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Some practical issues in the computational design of airfoils for the helicopter main rotor blades

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Abstract

Very important requirement for the helicopter rotor airfoils is zero, or nearly zero moment coefficient about the aerodynamic center. Unlike the old technologies used for metal blades, modern production involving application of plastic composites has imposed the necessity of adding a flat tab extension to the blade trailing edge, thus changing the original airfoil shape. Using computer program TRANPRO, the author has developed and verified an algorithm for numerical analysis in this design stage, applied it on asymmetrical reflex camber airfoils, determined the influence of angular tab positioning on the moment coefficient value and redesigned some existing airfoils to include properly positioned tabs that satisfy very low moment coefficient requirement.

Key words: helicopter, composite rotor blade, airfoil design, tab, numerical modelling.

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Nomenclature

α	angle of attack
C_L	airfoil lift coefficient
Cm_{ac}	airfoil moment coefficient about the aerodynamic center
$Cm_{1/4}$	airfoil quarter chord moment coefficient
C_{D}	airfoil drag coefficient
C_{P}	pressure coefficient
M	local Mach number
M_∞	free stream Mach number
Re	Reynolds number for the unit chord length
R_θ	Reynolds number defined by boundary layer momentum thickness as characteristic length
V	local velocity
a	local speed of sound
x, z	physical space coordinates
ξ, η	calculation space coordinates
f	$d\xi/dx$
g	$d\eta/dz$
u	local velocity component in x direction
w	local velocity component in z direction
$\bar{\phi}$	nondimensional velocity perturbation potential
c	airfoil chord length (unit)
x/c	relative chordwise coordinate
c_t/c	relative tab chord length
H	boundary layer shape factor
$\bar{\mathrm{H}}$	compressibility corrected boundary layer shape factor
M_{e}	local Mach number at the outer edge of the boundary layer
u_e	local velocity component at the outer edge of the boundary layer
ρ_e	local density at the outer edge of the boundary layer
Θ	boundary layer momentum thickness
θ	assumed reference tab angle with respect to the chord, from which tab angular deflection is defined
δ	optimum tab position, for estimated or true $Cm_{ac} = 0.0$
δ^*	boundary layer displacement thickness
τ	tab angular deflection (constant in this paper) from θ

Subscripts

$[n]$	superscript denoting the "n"-th iteration cycle value
(te)	subscript denoting a parameter value on the trailing edge
U	subscript denoting a parameter value on the upper airfoil surface
L	subscript denoting a parameter value on the lower airfoil surface

1 Introduction

Numerical aerodynamic design of the appropriate airfoils for the contemporary helicopter main rotor blades is probably one of the most challenging and demanding areas of the computational fluid mechanics. Helicopter main rotor blades in progressive forward flight are subjected to combined cyclic pitching, flapping and leading – lagging motions, at some 250 $\div$ 400 cycles (revolutions) per minute. Local blade section velocities are vector sums of the local tangential velocity of the rotating motion and the velocity of the progressive flight, so their angles of attack must change from some -5^{o} at advancing positions to even +20^{o} for the retreating ones, in order to obtain the resultant lifting force of the rotor disc in the plane of symmetry of the helicopter. The most important consequences of such airflow conditions are: (1) the tips of the advancing blades reach transonic speeds; (2) the tips of the retreating blades are stalled, and (3) the root sections of the retreating blades are subjected to the inverted flow, coming from the direction of their trailing edge. Unlike aircraft propellers, helicopter rotor blades are very flexible and they are kept spread in flight only by the centrifugal force acting normal to their axis of rotation. Also, the aerodynamic center, center of gravity and aeroelastic axis of all sections along the blade should coincide, or should be at least very close to each other.

Considering that, some of the most important aerodynamic requirements for the helicopter rotor airfoils are: (a) aerodynamic moment coefficient about the aerodynamic center $\mathrm{Cm}_{\mathrm{ac}}$ must be equal or very close to zero, to prevent the induction of the torsional moment along the blade and too large collective pitch control system forces; (b) critical (drag, moment, etc. divergence) Mach numbers should be reasonably high, to prevent excessive transonic effects at the tips of the advancing blades; (c) maximum lift coefficient and critical angle of attack should be large enough to prevent substantial loss of lift at the tips of the retreating blades (luckily, very quick changes in pitch of the blades enable achievement of higher maximum lift coefficients than in "static" cases, such as on airplane wings or in usual wind tunnel tests, because of the inability of flow to separate so quickly); d) profile drag should be as small as possible with previous requirements satisfied, etc.

2 Problem Definition

In this paper analyses will be focussed on the first requirement for main rotor airfoil design, stating that the aerodynamic moment about the aerodynamic center must be equal or very close to zero. In practice, two general categories of airfoils are used for helicopters. The first category are symmetrical airfoils, for which this requirement is readily satisfied (Fig. 1(a)), with some minor exceptions at higher angles of attack. The second group are the asymmetrical - reflex camber airfoils (Figures 1(b) $\div$ 1(d)), whose positive camber in the nose domain is used to improve airfoil characteristics at high angles of attack. They are characterized by convex shape of the front portion of mean line, which locally generates pitch-down aerodynamic moment. Thus the rear portion of the mean line must be concave, to balance the moment about the aerodynamic center and bring it as close to zero as possible. According to ref. [1], which currently contains airfoil data for more than 240 existing helicopters, these two categories of airfoils have been almost evenly used in helicopter industry.

Modern technologies imply the use of composite materials in the helicopter blade production, instead of metal (or sometimes even wooden) blades that were used extensively on the helicopters of earlier generations. In order to properly polymerize and merge the plies of the upper and lower composite blade surfaces, a small thin flat tab, of some 5% $\div$ 10% relative chord, is added at the trailing edge of the airfoil. For the metal or wooden blades that requirement did not exist. The direction of the tab extension in case of symmetrical airfoils is straight behind, at the zero angle with respect the chord of the original airfoil, and thus the zero Cm_{ac} requirement is satisfied.

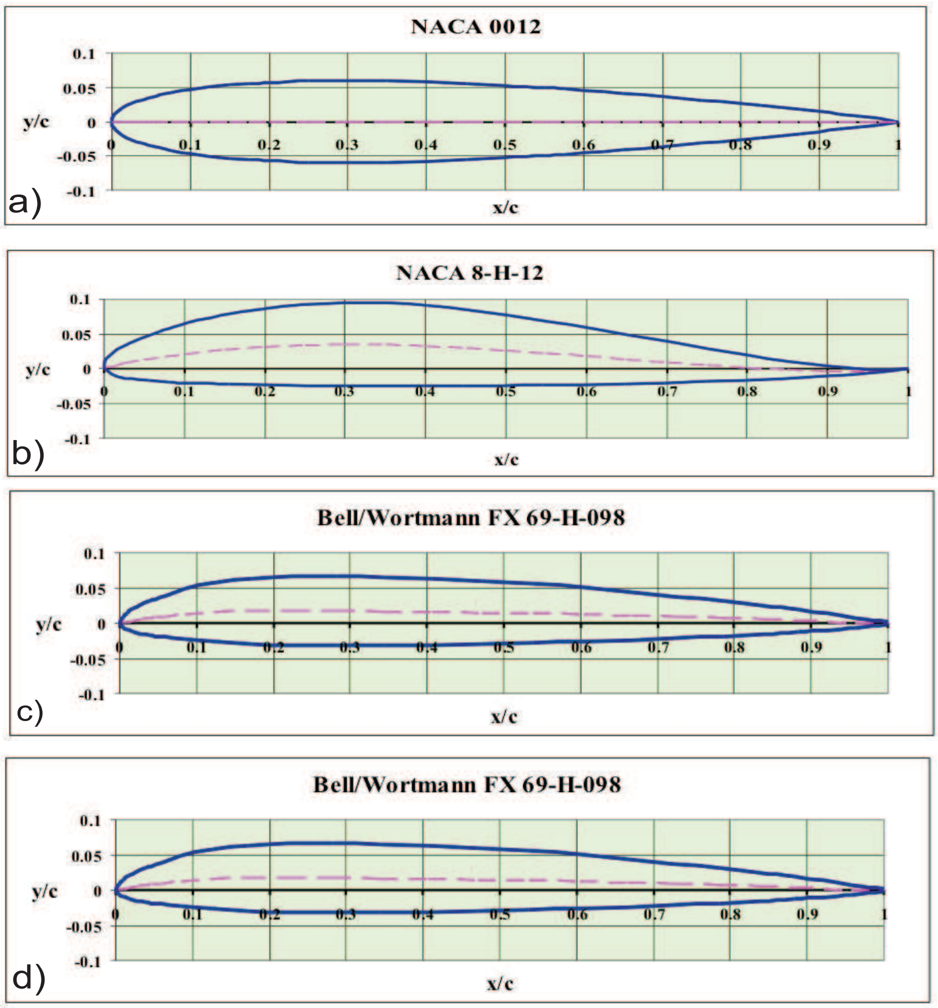

Figure 1: Some of the commonly used airfoils for main helicopter rotor blades

In case of reflex camber airfoils applied on composite blades (example is shown in Figure 2), determination of optimum angular tab position is more complex. Improper selection of the tab angle with respect to the blade chord could alter initially small enough Cm_{ac} of the basic airfoil to larger values that might induce excessive elastic torsion of the blade and change the designed twist angle distribution. The consequence would be the reduction of the overall rotor effectiveness, while in the most extreme cases, large Cm_{ac} could even induce aeroelastic blade divergence and fatal outcome of the flight. This problem could be partially solved by applying additional number of plies and strengthening the blade skin (but also increasing its mass). Even in that case, the increased control forces due to pitching moment could still remain an inevitable problem. So keeping the Cm_{ac} small enough is a necessity.

