic boundary-layer transition on a flared cone," *Acta Mechanica Sinica*, vol. 29, 2013, pp. 48–54.

[7] Zhang, C., Zhu, Y., Chen, X., Yuan, H., Wu, J., Chen, S., Lee, C., and Gad-el-Hak, M., "Transition in hypersonic boundary layers," *AIP Publishing* Available: https://aip.scitation.org/doi/10.1063/1.4935019.