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FLYGTEKNISKA FÖRSÖKSANSTALTEN

The Aeronautical Research Institute of Sweden

COORDINATES AND CALCULATIONS FOR THE FFA-W1-*xxx*, FFA-W2-*xxx* AND FFA-W3-*xxx* SERIES OF AIRFOILS FOR HORIZONTAL AXIS WIND TURBINES

by

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SUMMARY

Airfoils for use on horizontal axis wind turbines have been designed. The airfoils are divided into three different series:

The first series, FFA-W1-xxx, constitutes airfoils with thickness to chord ratios from 12.8% to 27.1%. The design lift coefficients for the FFA-W1-xxx series range from 0.9 for the 12.8% airfoil, 1.05 for a 15.2% airfoil to 1.2 for the 27.1% airfoil.

Two airfoils in a second series FFA-W2-xxx are designed with design lift coefficients approximately 0.15 units lower than that for the FFA-W1-xxx series.

The third series of airfoils constitutes airfoils with thickness to chord ratios ranging from 19.5% to 36%. The 21.1% and 19.5% thick airfoils are designed to conform to thinner NACA 63-600 airfoils which are then to be used for the outer parts of the wind turbine blade. The thicker airfoils are designed to offer better aerodynamic performance for a given thickness to chord ratio than thick NACA 63-600 airfoils.

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1. INTRODUCTION

A number of airfoils for Horizontal Axis Wind Turbines, HAWTs, have been designed at FFA. The airfoils were initially designed for a 45 meter diameter HAWT. That wind turbine were to be operated at constant tip speed ratio and to be power controlled by yawing out of wind. With this rotor in mind the airfoils in the FFA-W1-xxx series were designed. Desirable characteristics of these airfoils are discussed in [1] and [2]. The FFA-airfoils are designed to produce structurally efficient airfoils with a design C_l that is higher than for common general aviation airfoils.

This technical note mainly provides coordinates and calculated drag polars for the FFA-airfoils. A short description is given of the computer programs that were used for the design process and calculations, and some of the characteristics of the airfoils are discussed. Only calculated aerodynamic data for the FFA airfoils are presented. A 2-D wind tunnel test of the 15.2% thick FFA-W1-152 airfoil has been carried out and results from this test are presented in [3].

2. ANALYTICAL TOOLS

The airfoil design and analysis codes ISES and XFOIL have been used for the design and analysis of the FFA-airfoils. ISES, developed at MIT by Drela and Giles [4,5], is based on a stream tube Euler equation solver for the inviscid flow field. A two equation boundary layer formulation is used to calculate the boundary layer which is coupled to the inviscid flow field via the boundary layer displacement thickness. Boundary layer transition is determined by the e^n method [5] (n in the e^n method is designated Acrit).

The ISES program is capable of calculating small regions of separated flow. The onset of separation on the airfoil suction side and the accompanying drag increase seem to be relatively well predicted. The de-cambering effect of a thick boundary layer or separation is also represented in the C_l/α-curve. For some airfoils and Reynolds numbers even C_{lmax} and stall seems to be rather well estimated.

Mark Drela has recently developed a new program XFOIL. XFOIL uses essentially the same boundary layer formulation as ISES, but the inviscid flow field is calculated by a linear-vorticity stream function panel method. Transonic flow analysis with shocks is not possible with the panel method of XFOIL but the panel method is appreciably faster to run than the streamline Euler method of ISES. Calculations for wind turbine airfoils are well into the subsonic regime were no shocks appear and calculations with XFOIL instead of ISES has offered a considareble reduction in computing time.

Both ISES and XFOIL have inverse design modes, where the velocity distribution over a part of the airfoil can be changed and the corresponding change in airfoil geometry calculated. XFOIL also has very nice features such as modifying the mean line by adding a ΔC_p (as function of the chord) to the pressure distribution.

2.1 Comparisons with Wind Tunnel Tests

Evangelista and Vemuru [7] show comparisons between ISES-calculations and wind tunnel results for an EPPLER 387, a NASA NLF(1)-1015 and a NASA NLF(1)-0416 airfoil. Comparisons between ISES-calculations and wind tunnel results for the FFA-W1-152 airfoil are found in [3].

Results from ISES and XFOIL were checked against wind tunnel results for a number of airfoils. Fig. 1-7 show XFOIL calculations compared with wind tunnel results for three airfoils from the NACA 63-series and a NASA LS(1)-0421 airfoil.

XFOIL calculations were made with free transition and forced transition. In reality transition is affected by such factors as surface quality, free stream turbulence and noise. Correlation with transition studies during low turbulence environments and very smooth wave free surfaces indicate that values of Acrit≈14 should be used to represent free flight aircraft conditions. A commonly used value of Acrit seems to be 9, which has been used for all free transition calculations presented in this report.

Fig.1-7 show that the minimum drag level, as calculated by XFOIL, is lower than the values taken from wind tunnel tests. The general characteristics of the drag polar is, however, rather well represented.

In order to carry out comparison of other airfoils with the FFA-airfoils calculations with XFOIL have been made for 15% and 21% thick airfoils from the NASA LS(1) series, the FX 84-W series and NACA 63-400 airfoils, see paragraph 4. Results with the one airfoil tested in different wind tunnels often show some variation. It therefore seems most just to compare different airfoils tested in the same way, or in this case to compare computations made with the same program using the same boundary layer parameters.

Calculations are also made using fixed transition. This is done in order to simulate a rough leading edge. The effect of roughness around the leading edge is dependent on the severeness, e.g. height, and the extension of the roughness. It is in no way obvious how to simulate roughness in calculations, and what degree of roughness that is to be simulated. One way of simulating roughness in calculations would be to introduce a jump in the boundary layer momentum thickness in addition to forcing transition. This has been practiced by Hill and Garrad [8] and seems sound. However, for the present study only forced transition was used. The transition has been forced at x/c=1% at the suction side and 10% at the pressure side. The desire has been to represent a "worst case" with XFOIL and ISES. As the angle of attack changes, the stagnation point will move around the leading edge. To be sure that the fixed transition position would be downstream of the stagnation point, for the angle of attack range used in the calculations, 10% was choosen at the pressure side and 1% at the suction side. Transition as far forward as 1% might not be realistic and has only been used to represent the mentioned "worst case".

The FFA-W1-152 airfoil has been wind tunnel tested with leading edge roughness. Roughness strips with a height of ≈0.7 mm at x/c=5% were used on an airfoil that had a 700 mm chord [3]. Results from this test compare rather well with ISES calculations with forced transition at 1% on the suction side and at 10% on the pressure side.

Calculated results with forced transition compared to wind tunnel test results with NACA standard roughness indicate that the detrimental effect on the drag level of the NACA standard roughness is worse than that of solely tripping the boundary layer. For the 15% thick NACA 63-airfoils a rather sharp suction peak occurs at the leading edge at high angles of attack. The free transition point therefore moves forwards almost to the 1% point for angles of attack around stall. Fixing transition at x/c=1% therefore has very little effect on the C_{lmax}. Other airfoils produce a distinct trailing edge separation subjected to an increase in angle of attack before a high leading edge suction peak appears.The maximum lift with free transition is then reached with the transition position rearward of the 1% point. The effect of forcing transition at x/c=1% is then large, which is shown in e.g. Fig.8 and 9.

Results are shown for the NASA LS(1)-0421 MOD airfoil in Fig.7. XFOIL calculations are made with fixed transition at x/c=7 % at both the suction and the pressure side. The wind tunnel tests with "roughness on" are made with roughness strips at x/c=7.5% at both sides [10]. This roughness is sized only large enough to provoke transition and not to simulate any heavy insect contamination. The drag increase due to the roughness strip (tripping strip) is rather well captured by XFOIL for this case. This also exemplifies how different the effects of the NACA standard roughness and the milder NASA roughness are. A review of the effects of roughness on airfoils has recently been given by Bragg and Gregoreck [12] were the lift loss due to roughness, on a NASA LS(1)-0413 airfoil, is shown to be substantial if tested with more severe roughness.

Fig.7 also shows that the maximum lift and stall characteristics are not correctly represented by the XFOIL calculations. The lift-curve slope, $\partial C_l/\partial\alpha$, decreases as the boundary layer gets thicker and the trailing edge separation progresses forward. This is quite well reproduced in XFOIL calculations, e.g. Fig.6. However, the magnitude of $\partial C_l/\partial\alpha$ taken from XFOIL (and ISES) calculations seem to be slightly higher than values obtained from wind tunnel tests. When the separated regions get more extensive and the lift reaches its maximum value or drop (stall), XFOIL calculations often appear to overestimate lift. In XFOIL, the wake trajectory for a viscous calculation case is taken from an inviscid solution at the same α. This is not strictly correct since viscous effects decrease lift and change the wake trajectory. It is believed that this effect is the cause of the differences shown in Fig.8.

2.2 Application of the Calculated Airfoil Data to Horizontal Axis Wind Turbines

Three-dimensional effects, such as span-wise pressure gradients, centrifugal forces and Coriolis forces will be present on a wind turbine blade. This will result in differences in, e.g., C_{lmax} and lift-curve slope. Differences between two- and three dimensional calculations are shown by Sørensen [13],[14]. Experimental work is done in Denmark [15],[16], in the U.S.A [17],[18], in England [19],[20] and in Holland [21]. Work is also currently done at FFA. A 5.35 meter diameter wind turbine has been wind tunnel tested in a large wind tunnel at CARDC in China [22]. One of the blades were equipped with 232 pressure taps so that C_p/x recordings could be obtained for 8 different radial stations. This blade has recently been tested during non-rotating conditions at FFA. Results from this test will be used to obtain quantitative and qualitative differences in e.g. C_l between the rotating and non-rotating case.

It seems to be well established that C_{lmax} is increased for the inner parts of the blade. Results from Madsen and Rasmussen [16] indicate highly reduced lift-curve slope and C_{lmax} towards the tip. The angle of attack in these data is derived from blade element-/momentum methods (Strip methods). The change in lift-curve slope between 2-D and 3-D data can therefore very well be originating from an incompleteness in the estimation of the angle of attack by the blade element/momentum method. Results from [18] show results from tests with a rectangular untwisted blade equipped with the S809 airfoil developed at SERI [23]. Pressure measurements were made at the 80% radius station of the blade and the local angle of attack was measured by a wind vane ahead of the blade. These measurements were compared to 2-D tests and relatively good agreement is shown between the 2-D and 3-D case.

Many of the studies of 3-D effects are only yet in the starting phase and reliable data are still to be awaited. In the light of the differences in, e.g., C_{lmax} between the 2-D case and the 3-D the uncertainty in C_{lmax} as predicted by XFOIL and the differences between XFOIL- and ISES calculations seem to be of less importance. For the FFA-W1 airfoils and the FFA-W2-152 airfoil calculations with ISES are made. Differences in C_{lmax} between ISES and XFOIL calculations are shown in Fig.31-36. It is believed that ISES calculations are more reliable. However, calculations at these high angles of attack must be viewed upon with somewhat critical eyes. For the thickest airfoils the uncertainty is the greatest, but then again, the uncertainty about the differences between 2-D data and 3-D data is large for the root region.

No attempts are made to make corrections to the polars so that they apply to the 3-D case of a wind turbine.

Further uncertain data are C_m. The calculated values of C_m drop (the absolute values) when the trailing edge separation progresses forward. This is seen in, e.g., Fig.3 where the pitching moment drops from ≈-0.08 for attached flow to ≈-.04 for high angles of attack. C_m taken from the wind tunnel test show the opposite trend. This drop in C_m which seems to be present of all XFOIL (and ISES) calculations of the FFA-airfoils and must be considered to be unrealistic.

At high angles of attack the C_d values must also be viewed with some caution.

3. CHARACTERISTICS OF THE AIRFOILS

Airfoils are designed with varying thickness to chord ratios reflecting the need for different stations on the blade.

The helix angle is small for the outer parts of the blade (the lift vector is directed almost perpendicular to the turbine plane) and the L/D ratio is of utmost importance. As the helix angle gets larger towards the root, the L/D ratio gradually looses importance. Towards the root there is instead, for structural resons such as high cross section height and reasonably small chords, a need for thick airfoils for operation at high lift coefficients.

The FFA-W1 series is designed with best L/D ratios at rather high lift coefficients. However, the 12.8% percent thick airfoil designed for the blade tip has a slightly lower design C_l than the 15.2% thick airfoil. The reason why the tip airfoil has a lower design C_l is mainly the following: If the blade is given, e.g., a linear taper plan form, then the local lift coefficient should, for optimum performance decrease towards the tip. Also, airfoil drag with a rough leading edge would be higher if the airfoil were designed for a higher design C_l, and low drag is of highest importance at the blade tip wher the helixangle is small and the velocities are large.

The underlying specifications set up for the design of the FFA-W1 and FFA-W2 airfoils are described in [2] and a full discussion of the airfoil characteristics will therefore not be given here.

3.1 The 15.2% Thick Airfoils FFA-W1-152 and FFA-W2-152

The 15.2% thick airfoils FFA-W1-152 and FFA-W2-152 are designed to have high L/D ratios. High L/D ratios are easiest attained by a relatively high C_l design. It is the product C_l*c that is the important characteristic at each blade radius of a wind turbine. Designing the blade for a lower C_l therefore results in larger chords. Larger chords give larger Reynolds numbers which favours the L/D ratio. Also, if the cross section height of the blade is kept constant, the relative thickness decreases with design C_l and thinner airfoils generally have lower drag. Together this makes L/D a rather weak function of design C_l.

