

Experimental study of UTM-LST generic half model transport aircraft

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Abstract. This paper presents the experimental results from the investigation carried out at the UTM Low Speed wind tunnel facility (UTM-LST) on a half model generic transport aircraft at several configurations of primary control surfaces (flap, aileron and elevator). The objective is to measure the aerodynamic forces and moments due to the configuration changes. The study is carried out at two different speeds of 26.1 m/s and 43.1 m/s at corresponding Reynolds number of 1×10^6 and 2×10^6 , respectively. Angle of attack of the model is varied between -2° to 20° . For the flaps, the deflection applied is 0° , 5° and 10° . Meanwhile, for aileron and elevator, the deflection applied is between -10° and 10° . The results show the differences in aerodynamic characteristics of the aircraft at different control surfaces configurations. The results obtained indicate that a laminar separation bubble developed on the surface of the wing at lower angles of attack and show that the separation process is delayed when the Reynolds number is increased.

1. Introduction

Semi-span aircraft model consists of only half of the complete wing or tail of an aircraft. As an aircraft is a symmetric vehicle, investigation using a semi span half model is acceptable since the effect will be similar on the other side of the wing during straight and level longitudinal flight conditions. In general, most wind tunnel testing research requires that the correct flight conditions is simulated and the most accurate data is obtained. These issues are important in order to develop the accurate performance characteristics, especially at low-speed flight conditions such as takeoff, approach and landing that are usually encountered by subsonic transport aircraft. Typically, the Reynolds numbers achievable at the speed appropriate for takeoff and approach conditions in the available testing facilities are below the desired full-scale Reynolds number [1-3]. The need to extend Reynolds number testing capabilities up to full-scale conditions can be satisfied with the development of a semi-span testing capability.

A semi-span model testing offers few advantages over full span testing. By fabricating a semi-span model, a larger and more detailed design can be used. Due to the larger model size provided by semi-span testing, not only is the desired increased Reynolds number testing capability is produced, but the larger model size also improves data quality due to improved model strength, stiffness and overall fidelity. Constructing only half the model yields further benefits in terms of reduced model's cost. The complex high-lift systems and controls will only need to be produced for one wing. Another advantage of semi-span testing is that it only requires simple mounting on the balance and turntable. One of the disadvantages of semi-span testing is the wind tunnel's wall interference effects due to the model size. In this study, the model is fabricated and mounted on a splitter plate, which is offset from the tunnel



wall, such that it is outside the wall's boundary layer. The height of the model is also increased to avoid the boundary layer effect [1-3]. One of the main advantages of the half model is because it has a large wing area. Therefore, the flow visualization experiments can be carried out more effectively. A generic half model transport aircraft has been built in Universiti Teknologi Malaysia (UTM). The main objective of this project is to investigate the effect of Reynolds number on the performance of the aircraft when the control surface is changed.

2. Test facility and model description

The investigation is conducted in Universiti Teknologi Malaysia Low Speed Wind tunnel (UTM-LST) Aeronautical Laboratory facility. This is a closed-circuit, single-return, atmospheric wind tunnel that is capable of producing a maximum speed of 80 m/s. Cross sectional dimension of the test section is 2 m (breadth) x 1.5 m (height) x 5.8 m (length). A generic half-span model of transport aircraft has been chosen for this experiment. The model is designed so that its cross-sectional profile is NACA 2213. This semi-span model is consisted of a port wing from the full span model and a semi-fuselage, which is fabricated from a mold of the full-span fuselage [5,6]. The wing employs control surfaces of flap and aileron system, and tail with horizontal stabilizer and moveable elevator. The model is fabricated without engine nor vertical stabilizer. The model's detail dimensions is shown in Table 1 and Figure 1.

Table 1. Specification of the model

Parameter	Dimension
Fuselage Length, L	2.362 m
Wing Area, S	0.252 m ²
Mean Aerodynamic Chord	0.339 m
Half-span, b/2	0.983 m
Wing Volume, V _{wing}	0.00072 m ³
Fuselage Volume, V _{fuselage}	0.58 m ³

