International Conference on Military Technologies 2019 (ICMT'19)
May 30 – 31, 2019, Brno, Czech Republic

Analysis of Basic Methods for Aerodynamic Characteristics Determination of Missile Structure's Components

P. Konečný*, H.B. Le*, H.M. Nguyen+, Q. T. Tran+, Q. H. Mai+ and T.S. Ngo*
* Department of Weapons and Ammunition, University of Defence, Czech Republic, email: *pavel.konecny@unob.cz*
+ Department of Weapons, Le Quy Don Technical University, Ha Noi, Viet Nam, email: *nguyenhaiminhtb2013@gmail.com*

***Abstract*—The process of the components aerodynamic characteristics determination of the missile airframe is one of the decisive processes in the missile design. So far, there are several methods of aerodynamic characteristics determination, each of them has its own properties and is often used in different stages of the missile airframe design. The article explores and analyzes the properties of some commonly used methods. For clarifying their applicability, a few available comparisons of the aerodynamic characteristic values that were calculated by using these methods are also presented in the article. In addition, the article applies a method to determine the aerodynamic characteristics of a particular airfoil and compare it with available data.**

Keywords-missile design; aerodynamic characteristics; missile airframe.

I. Introduction

In the stages of missiles designs, the identification of the missile airframe components aerodynamic characteristics is always necessary. From preliminary design to detailed design and experiment, designers must calculate and determine the optimal missile airframe configuration. At different stages of the design process, the different methods are often used to determine aerodynamic characteristics with different accuracy. Each method has different mathematical essentials and uses different assumptions. Analysis of the essentials and capabilities of the methods, the comparison of calculation results using the selected methods can help designers to have a deeper assessment of these methods.

II. The missile airframe types

Missiles are made up of various components, in which the missile airframe is loaded by aerodynamic forces. The airframe must ensure stable missile movement and precise steering.

The major components that form the missile airframe include: nose, body, fixed and movable aerodynamic surfaces. Aerodynamic fixed surface has functions of maintaining missile stability, typically fins or stable wings. Aerodynamic movable surfaces have functions of controlling missile to follow the target, typically tail, wing or canard control. Commonly, the airframe is symmetrically cruciform, with four fixed fins and four movable control surfaces, as shown in Figure 1.

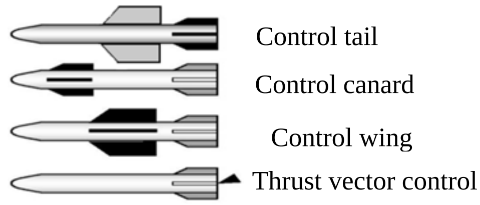

Figure 1. Four main categories of missile controls [1]

All aerodynamic surfaces belong to the wing family and have some of the same basic aerodynamic characteristics. Depending on their location and function on missiles they have different names. Depending on the design and function they may play a stabilizing or control role. An overview of the functions and characteristics of the missile's aerodynamic components is introduced below.

A. Nose

The drag on the missile depends on many factors. One of the most important of these factors is the shape of the nose. The nose is shaped to offer the minimum drag and the ideal aerodynamic nose shape is primarily related to the velocity of the missiles. An aggregation chart of response levels in flight modes of different nose shapes is shown in Figure 2 [2].

Ogive
Cone
LV-HAACK
Von Karman
Parabola
¾ Parabola
½ Parabola
$x^{3/4}$ Power
$X^{1/2}$ Power

0.8 1.0 1.2 1.4 1.6 1.8 2.0 M

Figure 2. Comparison of drag characteristics of various nose shapes in the transonic regions [2]
Rankings are: superior (1), good (2), fair (3), inferior (4)

B. Body

The missile role is to transport important missile components such as combat parts. Apart from that function, the missile profile also plays an important role in the missile airframe. The missile usually consists of cylindrical sections, between the sections with transition lines to remove the vortex resistance.