The choice of design C_l can therefore be made on structural considerations. To support the blade designer with airfoils optimized at different design C_l the aim has been to design different series of airfoils characterized by the best L/D at different C_l. So far only a 15.2% and 21% thick section is designed in the lower C_l design series FFA-W2-xxx.

If a blade element/momentum method is used to obtain optimum chord and twist distribution it is in fact the product C_l*c that is determined at each blade radius. A large chord can be choosen if a low design C_l is choosen, or the chord can be decreased if the design C_l is increased (more cambered airfoils or local twist towards a higher angle of attack). Different choises of design C_l, i.e. design angle of attack, for the same airfoil will result in different values of the local blade chord, which will result in different Reynolds numbers for the design case (Wind speed and tip speed ratio). Therefore, calculations should be carried out at constant C_l*c for determination of at what lift coefficient best L/D will be obtained for a given airfoil for use on horizontal axis wind turbines.

Fig.10 and 11 show calculations for the 15.2% thick airfoils at both constant Re, and with Re varying inversely with C_l. It is seen that by making the the calculations at constant C_l*c; i.e. at constant C_l*Re, in comparison to a constant Re, the angle of attack range for a given high L/D is increased. Fig.12 shows that a high L/D can be achieved both for the lower design C_l of the FFA-W2 airfoil as well as for the FFA-W1 airfoil. Fig.13 shows both airfoils plotted superimposed on each other. (The design C_l for the FFA-W2 airfoils can not be characterized as low, but their design C_l is lower than that for the FFA-W1 airfoils)

If airfoil surfaces are sufficiently smooth, the laminar boundary layer will undergo separation in the adverse pressure gradient before transition occurs. Transition then occurs in the free shear layer and a transitional bubble is formed. This is the normal way transition occurs at airfoils at low Reynolds numbers (Re less than

≈ 5 million). The main problem with this bubble is the associated drag increase. This drag increase is proportional to the height of the bubble (δ^*) and the magnitude of the pressure jump at the rear part of the bubble where reattachment occurs (see e.g. [5] and [25]).

For the thinner FFA-airfoils, where low drag is important, an "instability range" has been incorporated on the airfoil suction side. An instability range is a region of weak adverse pressure gradients that will promote the growth of boundary layer instability and transition, but were the gradients are small enough to avoid laminar separation. The "instability range" on the 15% airfoils are designed to minimize drag at Reynolds numbers of the order of 3 millions. The instability range is rounded off at the aft part but extends some distance beyond the point of laminar separation. In this way the pressure jump at the rear of the bubble, and thus the drag increase, will be kept small even if transition does not occur before separation. If the Reynolds number is decreased, the length of the laminar bubble will increase, and the pressure jump will be larger and cause a drag penalty. For the smooth surface condition of the airfoil, this effect sets the lower Re range for the airfoil to Re≈2 million.

One purpose was to design the 15.2% thick airfoils so that high L/D ratios can be realised. Operational experience with wind turbine blades, however, indicates that the problem with too long transitional bubbles will not be the primary problem. Surface manufacturing irregularities, erosion of the leading edge, or the accumulation of dirt or insects are likely to cause premature transition. A study of the effects of different L/D ratios on the power coefficient [2], shows the utmost importance of an airfoil for which the degradation of the L/D ratio with, e.g., a rough leading edge is kept to a minimum. Care is therefore taken to give the airfoils good, or at least acceptable, performance as "rough airfoils".

The lower surface is designed so that ≈90% laminar flow should be achievable if the surface is smooth. The upper surface is designed with a roof top followed by the "instability range" and a main pressure recovery to the trailing edge pressure. The length of the roof top is, however, matched in such a way that even if the boundary layer is subject to premature transition and thickening due to a rough surface, the pressure recovery should be mild enough to avoid boundary layer separation for lift coefficients ≈0.1 units above the design lift coefficient. It is believed that this gives the best combination of high L/D due to laminar flow for the smooth surface case, and acceptable L/D with rough surfaces.

It is possible that better rough airfoil performance can be achieved if the airfoil is given more aft loading like e.g. a NACA 63-615 (a=1.0) airfoil. Introducing a certain aft loading will decrease the velocity level of the roof top and reduce the average adverse pressure gradients. This is however incompatible with the requirements originally set up for design of the ≈15% thick FFA-airfoils to keep C_m more positive than ≈-0.06 (see [2]).

The FFA-W1-152 is designed for a higher design C_l than the NACA 63-615 airfoil. However, C_{lmax} is kept at the same level as, e.g., the NACA 63-615. The relative drop in C_{lmax} between smooth and rough airfoil is attempted to be kept as small as possible for the thinner FFA airfoils.

Care was also taken to avoid too abrupt stall characteristics. The pressure recovery has therefore been formed to have the shape parameter, H, gradually increasing. It is hoped that this will lead to a separation gradually moving forward. A Stratford type of pressure recovery (with a constant value of H) has been avoided. It is also likely that the existence of a roof top introduces gentler stall characteristics. This is seen for e.g. the NASA LS(1) and NASA LS(1) MOD airfoils [10]. The modification of the NASA LS(1) to obtain the MOD airfoil consisted in reducing the magnitude of the adverse pressure gradients aft of a roof top at C_l=0.4. This modification enhanced the L/D ratio at moderate to high lift coefficients (≈C_l>0.6) and at low Re, but it also made the stall more abrupt. The adverse pressure gradients aft of the kink at x/c≈0.5 is detrimental in the way that trailing edge separation starts at rather low angles of attack. However, the somewhat milder gradients in front of the kink sees to it that the separation movement is not too rapid and a gentle stall is obtained. For the modified airfoil, almost fully attached flow for angles of attack up to C_{lmax} is realised. Increasing the angle of attack further however leads to a rather quick movement of the separation point to x/c≈0.4 and a substantial loss in lift. Comparisons between the NACA 23015, which also lacks a roof top and a kink at higher α, and e.g 15% thick NACA 6-series airfoils also confirm that a roof top type of velocity distribution is favourable in order to avoid an abrupt stall.

The estimated C_{lmax} level from calculations and a rather constant post stall lift level were confirmed during the wind tunnel test of the FFA-W1-152 airfoil [3], which is shown in Fig.9. (The airfoil that was wind tunnel tested is designated AK15v4, but is essentially the same airfoil as FFA-W1-152.)

3.2 Thicker Airfoils in the FFA-W1 Series

Airfoils with thickness to chord ratios up to 27.1% have been designed in the FFA-W1 series. (The airfoils in the FFA-W3 series are used for thicker airfoils and the ≈21%, 24% and 27% airfoils from the FFA-W3 series can in principle replace the airfoils with equivalent thickness in the FFA-W1 series.) The design C_l has been increased from 1.05 for the 15.2% thick section to C_l design ≈1.2 for the 21.1% thick section. For the thicker sections the objective has been to achieve as high as possible L/D with a rough surface.

There exist several series of airfoils with varying thickness to chord ratios, such as e.g.the NACA 4-,5- and 6 digit series [9], the NASA LS series [10] and the FX 84 series [24]. Varying thicknesses for these airfoils are in principle achieved by scaling the coordinates around the mean camber line. For airfoils with a t/c ratio of say 15% it is the suction side of the airfoil that is critical to boundary layer separation. If these airfoils are thickened by adding equal thickness on both the upper and lower side of the mean line, then equal thickness induced velocities are added on both the suction and pressure sides of the airfoil. The increased maximum velocities and adverse pressure gradients on the pressure side do not normally introduce separation problems. However, the additionally increased adverse pressure gradients on the suction side are quite harmful. The margin to separation will be decreased and the drag rise will start at rather moderate lift coefficients, especially if the boundary layer is subject to premature transition. This behaviour is shown for all NACA airfoils and also for the FX 84 airfoils. The NASA LS(1) MOD airfoils show acceptable performance for t/c ratios up to 21%.

With modern design codes for airfoils it is easy to view the airfoil as an upper and lower side with corresponding velocity distributions. Fig. 14 shows the FFA-W1 airfoils plotted superimposed on each other. It is assumed that the maximum thickness is warranted for x/c positions around 0.3. The thickness distribution is such that an ellipsoid like shaped beam, centred at $x/c \approx 0.3$ will fill much of the airfoil. The thickness has been added mainly to the lower surface to avoid increasing the maximum velocity and adverse pressure gradients on the airfoil suction side too much. On the upper surface the thickness has been moved forward to reduce the adverse pressure gradients further.

Variation of the geometry:

Calculations were made with slightly different geometries. The starting position of the pressure recovery on the suction side was varied and it seemed as if the best L/D ratios, especially with premature transition, were obtained with the shortest possible roof top. A gentle stall is hoped to be obtained anyway for the thicker airfoils (t/c>21%). The stall characteristic requirements are also assumed to be lower towards the blade root.

In order to compensate for the loss in lift due to the low pressure at the maximum thickness position on the pressure side, the airfoils are given more aft loading.

If too much thickness is added to the upper side, this will cause too early separation on the suction side. Adding too much thickness to the lower side, together with the relatively aft back maximum thickness position on the lower side, will increase the adverse pressure gradients on the lower side too much. This will result in lower surface separation at low angles of attack as indicated in Fig.15b.

When an angle of attack range, in which the airfoil is to operate, is defined the aim is that the margin to separation on the lower surface side at low angles of attack should be about the same as the margin to separation on the upper surface at high angles of attack. This is one basic idea of the design of the thick FFA-airfoils

The adverse pressure gradients on the lower side will be rather large for the thicker FFA-airfoils. If the airfoil surface is smooth enough the laminar separation bubble can be relatively thick since no sufficient instability range is present on the pressure side of the thicker airfoils. One risk, that is not verified, is that a laminar bubble will be an undesirable noise generator. This together with the separation risk at low angles of attack with a rough leading edge led to the differences between the ≈24% and 27% thick airfoils of the FFA-W3 and the equivalent airfoils of the FFA-W1 series. The adverse gradients on the lower side has been relaxed somewhat for the FFA-W3 series airfoils. Moving the maximum thickness forward on the lower side would also reduce the magnitude of the pressure gradients there. This would, however, reduce the building height of the airfoil around $x/c \approx 0.3$.

3.3 The FFA-W2 Series

The objective was to design different series of airfoils with different design C_l, something like the NACA 6-series of airfoils. The FFA-W2-152 is designed with a lower design C_l than the FFA-W1-152 airfoil. The W2 airfoil is also designed to have a lower C_{lmax} than the W1 airfoil. The FFA-W2-210 airfoil is designed to have a lower design C_l and lower C_{lmax} than FFA-W1-211. No further airfoils are yet made in this series. However, such airfoils can be made relatively easily by reducing the camber of the FFA-W1 airfoils.

3.4 The FFA-W3 Series

The NACA 63-615 and 63-618 airfoils are believed to have rather good performance both as rough and smooth airfoils. The NACA 63 airfoils were considered during a study of a pitch regulated turbine with demands on slightly lower C_l design of the thinner airfoils than the FFA-W1-152 and FFA-W1-182 have. For a pitch regulated turbine it seems that the requirements put up for the thinner FFA-W1 and FFA-W2 airfoils such as to have the stall to occur at C_{lmax} rather close to the design C_l is not applicable. Instead it is desirable to have a certain margin to C_{lmax} in order to avoid the situation were most of the blade has angles of attack above stall. Pithing out of wind then increases power, which is the opposite of the normal operation mode of the regulation system.

As the thickness is increased the rough airfoil performance of the NACA 6-series airfoil declines. An improvement in performance is believed to be attainable if airfoils of the FFA-W1 type are used instead of scaling up NACA 63-600 airfoils to large thicknesses. If airfoil surfaces are perfectly smooth the difference might be small or even favouring the NACA airfoils of t/c around 21-25%. But if the airfoil surfaces are rough or if premature transition occurs, then the upper side boundary layer of the the NACA airfoils are likely to separate at too low angles of attack. This will cause high drag levels but also a highly reduced C_{lmax}.

The FFA-W1-211 airfoil was slightly modified to better conform to the thinner NACA 63-618 airfoil. The airfoil was given more aft loading and designated FFA-W3-211. Calculations with XFOIL give the FFA-W3-211 slightly higher C_{lmax} than the corresponding FFA-W1 airfoil. The ≈24% and 27% thick FFA-W3 airfoils have slightly more camber than the corresponding FFA-W1 airfoils. The adverse pressure gradients aft of maximum thickness on the lower side are somewhat reduced. The FFA-W3 airfoils have slightly larger pitching moment coefficients.

The three 30% to 36% thick airfoils are made in such a way that an ellipsoid like shaped beam centred at x/c≈0.3 is surrounded by, what is believed to be, an aerodynamic shape as advantageous as possible. The airfoils are given a finite thickness at the trailing edge since this is thought to enhance performance at high C_l. The upper side consists of almost a straight line aft of x/c≈0.65 (it can be made straight without changing the aerodynamics noticeably). The lower side should be given a downwards pointing trailing edge, which could be made more marked than for the the given airfoil coordinates if "low" drag at high C_l is wanted. Results with e.g. "gurney" flaps [26] and split flaps contra plain flaps [9] indicate that a finite trailing edge thickness and a downwards pointing lower trailing edge should be favourable for thick wind turbine airfoils.

A 19.5% thick airfoil, FFA-W3-195, is produced by interpolating the mean lines and thickness distributions of the FFA-W3-211 airfoil and a NACA 63-618 (a=1.0) airfoil. (Actually, the original NACA airfoil coordinates were rotated 0.3° anti-clock wise before the interpolation, so that the leading edges of the NACA airfoil better should conform to the 21% thick FFA airfoil.)