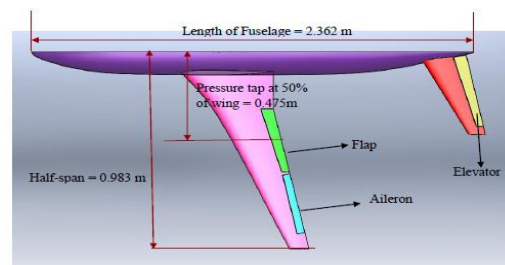


Figure 1. Dimension and location of the control surfaces of UTM-LST Half Model

There are two main external balances used in Aerolab to measure the aerodynamic characteristics of the half-span model. The first one is called *semi span balance*, which has a high capacity of up to 2000 N. Several previous projects [5, 6] applied this balance to measure aerodynamic characteristics but Aerolab found the balance to be unsuitable for relatively light aircraft model. Therefore, a new balance has been purchased called *JR3*. The capacity of this balance is about 1000 N in each channel. This is the first project to measure aerodynamic characteristics of UTM half-span model using JR3. The model is installed on JR3 balance to measure the forces and moments due control surfaces at different deflection angles. The full model was mounted on a 1.6-meter diameter turntable on the floor of the tunnel, approximately 1.9 meter aft of the tunnel floor's boundary layer (upstream) removal system (BLRS). The modified installation is depicted in Figure 2. Additionally, in order to further understand the flow physics involved in semi-span testing, surface pressure measurements and simple flow visualization using tuft are conducted. The model has 18 pressure ports (9 on each side to and bottom) located 50% of the wingspan for the pressure measurement.

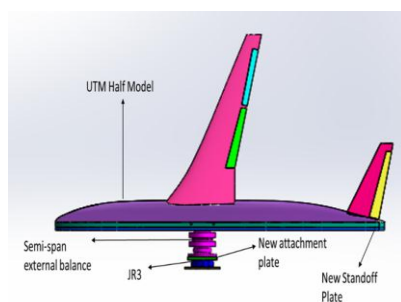


Figure 2. Modified installation with standoff

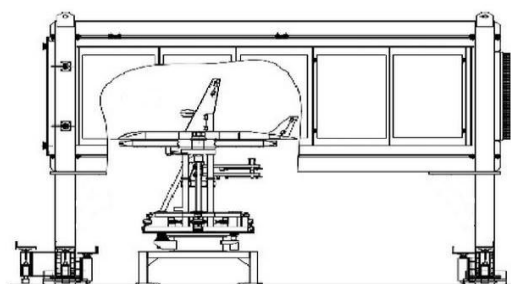


Figure 3. Original installation of UTM-LST half-span model with semi-span balance in UTM-LST wind tunnel

A standard wall correction as discussed in [7] is applied in this project. The wall correction consists of solid blockage and wake blockage. Solid blockage occurs due to the presence of the model inside the section, which creates velocity increase across the model. Calculation of the solid blockage is done using the simple equation for solid-blockage correction for two dimensional case as in Equation 1 [7].

$$\varepsilon_{sb} = \frac{K_1(\text{model volume})}{c^2} \quad (1)$$

where K_1 is a spanning configuration factor which is value of 0.52 for a model spanning across the tunnel height or in vertical direction. The model volume is 0.00073 m^3 and c is the tunnel parameters. Meanwhile, the wake blockage correction is also done in two dimensional case, using the simplified equations as in Equation 2, Equation 3 and Equation 4.

$$\varepsilon_{wb} = \frac{c/h}{2} C_{du} \quad (2)$$

$$\varepsilon_{wb} = \frac{0.5/2}{2} C_{du} \quad (3)$$

$$\varepsilon_{wb} = 0.125 C_{du} \quad (4)$$

where C_{du} is the uncorrected drag coefficient from the balance reading while h is the wind tunnel height, which is 1.5 meter. The total blockage factor for this model is about 12% at the angle of attack of $\alpha = 23^\circ$.

3. Wind tunnel testing

The installation of the UTM-LST half-span model in the test section is shown in Figure 4. The wind tunnel experiment is conducted in the UTM Low Speed Tunnel at Universiti Teknologi Malaysia's Aeronautics Laboratory with cross sectional 2.0m width x 1.5m height. The model was tested at 26.1 m/s and 43.1 m/s that corresponds to 0.5×10^6 and 1.0×10^6 Reynolds number based on the mean aerodynamic chord, respectively. The model has an overall length of 2.362 m and the wing area of 0.252 m^2 . During the experiment, angle of attacks were varied from -2° to 20° . The experiments are conducted in two phases: first, the model with no control surfaces deflection is tested and then the model with several control surfaces deflection is tested. Three measurements techniques are employed on the model, surface pressure measurement and steady balance data. In the final stage, threaded tuft flow visualization method is employed on the model at angle of attack varies from -2° to 20° .