C. Aerodynamic surfaces

Fins belong to fixed aerodynamic surfaces. The main purpose of using fins on a missile is to reach stability during flight, that is, to allow the missile to maintain its orientation and intended flight path. Fitting fins on a missile serve to shift the center of pressure behind the center of gravity. If the fins have the suitable airfoil shapes, they will reduce the drag, increase the lift at small attack angles, thereby increasing the missile stability.

The small control surfaces mounted on the missile nose are called canards.

Control wings and fixed wings belong also to aerodynamic control surfaces. The main role of the missile wings is to create lift and control the missile towards the target.

Aerodynamic surfaces often have planform shapes and airfoils similar to each other. Some basic planform shapes of the aerodynamic surfaces are shown in Figure 3.

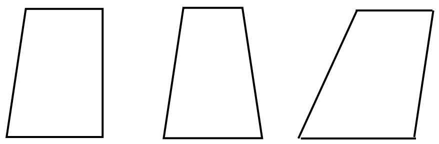

Figure 3. Some basic planform shapes of the aerodynamic surfaces

Some forms of airfoil are used for aerodynamic surfaces which can be seen in Figure 4.

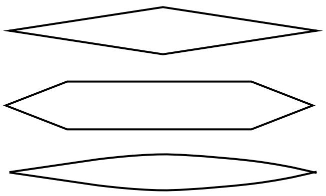

Figure 4. Supersonic airfoils

III. THE AERODYNAMIC CHARACTERISTICS OF THE MISSILE STRUCTURE COMPONENTS

A. Nose, body

The most important aerodynamic characteristic of nose and body is the drag coefficient.

At subsonic velocity, the drag includes skin friction and form drag, together called profile drag. The drag depends mainly on the shape of the nose and body of the missile.

When rockets move at supersonic velocity, the drag is mainly wave drag, where the wave drag arising due to formation of shock-waves is the deciding factor. The missile with a sharper nose tip, the shock will be oblique and will generate smaller drag. The Fig. 5 shows a model of the missile nose at supersonic velocity.

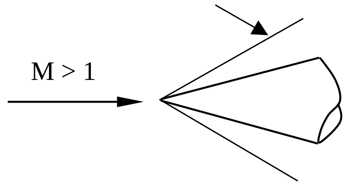

Figure 5. Supersonic nose

B. Aerodynamic surfaces (wings, fins)

The cross-section of the aerodynamic surfaces of missiles often uses the form of supersonic airfoils. The Figure 6 shows the general model of supersonic airfoil and shock wave phenomena. These phenomena generate lift and drag primarily on missiles.

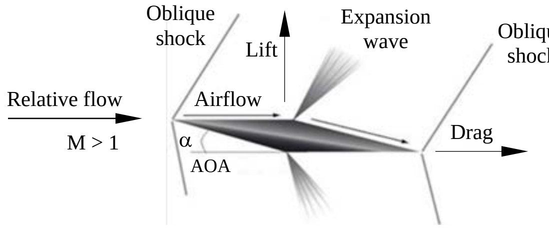

Figure 6. Airflow around a supersonic fin

Wings or fins have a lot of aerodynamic characteristics, but the most important aerodynamic characteristics are coefficient of drag C_D and coefficient of lift C_L.

These coefficients can be determined by the formula:

$$C_D = D/qS;\ C_L = L/qS,$$

where:

D - drag on wing (fin); L - lift on wing (fin);

q - free-stream dynamic pressure, $q = 1/2\rho V^2$,

S - reference area; V - free-stream velocity; ρ - atmospheric density;

IV. OVERVIEW OF THE METHODS FOR AERODYNAMIC CHARACTERISTICS DETERMINATION

There are several methods for the aerodynamic characteristics determination, each of them has its own advantages and disadvantages, and is often used in various stages of the missile design process. Some commonly used methods are shown below.