4. COMPARISON WITH OTHER AIRFOILS

Calculations with XFOIL were done for a number of ≈15% thick and ≈21% thick airfoils. Beside the FFA-W1 airfoils the NACA 63-600, the NASA LS(1) and the FX 84 airfoils were choosen for a comparison.

The FX 84 series has good performance with smooth surfaces and gentle stalling characteristics but has poor performance with leading edge roughness. Especially for the 21% thick section the drag increase and loss in lift due to premature transition is marked.

The NACA LS(1) MOD airfoils have good performance with premature transition[1] but has a very high C_{lmax} and sharp stall characteristics. Performance with smooth surfaces is rather modest.

The thinner NACA 63 airfoils are rather good airfoils both with smooth surfaces as well as with a roughened leading edge[2]. However, the performance with a rough leading edge decline as the thickness to chord ratios grow.

For the SERI airfoils [23] and the airfoils developed in England [8] no coordinates were available.

Calculations were done at Re=3 million with free transition and with fixed transition. Transition was fixed at x/c=1% at the suction side and at x/c=10% at the pressure side for the fixed transition case. All calculations were done at a Mach number of 0.15. The results are shown in Fig.16 and 17. Airfoil coordinates were obtained for an FFA-airfoil by taking 50% of the FFA-W1-211 and 50% of the FFA-W1-242 airfoil coordinates. This airfoil is considered to be more structurally equal to the ≈21% thick NASA, FX and NACA airfoils to which it is aerodynaically compared than the somewhat thinner FFA-W1-211 airfoil. The ≈21% thick airfoils are shown in Fig.18.

From Fig.16 it is seen that the FFA-W1-152 airfoil has the best L/D ratio for the free transition case. This is much due to the high value of C_l for which it is optimized. The FX airfoil has good performance with free transition. However, as the transition is fixed at x/c=1% at the suction side there is a substantial drag increase and substantial loss in lift for C_l greater than ≈0.7.

Fig.17 shows that the ≈22% thick FFA airfoil in comparison to the other airfoils have favourable performance at high lift coefficients with fixed transition. With free transition and for lift coefficients below unity the NACA and FX airfoils have lower drag than the FFA airfoil, but with fixed transition these airfoils have a substantial drag increase and loss in lift due to boundary layer separation on the suction side. The highest C_{lmax} value with fixed transition is also indicated for the FFA airfoil.

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[1] If comparisons with airfoils tested in [9] are done it should be observed that the roughness used in wind tunnel tests of the LS airfoils [10] is not as severe as the NACA standrad roughness [9].

[2] Hill & Garrad [8] come to the conclusion that the NACA 63 or 64 airfoils are hard to improve upon for the design of a low lift tip airfoil.

5. PRESENTATION OF DATA OF THE FFA AIRFOILS

Data is presented in the following way:

1. Figures of the airfoils. (Fig.18-22)

2. Inviscid C_p distributions for the airfoils at lift coefficients $C_l=0$ to $C_l=1.8$. For each airfoil the C_p distribution is displayed for an inviscid case. An angle of attack close to design C_l is choosen for these calculations. The dashed curves show the inviscid C_p distribution and the full curves show the viscous C_p distributions. (Fig. 25-27)

3. Drag polars calculated with XFOIL and ISES. Each page consists of three plots. In the left plot the drag coefficient, C_d, is plotted versus the lift coefficient, C_l. In the middle plot C_l and the pitching moment coefficient, C_m, is plotted versus the angle of attack, α. In the right plot the transition position is plotted on the x-axis and C_l on the y-axis.

   The different calculated cases are named in the following way: POL.Airfoil_rr_tt or POLAR.Airfoil_rr_tt. The Reynolds number is indicated by two digits where $rr=Re*10^{-5}$. The number or letters tt indicate the transition mode. tt=9 means free transition with Acrit=9, tt=TU1 means that transition has been forced at x/c=1% at the suction side and x/c=10% at the pressure side (Xtrs=0.01 and Xtrp=0.1). POL stands for XFOIL calculations and POLAR stands for ISES calculations. All calculations are made at a Mach number of 0.15.

   The designation for the airfoils in the FFA-W1 series is a Wxxx, where xxx is the thickness number for the airfoil. For the FFA-W2 airfoils the designation is W2-xxx and for the FFA-W3 series the designation is W3-xxx.

4. Airfoil coordinates

5. Data for the calculated polars in digit form (contrary to plotted form) submitted in appendix.1

Drag polars are calculated only for Re=1,2 and 3 million. The airfoils are designed for the Reynolds number regime above Re=2 million and upwards. From a boundary layer separation point of view the lowest Reynolds numbers are the most critical. As the Reynolds number increases the drag decreases and C_{lmax} is possibly increased. In order to reduce the number of calculations to be made, it was considered as most important to do the calculations at Reynolds numbers up to Re=3 million.

Fig 26 shows the lift/drag ratio versus the lift coefficient for the FFA-W1 airfoils at Re=3 million. The calculations are made with

XFOIL for both free transition and with fixed transition. Fig 27 shows a comparison between the 15.2% and 21.1% thick FFA-W1 airfoils and the FFA-W2 airfoils at Re=3 million.

Fig.28 and fig.29 show drag polars for the FFA-W1 airfoils at Re=3 million with the polars for all airfoils plotted in the same diagram.

Fig.30-35 show polars for the FFA-W1 airfoils

Fig.36-37 show polars for the FFA-W2 airfoils

Fig.38-45 show polars for the FFA-W3 airfoils

6.CONCLUDING REMARKS

The two-dimensional analysis and design codes ISES and XFOIL [4],[5],[6] have been used for the design of a number of airfoils for wind turbine applications.

The first series, FFA-W1-xxx, constitutes airfoils with thickness to chord ratios from 12.8% to 27.1%. These airfoils are designed with a relatively high design lift coefficient. This will meet the requirements for e.g. wind turbines designed for low tip speed ratios. The thinner airfoils are designed with requirements that high lift/drag ratios be realized with smooth surfaces and laminar flow, but with the constraint that good performance should be achieved even with a rough leading edge with insect contamination or/and manufacturing irregularities. The thicker airfoils are designed to overcome the weakness of high drag and low C_{lmax} that thick airfoils commonly exhibit with rough leading edges. Calculations with XFOIL indicate that the rough airfoil characteristics of the 21.1% thick FFA-airfoil is superior to the NASA LS(1)-0421 MOD, the NACA 63-621, and the FX 84-W-218 airfoils. A wind tunnel test of the 15.2% thick airfoil has been carried out in which much of the computed performance was verified.

A 15.2% and a 21.1% thick airfoil have been designed with the same requirements as the FFA-W1-xxx airfoils, but with a slightly lower design lift coefficient.

The FFA-W3 series has been designed to provide airfoils for pitch regulated turbines. The FFA-W3 series include airfoils with thickness to chord ratios up to 36%. A 19.5% thick airfoil have been made to conform with a 18% thick NACA 63-618 airfoil. It is assumed that NACA 63-6xx airfoils are used for thickness to chord ratios below 18%.

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TABLE 1. Profile coordinates.

Airfoil:FFA-W1-128

x/c	yu/c	yl/c
0.00006	0.00111	0.00111
0.00132	0.00573	-0.00443
0.00508	0.01250	-0.00832
0.01134	0.01967	-0.01190
0.02006	0.02717	-0.01499
0.03121	0.03494	-0.01753
0.04471	0.04284	-0.01962
0.06052	0.05076	-0.02132
0.07856	0.05860	-0.02264
0.09873	0.06623	-0.02360
0.12095	0.07353	-0.02427
0.14510	0.08039	-0.02465
0.17107	0.08667	-0.02477
0.19874	0.09223	-0.02466
0.22797	0.09696	-0.02436
0.25862	0.10074	-0.02391
0.29055	0.10344	-0.02333
0.32361	0.10491	-0.02263
0.35762	0.10507	-0.02184
0.39244	0.10386	-0.02096
0.42789	0.10131	-0.02004
0.46380	0.09751	-0.01908
0.50000	0.09259	-0.01809
0.53633	0.08676	-0.01704
0.57259	0.08020	-0.01595
0.60863	0.07312	-0.01481
0.64426	0.06573	-0.01363
0.67932	0.05820	-0.01241
0.71363	0.05069	-0.01112
0.74704	0.04337	-0.00976
0.77938	0.03636	-0.00837
0.81050	0.02978	-0.00698
0.84025	0.02373	-0.00561
0.86848	0.01829	-0.00430
0.89507	0.01353	-0.00315
0.91987	0.00950	-0.00220
0.94278	0.00621	-0.00148
0.96368	0.00366	-0.00101
0.98248	0.00183	-0.00078
0.99908	0.00076	-0.00080

TABLE 2. Profile coordinates.

Airfoil:FFA-W1-152

x/c	yu/c	yl/c
0.00000	0.00000	0.00000
0.00126	0.00666	-0.00552
0.00503	0.01416	-0.01002
0.01130	0.02239	-0.01380
0.02003	0.03102	-0.01717
0.03118	0.03988	-0.02016
0.04470	0.04886	-0.02276
0.06052	0.05784	-0.02499
0.07857	0.06668	-0.02693
0.09877	0.07526	-0.02861
0.12101	0.08344	-0.03006
0.14518	0.09110	-0.03128
0.17118	0.09810	-0.03229
0.19888	0.10431	-0.03308
0.22814	0.10958	-0.03367
0.25882	0.11372	-0.03404
0.29078	0.11662	-0.03422
0.32386	0.11818	-0.03416
0.35791	0.11837	-0.03388
0.39276	0.11711	-0.03337
0.42825	0.11432	-0.03263
0.46419	0.11006	-0.03167
0.50043	0.10457	-0.03047
0.53679	0.09810	-0.02903
0.57309	0.09088	-0.02735
0.60917	0.08310	-0.02548
0.64483	0.07497	-0.02341
0.67992	0.06667	-0.02112
0.71427	0.05838	-0.01865
0.74771	0.05028	-0.01605
0.78009	0.04249	-0.01341
0.81124	0.03513	-0.01082
0.84101	0.02830	-0.00841
0.86927	0.02212	-0.00626
0.89588	0.01665	-0.00442
0.92071	0.01194	-0.00294
0.94364	0.00803	-0.00185
0.96457	0.00492	-0.00115
0.98338	0.00258	-0.00087
1.00000	0.00098	-0.00098

TABLE 3. Profile coordinates.

Airfoil:FFA-W1-182

x/c	yu/c	yl/c
0.00002	0.00077	0.00077
0.00128	0.00712	-0.00623
0.00505	0.01504	-0.01115
0.01131	0.02339	-0.01533
0.02004	0.03216	-0.01959
0.03119	0.04121	-0.02388
0.04471	0.05039	-0.02825
0.06054	0.05955	-0.03266
0.07859	0.06855	-0.03708
0.09878	0.07721	-0.04143
0.12102	0.08542	-0.04563
0.14520	0.09305	-0.04960
0.17120	0.09996	-0.05321
0.19889	0.10601	-0.05639
0.22815	0.11107	-0.05904
0.25883	0.11501	-0.06105
0.29079	0.11768	-0.06232
0.32387	0.11897	-0.06278
0.35792	0.11874	-0.06244
0.39277	0.11689	-0.06130
0.42826	0.11356	-0.05940
0.46420	0.10892	-0.05675
0.50044	0.10323	-0.05332
0.53680	0.09673	-0.04905
0.57310	0.08962	-0.04406
0.60917	0.08209	-0.03852
0.64484	0.07430	-0.03260
0.67993	0.06641	-0.02660
0.71428	0.05855	-0.02086
0.74772	0.05086	-0.01575
0.78009	0.04344	-0.01144
0.81124	0.03641	-0.00797
0.84102	0.02984	-0.00528
0.86928	0.02380	-0.00327
0.89588	0.01837	-0.00187
0.92071	0.01358	-0.00101
0.94365	0.00946	-0.00060
0.96457	0.00604	-0.00057
0.98338	0.00329	-0.00081
1.00000	0.00115	-0.00115

TABLE 4. Profile coordinates.

Airfoil:FFA-W1-211

x/c	yu/c	yl/c
0.00001	-0.00052	-0.00052
0.00127	0.00779	-0.00651
0.00504	0.01619	-0.01293
0.01130	0.02504	-0.01937
0.02003	0.03421	-0.02603
0.03119	0.04358	-0.03292
0.04471	0.05304	-0.03996
0.06053	0.06244	-0.04702
0.07858	0.07163	-0.05403
0.09878	0.08046	-0.06087
0.12101	0.08878	-0.06739
0.14519	0.09642	-0.07346
0.17119	0.10323	-0.07893
0.19888	0.10905	-0.08365
0.22814	0.11372	-0.08742
0.25882	0.11706	-0.09007
0.29079	0.11891	-0.09149
0.32387	0.11920	-0.09152
0.35792	0.11795	-0.08994
0.39277	0.11535	-0.08657
0.42825	0.11156	-0.08141
0.46420	0.10677	-0.07468
0.50044	0.10114	-0.06665
0.53680	0.09483	-0.05771
0.57310	0.08799	-0.04832
0.60917	0.08074	-0.03903
0.64483	0.07319	-0.03033
0.67993	0.06557	-0.02255
0.71427	0.05802	-0.01588
0.74772	0.05066	-0.01042
0.78009	0.04358	-0.00611
0.81124	0.03686	-0.00288
0.84101	0.03054	-0.00064
0.86927	0.02468	0.00071
0.89588	0.01932	0.00130
0.92071	0.01450	0.00129
0.94364	0.01027	0.00089
0.96457	0.00663	0.00024
0.98338	0.00362	-0.00057
1.00000	0.00114	-0.00148

TABLE 5. Profile coordinates.

