APPLICATION OF AN IMPROVED INTEGRAL TURBULENT BOUNDARY LAYER MODEL WITH MODERATE SEPARATION CORRECTION ON *NACA* SERIES AND LOW SPEED SUPERCRITICAL AIRFOIL DERIVATIVES

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Abstract

The paper presents latest results in the development of a method for two-dimensional integral turbulent boundary layer analysis with moderate separation correction on general aviation airfoils. Initially it was based on a NASA's computer program and some empirical results, which gave general suggestions of how to solve such category of problems, but not with quite acceptable level of accuracy, specially in the case of lower speed wing section analyses. After introducing some minor, but relevant changes in the basic turbulent boundary layer model and some more remarkable changes in the moderate separation correction part of the analysis, an accurate algorithm for a large number of airfoils of 12-18% relative thickness was obtained. These results have covered range of lift coefficients that might be encountered on airfoils in cruising flight conditions and moderate maneuvers.

Results obtained by the modified numerical model have been proved by comparison with NACA/NASA experimental data for the analysed airfoils. So far, the modified model has been confined to lower Mach number verification with the data for existing airfoils, and the introduced algorithms are connected with particular categories and subcategories of airfoils. The aim is to spread its use to newly designed airfoils.

Symbols

C_L airfoil lift coefficient
C_D airfoil profile drag coefficient
Cp pressure coefficient
Cp_{base} base pressure coefficient used in separation correction model
Cpl_{max} maximum value of pressure coefficient on the lower surface of the airfoil in the vicinity of the trailing edge
α angle of attack
α_i angle of attack at which the separation correction is initialised
Cp_{sep} pressure coefficient at separation point
$Xsep_i$ separation point position corresponding the α_i angle of attack
$Xsep(\alpha)$ separation point position for angles of attack higher than α_i
$CCp(Xsep, \alpha)$ base pressure coefficient correction factor
$\Delta CCp(Xsep, \alpha)$ base pressure coefficient correction factor gradient
$CCp(Xsep_i, \alpha_i)$ base pressure coefficient correction factor at α_i angle of attack
α_i^* non-standard α_i, used in cases when transition from attached flow calculation to modified separation model can not be done through α_i that belongs to introduced separation correction algorithm
$Xsep_i^*$ separation point position corresponding the α_i^* angle of attack
CCp^* base pressure correction factor at α_i^*
δ^* boundary layer displacement thickness
θ boundary layer momentum thickness
H boundary layer shape factor
M_e local Mach number outside the t.b.l.
u_e local velocity outside the t.b.l.
ρ_e local density outside the t.b.l.

Introduction

This paper describes the results of an investigation aimed to modify and improve the turbulent boundary layer model such as the one applied in computer program *Trandes*[1] developed for NASA, widely and quite successfully used in transonic airfoil analysis and design. Unfortunately, when applied at lower subsonic speeds and on airfoils of higher relative thickness such as GA(W)-1, it misses experimental results for drag coefficient of the order of 7-10%, which is unacceptable even for preliminary engineering purposes. An attempt to spread the successful use of this program to lower Mach and moderate Reynolds numbers by improving its boundary layer model was quite a challenge.

The key problem was that specific rectangular grid used in this program enforced the application of an integral model for boundary layer calculations, with adjacent approximations and averaging as possible sources of the uncertainty of the results. The flow over the displacement surface is calculated by full potential model. The boundary layer transition was fixed to 6% chord, corresponding to 'standard roughness' conditions, and momentum integral equation is solved for the momentum thickness using the equations of Nash and Macdonald for skin friction and shape factor. These equations contain correction factors for compressibility effects on boundary layer, but their influence at small Mach numbers becomes negligible and they can not be the cause of any mistakes. The boundary layer displacement thickness is smoothed everywhere and added to the airfoil ordinates to obtain the body boundary conditions for potential flow. For separated flow at higher angles of attack, it is suggested that from the point of separation on to the trailing edge an inverse approach is to be used. The initial value of displacement thickness should be determined by increasing the Cp distribution linearly from Cp_{sep} at separation point up to the value Cp_{base} at the trailing edge (which for the aft-cambered airfoils corresponds to Cp_{max} on the rear part of the lower camber, according to the original model) and then the real displacement surface and Cp distribution in separated region is calculated during the iteration process.

Original model, in case of the analyses mentioned above, in the minimum drag region misses the experimental polar quite remarkably. When an attempt is made to calculate the region of moderate separation (as originally suggested), it is not possible to join the part of the polar without correction smoothly, and beside that, it corresponds to the experiment in a very small domain.

Many attempts to improve the results, in the beginning, only seemed to make them worse. The turbulent boundary layer model was very sensitive to any changes, producing 'oscillating' polars and lift curves. Finally, by simply altering the last constant in the classic equation of Nash that relates Clauser parameters G and βp for nonequilibrium boundary layer to a certain higher value (the idea which in the beginning seemed like a blaspheme) everything came to the proper place. Besides on GA(W)-1 airfoil, this change gave good adjustment of polar curves also on NACA 4 and 5 digit series airfoils for standard roughness conditions.