A. Linear theory

In this method, the wing is divided into a grid system, the partial grid elements approximate the actual wing planform and surface shape. In linear theory, the wing has negligible thickness, and lies essentially in the z = 0 plane. According to [3], the grid elements identified by L and N, are arranged such that L is numerically equal to x and N is numerically equal to βy where x and βy take on only integer values. These grid elements are used to permit a closer approximation to the actual wing planform. With respect to a specified point (x, y) the upstream region of influence τ is approximated by the shaded grid elements in Figure 7.

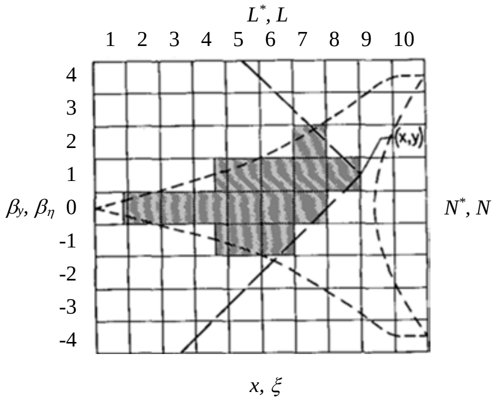

Figure 7. Wing coordinate system [3]

Each of these upstream region elements is conected with an influence factor $\overline{R}$. The factor $\overline{R}$ is determined from approximate solution to the linear theory of integrals over the bound region. The variation of $\overline{R}$ is illustrated in Figure 8 with grid element (L^*, N^*) [3].

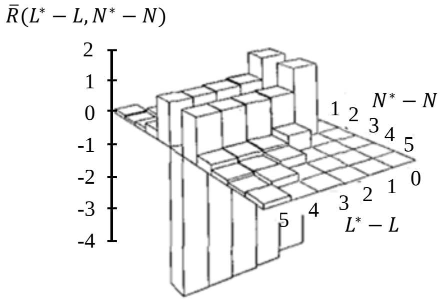

Figure 8. Numerical solution of linearized theory [3]

According to this method, given load distribution:

$$\frac{\partial z_c}{\partial x}=\frac{-\beta}{4}\cdot\Delta c_p\left(L^*,N^*\right)+ \\ +\frac{\beta}{4\pi}\cdot\sum\Delta c_p\left(L,N\right)\cdot A\left(L,N\right)\cdot\overline{R}\left(L^*-L,N^*-N\right), \tag{1}$$

where:

Δc_p - lifting-pressure coefficient,

c - local wing chord,

The brackets [x] designate the whole-number part of the quantity; l_e - leading edge; t_e - trailing edge,

A(L,N) - leading-edge-element weighting factor for influence summations,

A(L,N*)* - leading-edge-element weighting factor for force and moment summations,

For selected surface shape we can write:

$$\Delta c_p=\frac{-4}{\beta}\cdot\frac{\partial z_c}{\partial x}\left(L^*,N^*\right)+ \\ +\frac{1}{\pi}\sum\Delta c_p\left(L,N\right)\cdot A\left(L,N\right)\cdot\overline{R}\left(L^*-L,N^*-N\right). \tag{2}$$

Lift coefficient and drag coefficient at any selected semispan location $y = N^*/\beta$:

$$c_l=\frac{1}{c}\cdot\sum_{L^*=1+[x_{le}]}^{L^*=1+[x_{te}]}\Delta C_p(L^*,N^*)\cdot A(L^*,N^*)\cdot B(L^*,N^*), \\ c_d=\frac{-1}{c}\cdot\sum_{L^*=1+[x_{le}]}^{L^*=1+[x_{te}]}\frac{\partial z}{\partial x}(L^*,N^*)\cdot\Delta C_p(L^*,N^*)\cdot A(L^*,N^*)\cdot B(L^*,N^*), \tag{3}$$

where:

B(L, N) - trailing-edge-element weighting factor for influence summations,

B(L, N*)* - trailing-edge-element weighting factor for force and moment summations.

The lift and drag coefficients of the wing or fin are done by equations:

$$C_L=\frac{2}{\beta\cdot S}\cdot\int_0^{b/2}c_l\cdot c\,dy, \\ C_D=\frac{2}{\beta\cdot S}\cdot\int_0^{b/2}c_d\cdot c\,dy, \tag{4}$$

where: $\beta=\sqrt{M^2-1}$; S - reference wing area.