Airfoil:FFA-W1-242

x/c	yu/c	yl/c
0.00003	0.00138	0.00138
0.00129	0.00984	-0.00794
0.00506	0.01953	-0.01578
0.01132	0.02945	-0.02367
0.02005	0.03946	-0.03178
0.03121	0.04953	-0.04038
0.04473	0.05958	-0.04935
0.06055	0.06939	-0.05854
0.07860	0.07877	-0.06783
0.09879	0.08752	-0.07708
0.12103	0.09545	-0.08608
0.14521	0.10243	-0.09461
0.17120	0.10834	-0.10242
0.19890	0.11310	-0.10928
0.22816	0.11662	-0.11491
0.25884	0.11884	-0.11905
0.29080	0.11975	-0.12145
0.32388	0.11939	-0.12191
0.35793	0.11779	-0.12026
0.39278	0.11501	-0.11652
0.42826	0.11117	-0.11077
0.46421	0.10640	-0.10305
0.50044	0.10087	-0.09354
0.53680	0.09475	-0.08263
0.57310	0.08820	-0.07057
0.60917	0.08136	-0.05786
0.64484	0.07436	-0.04519
0.67993	0.06732	-0.03359
0.71428	0.06034	-0.02345
0.74771	0.05352	-0.01496
0.78009	0.04695	-0.00828
0.81123	0.04065	-0.00331
0.84101	0.03464	0.00013
0.86927	0.02894	0.00217
0.89588	0.02359	0.00297
0.92071	0.01862	0.00272
0.94364	0.01409	0.00165
0.96456	0.01002	0.00008
0.98337	0.00639	-0.00164
0.99999	0.00315	-0.00316

TABLE 6. Profile coordinates.

Airfoil:FFA-W1-271

x/c	yu/c	yl/c
0.00000	0.00000	0.00000
0.00126	0.01163	-0.01158
0.00503	0.02286	-0.02134
0.01130	0.03384	-0.03083
0.02003	0.04457	-0.04069
0.03118	0.05518	-0.05111
0.04470	0.06565	-0.06196
0.06052	0.07585	-0.07297
0.07857	0.08559	-0.08397
0.09877	0.09468	-0.09476
0.12101	0.10293	-0.10505
0.14518	0.11019	-0.11465
0.17118	0.11631	-0.12330
0.19888	0.12118	-0.13072
0.22814	0.12475	-0.13664
0.25882	0.12699	-0.14077
0.29078	0.12787	-0.14288
0.32386	0.12746	-0.14278
0.35791	0.12584	-0.14036
0.39276	0.12308	-0.13561
0.42825	0.11930	-0.12861
0.46419	0.11461	-0.11957
0.50043	0.10913	-0.10872
0.53679	0.10301	-0.09638
0.57309	0.09639	-0.08300
0.60917	0.08939	-0.06912
0.64483	0.08210	-0.05540
0.67992	0.07470	-0.04241
0.71427	0.06734	-0.03071
0.74771	0.06011	-0.02075
0.78009	0.05312	-0.01273
0.81124	0.04646	-0.00675
0.84101	0.04016	-0.00287
0.86927	0.03426	-0.00096
0.89588	0.02879	-0.00059
0.92071	0.02367	-0.00132
0.94364	0.01894	-0.00269
0.96457	0.01462	-0.00434
0.98338	0.01073	-0.00599
1.00000	0.00730	-0.00734

TABLE 7. Profile coordinates.

Airfoil:FFA-W2-152

x/c	yu/c	yl/c
0.00000	0.00000	0.00000
0.00126	0.00544	-0.00556
0.00503	0.01174	-0.01042
0.01129	0.01892	-0.01471
0.02001	0.02664	-0.01853
0.03115	0.03467	-0.02202
0.04466	0.04281	-0.02522
0.06047	0.05096	-0.02816
0.07850	0.05901	-0.03086
0.09868	0.06685	-0.03334
0.12090	0.07441	-0.03556
0.14505	0.08159	-0.03748
0.17103	0.08830	-0.03909
0.19870	0.09440	-0.04037
0.22793	0.09977	-0.04131
0.25859	0.10426	-0.04191
0.29052	0.10775	-0.04216
0.32357	0.11011	-0.04208
0.35759	0.11119	-0.04169
0.39241	0.11092	-0.04100
0.42786	0.10920	-0.04004
0.46378	0.10602	-0.03880
0.49999	0.10143	-0.03730
0.53631	0.09563	-0.03550
0.57258	0.08882	-0.03344
0.60862	0.08130	-0.03114
0.64425	0.07333	-0.02862
0.67931	0.06515	-0.02590
0.71363	0.05697	-0.02301
0.74704	0.04899	-0.01998
0.77939	0.04135	-0.01691
0.81051	0.03415	-0.01389
0.84026	0.02750	-0.01102
0.86849	0.02147	-0.00837
0.89508	0.01614	-0.00602
0.91989	0.01155	-0.00407
0.94280	0.00774	-0.00257
0.96370	0.00472	-0.00156
0.98250	0.00249	-0.00102
0.99910	0.00095	-0.00093

TABLE 8. Profile coordinates.

Airfoil:FFA-W2-210

x/c	yu/c	yl/c
0.00000	0.00000	0.00000
0.00126	0.00775	-0.00705
0.00503	0.01559	-0.01320
0.01130	0.02381	-0.01951
0.02003	0.03228	-0.02625
0.03118	0.04094	-0.03338
0.04470	0.04967	-0.04071
0.06052	0.05835	-0.04810
0.07857	0.06688	-0.05545
0.09877	0.07511	-0.06266
0.12101	0.08291	-0.06955
0.14518	0.09015	-0.07598
0.17118	0.09669	-0.08181
0.19888	0.10238	-0.08689
0.22814	0.10708	-0.09103
0.25882	0.11063	-0.09408
0.29078	0.11290	-0.09596
0.32386	0.11376	-0.09660
0.35791	0.11315	-0.09591
0.39276	0.11109	-0.09381
0.42825	0.10766	-0.09031
0.46419	0.10298	-0.08549
0.50043	0.09723	-0.07947
0.53679	0.09063	-0.07243
0.57309	0.08342	-0.06456
0.60917	0.07578	-0.05617
0.64483	0.06795	-0.04762
0.67992	0.06015	-0.03925
0.71427	0.05254	-0.03137
0.74771	0.04524	-0.02425
0.78009	0.03838	-0.01810
0.81124	0.03201	-0.01302
0.84101	0.02618	-0.00901
0.86927	0.02090	-0.00600
0.89588	0.01618	-0.00385
0.92071	0.01205	-0.00243
0.94364	0.00850	-0.00157
0.96457	0.00553	-0.00114
0.98338	0.00315	-0.00106
1.00000	0.00129	-0.00129

TABLE 9. Profile coordinates.

Airfoil:50% NACA 63-618 AND 50% FFA-W3-211

x/c	yu/c	yl/c
0.00000	0.00000	0.00000
0.00126	0.00761	-0.00669
0.00503	0.01549	-0.01302
0.01130	0.02388	-0.01925
0.02003	0.03251	-0.02549
0.03118	0.04137	-0.03154
0.04470	0.05038	-0.03752
0.06052	0.05935	-0.04339
0.07857	0.06813	-0.04908
0.09877	0.07659	-0.05453
0.12101	0.08460	-0.05961
0.14518	0.09205	-0.06425
0.17118	0.09880	-0.06839
0.19888	0.10474	-0.07193
0.22814	0.10972	-0.07471
0.25882	0.11359	-0.07666
0.29078	0.11624	-0.07769
0.32386	0.11758	-0.07768
0.35791	0.11754	-0.07647
0.39276	0.11616	-0.07394
0.42825	0.11355	-0.07012
0.46419	0.10985	-0.06505
0.50043	0.10516	-0.05881
0.53679	0.09963	-0.05173
0.57309	0.09341	-0.04418
0.60917	0.08660	-0.03644
0.64483	0.07934	-0.02883
0.67992	0.07183	-0.02162
0.71427	0.06420	-0.01504
0.74771	0.05660	-0.00924
0.78009	0.04916	-0.00431
0.81124	0.04200	-0.00033
0.84101	0.03523	0.00266
0.86927	0.02885	0.00466
0.89588	0.02293	0.00574
0.92071	0.01754	0.00597
0.94364	0.01265	0.00532
0.96457	0.00822	0.00385
0.98338	0.00425	0.00178
1.00000	0.00065	-0.00065

TABLE 10. Profile coordinates.

Airfoil:FFA-W3-211

x/c	yu/c	yl/c
0.00000	0.00000	0.00000
0.00126	0.00778	-0.00652
0.00503	0.01616	-0.01295
0.01130	0.02500	-0.01939
0.02003	0.03416	-0.02606
0.03118	0.04352	-0.03298
0.04470	0.05296	-0.04003
0.06052	0.06234	-0.04711
0.07857	0.07151	-0.05414
0.09877	0.08031	-0.06099
0.12101	0.08860	-0.06753
0.14518	0.09621	-0.07360
0.17118	0.10299	-0.07908
0.19888	0.10878	-0.08379
0.22814	0.11342	-0.08757
0.25882	0.11673	-0.09030
0.29078	0.11854	-0.09185
0.32386	0.11880	-0.09204
0.35791	0.11752	-0.09070
0.39276	0.11489	-0.08773
0.42825	0.11108	-0.08306
0.46419	0.10627	-0.07667
0.50043	0.10062	-0.06865
0.53679	0.09429	-0.05950
0.57309	0.08746	-0.04979
0.60917	0.08021	-0.03995
0.64483	0.07268	-0.03045
0.67992	0.06509	-0.02169
0.71427	0.05758	-0.01398
0.74771	0.05029	-0.00747
0.78009	0.04331	-0.00219
0.81124	0.03675	0.00178
0.84101	0.03066	0.00447
0.86927	0.02501	0.00600
0.89588	0.01984	0.00647
0.92071	0.01515	0.00599
0.94364	0.01096	0.00470
0.96457	0.00729	0.00284
0.98338	0.00414	0.00074
1.00000	0.00131	-0.00131

TABLE 11. Profile coordinates.

Airfoil:FFA-W3-241

x/c	yu/c	yl/c
0.00004	0.00160	0.00160
0.00130	0.00956	-0.00903
0.00507	0.02005	-0.01700
0.01133	0.03033	-0.02550
0.02006	0.04087	-0.03427
0.03122	0.05135	-0.04334
0.04474	0.06170	-0.05258
0.06056	0.07176	-0.06188
0.07861	0.08139	-0.07104
0.09880	0.09044	-0.07985
0.12104	0.09873	-0.08811
0.14522	0.10613	-0.09560
0.17121	0.11247	-0.10213
0.19891	0.11764	-0.10754
0.22817	0.12153	-0.11167
0.25885	0.12409	-0.11441
0.29081	0.12526	-0.11567
0.32389	0.12504	-0.11531
0.35794	0.12350	-0.11324
0.39279	0.12076	-0.10938
0.42827	0.11696	-0.10370
0.46422	0.11223	-0.09623
0.50045	0.10670	-0.08706
0.53681	0.10052	-0.07651
0.57311	0.09382	-0.06502
0.60918	0.08672	-0.05308
0.64485	0.07935	-0.04127
0.67994	0.07187	-0.03015
0.71428	0.06441	-0.02024
0.74772	0.05707	-0.01183
0.78010	0.04996	-0.00514
0.81124	0.04316	-0.00030
0.84102	0.03673	0.00291
0.86928	0.03070	0.00473
0.89589	0.02510	0.00528
0.92072	0.01996	0.00466
0.94365	0.01529	0.00311
0.96457	0.01107	0.00096
0.98338	0.00727	-0.00140
1.00000	0.00391	-0.00360

TABLE 12. Profile coordinates.

Airfoil:FFA-W3-270

x/c	yu/c	yl/c
0.00005	0.00209	0.00209
0.00131	0.01074	-0.01038
0.00508	0.02251	-0.02044
0.01135	0.03420	-0.03030
0.02009	0.04593	-0.04067
0.03125	0.05757	-0.05129
0.04478	0.06896	-0.06215
0.06062	0.07996	-0.07304
0.07869	0.09043	-0.08375
0.09890	0.10018	-0.09400
0.12115	0.10904	-0.10356
0.14535	0.11683	-0.11215
0.17137	0.12342	-0.11957
0.19909	0.12868	-0.12563
0.22837	0.13253	-0.13019
0.25908	0.13495	-0.13312
0.29107	0.13592	-0.13432
0.32418	0.13548	-0.13371
0.35825	0.13370	-0.13121
0.39313	0.13070	-0.12675
0.42864	0.12658	-0.12032
0.46462	0.12147	-0.11196
0.50089	0.11550	-0.10181
0.53728	0.10882	-0.09010
0.57361	0.10156	-0.07723
0.60971	0.09388	-0.06375
0.64541	0.08594	-0.05030
0.68053	0.07788	-0.03755
0.71490	0.06985	-0.02610
0.74837	0.06197	-0.01636
0.78077	0.05434	-0.00859
0.81195	0.04704	-0.00298
0.84175	0.04014	0.00076
0.87003	0.03368	0.00299
0.89666	0.02770	0.00377
0.92151	0.02220	0.00324
0.94446	0.01721	0.00168
0.96540	0.01269	-0.00052
0.98423	0.00858	-0.00295
1.00086	0.00495	-0.00517

TABLE 13. Profile coordinates.