Keeping this change, the domain of moderate separation was tested including original correction model. The new troubles appeared, until a certain relation between the position of separation Xsep and fractions of Cpbase used for different angles of attack was established for certain categories and subcategories of airfoils. These results gave unique solutions for Cl and Cd that properly and smoothly matched the experimental curves. The mentioned relation between Cpbase fraction and Xsep is a very simple function both in case of GA(W)-1 airfoil and NACA 4 and 5 digit series of airfoils. Keeping this logic, good polars were calculated for some GAW airfoil derivatives designed at the Belgrade Faculty of Mechanical Engineering[8,9].

Description of the method

The method that will be described is developed from the Nash-Macdonald turbulent boundary layer model with included moderate separation correction such as the one applied in *Trandes* computer program. This program applies the full potential airflow calculation around the airfoil both in inverse design and airfoil analysis cases on series of specifically designed rectangular grids which enable very quick and efficient convergence of solution. The viscous interaction effects can be included as an optional mode while in that case the airflow is calculated about the surface that is obtained by adding the calculated local displacement thickness to the airfoil coordinates. Although nowadays there are more profound methods for calculation of airflow, like those based on Navier-Stokes equations, this program category has still kept its firm positions in many aerospace companies and institutions, although in many cases in some upgraded forms, since it requires very short computer run-time and gives good results. (Such a program, named *Tranpro*[2], was developed at Belgrade Faculty of Mechanical Engineering from *Trandes,* but with completely automated trailing edge closure and some additional airfoil design input options. Results that will be presented in the paper were obtained by *Tranpro*'s recently upgraded turbulent boundary layer model calculations).

Original turbulent boundary layer model

In brief, the original boundary layer model used in *Trandes* and initially in *Tranpro* computer programs is based on calculation of the distribution of displacement thickness δ^* on upper and lower camber of the airfoil by solving the momentum integral equation:

$$\frac{d\theta}{dx}+(H+2-M_e^2)\frac{\theta}{u_e}\frac{du_e}{dx}=\frac{\tau_w}{\rho_e u_e}=\frac{1}{\zeta^2} \qquad (1)$$

for the momentum thickness θ. In equation (1) subscript 'e' denotes the local values of Mach number, density and velocity just outside the boundary layer, while $H=\delta^*/\theta$ is the shape factor. Parameter ζ from the same equation is calculated as:

$$\zeta = f1(M_e)\{2.4711 \quad \ln[f2(M_e,u_e,\theta]+4.75\}+$$

$$+1.5G+\frac{1724}{G^2+200}-16.87 \qquad (2)$$

where f1 and f2 are known functions. Clauser parameters G and β_p are defined by equations (3) and (4):

$$G=\frac{H-1}{H}\zeta \quad (3)\ ; \qquad \beta_p=-H\zeta^2\frac{\theta}{u_e}\frac{du_e}{dx} \quad (4)$$

and related by the empirical equation of Nash:

$$G = 6.1\ (\beta_p + 1.81)^{1/2} - 1.7 \qquad (5)$$

For calculated values of G and ζ, shape factor H is obtained from equation (3), and corrected for compressibility effects (correction is relevant only for higher subsonic Mach numbers). Then from the value of $(d\theta/dx)$ calculated by equation (1), θ and δ^* are obtained. This computation is done on the same grid spacing as for the inviscid part of calculation. The local values of velocity over the displacement surface are calculated in the relaxation cycles converging to final values simultaneously with the boundary layer solution.

The calculated displacement surface is then smoothed(1,2) over the upper and lower airfoil camber and extrapolated at the trailing edge. After that values of θ for upper and lower camber trailing edge are recalculated. Final solution of airflow is obtained using the smoothed displacement surface coordinates. Airfoil profile drag is calculated by the Squire-Young formula using final values of θ at the trailing edge.

Original separated flow correction

Previously described boundary layer calculation can be used in case of attached flow over the whole surface of the airfoil, i.e. in a narrow range of angles of attack corresponding to the minimum drag coefficient Cd_{min}. If the angle of attack is increased beyond that range, separation occurs and in the separated region boundary layer calculation can not be applied. Still, in case of not to strong separation it is possible to prescribe some additional, in most cases purely empirical equations, which could lead to satisfactory solutions when combined with partly restricted boundary layer equations in the domain of separated flow. Strictly theoretically speaking, such an approach might not be exactly correct, but for engineering purposes it could give quite satisfactory results.

Such a moderate separation model was suggested in *Trandes* program. At the moment the program was issued, this model was suggested for use with caution, since not too much experience with its use had been gained.

The suggested model was based on the following:

(a) Since the turbulent boundary layer model can not be applied in the separated flow region, an additional relation must be introduced to 'close' the calculation cycles, by prescribing the initial values of displacement thickness in this case. Some experience showed that by *setting a new, linear* Cp *distribution from* Cp_{sep} *at* Xsep *to the value* Cp_{base} *at the trailing edge* on the upper airfoil camber can 'artificially thicken' the initial δ^* values, which after the iteration process is completed may quite well converge to the real distribution (if Xsep position has been correctly chosen). Such Cp distribution is only used to initialize the δ^* values and it by no means is an attempt to give a 'smart guess' of what Cp should be like in that region.