B. PAN AIR

PAN AIR is a higher order panel method to solve boundary value problems involving the Prandtl-Glauert equation for subsonic and supersonic potential flows. This equation for supersonic flow is written in form [4]:

$$-\overline{\nabla}^2\phi=\left(M_\infty^2-1\right)\cdot\phi_{xx}-\phi_{yy}-\phi_{zz}=0\,, \tag{5}$$

where M is the free-stream Mach number and is the perturbation velocity potential.

The wing or fin surface is divided into quadrilateral panels, the parameters on each panel are shown as shown in Figure 9.

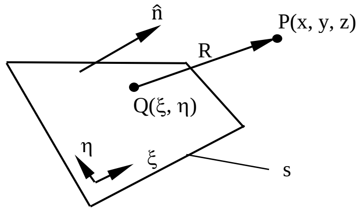

Figure 9. Geometry structure of i[th] panel [5]

The boundary condition at i[th] control point P:

$$\vec{V}_i \cdot \hat{n}_i = \left(\vec{V}_\infty + \sum_{j=1}^{N} \vec{v}_{ij}\right) \cdot \hat{n}_i = 0, \tag{6}$$

where: $\vec{v}_{ij}$ denotes the velocity at control point i[th], owing to the source-doublet distributions at panel j.

The boundary conditions application at these N control points leads to the equation:

$$[\mathrm{AIC}]\{\lambda\} = \{b\}, \tag{7}$$

where [AIC] is called the matrix of aerodynamic influence coefficients, $\{\lambda\}$ is the vector of unknown singularity parameters and the elements of {b} are known from boundary conditions.

The linear system of equations to determine the velocity and pressure distribution on each panel is solved. This allows to determine the aerodynamic characteristics of the wing.

C. *Combine Euler, Navier-Stockes equations and finite element method*

The Euler equations for an ideal incompressible fluid without external forces [6]:

$$\frac{\partial}{\partial t} \cdot \iiint_{\Omega} Q dV + \iint_{\partial\Omega} F(Q) \cdot n dS = 0, \tag{8}$$

where Ω represents the physical domain with a boundary $\partial\Omega$:

$$Q = \begin{Bmatrix} \rho \\ \rho u \\ \rho v \\ \rho w \\ e \end{Bmatrix}, \; F(Q) \cdot n = (V \cdot n) \begin{Bmatrix} \rho \\ \rho u \\ \rho v \\ \rho w \\ e + p \end{Bmatrix} + p \begin{Bmatrix} 0 \\ n_x \\ n_y \\ n_z \\ 0 \end{Bmatrix}, \tag{9}$$

n_x n_y n_z are the Cartesian components of the external surface unit normal *n* on the boundary $\partial\Omega$; *u, v, w* - velocity components in *x, y,* and *z* directions; ρ - density; *e* - total energy per unit volume.

The Navier-Stokes equations describe the motion of fluids and result from the Newton's second law of motion for fluids. In the case of a compressible Newtonian fluid, this yields:

$$\underbrace{\rho \cdot \left(\frac{\partial u}{\partial t} + u \cdot \nabla u\right)}_{1} = \underbrace{-\nabla p}_{2} + \underbrace{\nabla \cdot \left(\mu \cdot \left(\nabla u + (\nabla u)^T\right) - \frac{2}{3}\mu \cdot (\nabla \cdot u) \cdot I\right)}_{3} + \underbrace{F}_{4}, \tag{10}$$

where *u* is the fluid velocity, *p* is the fluid pressure, ρ is the fluid density, and μ is the fluid dynamic viscosity. The different terms correspond to the inertial forces (1), pressure forces (2), viscous forces (3), and the external forces applied to the fluid (4).

The flow is divided into a grids system. These methods use finite element method to solve Euler or Navier-Stokes equations with boundary conditions which include shock wave faces appearing on the wings or fins.