One of the obvious approaches to this problem is undertaking the extensive wind tunnel tests during which, among the other results, the optimum angular tab position for the required chord length is determined. This is the usual approach in case the high budget helicopter design projects, applied by the companies such as Boeing, Hughes, Aerospatiale, etc. Some of the reflex camber airfoils with the tab included in the geometry shape, obtained by such wind tunnel tests, have been published (and often patent protected, Figure 3). Unfortunately, according to the "UIUC Airfoil Coordinates Database Ver. 2.0", published by the University of Illinois [2], which currently contains data for more than 1550 airfoils, the number of available rotorcraft airfoils with tabs included is still proportionally very small compared with the number of the "pure" rotorcraft airfoils that do not contain tab.

On the other hand, in low budget development programs, characteristic for presently very popular light helicopters, extensive airfoil wind tunnel tests could sometimes overload the available funds. Engineers are often forced to select optimal airfoils for the given helicopter, whose geometry does not include tab, and then add the tab in some of the design stages. Instead of the wind tunnel tests, in such projects it might be less expensive to produce several different blade sets and test them directly on the helicopter prototype, until apparently satisfactory blade behavior from the aspect of aerodynamic moment is reached.

Proper initial estimate of the tab angular position can be one of the factors that could remarkably reduce the overall cost both of the wind tunnel and/or the prototype tests. In the rest of this paper numerical simulation and analysis of the above mentioned initial design stage, assuming static flow conditions, is presented. Tabs of usual relative

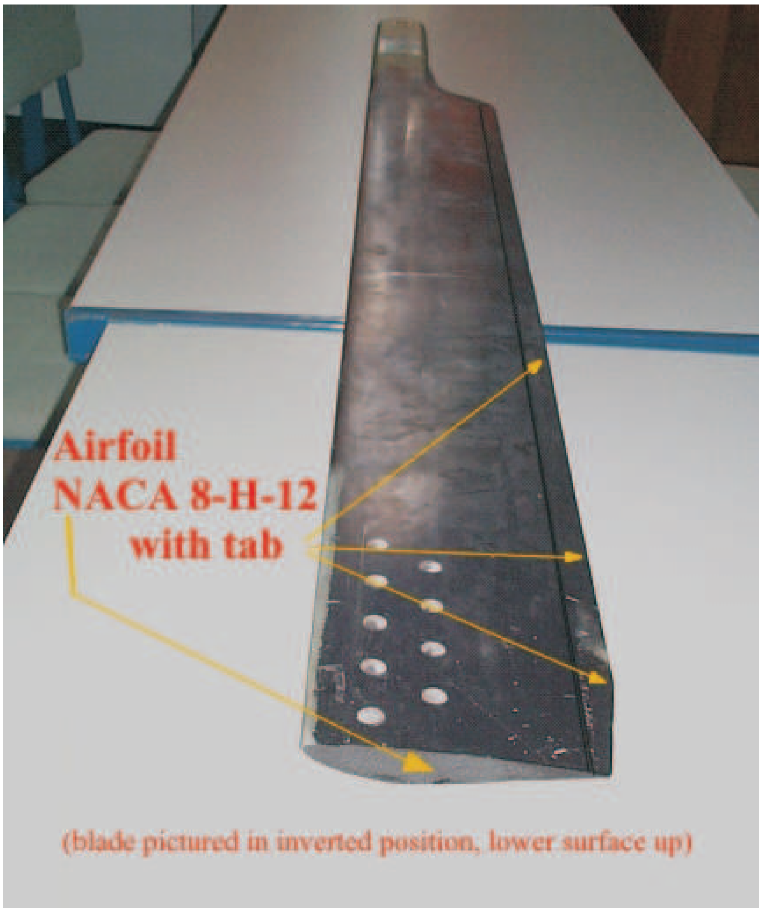

Figure 2: Prototype of a light helicopter composite rotor blade; tab is added to the "pure" airfoil shape because of the composite production requirements

chord lengths have been added to several "pure" airfoils. The attention is paid to the analysis of influence of the tab angular position on the achieved the Cm_{ac}-s, that may raise beyond the assumed low value limits. Some existing asymmetrical helicopter airfoils have then been redesigned to include properly positioned tabs that satisfy low moment coefficient requirement. Finally, the accuracy of the whole presented numerical modeling algorithm was tested and verified on a symmetrical airfoil, for which the tab optimum position is readily known ($\delta = 0^o$),

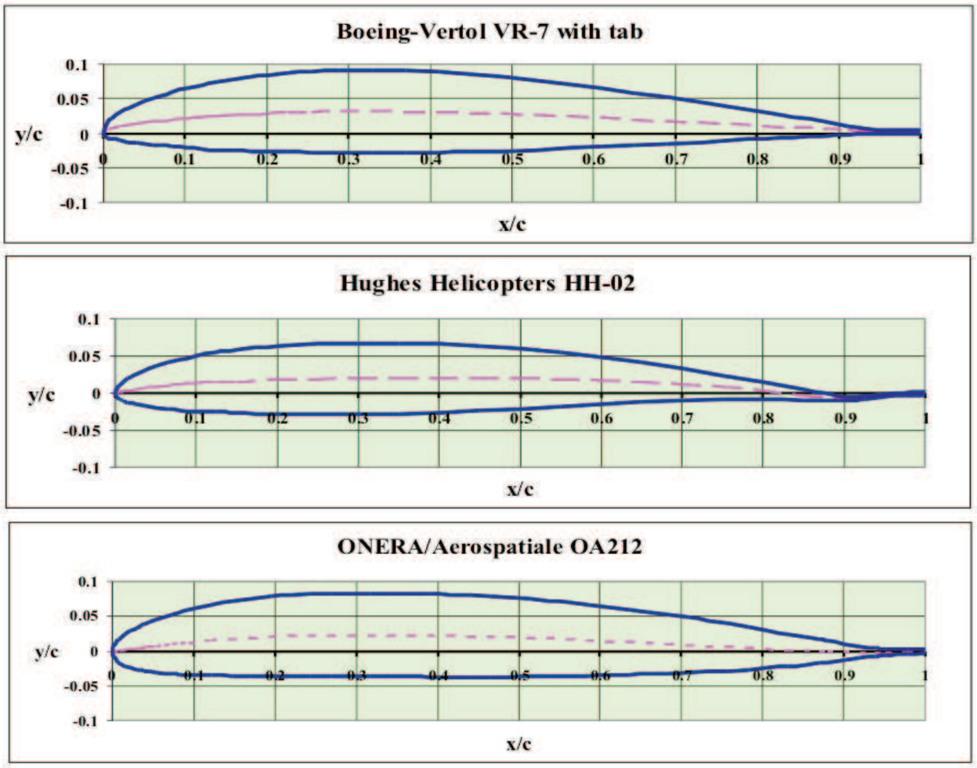

Figure 3: Several rotorcraft airfoils which include tab within their original geometry; the number of such available airfoils is still relatively small.

since it has to preserve the airfoil's symmetry and the zero Cm_{ac} value.

3 Concise description of the applied numerical model

Calculations presented in this paper are the results of the computer program TRANPRO [3], an upgraded version of the Trandes [4] computer program. The Trandes was developed by Prof. L.A.Carlson of the Texas A& M Univ. USA, under the contract for the NASA agency, in the late seventies. Very soon the Trandes has also been accepted by many leading aircraft corporations and universities throughout the world. This program was used for the aviation airfoil analysis and design, for subsonic and lower transonic speed domains. Although the state of the art software of its time due to many qualities it possessed, it has also been the subject to some critics in scientific papers ever since the time of it's issue, and many of its users have developed their own upgraded versions of this program. The author of this paper has had a chance to use Trandes extensively, and from this experience, the TRANPRO computer program has been developed.

Both in the Trandes and the TRANPRO computer programs, the zonal approach in airflow calculation is applied. Although nowadays quite classical, such approach is still very successful in many operational engineering applications. The advantage of this approach lies mostly in its extremely high computer resource and time efficiency, while at the same time the accuracy of the results can be brought to more than satisfactory level. In order to achieve this goal with TRANPRO and make it compatible with contemporary commercial software packages, the author has introduced some important changes and modifications to the basic Trandes model, and established algorithms of its optimum use. Some of them are briefly described in section 3.3, while considering mentioned specific aspects of helicopter airfoil design, detailed descriptions and analyses are presented in sections 4 and 5.