(b) Base pressure coefficient Cp_{base} *has a typical value of* Cp_{base} = 0.6 (approximately) *for classical airfoils while for aft-cambered airfoils it should be set to* $Cp_{base} = Cpl_{max}$ where this value represents the maximum Cp value close to the trailing edge at the lower surface of the airfoil.

(c) Not only at separation point, but in the whole region τ_w is set to zero in order to retain numerical consistency of the model (since in separated region it is negative). In that case, ζ and also β_p become infinite. Instead of that, β_p *is set to high enough value of 10000,* and other parameters calculated based on this value.

(d) The equations of the *Nash-Macdonald model with mentioned restrictions as for the boundary layer are applied in the separated region.*

(e) So calculated displacement thickness distribution is used only to determine the *slope of the displacement surface at* Xsep. *This value is then used for extrapolation of new* δ^* *distribution which is linear from* Xsep *to the trailing edge.*

Both in the calculation of separated flow and the turbulent boundary layer calculation, a certain sensitivity to grid spacing is reported. Since the computer program uses a specific series of rectangular grids with constant spacing in x - direction in the domain of the airfoil, some uncertainties may occur in the region of leading and trailing edge, where closer grid spacing in some cases might be necessary.

Modified turbulent boundary layer model

In order to satisfy some project requirements, a computer program *Tanpro* was used on PC486 category computers for the analyses of airfoils for general aviation use, 12 - 18% thick, at low Mach and moderate Reynolds numbers. (Initially, some minor adjustments of computer listing were done in order to match the results of *Trandes* test samples in the domain of transonic analysis and design, due to the different processor types of computers on which *Trandes* was originally run). *Tranpro* computer program in the beginning had the same viscous interaction model as *Trandes*, described in the two previous sections. In order to confirm the solutions for the mentioned airflow domain, results for some usual airfoils with transition fixed at 6% chord were compared with the experimental data. Surprisingly, quite remarkable discrepancy with the experiment was observed (figures 7. - 15.)

In all cases, program results gave drag coefficients 7 - 10% lower, and lift coefficients slightly higher than the experiment. Since drag coefficient was obtained through the boundary layer calculation, many steps were taken to detect the source of the problem there.

Table 1.

Variation of Nash parameters (*GA(W)-1 test example*): α = 0 deg. MRe=6.3 M=0.15 NASA stand.rough. experimental - target values: C_D =0.0109, C_L = 0.48		
Variation of A_N		
A_N	C_L	C_D
6.1	*0.4997*	*0.00940*
5.6	0.4963	0.00978
5.1	0.4974	0.01019
4.6	0.5078	0.01068
4.1	0.5216	0.01122
Variation of B_N		
B_N	C_L	C_D
1.81	*0.4997*	*0.00940*
1.61	0.4976	0.00974
1.31	0.4941	0.01027
0.81	0.4916	0.01134
0.61	0.4911	0.01195
Variation of C_N		
C_N	C_L	C_D
1.7	*0.4997*	*0.00940*
1.9	0.4989	0.00956
2.7	0.4952	0.01002
3.7	0.4903	0.01068
5.7	0.4794	0.01225

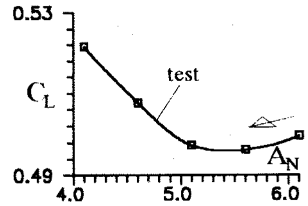


Figure 1.

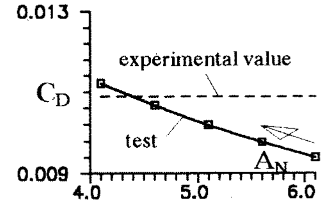


Figure 2.

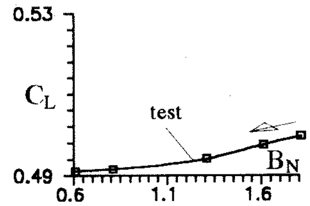
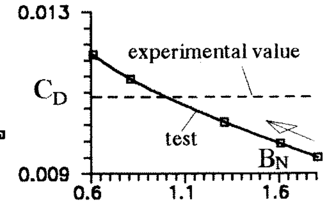


Figure 3.



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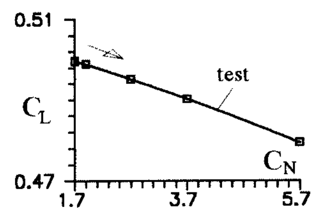


Figure 5.

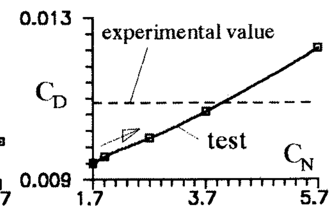


Figure 6.