There are several commercial and open-source software packages which facilitate solutions to the Navier-Stokes equations. The best known of them are ANSYS Fluent, OpenFoam and StarCCM+.

D. *Supersonic Wind Tunnel Testing*

A supersonic wind tunnel is an equipment that produces air flow at supersonic velocity (1.2 < M < 5). The flow Mach number is determined by the nozzle geometry. The Reynolds number is varied by changing the density level (pressure in the setting chamber). A supersonic wind tunnel has a large power demand, so most of them are designed for intermittent instead of continuous operation.

Even the most sophisticated computational approaches cannot substitute for actual test data. Properly designed and conducted wind tunnel tests can provide chordwise loading and spanwise loading as well as global forces and moments. While undeniably accurate, wind tunnel testing is both time consuming and expensive. It is a reason why wind tunnel testing is never carried out in the preliminary design phase. Figure 10 shows a supersonic wind tunnel model.

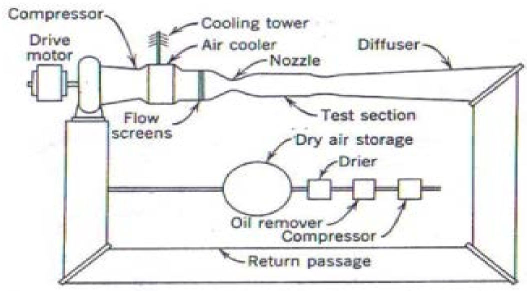

Figure 10. Continuous Wind Tunnel [8]

V. COMPARISON OF SEVERAL METHODS

For determining the aerodynamic characteristics of the missile wings (fins), the above methods have ignored several influencing factors to simplify the aerodynamic model. Whichever method that ignores many influencing factors is less accurate. For seeing the correlation of accuracy between methods, some comparisons of the results that are calculated by the above methods are presented in the article. In addition, the comparison between available results [7] and new calculated results is also presented. For the above comparisons, some analyzes and assessments of correlations among methods have been carried out.

A. *A comparison between Linear Theory and Supersonic Wind Tunnel Testing*

In [3], Linear theory was used to calculate some aerodynamic characteristics of a wing-body with uncambered wing. The relationship between ΔC_D (drag coefficient due to lift) and C_L (lift coefficient) is shown in the Figure 11.

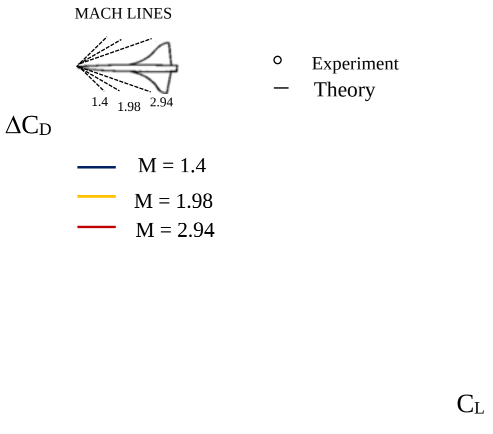

Figure 11. Comparison of merical method and experiment [3]

Experimental data and numerical results agree quite well.

B. A comparison between PAN AIR method and Supersonic Wind Tunnel Testing

In [4], PAN AIR method was used to calculate some aerodynamic characteristics of some aircraft models. The relationships between the lift coefficient with the angle of attack and the moment coefficient at M = 1.2 which are solved by different methods is shown in the Figure 12.

C_L

C_M

Figure 12. Comparisons of PAN AIR and experimental values of lift and pitch moment coefficients [4]

The agreement between the calculated and experimental results are very good. PAN AIR gives better agreement with experimental data than either the constant-pressure panel method or the Mach box method, because they can't provide accurate modeling of the canopy and inlet.