3.1 Calculation of the inviscid part of the flow

The inviscid part of the flow is calculated over the displacement surface of the airfoil, i.e. the airfoil contour increased by the numerically smoothed local distribution of the δ^*. In the TRANPRO, this calculation is done by the same general algorithm as the one applied in Trandes. It is based on the solution of the full nondimensional perturbation potential $\bar{\phi}$ nonlinear partial differential equation, which in the physical $x - z$ space for the unit airfoil chord length takes the form:

$$\left(a^2 - u^2\right) \overline{\phi}_{xx} + \left(a^2 - w^2\right) \overline{\phi}_{zz} - 2\, uw\, \overline{\phi}_{xz} = 0 \tag{1}$$

while, applied in the calculation space $\xi - \eta$, it changes to:

$$\left(a^2 - u^2\right)\ f\left(\overline{\phi}_\xi\right)_\xi + \left(a^2 - w^2\right)\ g\left(\overline{\phi}_\eta\right)_\eta - 2\ uw\ \ fg\ \overline{\phi}_{\xi\eta} = 0 \tag{2}$$

where $f = \mathrm{d}\xi\ /\ \mathrm{d}x$ and$g = \mathrm{d}\eta\ /\ \mathrm{d}z$. Specially, in the local supersonic domain, where Jameson's rotated finite difference $s - n$ scheme is used, the governing equation takes the form [3, 4]:

$$(1 - \mathrm{M}^2)\overline{\phi}_{ss} + \overline{\phi}_{nn} = 0 \tag{3}$$

in which:

$$\overline{\phi}_{ss} = \frac{1}{V^2}\left[u^2 f(f\overline{\phi}_\xi)_\xi + 2\,uw\,fg\overline{\phi}_{\xi\eta} + w^2 g(g\overline{\phi}_\eta)_\eta\right] \tag{4}$$

$$\overline{\phi}_{nn} = \frac{1}{V^2}\left[w^2 f(f\overline{\phi}_\xi)_\xi - 2\,uw\,fg\overline{\phi}_{\xi\eta} + u^2 g(g\overline{\phi}_\eta)_\eta\right] \tag{5}$$

Very quick convergence of the solution is obtained by calculating the flow on the series of rectangular grids, starting with 13 x 7, then 25 x 13, 49 x 25 and 97 x 49. Very often the final solution is obtained on the 49 x 25 grid, so the finest grid need not be applied, which reduces the computation time.

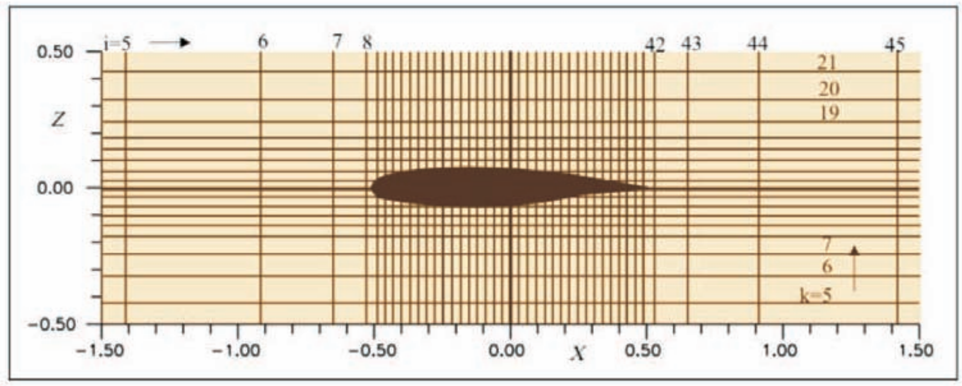

Figure 4: Grid 49 x 25 in the vicinity of the airfoil

3.2 Calculation of the viscous effects

In this paper only turbulent boundary layer case with transition point fixed close to the leading edge will be discussed (reasons for that will be explained in the section 4). In Trandes, the Nash-Macdonald integral turbulent boundary layer calculation is used, while in Tranpro, the modified [5, 6] version of this model is applied. The momentum integral equation [7,8]:

$$\left(\frac{\mathrm{d}\theta}{\mathrm{d}x}\right)^{[n]} = -\frac{(\theta^{+})^{[n-1]}}{u_e^{+}}\frac{\mathrm{d}u_e}{\mathrm{d}x}\left(\mathrm{H}+2-\mathrm{M}_{\mathrm{e}}^2\right)+\frac{1}{\left(\zeta^{[n]}\right)^2} \tag{6}$$

is solved for the momentum thickness θ. In (6), "e" denotes the values on the outer edge of the turbulent boundary layer, while $[n]$ denotes a certain iteration cycle value. Parameter ζ is defined by:

$$\begin{aligned}\zeta^{[n]} &= \mathrm{F_C}\left[2.4711\cdot\ln\left(\mathrm{F_R}\,\mathrm{R}_\theta^{[n-1]}\right)+4.75\right]+\\ &\quad 1.5\,\mathrm{G}^{[n-1]}+\frac{1724}{\left(\mathrm{G}^{[n-1]}\right)^2+200}-16.87\end{aligned} \tag{7}$$

in which:

$$\mathrm{F_C} = 1+0.066\ \left(\mathrm{M_e}\right)^2-0.008\ \left(\mathrm{M_e}\right)^3 \tag{8}$$

$$\mathrm{F_R} = 1-0.134\ \left(\mathrm{M_e}\right)^2+0.027\ \left(\mathrm{M_e}\right)^3 \tag{9}$$

where G is the Clauser parameter. Shape factor $\mathrm{H}=\delta^*/\theta$ is calculated by:

$$\bar{\mathrm{H}}^{[n]} = \frac{1}{1-\mathrm{G}^{[n-1]}\,(1/\zeta)} \tag{10}$$

and

$$\mathrm{H}^{[n]} = \left(\bar{\mathrm{H}}^{[n]}+1\right)\ \left[1+0.178\ \left(\mathrm{M_e^{+}}\right)^2\right]-1 \tag{11}$$

In the modified TRANPRO's numerical model [5, 6], the Clauser parameters G and β_{p} are related by :

$$G^{[n]} = 6.1\sqrt{\beta_{\mathrm{p}}^{[n]} + 1.81} - 4.1 \tag{12}$$

(while in the original Carlson/Nash-Macdonald model used in Trandes, the usual equation of Nash [1, 10] is applied, giving quite inaccurate results). Once θ is determined, turbulent boundary layer displacement thickness is calculated by $\delta^* = \mathrm{H} \cdot \theta$. Finally, the distribution of δ^* is smoothed [3, 4] over the airfoil, and so the airfoil displacement surface is obtained.

For the profile (joined pressure and friction) drag coefficient calculations, the modified Squire-Young formula, with the separate trailing edge values for upper and lower surface, is used:

$$C_{\mathrm{DP}} = 2 \cdot \left[\theta_{(\mathrm{te})\,\mathrm{U}} \left(\frac{u_{e(\mathrm{te})}}{u_\infty} \right)_{\mathrm{U}}^{\frac{\overline{\mathrm{H}}_{(\mathrm{te})\mathrm{U}}+5}{2}} + \theta_{(\mathrm{te})\,\mathrm{L}} \left(\frac{u_{e(\mathrm{te})}}{u_\infty} \right)_{\mathrm{L}}^{\frac{\overline{\mathrm{H}}_{(\mathrm{te})\mathrm{L}}+5}{2}} \right] \tag{13}$$

In case of the transonic (supercritical) flow, it is necessary to calculate and add the wave drag coefficient to this value [3, 4, 9]. Otherwise, airfoil drag coefficient is $C_{\mathrm{D}} = C_{\mathrm{DP}}$.

3.3 Overview of some specific characteristics of the applied software

Compared with Trandes, program TRANPRO developed by the author of this paper, has many additional modules and modifications that give it substantial advantages over the source program, such as:

1) Completely automated trailing edge closure control in the inverse stage of the introduced inverse-direct airfoil design algorithm, based on controlled corrections of the initial pressure distributions initially defined by the user; purely inverse approach of Trandes was prone to giving "U" or "gamma" airfoil shapes, and required manual corrections in pressure coefficient inputs by the user, and the successful outcome was largely influenced by the user's experience in that [3].

2) Capability of additional automatic corrections of the mean line shape in the direct design stage, in order to generate airfoils that will

satisfy required quarter chord moment coefficient value or indirectly the prescribed lift coefficient; this option did not exist in Trandes [3].

3) Modified boundary layer calculation model, and thus largely improved accuracy in profile drag calculations, giving good agreements with experimental results in wide range of angles of attack; Trandes had problems with profile drag accuracy even at smaller angles of attack [5], [6].

4) Improved transonic flow algorithm, that gives stable and unique solutions for the wave drag and does not rely on the "user's experience" factor; on the other hand, Trandes could in some cases give even negative wave drag coefficients, and user had to eliminate this problem by some suggested approaches, whose final outcome also depended very much on his skills [9], etc.

Application of rectangular grids (Figure 4) has advantages, because the generation of grids is performed using rather simple algorithms, they do not depend on airfoil shape and the results on coarser grids are used to interpolate initial values on finer ones. Such approach gives very quick convergence of the final solution, and thus, on modern computers, TRANPRO's CPU time is reasonably short even for more complex analyses, involving calculations for wide ranges of angles of attack and Mach numbers in a single program run (Trandes could only analyze one angle of attack for one given Mach number in a single run).