The integral boundary layer model does not give any insight in the internal boundary layer physicality, so the consequences of variation of some empirical relations used in the model could not have been examined directly. On the other hand, improper variation of some of them, as an example, caused 'oscillating' polars with the change of α, which meant that physical consistency of the model was disturbed, and such modifications were abandoned.

At last, when analysis was focused to the equation of Nash *(5)*, some promising results appeared. The equation was rewritten in the form:

$$G = A_N (\beta_p + B_N)^{1/2} - C_N \qquad (6)$$

and the introduced coefficients were varied separately for the fixed values of α approximately corresponding C_{Dmin} for various airfoils. Example of such an analysis is given in Table 1. and on figures 1. - 6. for GA(W)-1 airfoil (in case of the other tested airfoils conclusions are more or less the same).

From data presented in Table 1. the following is obvious:

- decreasing of only A_N (figures 1. & 2.) causes uniform increase of drag coefficient, but non-uniform behavior of lift coefficient (first decreases, than gradually increases), which eliminated this parameter from further consideration;

- decreasing of only B_N (figures 3. & 4.) causes an uniform increase of drag coefficient and uniform decrease of lift coefficient at the same time, which is a desired tendency;

- increasing of only C_N (figures 5. & 6.) also causes an uniform increase of drag coefficient and uniform decrease of lift coefficient at the same time.

Both B_N and C_N variations can reach desired values of drag coefficient. Beside that, C_N gives good match of lift coefficient, very 'stable' polar curve and good agreement with experiment in the range of α in which turbulent boundary layer model without separation correction may be applied, so C_N was the final choice for further analyses. In case of a wide number of tested airfoils including GA(W)-1, the value of C_N = 4.1 proved to give the best results (see C_{Dmin} region of the polars of airfoils at figures 7. to 15.). Thus for the modified equation of Nash the following form was accepted:

$$G = 6.1\,(\beta_p + 1.81)^{1/2} - 4.1 \tag{7}$$

In this paper no further efforts will be involved in trying to explain the physical background of the introduced change. The explanation that 'it simply works well' in practice may be enough.

Table 2.

Airfoil designation	α_i^* deg..	$Xsep_i^*$	CCp^*	α_i deg.	$Xsep_i$	$CCp(Xsep,\alpha)$	$Xsep(\alpha)$
1	2	3	4	5	6	7	8
NACA 2412	/	/	/	2	0.90	$1.45 + 0.25(\alpha - 2)$	$0.90 - 0.03(\alpha - 2)$
NACA 2415	/	/	/	2	0.90	$1.10 + 0.25(\alpha - 2)$	$0.90 - 0.03(\alpha - 2)$
NACA 4412	/	/	/	2	0.90	$0.90 + 0.15(\alpha - 2)$	$0.90 - 0.03(\alpha - 2)$
NACA 4415	/	/	/	2	0.90	$0.95 + 0.15(\alpha - 2)$	$0.90 - 0.03(\alpha - 2)$
NACA 4418	/	/	/	2	0.90	$1.00 + 0.15(\alpha - 2)$	$0.90 - 0.03(\alpha - 2)$
NACA 63_2 - 215	/	/	/	3	0.90	$1.10 + 0.15(\alpha - 3)$	$0.90 - 0.03(\alpha - 3)$
NACA 63_3- 218	/	/	/	3	0.90	$1.10 + 0.15(\alpha - 3)$	$0.90 - 0.03(\alpha - 3)$
NACA 64_2 - 215	/	/	/	3	0.90	$1.10 + 0.15(\alpha - 3)$	$0.90 - 0.03(\alpha - 3)$
NACA 64_3 - 218	/	/	/	3	0.90	$1.10 + 0.15(\alpha - 3)$	$0.90 - 0.03(\alpha - 3)$
NACA 65_2 - 215	/	/	/	3	0.90	$1.10 + 0.15(\alpha - 3)$	$0.90 - 0.03(\alpha - 3)$
NACA 65_3 - 218	/	/	/	3	0.90	$1.10 + 0.15(\alpha - 3)$	$0.90 - 0.03(\alpha - 3)$
NACA 63_1 - 412	2	0.90	0.85	3	0.90	$1.10 + 0.10(\alpha - 3)$	$0.90 - 0.03(\alpha - 3)$
NACA 63_2- 415	2	0.90	0.85	3	0.90	$1.10 + 0.10(\alpha - 3)$	$0.90 - 0.03(\alpha - 3)$
NACA 63_3 - 418	2	0.90	0.85	3	0.90	$1.10 + 0.10(\alpha - 3)$	$0.90 - 0.03(\alpha - 3)$
NACA 64_1 - 412	2	0.90	0.85	3	0.90	$1.10 + 0.10(\alpha - 3)$	$0.90 - 0.03(\alpha - 3)$
NACA 64_2 - 415	2	0.90	0.85	3	0.90	$1.10 + 0.10(\alpha - 3)$	$0.90 - 0.03(\alpha - 3)$
NACA 64_3 - 418	2	0.90	0.85	3	0.90	$1.10 + 0.10(\alpha - 3)$	$0.90 - 0.03(\alpha - 3)$
NACA 65_1 - 412	2	0.90	0.85	3	0.90	$1.10 + 0.10(\alpha - 3)$	$0.90 - 0.03(\alpha - 3)$
NACA 65_2 - 415	2	0.90	0.85	3	0.90	$1.10 + 0.10(\alpha - 3)$	$0.90 - 0.03(\alpha - 3)$
NACA 65_3 - 418	2	0.90	0.85	3	0.90	$1.10 + 0.10(\alpha - 3)$	$0.90 - 0.03(\alpha - 3)$
NACA 63_2 - 615	/	/	/	3	0.90	$0.65 + 0.15(\alpha - 3)$	$0.90 - 0.03(\alpha - 3)$
NACA 63_3 - 618	/	/	/	3	0.90	$0.75 + 0.15(\alpha - 3)$	$0.90 - 0.03(\alpha - 3)$
NACA 65_3 - 618	/	/	/	1	0.90	$0.60 + 0.15(\alpha - 1)$	$0.90 - 0.03(\alpha - 1)$
GA(W)-1	/	/	/	5	0.93	$0.555 + 0.075(\alpha - 5)$	$0.93 - 0.03(\alpha - 5)$
NACA 23012	/	/	/	3	0.96	$1.4 + 0.30(\alpha - 3)$	0.96 = const.