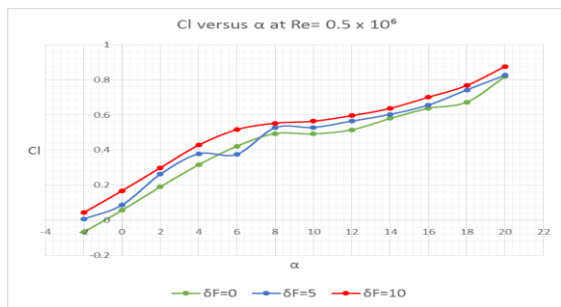


Figure 4. The installation of half-span model in UTM-LST

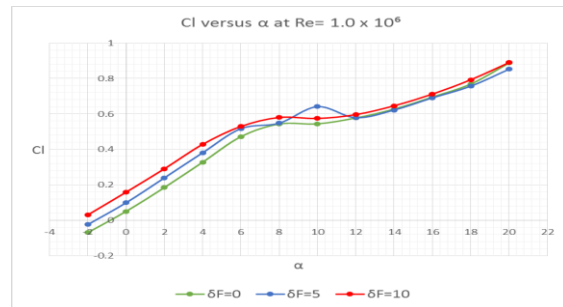
4. Results

4.1. Steady balance data

Figure 5, Figure 6 and Figure 7 show the results obtained from JR3 balance experiments. Blockage correction factor of 13% has been taken into consideration in the calculation. Lift coefficient in Figure 5 shows that the change of the Reynolds number has no impact on the results of the flap, aileron and stabilizer. The lift coefficient often increases once the angle of flap, aileron and stabilizer is increased. The results obtained show that stalling angle for this aircraft is at about $\alpha = 8^\circ$. However, increasing the stabilizer angle can give an impact on lift (see Figure 5(iii)). The characteristics of drag and pitching moment are shown in Figure 6 and Figure 7, respectively. Similar trend is observed for the drag and pitching moment coefficients. In all cases, δ_F is the angle of flap deflection, δ_{Ae} is the aileron angle while δ_{St} is the stabilizer angle.

