The Aerodynamics Analysis on Cambered Fuselage Model

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Abstract. There various factors gives influence in determining the fuselage shapes, such as the payload, cockpit, wing and tail placements or in manner up and down loading the payload for a cargo aircraft. These factors may come up the fuselage is no longer as symmetrical fuselage but represent as a cambered fuselage. As results the lift coefficient as well as its pitching moment coefficient is no longer equal to zero as the angle of attack goes to zero. Basically the manner how to determine the fuselage aerodynamics characteristics for cambered fuselage can be done in similar way as in the case of symmetrical fuselage by simply replacing the angle of attack \( \alpha \) term with \( (\alpha - \alpha_{L=0}) \), where \( \alpha_{L=0} \) represent the angle of attack at zero lift. The present work use a similar manner in determining the zero lift angle of attack as it had been used in DATCOM software. To investigate the effect of camber on the aerodynamics characteristic fuselage, the present work use a fuselage model with a circular cross section where the location of center of the circle placed along the fuselage’s camber line. The fuselage’s camber line defined according to the definition of camber line of NACA airfoils. Aerodynamics analysis on over various fuselage models indicate that the maximum camber line thickness and their position give a significant influent to the fuselage aerodynamics characteristics.

1. Introduction

The aerodynamics characteristics for unsymmetrical or cambered fuselage may be estimated by using the same aerodynamics equation applied for symmetrical fuselage, just simply by replacing the angle of attack \( \alpha \) which appeared in that equation with \( (\alpha - \alpha_{L=0}) \). Here \( \alpha_{L=0} \) represents the zero lift angle of attack of the fuselage body. In addition to this, one needs to convert the fuselage body of unsymmetrical fuselage to become its equivalence symmetrical body. Basically the difficulty in solving aerodynamics problem of unsymmetrical fuselage, is in determining the zero lift angle of attack of the corresponding fuselage body \( \alpha_{L=0} \). Figure 1 shows an aircraft model had been used The NASA’s Technical Note, NASA TN D-6800[1]. This report gives aerodynamics analysis a complete aircraft by using a semi empirical aerodynamics method. To estimate the aerodynamics characteristics, this report defined the zero lift angle of attack of the fuselage \( \alpha_{L=0} \) just set to equal -3°, with out any clarification how to obtain that value.

Fig. 1: The side view of NASA aircraft model
The present work uses the approach for determining the zero lift angle of attack by adopting the algorithm that had been used by DATCOM software [2] applied to the cambered fuselage model created by use of the coordinate as one defines coordinate airfoil NACA four, five, or six digit series.

2. Camberline Equation. To investigate the effect of aerodynamic characteristics on the fuselage due to camberline, here one may use the definition of camber line applied to the airfoil NACA series. The camber line for NACA four digits series is given by [3]:

\[
\frac{y_c}{c} = \begin{dcases}
\frac{m}{p^2} \left[2p \left(\frac{c}{c}\right) - \left(\frac{c}{c}\right)^{2}\right]; & 0 \leq \left(\frac{c}{c}\right) < p \\
\frac{m}{(1-p)^2} \left[1 - 2p + 2p \left(\frac{c}{c}\right) - \left(\frac{c}{c}\right)^{2}\right]; & p \leq \left(\frac{c}{c}\right) < 1
\end{dcases}
\]  

(1)

Where:
- \( m \): the maximum camber line in 1/100 of chord
- \( p \): the position of the maximum camber line in 1/10 chord

The NACA six digits series can be classified into three groups. The first group of NACA six digits series are having a camber line distribution defined as:

\[
\frac{y_c}{c} = \frac{C_{L1}}{4\pi} \left[\ln \left(1 - \left(\frac{x}{c}\right)\right)\ln \left(1 - \left(\frac{x}{c}\right)\right) + \left(\frac{x}{c}\right)\ln \left(\frac{x}{c}\right)^3\right] 
\]  

(2)

The \( C_{L1} \) in above equation represent the design lift coefficient which one can choose any value. The designed airfoil use above relation will produce an airfoil with a uniform loading along the entire chord length. The second group of NACA six digits series is the airfoil designed to have a uniform loading occurred from the leading edge \( x = 0 \) up to any point along chord \( a \), where \( a \) is less than 1 and greater than 0. The camber line for this type of airfoil NACA six digits series is given as:

\[
\frac{y_c}{c} = \frac{C_{L1}}{2\pi(1+a)} \left[B - \left(\frac{x}{c}\right)\ln \left(\frac{x}{c}\right) + g - h \left(\frac{x}{c}\right)\right] 
\]  

(3)