Modified separated flow correction

In the next step, the original separated flow correction was applied with the modified boundary layer model in the analyses of airfoils at higher angles of attack. Many tests were done, and the following two main problems appeared: (1) the part of the polar with the applied correction could not join the part of the polar without correction smoothly (in many cases it could not join it at all), and (2) the part of the polar with the applied correction matched the experimental polar curve in a very small domain, actually, it just crossed it at a certain C_L value.

After many introduced changes in order to make a model that will work well in a range of angles of attack where moderate separation occurs, an algorithm which gave very good results has been derived. This model is based on the following:

(A) Recommendations *(a) and (c) of the original model should be accepted completely.*

(B) Instead of the original equation of Nash, in the restricted boundary layer model equations set, *the modified equation* (7) *should be used.*

(C) For categories and/or subcategories of airfoils, the *angle of attack* α_i from which moderate separation correction is to be introduced, should be defined (Table 2.)

(D) For categories and/or subcategories of airfoils, the *relative position of initial separation point* $Xsep_i$ corresponding the α_i angle of attack should be defined (Table 2.)

(E) *Equation which defines the separation point position* Xsep for angles of attack α beyond α_i, for categories and/or subcategories of airfoils should be defined. For all tested airfoils (except the NACA 230xx category) it is given by the relation:

$$Xsep(\alpha) = Xsep_i - 0.03 \cdot (\alpha - \alpha_i) \qquad (8)$$

(F) *Base pressure coefficient* Cp_{base} *must be defined* as a function of Cpl_{max}:

$$Cp_{base} = CCp(Xsep,\alpha) \cdot Cpl_{max} \qquad (9)$$

where Cpl_{max} represents the maximum Cp value close to the trailing edge at the lower surface of the airfoil, and $CCp(Xsep,\alpha)$ is the base pressure coefficient correction factor.

(G) *Base pressure coefficient correction factor* $CCp(Xsep,\alpha)$ *must be determined* for categories and/or subcategories of airfoils (Table 2.) as a function of the angle of attack or the separation point relative position:

$$CCp(Xsep,\alpha) = CCp(Xsep_i,\alpha_i) + $$
$$+ \Delta CCp(Xsep) \cdot (Xsep - Xsep_i) \qquad (10.a)$$

or

$$CCp(Xsep,\alpha) = CCp(Xsep_i,\alpha_i) + $$
$$+ \Delta CCp(\alpha) \cdot (\alpha - \alpha_i) \qquad (10.b)$$

where $\Delta CCp(Xsep)$ or $\Delta CCp(\alpha)$ represent the base pressure coefficient correction factor gradient, and $CCp(Xsep_i,\alpha_i)$ base pressure coefficient correction factor at α_i angle of attack.

(H) In case of some airfoils, transition from attached flow calculation to modified separation model can not be done through α_i that belongs to introduced separation correction algorithm. In that case, the *non-standard values of* α_i*, $Xsep_i$* and CCp* *must be determined* (Table 2.).

(I) Values of δ^* and θ calculated by presented modified separated flow correction and smoothed should be accepted as final (i.e. recommendation '(e)' of the original model must not be applied).

Described model when separated correction must be applied gives very good results compared with the experimental data (results for higher C_L-s, figures 7. to 15.). All parameters needed for its application which are different from the original model are given in Table 2.

Presentation of the results

Results obtained by the modified model and compared with the results of the original model and experiment are given in figures 7. - 15. At high C_L-s slight divergence of the modified model results from the experiment announces that moderate separation changes to massive flow separation, which can not be treated by the models based on momentum integral equation (1) and separated flow corrections, described in this paper.