On the other hand, the application of such grids also has some disadvantages, that come out from the fact that the airflow parameters on the effective airfoil shape (geometrical shape plus local displacement thickness) must be interpolated, since grid points generally do not coincide with that contour. Because of the applied calculation scheme (Figure 4), boundary layer might be slightly "numerically" thickened on one airfoil surface and thinned on the other, compared with their actual distribution, which than induces proportional redistribution of the calculated pressure coefficient, and vice versa, during the viscous-inviscid closed loop iteration procedure. Owing to that, the results are sometimes systematically "shifted" with respect to the experimental values for a given true angle of attack, depending on the basic airfoil shape and airflow parameters. In literature it is known as "the angle of attack systematic error", or "shift", which sometimes might be of the order of $1 \div 2$ degrees. In case of lift and drag calculations, it can be easily overcome by comparing drag coefficient directly with the lift coefficient, i.e. by polar curve presentation (C_D versus C_L), because both parameters are proportionally shifted. In such representation of the results, TRANPRO gives good agreements with the appropriate experimental data [5].

It should be noted that for a given basic airfoil shape and airflow conditions, eventual shift of the results is quite steady and uniform in a wide range of angles of attack. That is the reason why this kind of numerical behavior is usually rather called shift than error, since the term error often implies quite stochastic and unpredictable scatter in numerical solutions, which is not the case here.

If it appears, the shift is also reflected in the moment coefficient results to a certain extent. Since by its definition the Cm_{ac} does not depend on the angle of attack or the lift coefficient, except at high angles of attack, shift in this particular case could not really be treated as "the angle of attack shift" (maybe "effective shape redistribution", or some similar formulation would suit it better). To calculate Cm_{ac} and the aerodynamic center position, values of $Cm_{1/4}$and C_L for two angles of attack must be used. Very important fact about the application of here applied numerical model is that, whatever pair of angles of attack is used, the calculated Cm_{ac} is always practically the same, and the shift in calculated value of this parameter is also quite constant and stable. This enables that, for certain engineering applications, algorithms could be established to eliminate the shift from the calculated value of the moment coefficient about aerodynamic center, and give results that can be used in airfoil design as very reliable for practical engineering purposes. Such an algorithm, introduced by the author of this paper and applied for the analysis of the helicopter airfoil design problem described in this paper, will be demonstrated in the following sections.

4 First stage of airfoil design – evaluation of the influence of tab geometry on the variation of Cm_{ac}

As mentioned in section 2 of this paper, trailing edge tab is a necessity in the composite blade production. Although this structure element is added because of the production technology requirements, its existence may affect the aerodynamic behavior of the blade and thus the whole rotor efficiency, if not properly designed and positioned on reflex camber airfoils. Relative chord lengths of the tabs (c_t/c) range from approximately 5% in case of large transport helicopters to some 10% for light helicopters. Such small relative chord lengths may sometimes lead the designers to underestimate the tabs' capacity to retrim the initially sufficiently small $\mathrm{Cm_{ac}}$ of pure reflex camber rotorcraft airfoils to values that might be improper for this purpose.

The problem of initial tab positioning will be introduced by an example of simple theoretical analysis, that will be presented for the NACA 8-H-12 helicopter airfoil (Figure 5). Initially we will suppose that an infinitely thin flat tab of 5% chord length should be added behind the original trailing edge and that it is necessary to define some limits between which the optimum tab position, that will give $\mathrm{Cm_{ac}} = 0.0000$, could be expected. If the mean line would be extrapolated following its trend in the vicinity of the trailing edge, point at 5% behind the airfoil connected with the trailing edge point would give a flat tab angle of some 8.3° above the chord. For airfoil without the tab, the rear - concave part of the mean line is designed balance the pitch-down moment of its front convex section, so the total moment about the aerodynamic center is nearly zero. With this modification, only the concave mean line portion is extended and the airfoil will be overbalanced nose-up, probably even to an extent that might be unacceptably large for helicopter applications.

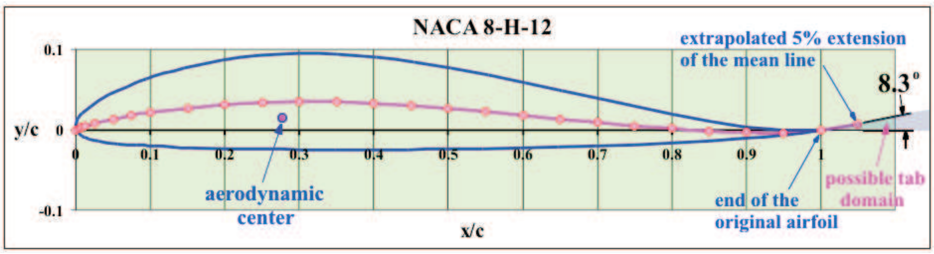

Figure 5: Determination of the possible domain of tab angular positions for 8-H-12 airfoil

On the other hand, if the tab would be positioned just straight behind as a chord extension, the natural slightly upward oriented flow just behind the trailing edge of the original airfoil in this case would be immediately forced to change direction. As a consequence, an upward oriented reaction force on the tab would be produced, causing overbalancing pitch-down moment about the aerodynamic center. So, the optimum position of the tab on the 8-H-12 airfoil should be somewhere between these two established limits. This means that a blade designer who wishes to use the 8-H-12 airfoil, when defining the initial shape for a wind tunnel test model or a prototype test blade, will have to assume (or maybe better – guess) the initial tab position somewhere in this range. The 8.3° range is quite large, and so are the chances that optimum tab position might be missed to an extent when Cm_{ac} would become too large for a rotor airfoil. That would certainly require production of a new test model or test blade, additional time and funds, etc.

In order to more profoundly evaluate the tab size and angular position influence on Cm_{ac}, the author has redesigned four airfoils widely used in helicopter industry, by fitting them with approximately 0.5% ÷ 0.9% thick flat tabs. Beside the three asymmetrical airfoils - NACA 23012 (ref. [10]), NACA 8-H-12 (ref. [11]) and Bell/Wortmann FX 69-H-098 (ref. [2]), symmetrical airfoil NACA 0012 (ref. [10]) has also been included in the analysis as a control case, in order to verify the accuracy of numerical calculations and conclusions obtained by them (neutral tab position and some of its influences are readily known for symmetrical airfoils). Airfoils were redesigned by extending them initially for 5% chord behind the original trailing edge (7.5% in case of FX 69-H-098) at an assumed reference angle θ with respect to the chord. Because of the finite thickness, tabs were also extended from trailing edge position towards the leading edge, until they blended with the original airfoil contours. Their heights were adjusted so that blending points were at the same longitudinal position considering upper and lower surfaces. By this, smooth aerodynamic shape at the airfoil-tab junction has been achieved for all airfoil modifications; such general approach is usual in composite blade design. It should be noted that in some cases, in order to simplify the production technology, designers form the tabs simply by squeezing the trailing edge domain of the original airfoil (by which Cm_{ac} is also altered). That approach generates grooves at the upper and lower surface tab junction and, as a consequence, the profile drag is slightly increased. Also, such airfoil forms with discontinuities could cause numerical problems in computational analysis, so they have not been considered in this paper.

Finally, the new airfoils with tabs were scaled to the unit chord length. After that, true tab relative chords were: 7.1% for NACA 0012 and NACA 23012; 9.5% for NACA 8-H-12 and 9.3% for Bell/Wortmann FX 69-H-098. For NACA 0012 initial tab angle with respect to the chord was $\theta = 0^o$, because it is symmetrical; the same angle θ was assumed for 23012 and FX 69-H-098, because their mean lines are practically tangential to the chord at the trailing edge. In case of 8-H-12, approximately the middle position of the range, shown in Figure 5, was selected, $\theta =$ -4.27^o. It was obtained by connecting points of the mean line at 95% chord and the trailing edge, extending it for another 5% chord length behind the original airfoil, and setting it to proper thickness. Shape of the initial 8-H-12 geometry with the tab added is shown in figure 6.

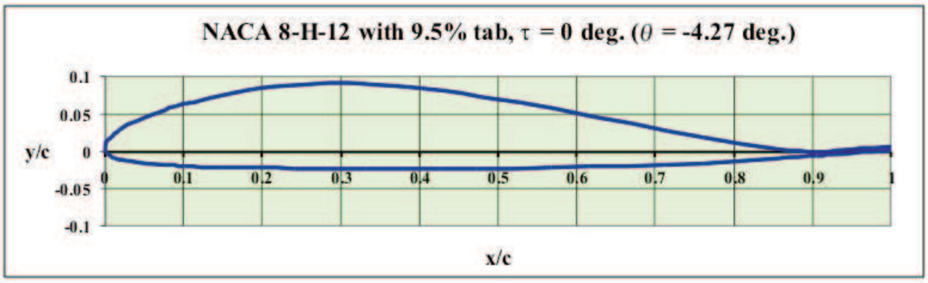

Figure 6: Shape of the NACA 8-H-12 helicopter airfoil in the initial design stage, with the tab added at the assumed reference position $\theta =$ -4.27^o from which the tab static deflections will be analyzed

The next stage was generation of airfoil modifications with tab positions altered from their initial angle θ in 1^o steps, in the selected range of $\tau = \pm$ 4^o (example of 8-H-12 modifications is shown in figure 7). Thus, for each of the four considered airfoils, nine modifications with the tabs were designed and the total of 40 airfoils were prepared for the analysis, including 36 new airfoils with tabs and 4 original airfoils.