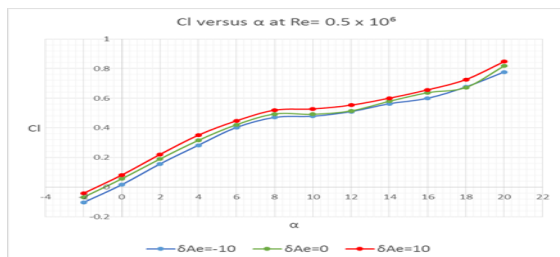


a) $R_{Mac} = 0.5 \times 10^6$

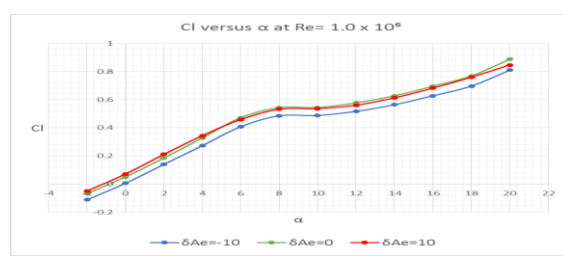


b) $R_{Mac} = 1.0 \times 10^6$

(i) Effect of flap deflections on C_L



a) $R_{Mac} = 0.5 \times 10^6$

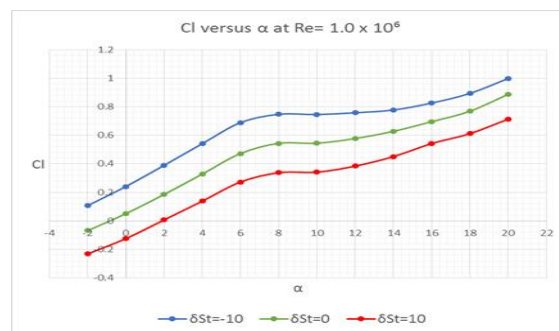


(b) $R_{Mac} = 1.0 \times 10^6$

(ii) Effect of aileron deflections on C_L



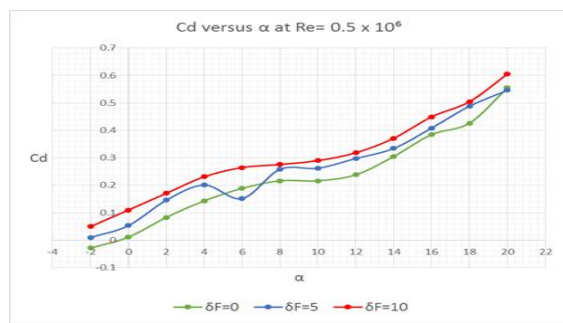
(a) $R_{Mac} = 0.5 \times 10^6$



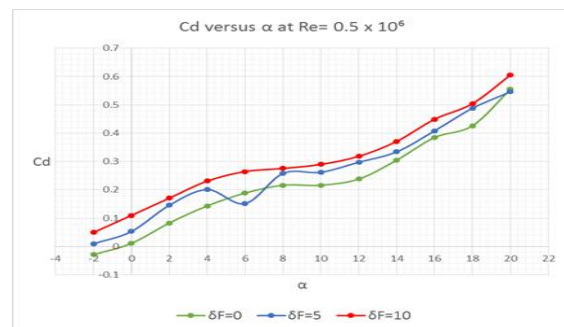
(b) $R_{Mac} = 1.0 \times 10^6$

(iii) Effect of stabilizer deflections on C_L

Figure 5. The effects of control surfaces on C_L

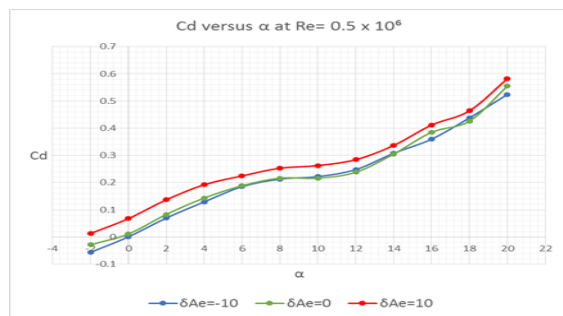


(a) $R_{Mac} = 0.5 \times 10^6$

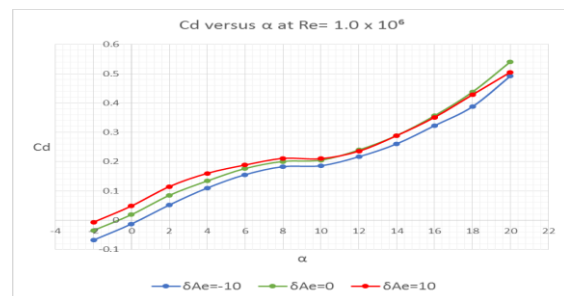


(b) $R_{Mac} = 1.0 \times 10^6$

(i) Effect of flap deflections on C_D

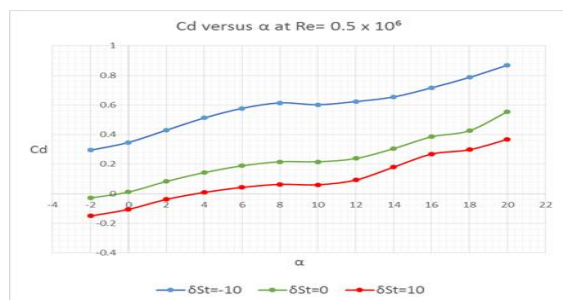


(a) $R_{Mac} = 0.5 \times 10^6$

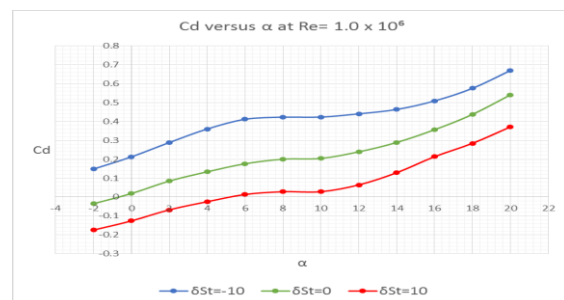


(b) $R_{Mac} = 1.0 \times 10^6$

(ii) Effect of flap Aileron on C_D



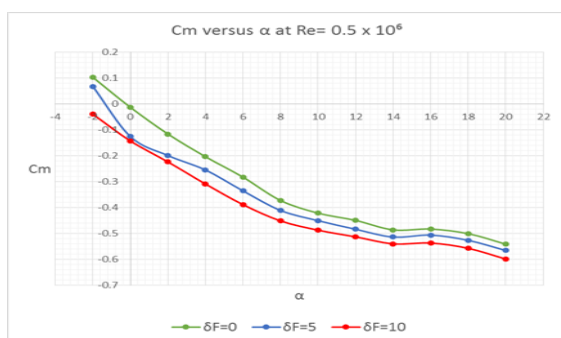
(a) $R_{Mac} = 0.5 \times 10^6$



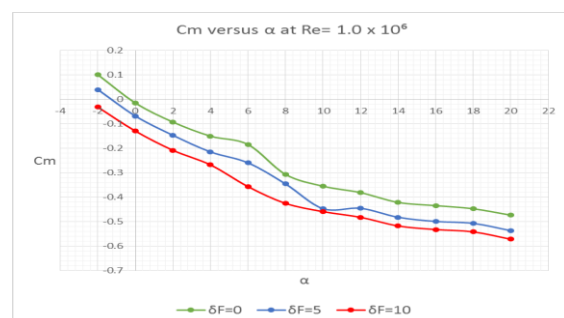
(b) $R_{Mac} = 1.0 \times 10^6$

(iii) Effect of flap Stabilizer on C_D

Figure 6. The effects of control surfaces on C_D



(a) $R_{Mac} = 0.5 \times 10^6$



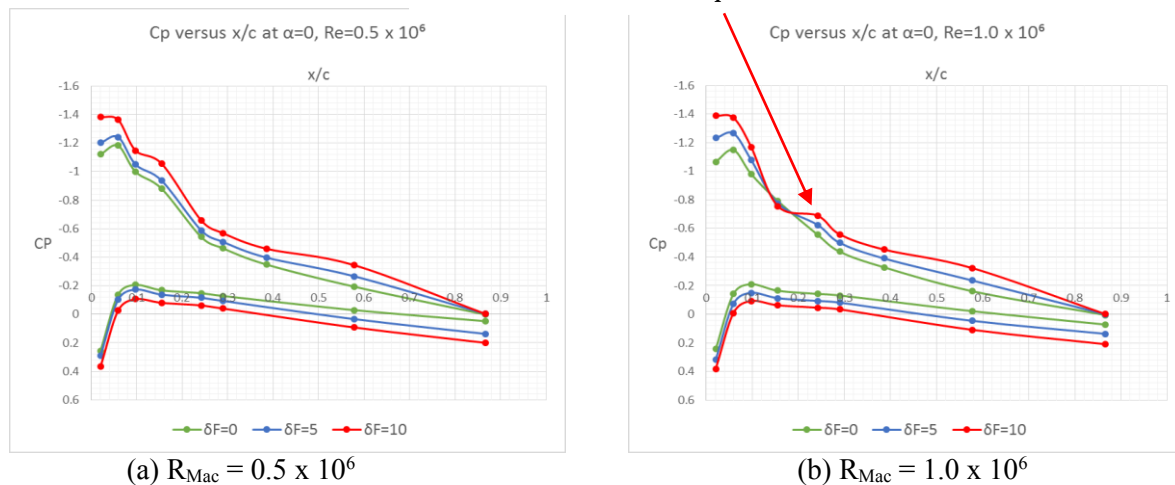
(b) $R_{Mac} = 1.0 \times 10^6$

Figure 7. C_M at Different Flap deflection

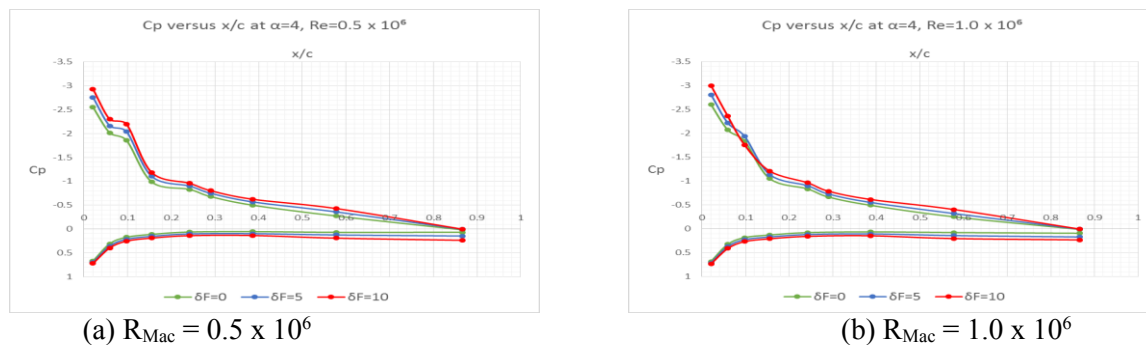
4.2. Surface pressure measurements data