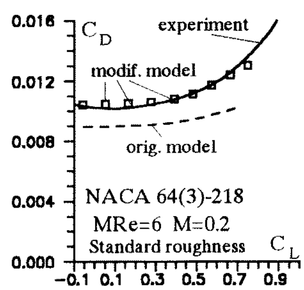
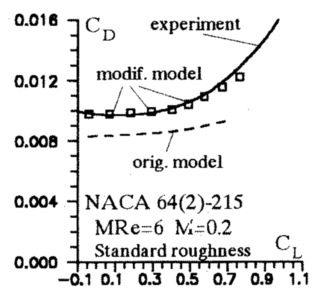


Figure 7.



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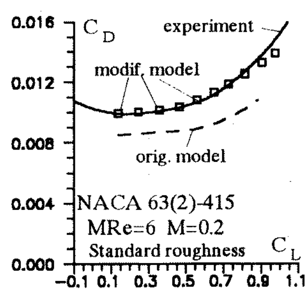

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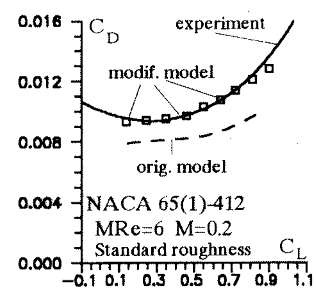

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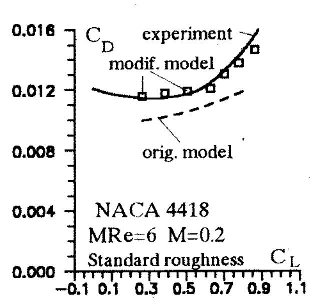

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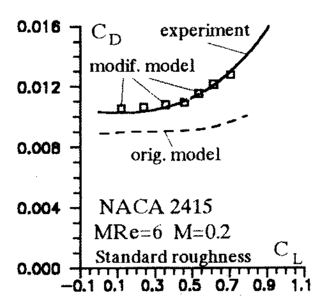

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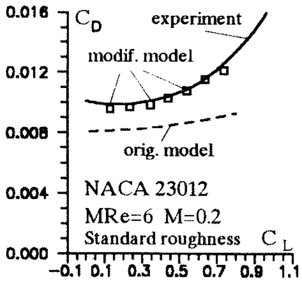

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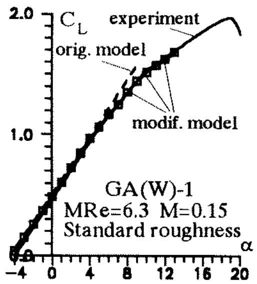

Figure 14.

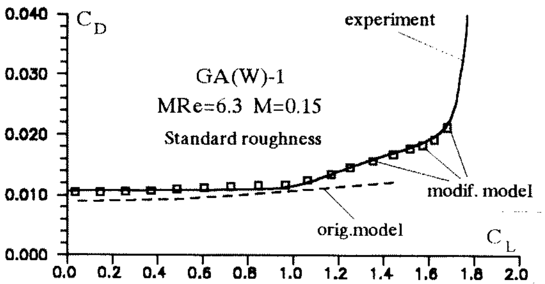

Figure 15.

All numerical results are obtained with transition fixed to 6% chord. Experimental data are with the standard roughness included, with transition fixed to 6 - 8% position. Mach and Reynolds numbers correspond to the airflow conditions for lower speed general aviation aircraft. All results were obtained according to algorithms presented in Table 2. Naca series airfoils presented in figures 7. - 13. have been selected by random choice from Table 2. list and they are represented only by their polars. As an airfoil of special interest, GA(W)-1 is presented by C_L-α and C_L-C_D curves. Twice increased distributions of δ* for this airfoil are presented by dotted lines in figures 16. - 20. (MRe=6.3 , M=0.15).

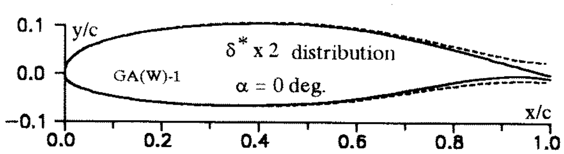

Figure 16.

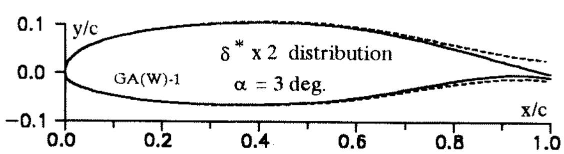

Figure 17.

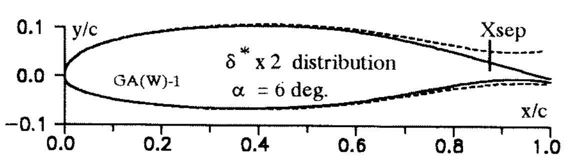

Figure 18.