After that, the reference airflow conditions were selected. A light helicopter was assumed, in cruising flight at the horizontal speed of 150 km/h. From the aspect of the largest absolute moment in [Nm] about aerodynamic center that can be achieved on rotor disc, the most critical the position is that of an advancing blade at an angle of 90° to the direction of flight. In this case the pitch angles, i.e. local blade angles of attack are small. According to that, for usual reference blade section at 75% of a light helicopter rotor radius, Mach and Reynolds numbers obtained were $M_\infty = 0.5$ and $Re = 2.3 \cdot 10^6$, while the angle of attack range was selected in the domain of $\alpha = 0^o \div 3^o$, with 1^o calculation step.

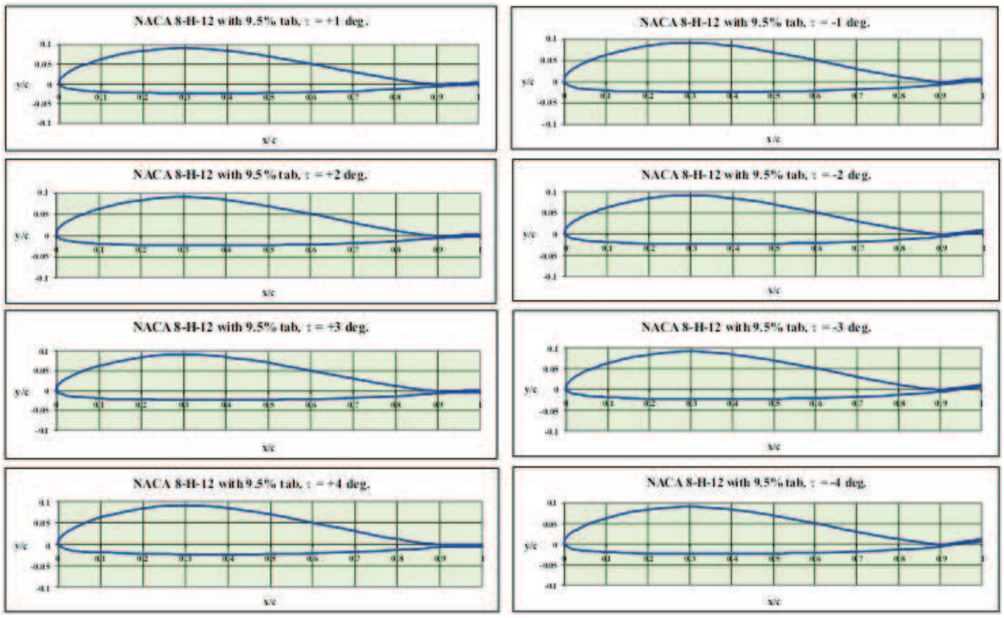

Figure 7: Modifications of the 8-H-12 airfoil with tab, obtained by deflecting the tab in the range of $\tau = \pm\ 4^{\text{o}}$ from the initially assumed $\theta =$ -4.27° reference angular position

As mentioned in the introduction of this paper, during the cruising flight, helicopter blades are exposed to very intensive dynamic motions including pitching, flapping and leading-and-lagging at some $4 \div 7$ cycles per second, causing vibrations that can readily induce an early transition from laminar to turbulent boundary layer. Although this paper represents the first stage of helicopter airfoil design, in which initially only the static airflow conditions are assumed, this fact was taken into account by fixing the transition point at approximately 5% from the

Table 1: Influence of tab position on $\mathrm{Cm_{ac}}$ calculated for 36 airfoil modifications with tabs

Tab deflection τ [deg]	**CONTROL CASE**		**DESIGN CASES**					
	NACA 0012		**NACA 23012**		**NACA 8-H-12**		**Bell/Wortmann FX 69-H-098**	
	$\theta = 0[deg]^*$, $c_t/c = 7.1\%$		$\theta = 0[deg]^*$, $c_t/c = 7.1\%$		$\theta = -4.27[deg]^*$, $c_t/c = 9.5\%$		$\theta = 0[deg]^*$, $c_t/c = 9.3\%$	
	Cm_{ac} calc.	ΔCm_{ac}	Cm_{ac} calc.	ΔCm_{ac}	Cm_{ac} calc.	ΔCm_{ac}	Cm_{ac} calc.	ΔCm_{ac}
-4	0.03859	0.02851	0.04114	0.02859	0.09137	0.04286	0.04444	0.04046
-3	0.03115	0.02107	0.03465	0.02210	0.07792	0.02941	0.03458	0.03060
-2	0.02404	0.01396	0.02790	0.01535	0.06800	0.01949	0.02442	0.02044
-1	0.01710	0.00702	0.01975	0.00720	0.05815	0.00964	0.01455	0.01057
0	0.01008	0.00000	0.01255	0.00000	0.04851	0.00000	0.00398	0.00000
1	0.00289	-0.00719	0.00522	-0.00733	0.03966	-0.00885	-0.00662	0.01060
2	-0.00485	-0.01493	-0.00241	-0.01496	0.02980	-0.01871	-0.01744	-0.02142
3	-0.01303	-0.02311	-0.01049	-0.02304	0.02018	-0.02833	-0.02863	-0.03261
4	-0.02190	-0.03198	-0.01800	-0.03055	0.01048	-0.03803	-0.03968	-0.04366
* - angle from the airfoil chord (negative upward) at which initial tab position $\tau = 0^0$ was assumed								

leading edge. Results obtained by TRANPRO for 36 airfoil derivatives with tabs, with all previously mentioned parameters applied, are shown in Table 1. It should be noted that term "tab deflection" in this paper implies angular differences between fixed tab positions of a certain airfoil modification and the basic modification, on which the tab is placed at an angle θ with respect to the original airfoil chord.

The analysis of the obtained results should be started with considering the "control case" values, obtained for the NACA 0012 airfoil with tab added. For symmetrical airfoils, zero tab deflection θ & $\tau = 0^o$ can be the only correct solution for which $\mathrm{Cm_{ac}} = 0.0$. On the other hand, the calculated value for $\tau = 0^o$ is $\mathrm{Cm_{ac}} = 0.01008$, which actually represents the numerical shift of the results for the given basic airfoil shape and airflow conditions, typical for rectangular grids, as explained in section 3.3.

When tab influence on moment coefficient should be analyzed, one of the obvious ways to eliminate shift in obtained $\mathrm{Cm_{ac}}$ results is to consider the differences between calculated $\mathrm{Cm_{ac}}$-s for certain tab deflections and the reference zero deflection, instead of considering the actual $\mathrm{Cm_{ac}}$ values (in Table 1 " $\mathrm{Cm_{ac}}$ calc."). Such differences, the $\Delta\mathrm{Cm_{ac}}$ values, should be quite "shift free" results because of the similarity of basic airfoil shapes (both tab sizes and deflections are small), which generate almost the same shifts for the same airflow conditions and angles of attack used in the analysis. For symmetrical airfoils, $\Delta \mathrm{Cm}_{\mathrm{ac}}$ for the same tab deflection in positive and negative direction must be exactly the same, only with the opposite sign. Table 1 shows that for NACA 0012 such pairs are very close by the absolute values, but not exactly the same. The largest difference in calculated results, obtained for the highest analyzed tab settings of $\tau = +4^{\circ}$ and -4°, is about 0.0035, and it represents the real order of TRANPRO's effective calculation error when such an approach is applied. This level of error is quite small even for $\mathrm{Cm}_{\mathrm{ac}}$ considerations that are subject of this paper, which shows that obtained results can be trustworthy.

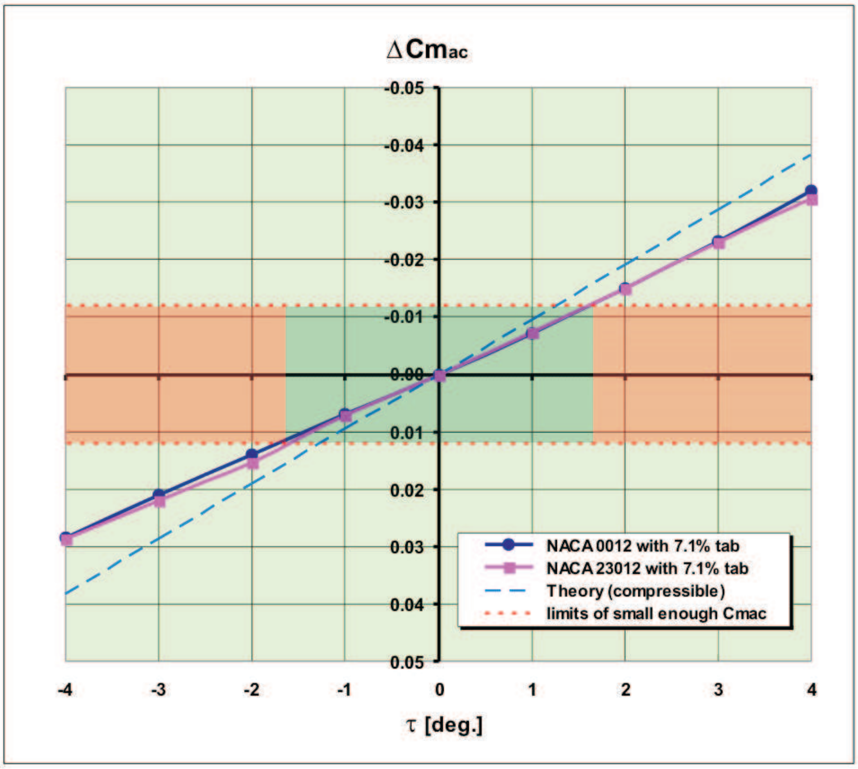

Figure 8: Variation of $\Delta \mathrm{Cm}_{\mathrm{ac}}$ for certain tab deflections from the assumed reference positions, for NACA 0012 (verification case) and NACA 23012 airfoils, both originally 12% thick ($\mathrm{M}_{\infty} = 0.5$ and $\mathrm{Re} = 2.3 \cdot 10^{6}$)