Sample data of surface pressure measurement at an angle of attack of 0° , 4° and 8° are shown in Figure 8. The data obtained shows that only the flap is affecting the flow separation compared to aileron and stabilizer. Therefore, only data involving flap is discussed here. Figure 8(i) shows the distribution of pressure at $\alpha = 0^\circ$. The diagram shows that the flow attaches to the wing on the entire wing. For both Reynolds number, it can be observed that the lift increases if the flap angle is increased. At Reynolds number 1×10^6 , it can be noticed that the laminar separation bubble might develop at 15% to 25% of the wing if the flap is set at 10° . Laminar separation bubble can be predicted to occur on the wing if gradient of C_p against x/c is zero. From Figure 8(i) at the Reynolds number of 1×10^6 , it may develop from $x/c = 15 - 25\%$ of the wing.

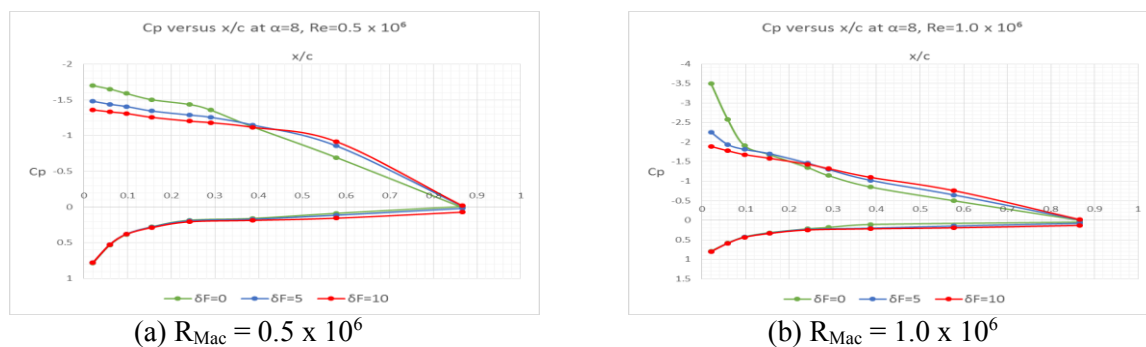
Laminar bubble separation



(i) Pressure coefficient at $\alpha = 0^\circ$



(ii) Pressure coefficient at $\alpha = 4^\circ$



(iii) Pressure coefficient at $\alpha = 8^\circ$

Figure 8. Surface pressure measurement data at $\alpha = 0^\circ$, 8° and 10°

When the angle of attack is increased to $\alpha = 4^\circ$, not much of a difference can be observed compared to the result obtained for $\alpha = 0^\circ$. The airflow is still attached to the wing to the wing surface. Laminar separation bubble might develop at position 8% of the wing at Reynolds number of 0.5×10^6 . There is a significant difference when the angle of attack is increased to $\alpha = 8^\circ$. Figure 8(iii) indicates that the flow separation has developed on the wings. At Reynolds number of 0.5×10^6 , it can be observed that the increase in flap angle has promoted the flow separation. At higher Reynolds number of 1×10^6 (see Figure 8(iii)), it can be observed that the flow separation has