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The flow separation development analysis in subsonic and transonic flow regime of the laminar airfoil

Robert Placek[a]*, Paweł Ruchała[a]

[a]*Institute of Aviation, Al.Krakowska 110/114, Warsaw 02-256, Poland*

Abstract

Wind tunnel tests of a laminar airfoil have been performed at the Institute of Aviation in Warsaw. The main goal of the investigation was to study the separation process development in subsonic and early transonic flow regime. The airfoil chord was 0.2m. During wind tunnel test the natural laminar-turbulent transition was applied. The Mach numbers were 0.3 and 0.7. Reynolds number were approximately equal to $1.22 \cdot 10^6$ and $2.85 \cdot 10^6$ respectively. The angle of incidence was increased up until the flow was fully separated. During the experimental research, chosen test methods such as pressure measurements and schlieren visualization were applied. Wind tunnel results were analyzed in terms of aerodynamic coefficients and flow separation type identification. The wind tunnel investigation revealed that separation phenomena at subsonic and transonic flow regime affected in a different manner on the airfoil aerodynamic performance. This was mainly because of the change of the flow pattern influencing on the separation process.

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Keywords: laminar airfoil, separation, experiment, subsonic, transonic

1. Introduction

The flow separation phenomenon is related to each aviation airfoil and is associated with the break-off of the thin layer (called boundary layer), right at the wing surface. The way in which the flow separation develops to the moment of the full separation occurrence, is strictly dependent on various factors: an airfoil thickness (thin, moderate, thick), an airfoil type (turbulent, laminar, supercritical), an airfoil surface quality (smooth or with roughness), the angle of attack, flow conditions (altitude and air turbulence), and Reynolds number. The exemplary

* Corresponding author. Tel.: +48-22-8460011 int. 360
E-mail address: robert.placek@ilot.edu.pl

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flow separation development with the angle of incidence rise at subsonic speed was presented by Talay (1975). The flow separation development appears on the upper airfoil surface and propagates upstream, into the leading edge direction. When it occurs, the increment of lift coefficient with angle of attack began to decrease until the maximum lift coefficient value is reached. Afterwards, when angle of attack is still increased, the lift decrease appears. This phenomenon is strictly related to the pitching moment change and a drag increase. For subsonic speeds, there are indicated characteristic types of the wing-section stall, according to Gault (1957), Whitford (1987) and Raymer (1992): the trailing-edge, the leading-edge, and an thin-airfoil (or combined: trailing-edge and the leading-edge).

For the transonic flow range, above critical Mach number value over an airfoil surface appears shock wave terminating the supersonic region. The shock wave interaction with boundary layer (laminar, transitional or turbulent) modify the way of the flow separation (Haynes et al., 1957), causing the boundary layer thickening or shock-induced separation of the boundary layer. The structure of the flow separation at transonic speeds depends additionally on: the free stream Mach number value, the shape and strength of the shock and pressure fluctuations in the flow (resulting the frequency and amplitude of shock oscillations). The proposition of the shock wave separation process classification (bubble separation) was presented by Pearcey (1968). Pearcey proposed classification of the separation types based on a stationary shock interacting with a turbulent boundary layer. Two models were specified: A) only rearward growth of laminar bubble and B) applicable when the rear separation is incipient or present. The B) model was detailed and divided in three groups: 1) a rear separation provoked by bubble, 2) a rear separation provoked by shock and 3) with a rear separation already present. Mundell and Mabey (1986) proposed classification of the shock wave boundary layer interaction and excitations on airfoils in unsteady transonic flow with shock oscillations. The following classification of the flow separation with increasing incidence was proposed: a) type 1: for a low angle weak shock thickens boundary layer, b) type 2: a stronger shock locally separates boundary layer for higher values of angles, c) type 3: for high values of the angle of incidence a very strong shock separates the boundary layer to the trailing edge.

The presented paper contains experimental results of the flow separation over laminar airfoil for chosen subsonic and transonic Mach numbers and various angles of incidence. The schlieren method was used for a visualization and pressure measurements were used for the lift and drag coefficients estimation (averaged values). The flow separation development influenced on a pressure coefficient distribution and airfoil aerodynamic characteristic values. Results showed different conditions of the flow separation for tested subsonic and transonic flow values.

Nomenclature

BL	boundary layer
c	airfoil chord
CD	drag coefficient
CL	lift coefficient
C_p	pressure coefficient
C_p*	critical pressure coefficient
LE	leading edge
M	Mach number
Re	Reynolds number
SW	shock wave
TE	trailing edge
αi	angle of incidence

2. Approach

The wind tunnel research was carried out in the trisonic N-3 wind tunnel at the Institute of Aviation.

The N-3 wind tunnel, described by Wiśniowski (2016), is a closed circuit blow-down type with a partial flow recirculation. Dimensions of the test section are: the cross-section 0.6x0.6m, the length 1.5m. The tested 2D airfoil model was of the laminar type with maximum thickness 15% c and the chord length 0.2 m. The V2C airfoil shape