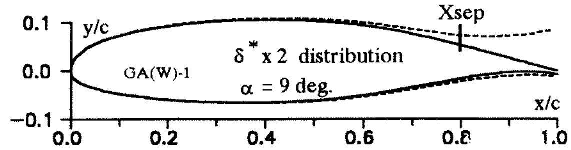

Figure 19.

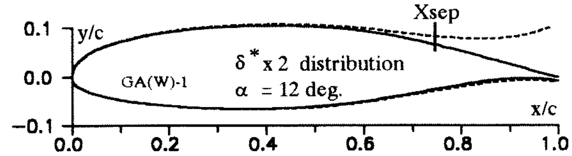

Figure 20.

Anlysis of the modified model results

Presented results show the obvious advantages of the modified model over the original. This model, however, is an intermediate development phase of another, more sophisticated model which would not be confined to a certain airfoil category but only to its geometry and airflow conditions. So far, this model can still be successfully used instead of the original in the design process of new airfoils, but correction for separated flow must be determined from the presented algorithms by placing the new airfoil in one of the existing categories according to its geometric characteristics. This might be a source of possible mistake for an not too experienced user and it should be done with caution.

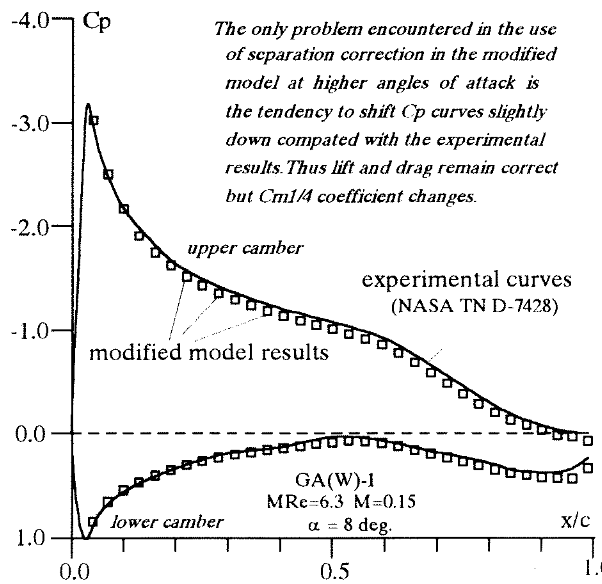

Figure 21.

When modified model with separated flow correction is applied, in some cases it shows a tendency to translate the Cp curves both for the upper and the lower camber of the airfoil just slightly down with respect to the experimental results, when applied at higher angles of attack (figure 21.). This difference is generally very small and it does not affect results for C_L and C_D, but it changes moment coefficient more remarkably. In that case, instead of $Cm_{1/4}$ values (quarter chord moment coefficient), the $Cm_{a.c}$ (moment coefficient about the aerodynamic center) should be recalculated from results for several smaller angles of attack where separated flow correction is not necessary, using modified boundary layer model. After that, the same $Cm_{a.c}$ value can be used for aerodynamic moment calculations at higher angles of attack. The source of this could also be in the specific rectangular grid type used in computer program.

It should be kept in mind that described model for separated flow correction can be used for the analyses of airfoils at which moderate separation starts at the trailing edge. That means that it can not cover cases when larger separation bubble with reattachment is formed somewhere on the airfoil (for instance on thin laminar airfoils at higher angles of attack close to the leading edge, or on airfoils with concave upper camber in the trailing edge region). Even with this exception, it can be applied on most of the airfoils suitable for use on low speed general aviation aircraft, since Table 2. lists only some of the airfoils on which the successful use of modified model was proved. Although, if an attempt to apply this model on an airfoil from the exempted category is done for any reason, no unique solutions for C_L and C_D will be obtained and a user will not be drawn to wrong conclusions.

Modified model in practice

The presented modified turbulent boundary layer model with moderate separation correction was included in *Tranpro* computer program. Among the other cases, it was used for the analyses of two interesting GA(W)-1 airfoil derivatives, designated GA(W)-NI10 and GA(W)-MI25, developed at the Belgrade Faculty of Mechanical Engineering[8,9]. These airfoils were designed for some practical project requirements with an intention to further improve the original airfoil's lift characteristics.

The airfoil GA(W)-NI10 was obtained by leading edge region modification of GA(W)-1 (figure 22.).

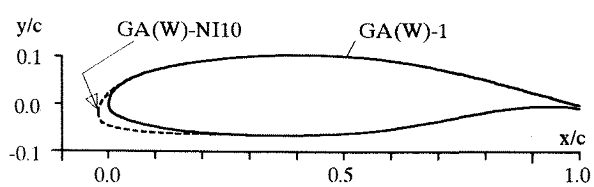

Figure 22.

Unlike some existing modifications of this kind, (for instance as the one developed by well known American designer Burt Rutan for one of his canard-configured light aircraft where a completely new nose was formed), on GA(W)-NI10 the original nose section of GA(W)-1 was 'cut-off' and rotated nose-down for 10° about a certain point, and then blended with the rest of the contour of the original airfoil.