been delayed. The attached flow still exist in front of the wing and this indicates that the increase of Reynolds number has delayed the process of separation. When Reynolds number is increased, this means that the viscosity of the airflow has also increased. The airflow becomes more turbulent in this situation. Stronger ability of turbulent boundary layer to sustain the adverse pressure gradient has delayed the separation process [8].

4.3. Thread flow visualization results

This section discussed the thread flow visualization that has been carried out on the model. During the experiment, a camera was placed on the roof of the UTM test section. Images of the flow separation were recorded for every five seconds. This process is done for all specified angles of attack. Sample images of surface flow visualization using thread performed on the clean wing configuration is shown in Figure 10. This experiment was performed at angles of attack between -2° to 20° and the Reynolds number of 0.5×10^6 . In Figure 9(a), it can be seen that the thread is fully attached to the surface of the wing. It shows in this position that the airflow is attached to the wing surface and separation is not observed. When the angle of attack is increased to $\alpha = 5^\circ$, the airflow in the leading edge is attached to the wing surface to a certain chordwise position. In the trailing edge region, it can be observed that the flow separation begins to develop. Increasing the angle of attack further causes the separation point to move forward as shown in Figure 9(c). Finally, at $\alpha = 5^\circ$, the entire wing is covered by the flow separation (see Figure 9(d)).