In figures 8 & 9 the TRANPRO's results were compared with the analytical values. They were obtained by compressibility corrected thin airfoil theory [12] for the calculation of $Cm_{1/4}$gradient with plain flap deflection and appropriate relative chord lengths, i.e. 7.1% in Figure 8 and 9.5% in Figure 9, and then applied for the same angles of deflection as in numerical analysis. Theory assumes thin airfoils and inviscid flow, so theoretical gradients are slightly larger than numerically obtained gradients for medium thickness airfoils with boundary layer effects included. Such results are expected, since boundary layers developed on reasonably thick airfoils decrease effectiveness of the thin flat tabs at small angles of deflection. Numerically calculated moment gradient of a thinner airfoil FX 69-H-098 (originally 9.8% thick, before tab was added) is also slightly larger than the gradient of a thicker 8-H-12 (originally 12% thick), for practically the same relative chord, but is still smaller than the thin airfoil theory values (Figure 9).

For each of the considered airfoils there always exists an optimum position of the tab for which moment coefficient about the aerodynamic center is exactly equal to zero. In the starting stage of helicopter airfoil design, in which tab should be added to the basic airfoil, that position is not known in advance, except for symmetrical airfoils. Thus, the angular position of a tab must be initially assumed, and the tolerance level of an error in estimating the optimum angular position of the tab can be evaluated from Figures 8 and 9. In order to do that, it was necessary to establish a limit Cm_{ac} value, below which moment coefficient could be considered small enough and suitable for applications on helicopter rotors. In the existing literature, such an explicit limit can hardly be found, because it may depend to a certain extent on the actual blade's structural and/or rotor control system design. Because of that, the author has applied a statistical approach to estimate this value. Considering the data for some 240 helicopters that have been in production and operational use until present time [1], one of the most widely applied medium thickness asymmetrical airfoils is NACA 23012. Its Cm_{ac} [10] is equal to -0.008 for $Re = 8 \cdot 10^6$ and approximately -0.012 for $Re = 3 \cdot 10^6$ (although theoretically Cm_{ac} should not be influenced by Reynolds number, in practice small variations exist). The latter value is quite appropriate for airflow conditions characteristic for helicopter rotors, and it has been adopted as a limiting value of small

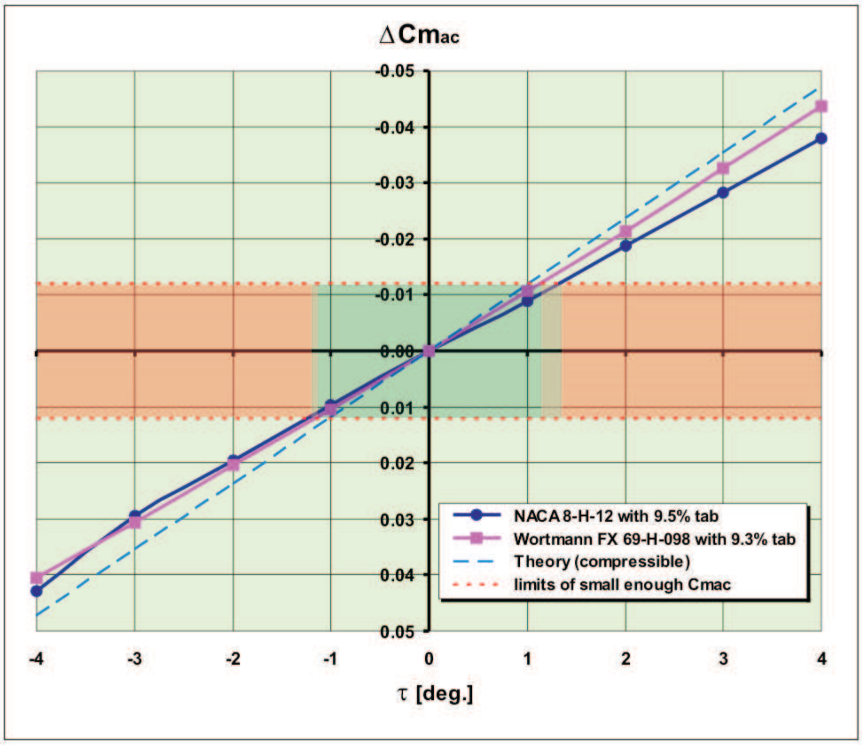

Figure 9: Variation of $\mathrm{Cm_{ac}}$ for certain tab deflections from the assumed reference positions, for 8-H-12, originally 12% thick and FX 69-H-098, originally 9.8% thick airfoils ($\mathrm{M_\infty} = 0.5$ and $\mathrm{Re} = 2.3 \cdot 10^6$)

enough $Cm_{ac} - s$ in this analysis.

Considering graphs in Figures 8 & 9, it is obvious that tolerance limits above and below the optimum tab angular position in initial design estimates are actually very small, and range between $1.2^o \div 1.6^o$ in one or the other direction, depending on the airfoil asymmetry, thickness and relative chord of the applied tab. Since the optimum tab position on asymmetrical airfoils can not be known in advance, there is a large probability that initial designs of wind tunnel test models or prototype blades with tabs, produced without any serious prior analyses, may not prove quite satisfactory in the sense of the moment coefficient.

One of the apparent ways to solve this problem is application of numerical analyses for helicopter airfoils redesigned to contain the tab.

On the other hand, since the Cm_{ac} requirement for helicopter airfoils is very strict, the use of computer programs whose accuracy is quite satisfactory in the analysis of the airplane wing airfoils, might sometimes be unreliable in this particular case. Even contemporary and very sophisticated commercial computer packages with very wide domains of application, often possesses a large variety of adjustable input control parameters that must be tuned and tested very carefully and thoroughly for each particular analysis problem, or otherwise their results might also be questionable in operational design work.

In the next section of this paper, the author has tested the capabilities of TRANPRO to be applied in the delicate task of actual determination of optimum tab position that generates sufficiently small moment coefficient about aerodynamic center and satisfies rotorcraft application requirements.

5 Second stage of airfoil design and examples of numerically optimized helicopter airfoils with tabs

5.1 Redesign of NACA 23012 and 8-H-12 airfoils with tabs fitted

For numerical design of helicopter airfoils to which tabs should be added and optimized to give small moment coefficients about the aerodynamic center, the NACA 23012 and NACA 8-H-12 public domain airfoils have been chosen (the "legal status" of Bell/Wortman airfoil is presently not known to the author, so its new geometry with tab could not be published in this paper anyway). The 8-H-12 has also been very widely used, specially in light helicopter production [1]. Before determination of the optimum tab position, procedure described in section 4 of this paper must be applied, giving the results as shown in Table 1. Taking into account already mentioned characteristics and capabilities of TRANPRO, the author has then introduced and applied the following algorithm:

Experimental values of $\mathrm{Cm}_{\mathrm{ac}}$ for the original airfoils were obtained [10, 11].

Using TRANPRO, Cm_{ac} was calculated for the same original airfoils *without the tab*. The calculated results were obviously different from the experimental values to a certain extent because of their shift, as discussed in section 3.

If the Cm_{ac} value of the original airfoil should have been preserved, the target value of calculated Cm_{ac} for airfoil redesigned to *contain the tab* should be the same as obtained in item (B). On the other hand, it was more appropriate to design the airfoil with tab that was supposed to give true Cm_{ac} exactly equal to zero. For that purpose, target value for the calculated moment coefficient was obtained as the difference of the values defined in (B) and (A), giving (C) = (B) – (A). This approach was based on the fact that the shift of TRANPRO's numerical results from experiment is stable and constant for a given basic airfoil shape and airflow conditions, and the *assumption* that shift of the calculated values for airfoil with the tab would be very close to that of the original airfoil.

For the target value calculated in (C), interpolation of tab position τ that will give same target Cm_{ac} was done.

Using the known initial tab angle θ (Table 1) and deflection τ calculated in (D), the angular position of the tab δ with respect to the chord, for which true Cm_{ac} of airfoil with the tab should be zero, was obtained as $\delta = \theta + \tau$.

This algorithm is established in order to eliminate the influence if any eventual inherent systematic errors, or shifts of the results, and give the airfoils whose true moment coefficient about the aerodynamic center will be very close to zero (Figures 10 (a) & (b)). It can be readily expected that beside the Cm_{ac}, other aerodynamic parameters of the original airfoils will not change very remarkably when tabs are fitted to them (see Table 4). On the other hand, existence of the tabs on rotor blades is practically inevitable when composite technologies are applied, and satisfying low moment coefficient requirement in this case is the primary task during the first stage of the airfoil design and analyses, which assumes "static" airflow conditions applied in presented analysis.

Coordinates of the airfoils shown in Figures 10 (a) & (b) are presented in the Appendix.