C. A comparison between Euler, Navier-stokes methods and Supersonic Wind Tunnel Testing

In [6], an unstructured Euler flow solver and a structured thin-layer Navier–Stokes solver were used to calculate some aerodynamic characteristics of a conventional missile model at supersonic velocity with high angles of attack. The results are compared with experimental data. The Euler solution uses USE3D tool. The Navier-Stokes solution uses FLU3M tool. The conventional missile model has the geometry with an ogive nose, a cylindrical body, and four clipped delta tail fins. The fin aifoil is double wedge.

Comparisons between the Euler and Navier-stock solutions with the experimental data on normal force coefficient are shown in the Figure 13.

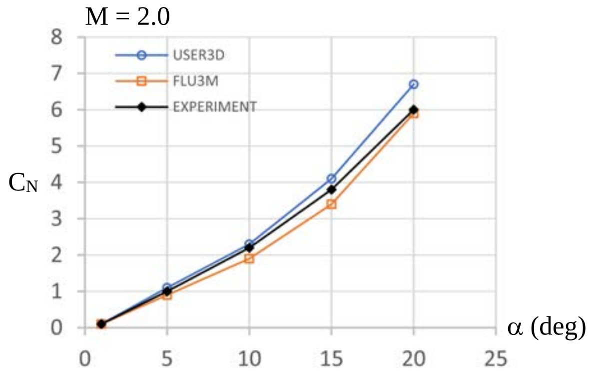

Figure 13. Variation of normal force coefficient with angle of attack [6]

According to Figure 13, all observed results are in very good agreement at small angles of attack, in which the present Euler results are much closer to the experimental data.

When the angle of attack increases, the difference from the experimental results increases. This can be attributed to viscous effects becoming dominant, especially at high angles of attack. At these angles, Navier-Stokes solutions are effective in predicting the viscous behavior of the flow field with obtained results quite close to the experimental results.

Results for the individual fin aerodynamic characteristics are shown in Figure 14. The agreement between the two computational methods and the experimental data are very good.

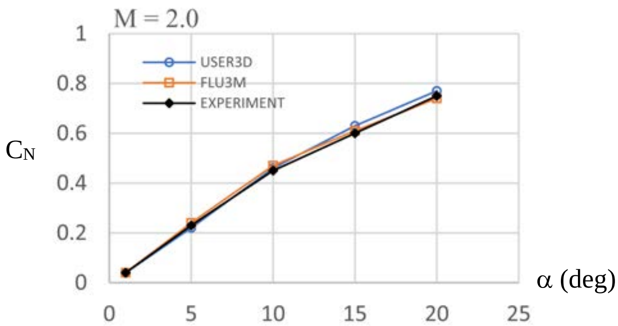

Figure 14. Variation of normal force coefficient for one fin with angle of attack [6]

D. A comparison between the available result of Shock-Expansion method and calculation result by Navier-Stockes method for Double edge airfoil.

The article proceeds calculating the lift coefficient of an airfoil model with the same geometric parameters and the same flow condition of a calculation by Shock-Expansion method in [7]. The article uses the Navier-Stokes method with Ansys Fluent tool.

Calculation results are compared with the result of Shock-Expansion method in [7].

The model of the airfoil chosen to be calculated according to [7] is a double wedge airfoil with geometric parameters as: chord = 0.0254 m; thickness/chord ratio is 0.1.

Flow conditions:

- Inlet: $M_\infty = 2.5$; $T_\infty = 218$ K; $p_\infty = 20265$ Pa;
- Airfoil: no slip;
- Wall: slip; $u_x = 600$ m/s; $u_y = 0$ m/s;
- Angle of attack: 0°; 2°; 4°; 8°; 12°.

Ansys Fluent calculation results are displayed on the same graph with the results of Shock-Expansion method as shown in Figure 15. Results of the two methods agree quite well.