Airfoil:FFA-W3-301

x/c	yu/c	yl/c
0.00001	-0.00119	-0.00119
0.00127	0.01326	-0.01256
0.00504	0.02667	-0.02502
0.01131	0.04001	-0.03764
0.02004	0.05339	-0.05035
0.03119	0.06655	-0.06312
0.04471	0.07938	-0.07582
0.06053	0.09167	-0.08825
0.07859	0.10323	-0.10018
0.09878	0.11385	-0.11136
0.12102	0.12332	-0.12155
0.14519	0.13146	-0.13052
0.17119	0.13814	-0.13810
0.19889	0.14325	-0.14411
0.22814	0.14673	-0.14847
0.25883	0.14866	-0.15108
0.29079	0.14908	-0.15188
0.32387	0.14807	-0.15080
0.35792	0.14578	-0.14780
0.39277	0.14230	-0.14283
0.42825	0.13776	-0.13591
0.46420	0.13229	-0.12710
0.50044	0.12601	-0.11652
0.53680	0.11909	-0.10442
0.57310	0.11169	-0.09116
0.60917	0.10391	-0.07722
0.64484	0.09587	-0.06318
0.67993	0.08766	-0.04966
0.71428	0.07939	-0.03722
0.74772	0.07116	-0.02631
0.78009	0.06309	-0.01715
0.81124	0.05528	-0.01002
0.84101	0.04780	-0.00479
0.86927	0.04074	-0.00134
0.89588	0.03417	0.00040
0.92071	0.02812	0.00058
0.94364	0.02266	-0.00051
0.96457	0.01775	-0.00261
0.98338	0.01333	-0.00558
1.00000	0.00937	-0.00891

TABLE 14. Profile coordinates.

Airfoil:FFA-W3-332

x/c	yu/c	yl/c
0.00001	0.00146	0.00146
0.00127	0.01519	-0.01525
0.00504	0.03032	-0.03041
0.01131	0.04539	-0.04555
0.02004	0.06035	-0.06064
0.03119	0.07505	-0.07553
0.04470	0.08930	-0.09006
0.06053	0.10288	-0.10402
0.07858	0.11553	-0.11717
0.09877	0.12701	-0.12927
0.12100	0.13709	-0.14010
0.14518	0.14557	-0.14945
0.17117	0.15233	-0.15717
0.19886	0.15729	-0.16315
0.22812	0.16044	-0.16731
0.25880	0.16185	-0.16959
0.29076	0.16162	-0.16998
0.32384	0.15989	-0.16843
0.35788	0.15680	-0.16493
0.39273	0.15253	-0.15946
0.42821	0.14729	-0.15205
0.46415	0.14117	-0.14277
0.50038	0.13430	-0.13177
0.53674	0.12683	-0.11928
0.57304	0.11890	-0.10563
0.60910	0.11064	-0.09123
0.64476	0.10219	-0.07660
0.67985	0.09366	-0.06230
0.71420	0.08514	-0.04888
0.74763	0.07675	-0.03683
0.78000	0.06855	-0.02624
0.81115	0.06064	-0.01761
0.84092	0.05308	-0.01088
0.86918	0.04593	-0.00620
0.89578	0.03926	-0.00351
0.92061	0.03309	-0.00262
0.94354	0.02745	-0.00324
0.96446	0.02229	-0.00524
0.98327	0.01754	-0.00875
0.99989	0.01325	-0.01319

TABLE 15. Profile coordinates.

Airfoil:FFA-W3-360

x/c	yu/c	yl/c
0.00012	0.00612	0.00612
0.00138	0.01755	-0.01459
0.00515	0.04140	-0.03624
0.01141	0.05950	-0.05330
0.02014	0.07716	-0.07006
0.03129	0.09405	-0.08587
0.04481	0.11001	-0.10064
0.06063	0.12474	-0.11440
0.07868	0.13804	-0.12695
0.09887	0.14982	-0.13821
0.12111	0.15996	-0.14818
0.14528	0.16844	-0.15683
0.17128	0.17512	-0.16402
0.19897	0.17994	-0.16970
0.22823	0.18291	-0.17375
0.25891	0.18414	-0.17603
0.29086	0.18370	-0.17643
0.32394	0.18171	-0.17488
0.35799	0.17825	-0.17134
0.39283	0.17347	-0.16578
0.42831	0.16749	-0.15816
0.46426	0.16048	-0.14848
0.50049	0.15262	-0.13705
0.53685	0.14408	-0.12416
0.57314	0.13505	-0.11009
0.60921	0.12568	-0.09522
0.64487	0.11611	-0.08000
0.67996	0.10647	-0.06500
0.71431	0.09687	-0.05083
0.74774	0.08740	-0.03807
0.78011	0.07815	-0.02713
0.81126	0.06920	-0.01816
0.84103	0.06063	-0.01120
0.86929	0.05251	-0.00636
0.89589	0.04489	-0.00361
0.92072	0.03783	-0.00273
0.94365	0.03137	-0.00342
0.96457	0.02543	-0.00555
0.98339	0.01997	-0.00928
1.00000	0.01503	-0.01393

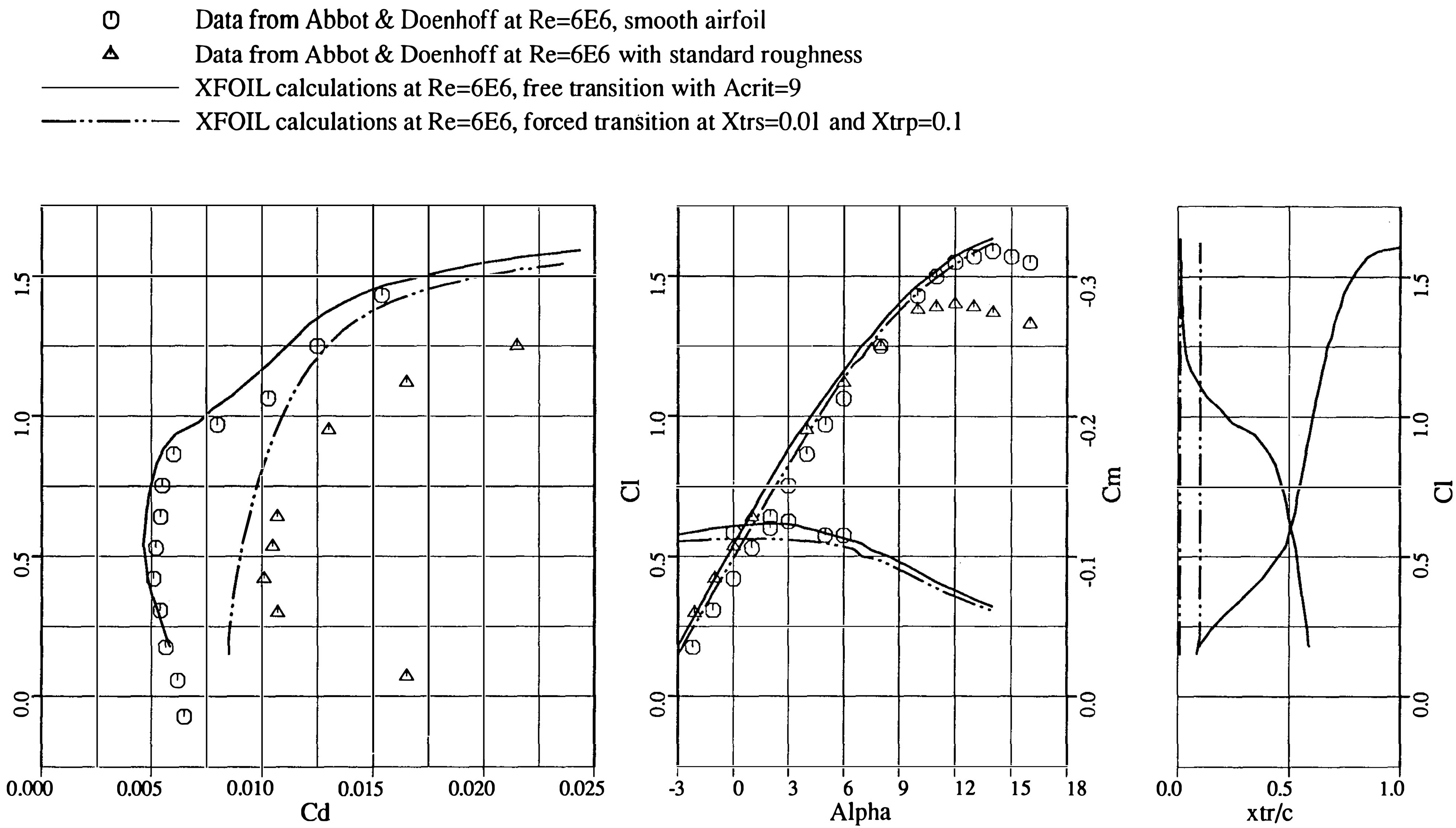


Fig.1 XFOIL calculations compared with wind tunnel test results from [9] for the NACA 63-615 airfoil.

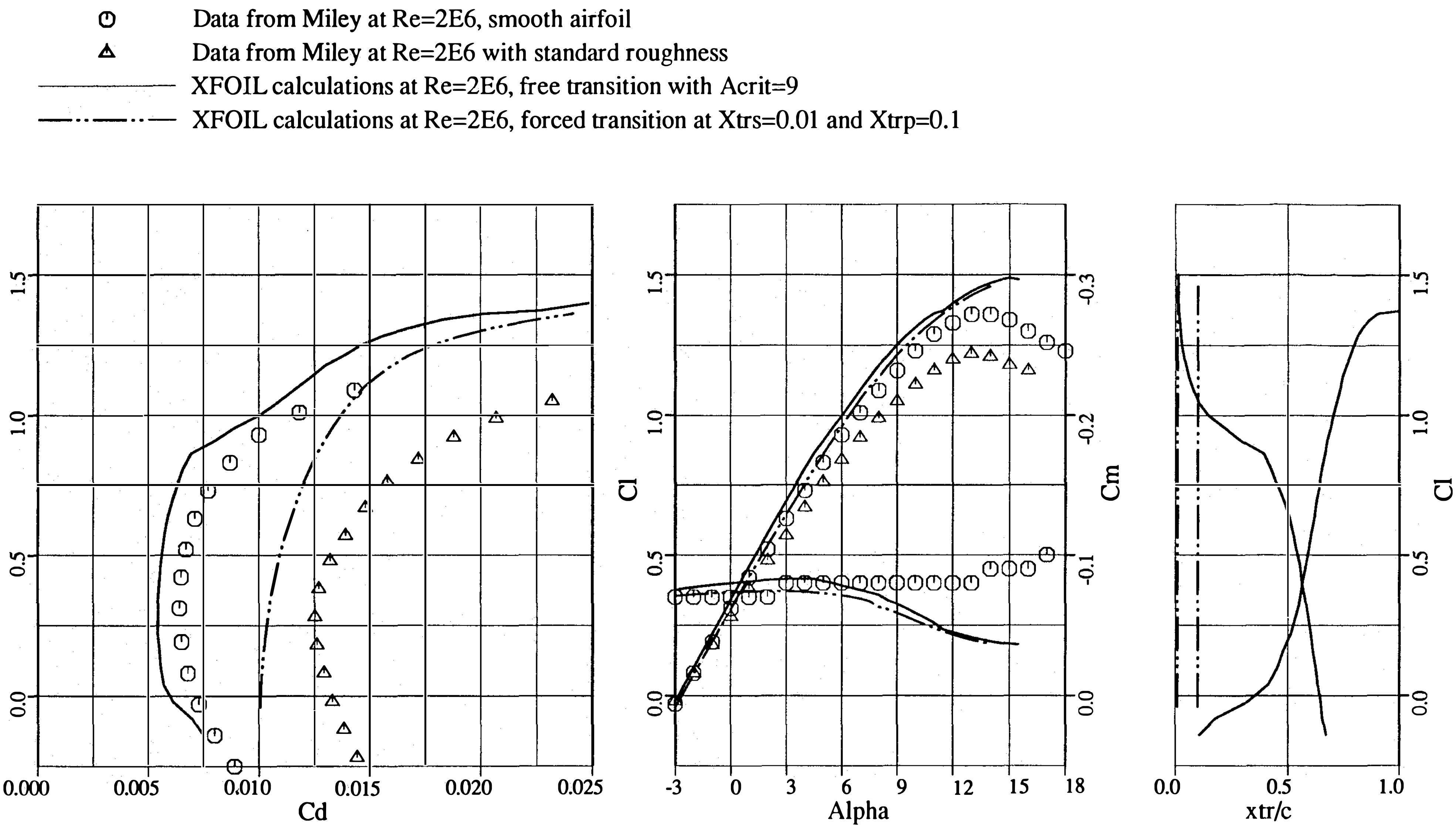


Fig.2 XFOIL calculations compared with wind tunnel test results from [11] for the NACA 63-615 airfoil.