Table 2: Second stage of airfoil design – determination of optimum angular position of the tab; prior to that, design work and analyses must be done as described in section 4 and Table1.

NUMERICAL DESIGN OF THE TWO ASYMMETRICAL AIRFOILS WITH TABS ADDED, for required $Cm_{ac} = 0.0$ **(values "**Cm_{ac} **calc." as presented in Table 1, must be calculated first)**	**NACA 23012** $c_t/c = 7.1\%$ $\theta = 0^o$	**NACA 8-H-12** $c_t/c = 9.5\%$ $\theta = -4.27^o$
(A) Experimental Cm_{ac} (airfoil without tab)	- 0.012	0.0050
(B) Calculated Cm_{ac} (airfoil without tab)	0.00570	0.03983
(C) Target Cm_{ac} to achieve true $Cm_{ac} = 0.0$ *with tab* ⇒ (B)-(A)	0.0177	0.03483
(D) Interpolated tab position using Table 1 (Cm_{ac} calc.) to obtain (C)	$\tau = -0.71^o$	$\tau = +1.49^o$
(E) **Calculated tab position from the chord** $\delta = \theta + \tau$ **for zero moment coefficient about the aerodynamic center**	$\delta = -0.71^o$	$\delta = -2.78^o$

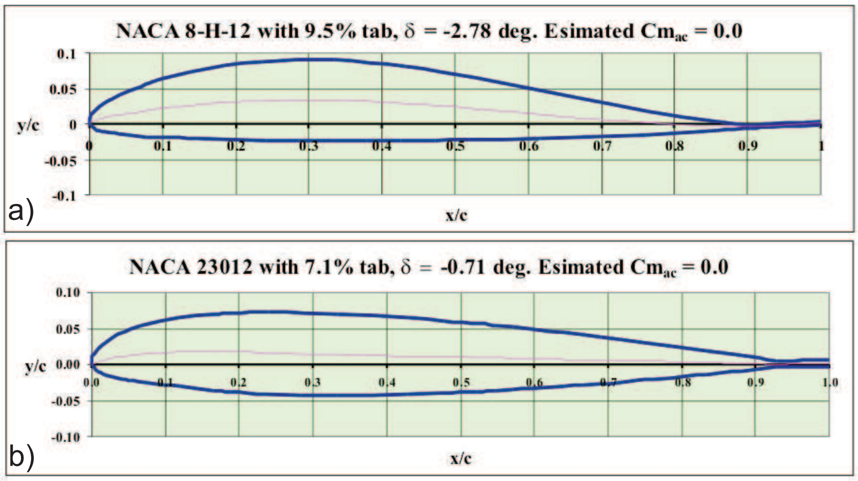

Figure 10: Airfoils fitted with tabs, designed with intention to generate very small moment about the aerodynamic center

5.2 Verification of the presented algorithm

The point which deserves most of the attention in the presented algorithm, contained in item (C), is the assumption that the shift of the calculated results from the true values for airfoils with tabs added will be very close to that of the original airfoils, considering the fact that basic airfoil shapes are quite similar. On the other hand, a small difference in those shifts will cause that the true Cm_{ac} of the airfoils with the tabs added will not be exactly equal to zero, as desired. Determination of the amount of that discrepancy can be used for evaluation the overall accuracy of the presented method.

In order to quantify the amount of expected difference in the calculated and true value of the zero moment coefficient about the aerodynamic center, the same procedure has been applied for the symmetrical airfoil NACA 0012. As already mentioned, the actual optimum tab position for this airfoil is $\delta = 0^o$. Results of this verification case are presented in Table 3 and Figure 11. Obtained result shows the error of 0.43^o in estimated optimum tab angle with respect to the already known true value. It also gives the $\Delta Cm_{ac} = 0.003$, interpolated using column "ΔCm_{ac}" of Table 1 for NACA 0012 airfoil. It should actually

Table 3: Evaluation of here presented algorithm shows that the error in determination of optimum tab position for NACA 0012 airfoil is less than half a degree

VERIFICATION CASE - DESIGN OF A SYMMETRICAL AIRFOIL WITH TAB. Known true tab position angle δ for symmetrical airfoils is $\delta = 0^o$ (values "Cm_{ac} calc." as presented in Table 1, must be calculated first)	NACA 0012 $c_t/c = 7.1\%$, $\theta = 0^o$
(A) Experimental Cm_{ac} (airfoil without tab)	0.000
(B) Calculated Cm_{ac} (airfoil without tab)	0.01305
(C) Target Cm_{ac} to achieve true $Cm_{ac} = 0.0$ with tab $\Rightarrow$ (B)-(A)	0.01305
(D) Interpolated tab position using Table 1 (Cm_{ac} calc.) to obtain (C)	$\tau = -0.43^o$
(E) Calculated tab position from the chord $\delta = \theta + \tau$ for zero moment coefficient about the aerodynamic center	$\delta = -0.43^o$
Numerical error in determination of optimum tab position using presented method	-0.43^o

correspond to the real moment coefficient about the aerodynamic center of this airfoil if a tab of a given geometry would be positioned at an angle of δ = - 0.43°.

Formal application of the presented algorithm, used in Table 3, could have been avoided in case of symmetrical airfoils, if only the divergence of calculated Cm_{ac} from the target true value should have been quantified. The value ΔCm_{ac} = 0.003 could be readily obtained as a difference between the calculated Cm_{ac} for the original airfoil (Table 3, (B)) and for the airfoil fitted with the tab at the already known optimum position (Table 1, $\tau = 0^o$). That difference is actually the difference between the TRANPRO's result shifts for airfoil without the tab and its modification with the tab added. When item (C) of the algorithm is applied, it generates a small error of – 0.43° in tab positioning. Mentioned value of Cm_{ac} is well within the posted limits of sufficiently small moment coefficients, and suggests that the expected true values of Cm_{ac}-s for the other two airfoils, presented in section 5.1, should also be sufficiently small and that the rigorous Cm_{ac} requirement would be satisfied to an extent that further adjustments in tab position in later test stages would most probably not be necessary.

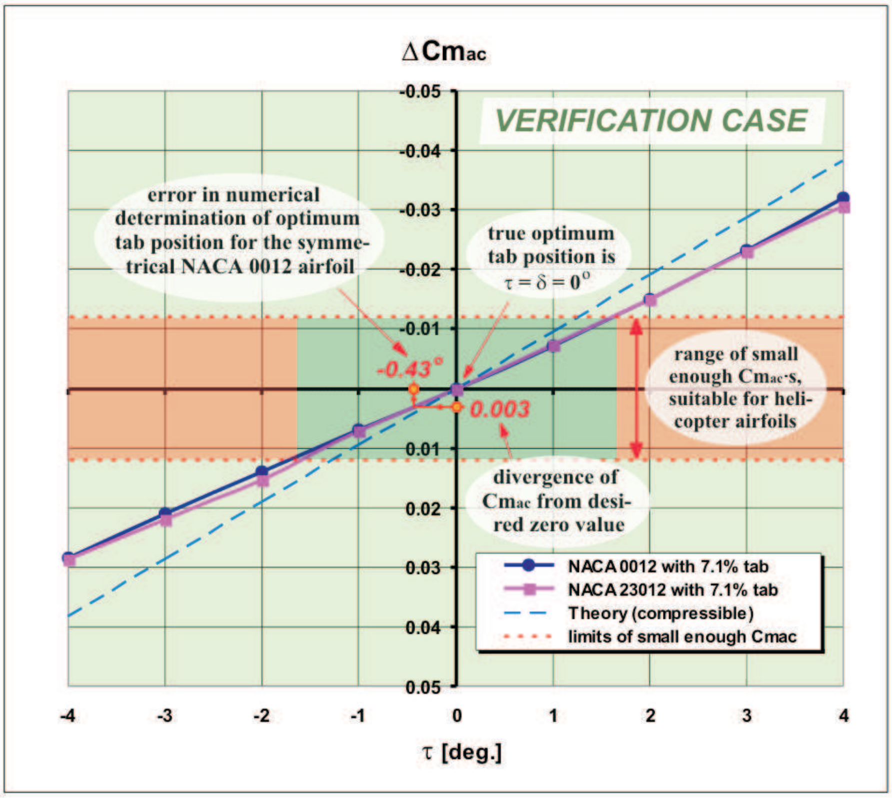

Figure 11: Difference between calculated and true optimum tab position for NACA 0012 airfoil in the verification case is well within the tolerance limits

5.3 Evaluation of the tab influence on drag coefficient

In this section only a brief insight in the tab influence on drag coefficient of the two newly designed airfoils, presented in section 5.1, will be given. Numerical drag tests were done using airflow conditions for which experimental data [10, 11] for leading edge roughness case were available. Results are presented in Table 4. TRANPRO's results for C_D coincide quite well with experiment for NACA 23012 original airfoil, while in case of the original NACA 8-H-12 airfoil, obtained drag coefficients are slightly larger than experimental (experimental polar in around-minimum drag domain shows noticeable irregularities, which is characteristic for many airfoils at small Reynolds numbers). Calculated drag coefficient values obtained for modifications with the tabs are all generally smaller than for the basic airfoils. When considering this fact, it must be remembered that new airfoils were obtained by extending the original ones, and then scaling the new shapes back to the unit chord length. Relative thickness of such airfoils is smaller than of the original, and is approximately 11.2% for both new airfoils (see the Appendix).