Where:
- \( B = \frac{1}{1+a} \left[\frac{1}{2} \left( a - \left(\frac{x}{c}\right) \right)^2 \ln \left( a - \left(\frac{x}{c}\right) \right) + \frac{1}{2} \left( 1 - \left(\frac{x}{c}\right) \right)^2 \ln \left( 1 - \left(\frac{x}{c}\right) \right) \right] \right)

(4)

- \( g = \frac{-1}{(1-a)^2} \left[ a^2 \left( \frac{1}{2} \ln a - \frac{1}{4} \right) - \frac{1}{4} \right] \)

(5)

- \( h = (1-a) \left[ \frac{1}{2} \ln (1-a) - \frac{1}{4} \right] + g \)

(6)

Another group of NACA six digits represents special case of NACA six digits series with setting in such away, so that the airfoil will have a uniform loading from the leading edge \( x = 0 \) to \( x = 0.8 \ c \). Small modification had been applied to this airfoil by setting that the uniform loading from the leading edge up to \( x = 0.87437 \) and the design lift coefficient design as \( C_{Lmod} = C_{Ldes} / 1.0209 \).

The camber line distribution along x-axis for such kind airfoil will be:
\[
\frac{y_c}{c} = \frac{C_Li}{2\pi(1+a)} \left[ B - \left( \frac{x}{c} \right) \ln \left( \frac{x}{c} \right) + g - h \left( \frac{x}{c} \right) \right] \text{ for } 0 \leq \frac{x}{c} \leq 0.87437 \quad (7)
\]

And
\[
\frac{y_c}{c} = 0.0302164 - 0.245209 \left[ \left( \frac{x}{c} \right) - 0.87437 \right] \text{ for } 0.87437 < \frac{x}{c} \leq 1.0 \quad (8)
\]

3. Discussion and Result. The aerodynamics analysis on cambered fuselage model here, use a fuselage having a circular cross section with the distribution of radius cross section is equal to the distribution of the airfoil thickness of the NACA series. The fuselage camber line defined according to the camber line of the NACA four digits series for fixed position of maximum camber at \( x = 0.1 \) \( c \) and varying the maximum camber from \( Y_{c_{\text{max}}} = 0.02 \) \( c \) to \( Y_{c_{\text{max}}} = 0.1 \) \( c \) as shown in the Figure 2. While Figure 3 show the same camber line but for the position maximum camber at \( y_{c_{\text{max}}} = 0.5 \) \( c \). The shape of camber line for the same maximum camber as previous but differ in term of their position of the maximum camber as shown in the Figure 4. While Figure 5 and 6 are shown the camber line shape for the NACA five digits series in standard form and in its reflexed camber line respectively. The other examples of camber line which defined according to the camber line definition for NACA six digits series are shown in the Figure 7.

Fig. 2: The shape of camber line for NACA four digits series for different values of maximum camber setting at a fixed position maximum camber \( x_{c_{\text{max}}} = 0.1 \) \( c \)

Fig. 3: The shape of camber line for NACA four digits series for different values of maximum camber setting at a fixed position maximum camber \( y_{c_{\text{max}}} = 0.5 \) \( c \)

Fig. 4: Shape camber line’s NACA four digits series with different position maximum thickness

Fig. 5: Distribution camber NACA five digits series for model 210., 220., 230., 240., 250..
The aerodynamics analysis to the fuselage with geometry as mentioned above carried out at Mach number M = 0.2 and Reynolds $R_L = 3. 10^6$. For the case fuselage geometry developed based on NACA four digits series, at fixed maximum value of camber line equal to 0.01 $c$ and varying the position of the maximum camber in term of lift coefficients as function of angle of attack is shown in the Figure 8a. While Figure 8b and Figure 8c are shown in term of drag and pitching moment coefficients respectively. Figure 9a, b and c for the same case, with the maximum camber is set equal to 0.02 $c$, while for the maximum camber 0.05$c$, their results as presented in the Figure 10.
The influence of position of maximum camber and position of it term of angle of attack at zero lift $\alpha_{\text{CL}=0}$ to the fuselage model created based on NACA four digits series as depicted in the Figure 11a, while associated drag and its pitching moment are shown in the Figure 11b and Figure 11c respectively.

Fig. 11a: The zero lift angle of attack variation on fuselage model based profile NACA four digits series

Fig. 11b: The drag coefficient at zero lift angle of attack variation on fuselage model based profile NACA four digits series
4. Conclusion. Investigation on fuselage model developed by use of NACA series geometry data indicates that the zero lift angle of attack decreasing as the position and the value of maximum camber lines are increasing. Similar manner occurred to their drag coefficient and the pitching moment coefficients.

References