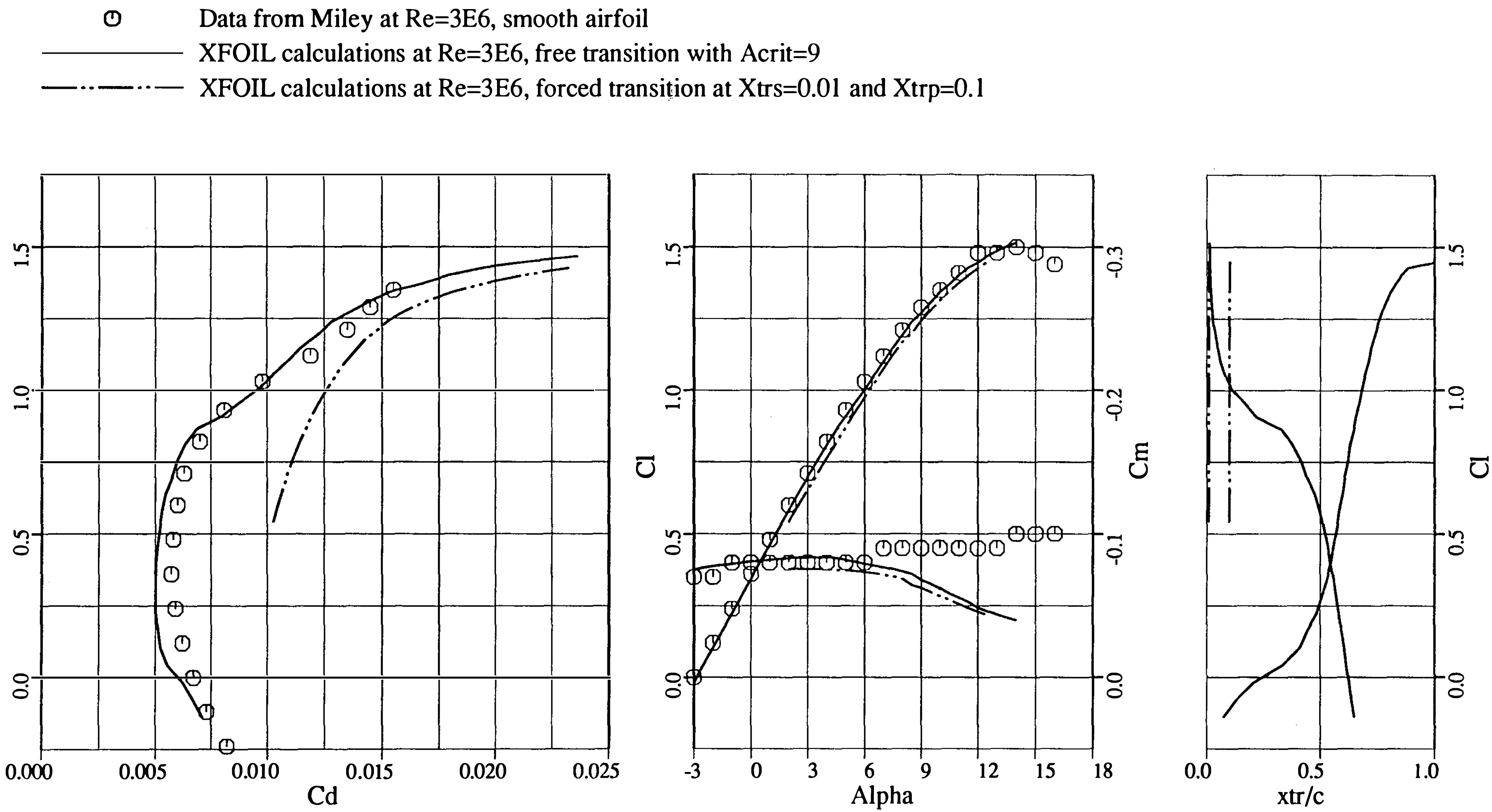


Fig.3 XFOIL calculations compared with wind tunnel test results from [11] for the NACA 63-415 airfoil.

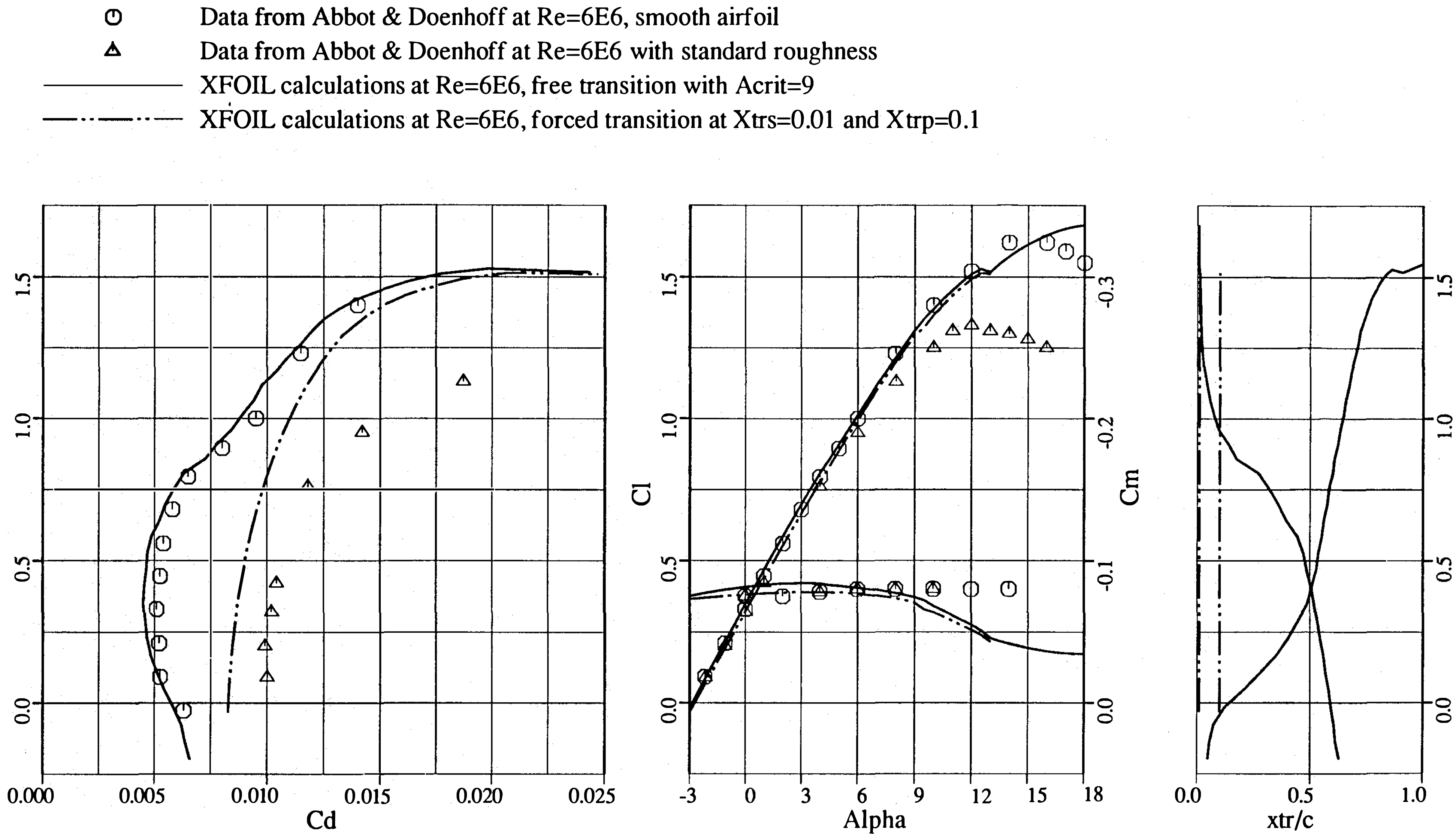


Fig.4 XFOIL calculations compared with wind tunnel test results from [9] for the NACA 63-415 airfoil.

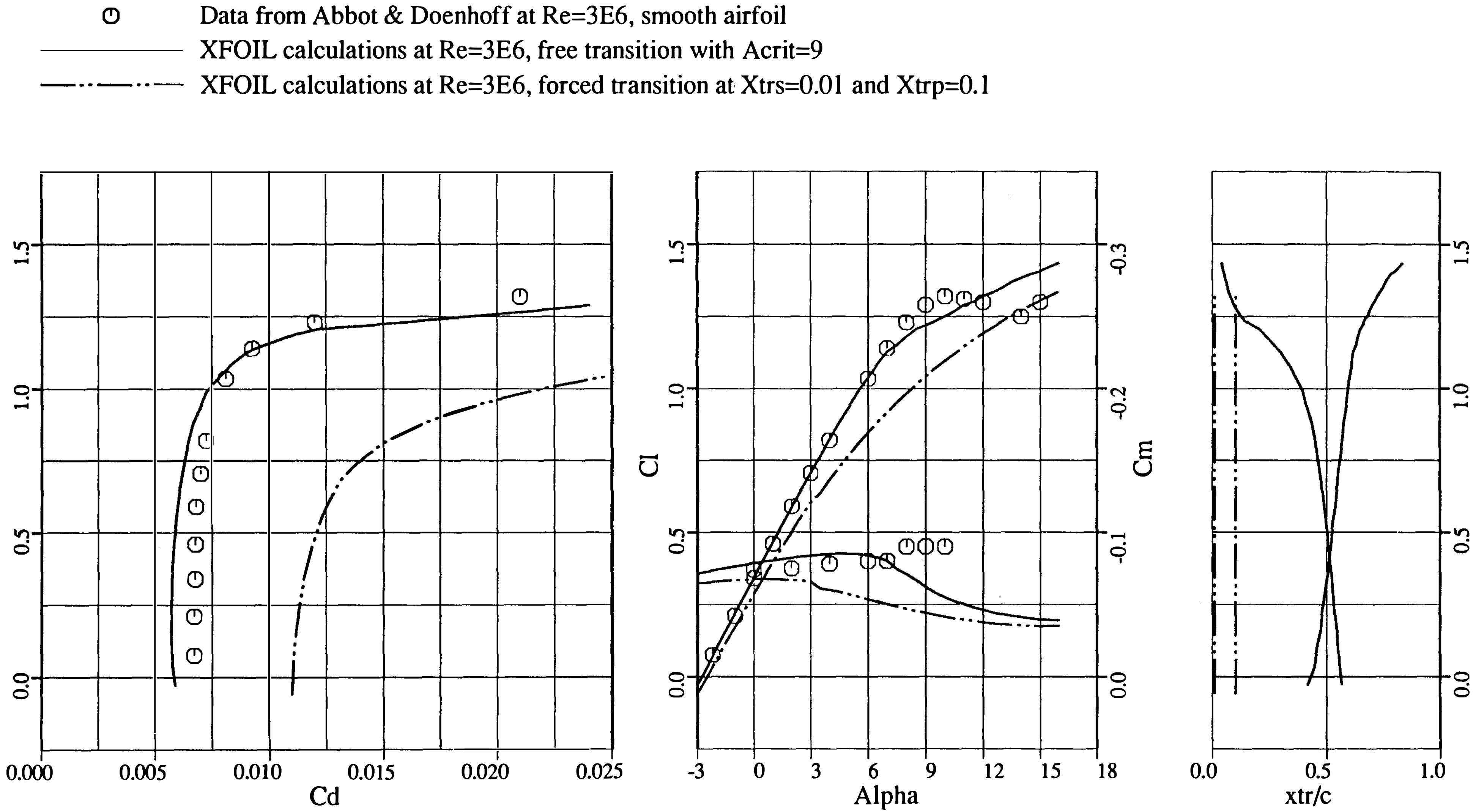


Fig.5 XFOIL calculations compared with wind tunnel test results from [9] for the NACA 63-421 airfoil.