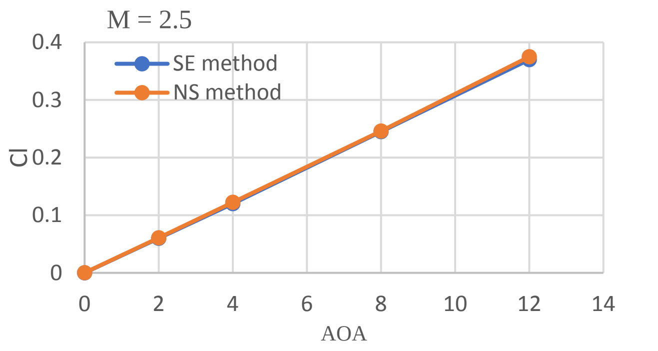

Figure 15. The comparison of lift coefficient results which are determined by two methods Shock-Expansion method and Navier-Stockes method

The distribution of the airflow velocity on the airfoil at angle of attack 12° is shown in the Figure 16:

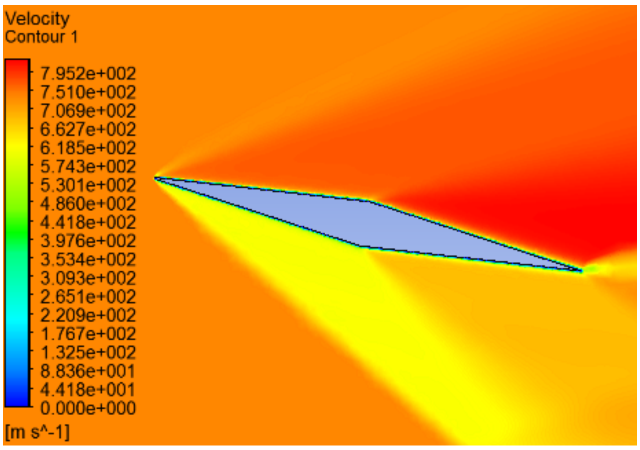

Figure 16. Distribution of airflow velocity around the airfoil at the angle of attack 12°

VI. Conclusion

Several numerical methods are introduced for comparison of the aerodynamic characteristics determination. There are now many quick and accurate numerical solution tools. The calculation results by these tools are compared with the experimental results. These comparisons show fairly high accuracy of the mentioned methods. However, each of these methods has different accuracy for selected mode of movement. This difference is due to the assumptions that these methods use for simplification of the solved problem.

Tools for identification of the aerodynamic characteristics are often used in the early stages of design to predict them before providing experiments in the wind tunnel.

Acknowledgment

This work has been supported by the specific project of the University of Defence in Czech Republic (Evaluation and assessment of weapon systems).

References

[1] "Missile control systems", available from: <http://www.aerospaceweb.org/question/weapons/q0158.shtml>.

[2] GARY A. CROWELL SR, "The descriptive geometry of nose cones", 1996.

[3] Wilbur D. Middleton and Harry W . Carlson, "A numerical method of estimating and optimizing the supersonic aerodynamic characteristics of wings of arbitrary operations meeting", AIAA transport aircraft design & operations meeting, Seattle, Washington August 10-12, 1964.

[4] Thomas Derbyshire, Kenneth W. Sidwell, "PAN AIR Summary Document", NASA Contractor Report 3250, 1982.

[5] R. L. Carmichael, L. L. Erickson, "PAN AIR - A Higher Order Panel Method for Predicting Subsonic and supersonic Linear Potential Flows About Arbitrary Configurations", AIAA 14th Fluid and Plasma Dynamics Conference, California, June 1981.

[6] E. Oktay, N. Alemdaroglu, E. Tarhan, P. Champigny and P. d'Espiney, "Euler and Navier–Stokes Solutions for Missiles at High Angles of Attack", JOURNAL OF SPACECRAFT AND ROCKETS Vol. 36, No. 6, November–December 1999.

[7] Ramesh Kolluru1, Vijay Gopal, "Numerical Study of Navier-Stokes Equations in Supersonic Flow over a Double Wedge Airfoil using Adaptive Grids", BMS College of Engineering, Bangalore, Karnataka, India.

[8] Pope A., Goin K, "High Speed Wind Tunnel Testing", New York: John Wiley & Sons; 1965.