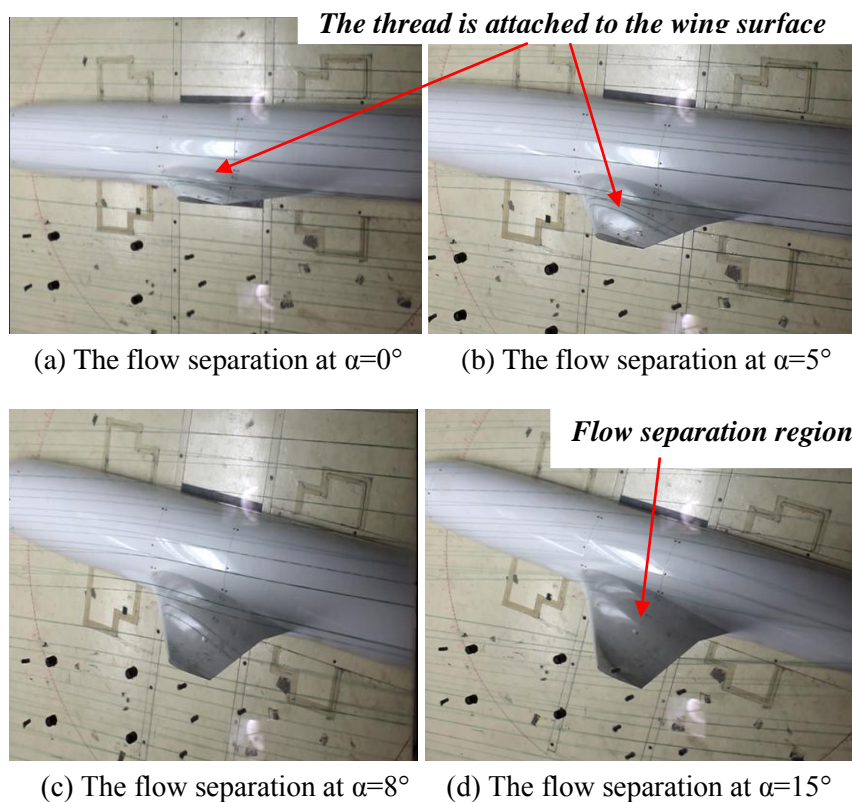


Figure 9. Thread tuft flow study at Reynolds number of 0.5×10^6

5. Conclusion

This experiment is conducted with the aim of providing a detailed explanation on the aerodynamic characteristics of half-span model of transport aircraft at certain control surface deflections. In this experiment, three measurement techniques have been carried out on the aircraft model, i.e. balance measurement, surface pressure measurement and flow measurement visualization. It is deduced that when flaps are deflected at larger angle, the flow is still attached and the lift increases with increasing angle of attack. This paper discusses in detail the effect of control surfaces on lift, drag and moment. The results obtained also show that the airflow is sensitive to Reynolds number. It is found that the separation process is delayed if the Reynolds number is increased.

Acknowledgements

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References

- [1] Yokokawa Y, Murayama M, Uchida H and Tanaka K 2010 *AIAA Aerospace Science Meeting*
- [2] Gatlin G M and McGheet R J 1997 *Journal of Aircraft* **34** 500-5
- [3] Milholen W E, Chokani N and McGhee R J 1996 *Journal of Aircraft* **33** 115-22
- [4] Mansor S 2008 *Low Speed Wind Tunnel* Universiti Teknologi Malaysia
- [5] Ishak I S, Mat S, Mat Lazim T and Awang M Z 2003 *International Conference on Mechanical Engineering*
- [6] Sunthrasakaran R 2015 *Journal of Aeronautical Science and Engineering* **1**
- [7] Pope A and Rae W 1984 *Low-speed wind tunnel testing* (New York: Wiley)
- [8] Mat S 2010 *The Analysis of Flow on Round-Edged Delta Wings* PhD Thesis University of Glasgow