This leads to conclusion that the obtained amount of reduction in profile drag must be influenced primarily by the decreased effective relative thickness of the new airfoils, while the influence of tabs on variation of the drag coefficient in this case is practically negligible. This supports the assumption stated in section 5.1 that, beside the Cm_{ac}, other aerodynamic parameters of the original airfoils, such as minimum drag, maximum C_L/C_D ratio etc, will not change remarkably when tabs are fitted to them.

6 Conclusion

In the process of design or redesign of asymmetrical rotorcraft airfoils that should be fitted with tabs for the composite blade production purposes, primary attention must be focused on preservation of nearly zero moment coefficient about the aerodynamic center. In the preliminary analysis stage, the author has fitted tabs to several airfoils operationally used in helicopter blade production, which initially did not have them and then, using computational analysis, quantified the influence of the tab's size and possible angular position ranges on the variation of their Cm_{ac} values. The tolerance limits of Cm_{ac} for rotorcraft airfoils, in this case established using statistical approach, proved to be very narrow. This analysis has shown that, although the usual relative chord lengths of the tabs are proportionally small, the domains of acceptable

Table 4: Review of the experimental drag coefficient values for original airfoils, values calculated for original airfoils and values calculated for airfoils with tabs, whose relative thicknesses are decreased to 11.2% .

DRAG COEFFICIENT RESULTS	NACA 23012		NACA 8-H-12	
Airflow parameters used both in experiment and in numerical analysis	$M = 0.2$, $Re = 6 \cdot 10^6$		$M = 0.2$, $Re = 2.6 \cdot 10^6$	
Nominal angles of attack, used in numerical analysis	$\alpha = 0^o$	$\alpha = 3^o$	$\alpha = 0^o$	$\alpha = 3^o$
Calculated and experimental lift coefficient C_L of the original airfoil	0.1287	0.4558	0.0407	0.3667
Experimental drag coefficient C_D of the original airfoil; leading edge roughness case (boundary layer transition forced close to the leading edge)	0.0099 ref.[10]	0.0104 ref.[10]	0.0100 ref.[11]	0.0112 ref.[11]
Calculated drag coefficient C_D of the *original* airfoil	**0.00959**	**0.01003**	**0.01144**	**0.01187**
Calculated lift coefficient C_L of the airfoil with the tab added	**0.0359**	**0.3881**	**0.0463**	**0.3913**
Calculated drag coefficient C_D of the airfoil *with the tab*	**0.00899**	**0.00902**	**0.01079**	**0.01118**

tab angular positions of only some $\pm$ 1.2 $\div$ 1.5^{o} around the optimum one. Since the optimum angular tab position, that will give exactly zero Cm_{ac} value, can not be known in advance for asymmetrical airfoils, these results have also proven that tab positions in such kind of airfoil design should by all means be first predefined by numerical analysis, involving special attention and care, before undertaking any further more expensive and time consuming design stages.

The obtained results also imply that the influence of eventual inherent numerical errors of applied software on the results must be reduced to an absolute minimum. For that purpose, in this paper an original approach has been established and applied using the author's computer program TRANPRO for redesign of the several asymmetrical airfoils. This program has many advantages over the source NASA-Trandes code, from which it has been derived, although both programs use rectangular grids for airflow calculations. Such grids are very favorable from the aspect of their generation, but sometimes they are prone to induce the "shift of the results" systematic numerical error. In the existing publications, ways to eliminate it for C_L and C_D coefficients are well known. On the other hand, for Cm_{ac} calculations, this problem becomes much more complex. Here presented original algorithm has been established in a way that eventual "shift" errors in subsequent steps practically cancel out, and the final results contain a very small inevitable numerical error, which to a certain extent exists in all kinds of software nowadays. Using this algorithm, the two asymmetrical rotorcraft airfoils have been fitted with the tabs with the optimum positions determined. Then, the same algorithm has been applied and verified on a symmetrical airfoil (control case), for which the optimum tab position is readily known, proving that the presented approach can give very useful and reliable results. Although here applied on a particular software, the general concept of presented approach can be used with any other computer program in order to achieve the highest possible accuracy, required in helicopter airfoil design.

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Appendix

Coordinates of the two airfoils suitable for application on composite helicopter rotor blades, with the tabs added using algorithm described in sections 4 and 5.1 of this paper and presented in figures 10 (a) & 10 (b).

Submitted on November 2004, revised on January 2005.

NACA 23012 with 7.1% tab				**NACA 8-H-12 with 9.5% tab**			
X_{upper}	Y_{upper}	X_{lower}	Y_{lower}	X_{upper}	Y_{upper}	X_{lower}	Y_{lower}
0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000
-0.00022	0.00771	0.00403	-0.00656	0.00140	0.01170	0.00812	-0.00780
0.00153	0.01259	0.00799	-0.00976	0.00341	0.01448	0.01088	-0.00901
0.00727	0.02083	0.01654	-0.01404	0.00766	0.01910	0.01615	-0.01074
0.01824	0.03061	0.02938	-0.01793	0.01886	0.02801	0.02876	-0.01348
0.02999	0.03819	0.04144	-0.02046	0.04213	0.04107	0.05310	-0.01653
0.04216	0.04441	0.05308	-0.02241	0.06585	0.05124	0.07701	-0.01829
0.06715	0.05397	0.07571	-0.02556	0.08978	0.05965	0.10070	-0.01961
0.09249	0.06071	0.09799	-0.02831	0.13807	0.07263	0.14765	-0.02135
0.11780	0.06536	0.12029	-0.03091	0.18673	0.08195	0.19422	-0.02239
0.14286	0.06842	0.14285	-0.03340	0.23575	0.08803	0.24044	-0.02302
0.16752	0.07033	0.16582	-0.03573	0.28542	0.09079	0.28601	-0.02338
0.19168	0.07146	0.18927	-0.03780	0.33499	0.08983	0.33168	-0.02371
0.23934	0.07234	0.23685	-0.04080	0.38373	0.08600	0.37817	-0.02375
0.28698	0.07187	0.28445	-0.04242	0.43200	0.08019	0.42514	-0.02358
0.33458	0.07031	0.33208	-0.04297	0.47990	0.07301	0.47248	-0.02320
0.38217	0.06787	0.37973	-0.04263	0.52750	0.06471	0.52012	-0.02264
0.42974	0.06470	0.42740	-0.04157	0.57484	0.05568	0.56802	-0.02181
0.47730	0.06092	0.47508	-0.03989	0.62201	0.04619	0.61609	-0.02074
0.52485	0.05662	0.52277	-0.03769	0.66905	0.03655	0.66429	-0.01937
0.54862	0.05429	0.54662	-0.03642	0.71604	0.02703	0.71253	-0.01771
0.57239	0.05186	0.57047	-0.03504	0.76303	0.01805	0.76078	-0.01567
0.59615	0.04933	0.59432	-0.03356	0.81010	0.00996	0.80895	-0.01318
0.61992	0.04671	0.61818	-0.03199	0.85730	0.00327	0.85699	-0.01001
0.64368	0.04400	0.64204	-0.03032	0.90471	-0.00113	0.90481	-0.00599
0.66744	0.04120	0.66590	-0.02858	0.92857	0.00003	0.92857	-0.00483
0.69119	0.03831	0.68976	-0.02674	0.95238	0.00119	0.95238	-0.00367
0.71495	0.03535	0.71362	-0.02483	0.97619	0.00235	0.97619	-0.00251
0.73870	0.03231	0.73749	-0.02284	1.00000	0.00352	1.00000	-0.00135
0.76246	0.02918	0.76135	-0.02077				
0.78621	0.02598	0.78522	-0.01862				
0.80996	0.02270	0.80909	-0.01639				
0.83370	0.01933	0.83296	-0.01408				
0.85745	0.01589	0.85684	-0.01168				
0.88119	0.01235	0.88071	-0.00920				
0.90493	0.00873	0.90459	-0.00663				
0.92867	0.00501	0.92847	-0.00396				
0.95238	0.00531	0.95238	-0.00367				
0.97619	0.00560	0.97619	-0.00337				
1.00000	0.00590	1.00000	-0.00308				

Neki praktični aspekti u kompjuterskom projektovanju aeroprofila za lopatice glavnih rotora helikoptera

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Jedan od veoma važnih zahteva koji aeroprofili rotora helikoptera treba da ispune jeste da koeficijent momenta oko aerodinamickog centra mora približno biti jednak nuli. Za razliku od starijih tehnologija, korišćenih u proizvodnji metalnih lopatica, savremene izvedbe koje se baziruju na primeni plastičnih kompozita zahtevaju da se na izlaznoj ivici doda ravan repić, čime se menja izvorni oblik aeroprofila. Uz pomoc kimpjuterskog programa TRANPRO, autor je razvio i verifikavao algoritam za ovu fazu projektovanja, namenjen promeni na nesimetričnim aeroprofilima sa srednjom linijom u obliku latiničnog slova "S", kvantifikavao globalni uticaj ugaonog polozaja repića na promenu momenta oko aerodinamičkog centra, a zatim reprojektovao nekoliko aeroprofila kojima je dodao repić na takav način da je zahtev za malom vrednošću ovog koeficijenta momenta u potpunosti ispunjen.