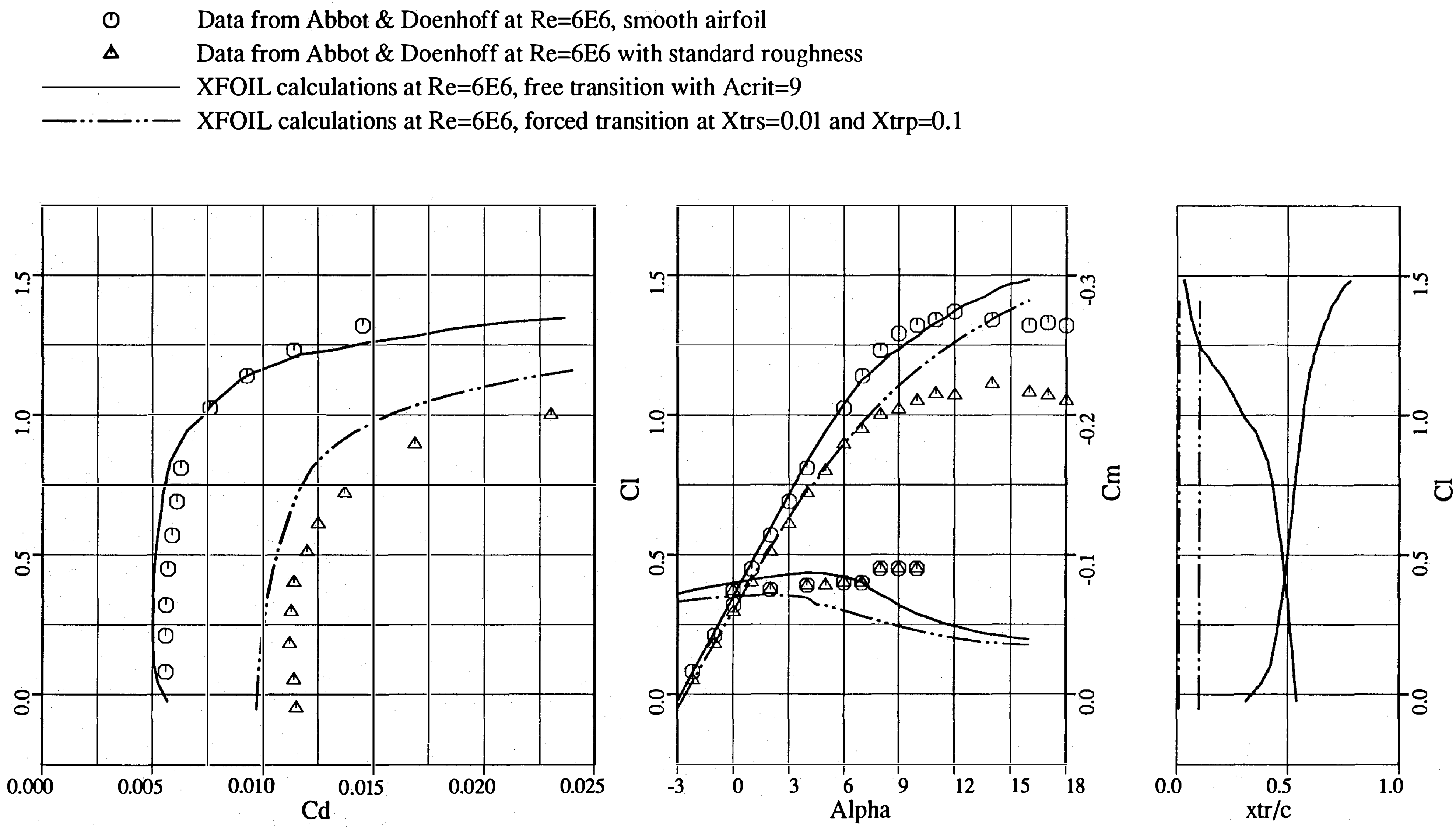


Fig.6 XFOIL calculations compared with wind tunnel test results from [9] for the NACA 63-421 airfoil.

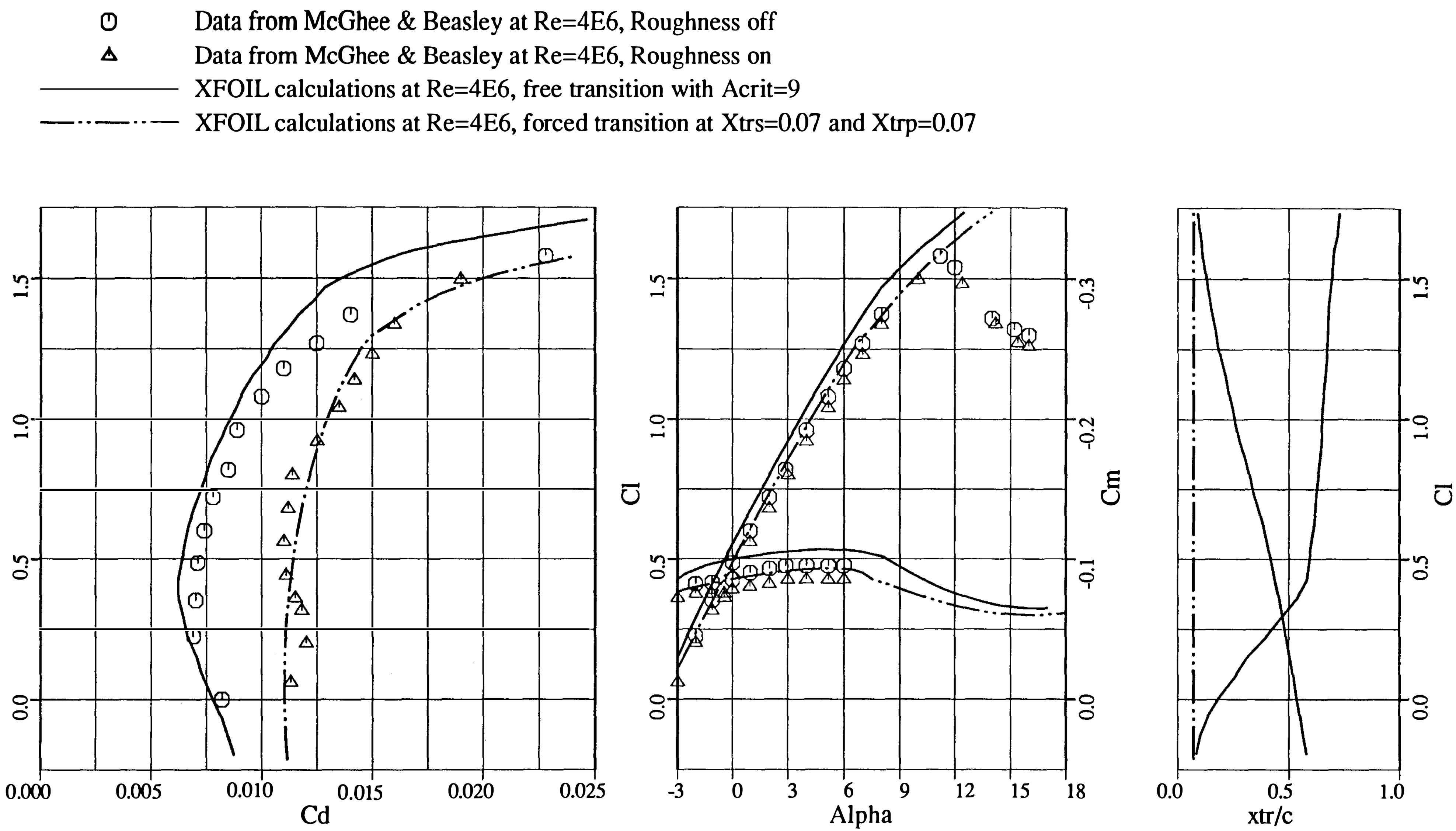


Fig.7 XFOIL calculations compared with wind tunnel test results from [10] for the NASA LS(1)-0421 MOD airfoil.

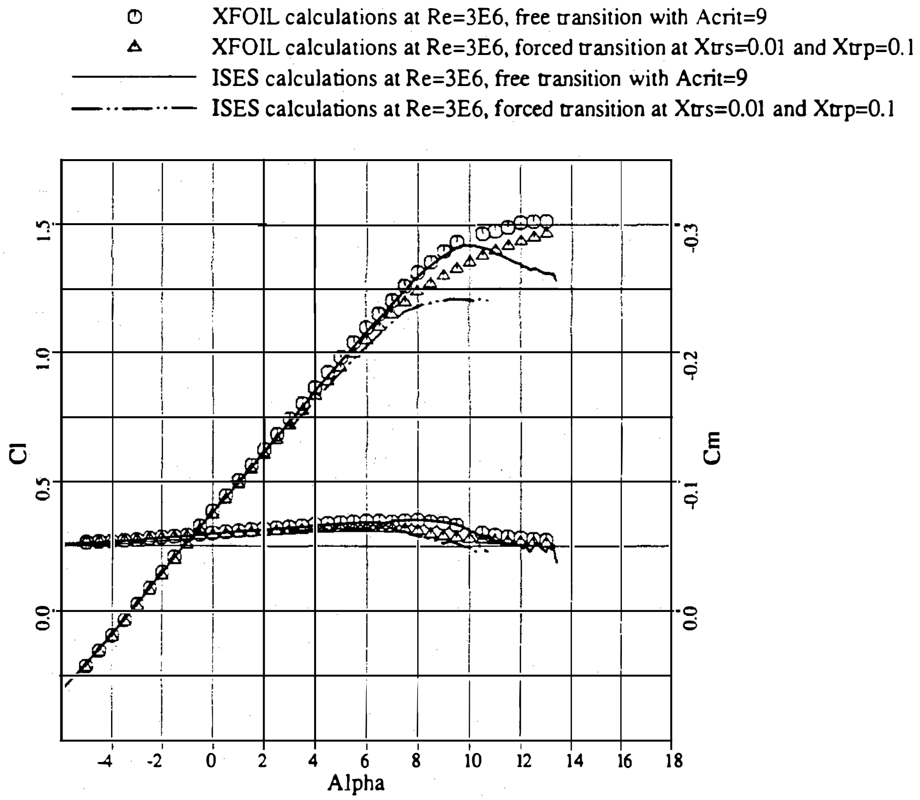


Fig.8 Comparison between ISES and XFOIL calculations for the AK15v4 airfoil (essentially the same airfoil as the FFA-W1-152).

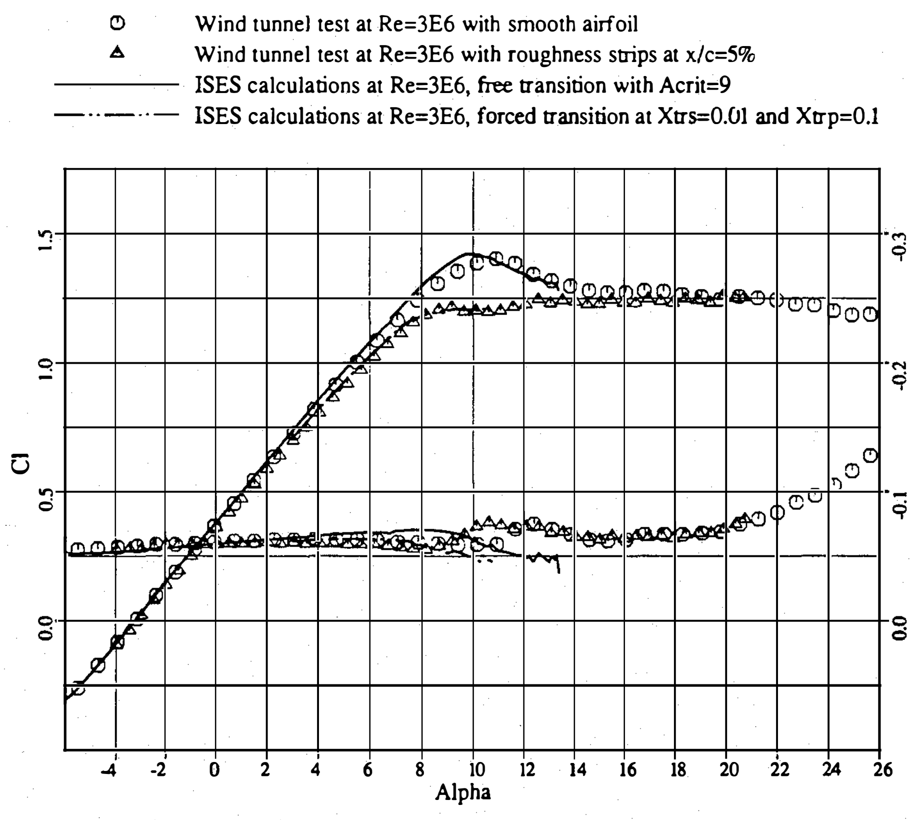


Fig.9 Comparison between ISES and wind tunnel results [3] for the AK15v4 airfoil.

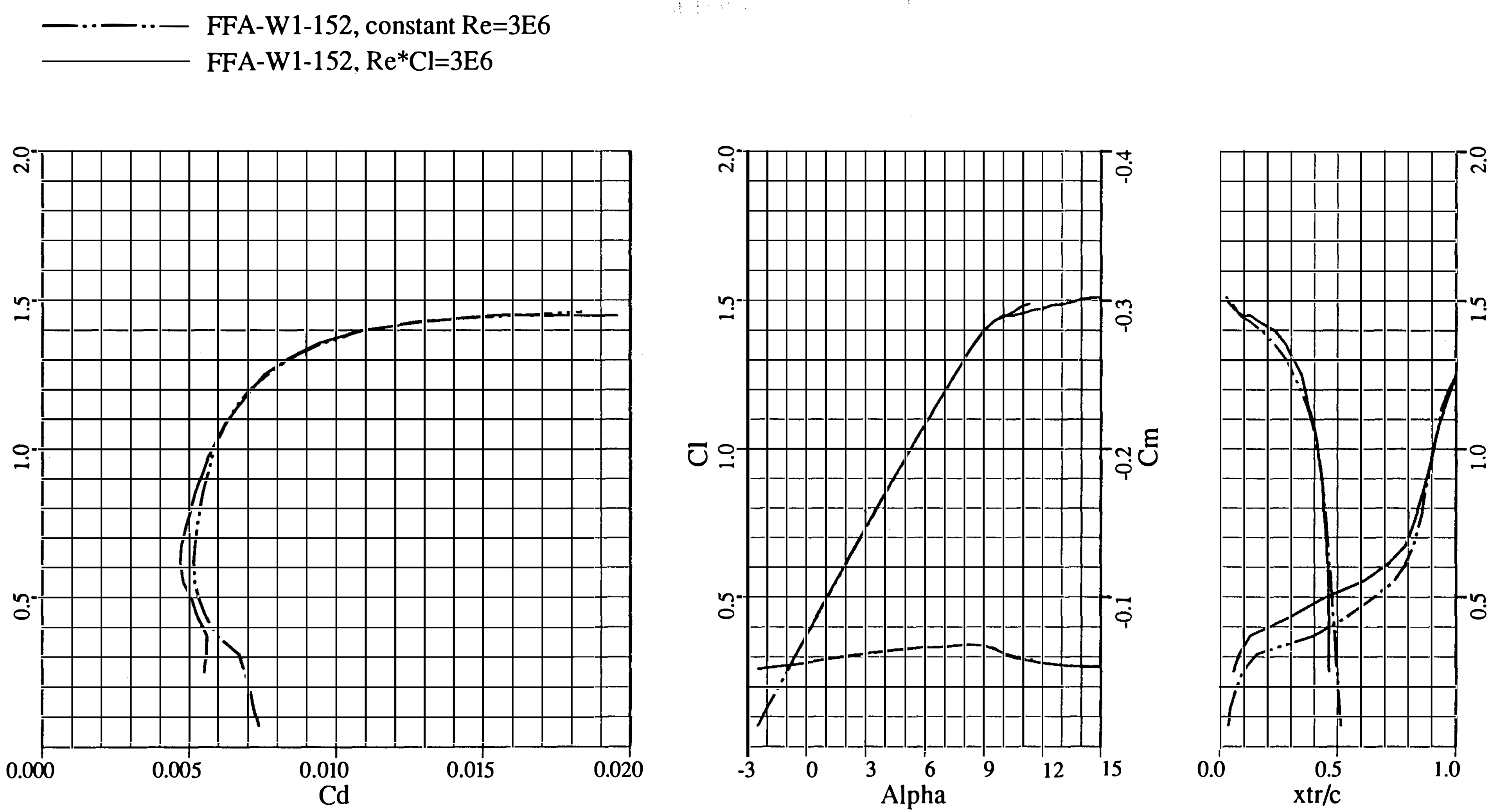


Fig.10a XFOIL calculations for the FFA-W1-152 airfoil at a constant Re=3 million and calculations at Re×C_L=3 million.