In case of GA(W)-MI25 airfoil, a specific trailing edge modification of the original airfoil was done (figure 23.)

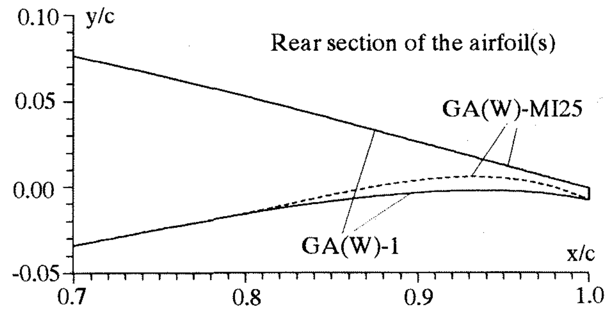

Figure 23.

Actually, only approximately last quarter of the lower camber was replaced by a curve with the trailing edge angle of -25° instead of -11.96° of the original airfoil.

Results obtained by *Tranpro* with the modified turbulent boundary layer model with moderate separation correction are given in figures 24 - 27.

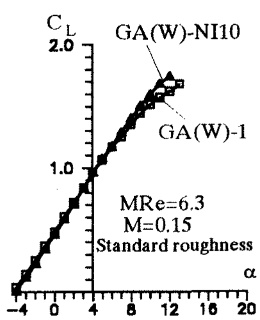

Figure 24.

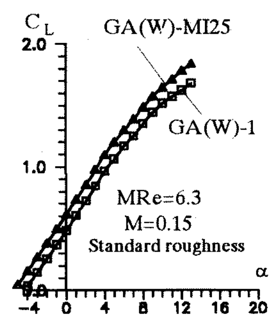

Figure 25.

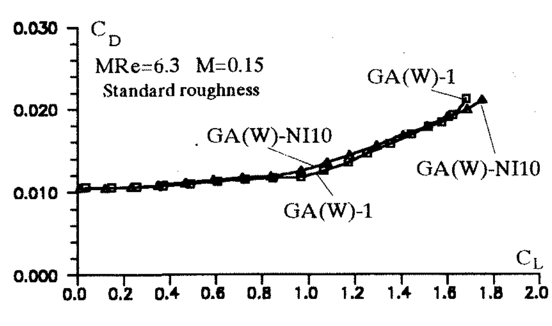

Figure 26.

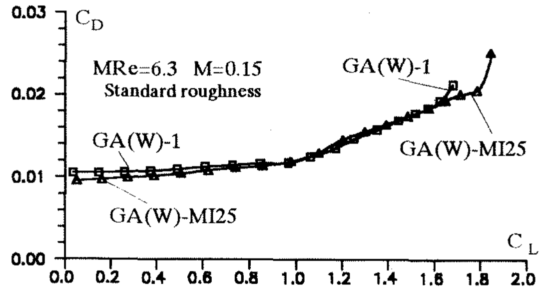

Figure 27.

In case of both of the new derivatives, a slightly altered GA(W)-1 modified model algorithm was applied.

The reason why these two airfoils were presented in this paper in such a detail requires an explanation. The primary task during the design of the two GA(W) derivatives was to further increase C_{Lmax} (although the original airfoil has very high value of maximum lift coefficient). As already mentioned, the described boundary layer model can not deal with massive separation i.e. with around-critical angles of attack. Still, at higher angles of attack with moderate separation, C_L-s of both GA(W) derivatives have overcome the original airfoil's lift curve, while their polars show that their drag divergence region, corresponding the end of moderate and start of massive separation, is moved to higher C_L values. Since the original nose shape was kept in both cases, no abrupt stall of this two derivatives is to be expected, and so an expectation of higher C_{lmax} in both cases sounds very reasonably. It is strongly believed that some future experiments will prove this statement.

Thus modified model can, in some cases, be used to give 'reasonable estimates' of airfoil behavior beyond the model's range of application, besides fulfilling its primary role. Mentioned GA(W) derivatives are believed to be such an example.

Conclusion

The modified turbulent boundary layer model with moderate separation correction was derived from the original model, applied in computer programs *Trandes* and initially in *Tranpro*. It has shown remarkable advantages over the original model in cases of application at low Mach and moderate Reynolds numbers, on most of the airfoils suitable for use on low speed general aircraft. In all presented analyses transition was fixed to 6% chord, which corresponds to the airflow conditions most often encountered in flight, even on composite structure wings when polluted by insects, dirt or rain and on metal wings with leading edge flaps, leading edge structure caps, de-icer boots, round head rivets extending over the airfoil surface close to the leading edge, etc.

The lift coefficient range available for the analyses is spread by the modified model to most of the values that appear in flight, except very high C_L-s during extreme maneuvers. Thus a computer program applying the modified model can be used in the airfoil selection or design phase of the aircraft design process for quick and efficient determination of optimal airfoil choice (but it certainly can not exclude the necessity for final experimental verifications).

So far, the modified model applies algorithms confined to airfoil categories and subcategories. The future aim is to connect them with airfoil geometry and airflow conditions only.

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