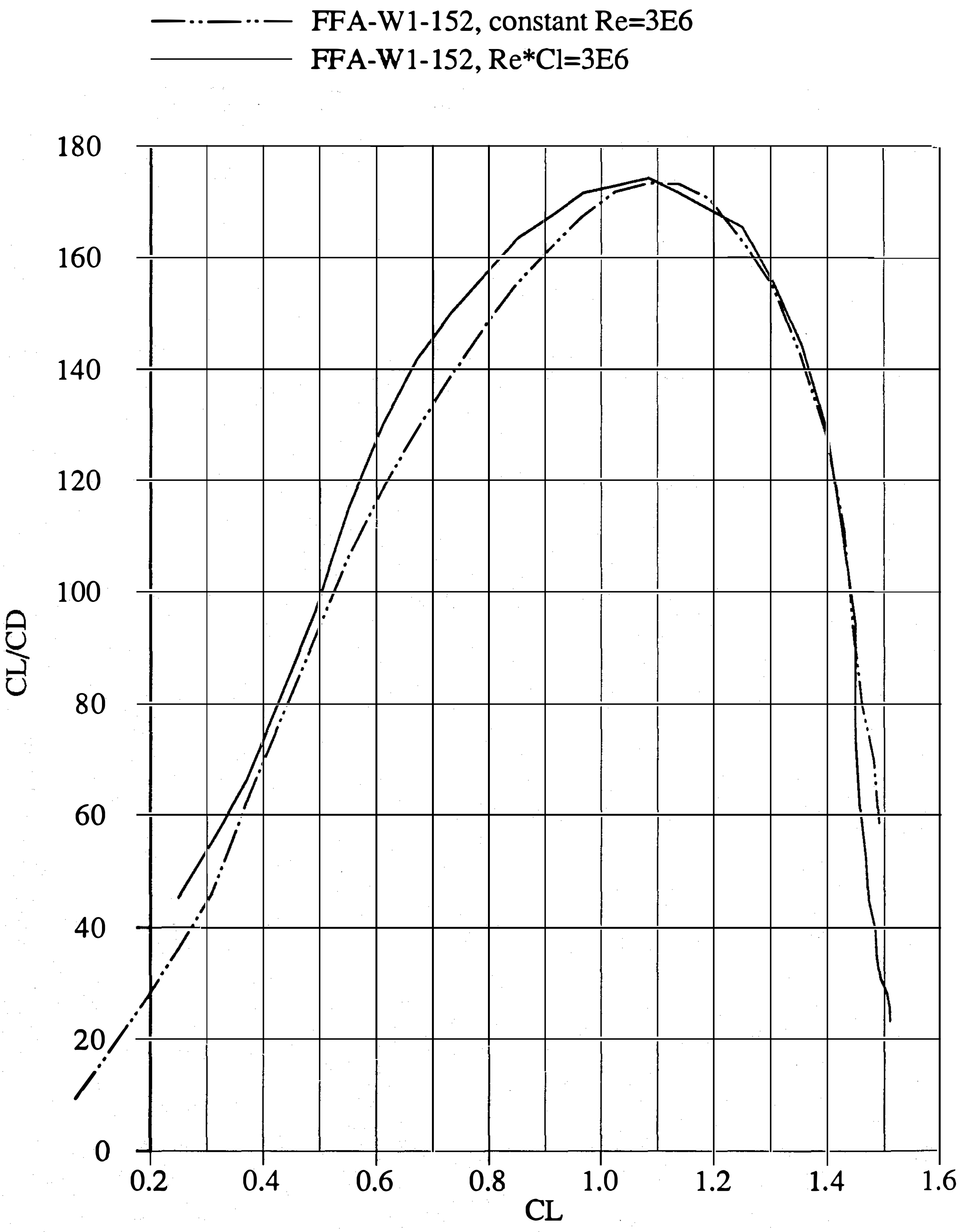


Fig.10b The lift/drag ratio versus C_l. XFOIL calculations for the FFA-W1-152 at a constant Re=3 million and calculations at $Re \times C_l$=3 million.

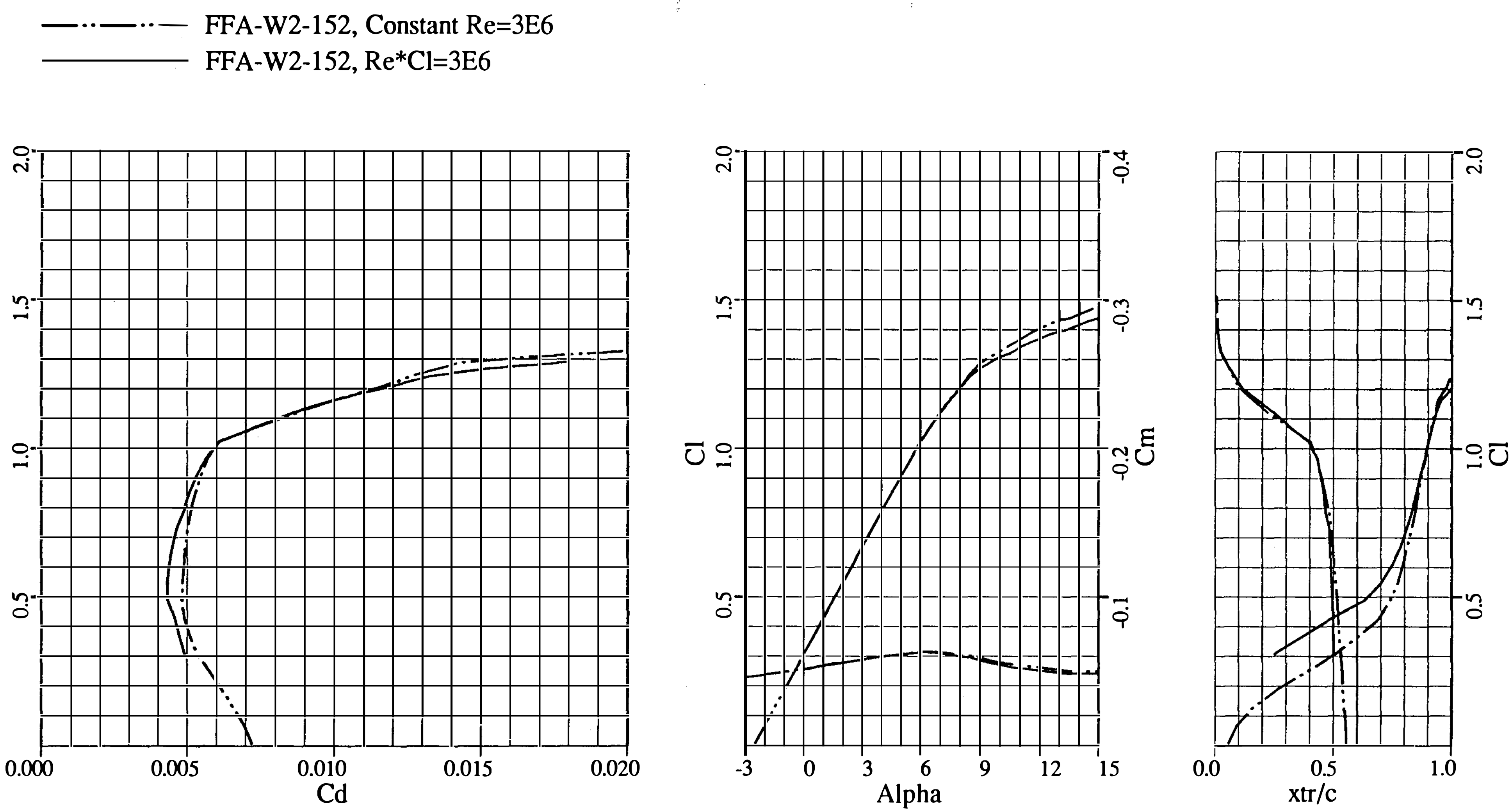


Fig.11a XFOIL calculations for the FFA-W2-152 airfoil at a constant Re=3 million and calculations at Re×C_l=3 million.

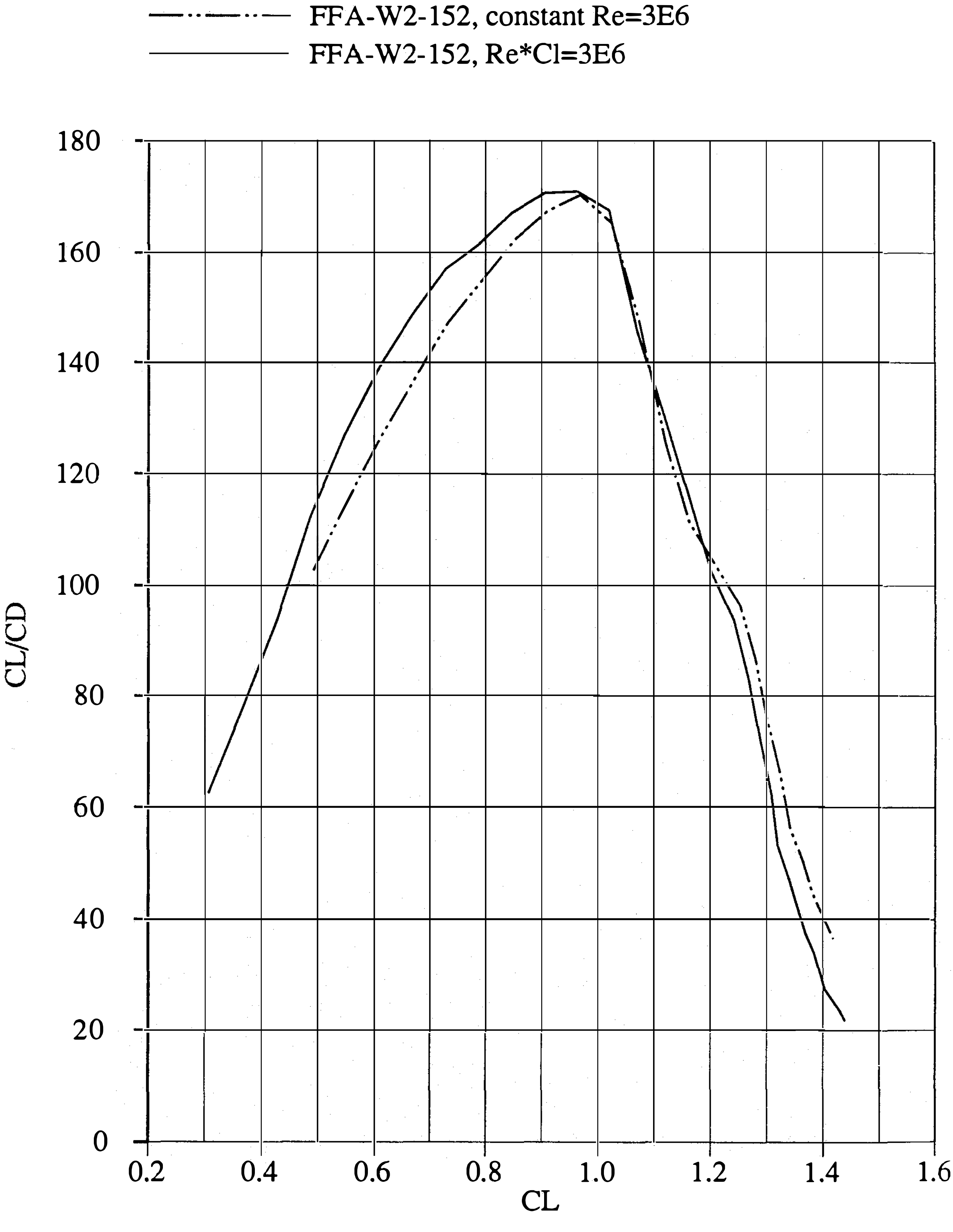


Fig.11b The lift/drag ratio versus C_l. XFOIL calculations for the FFA-W2-152 airfoil at a constant Re=3 million and calculationd at $Re \times C_l$=3 million.

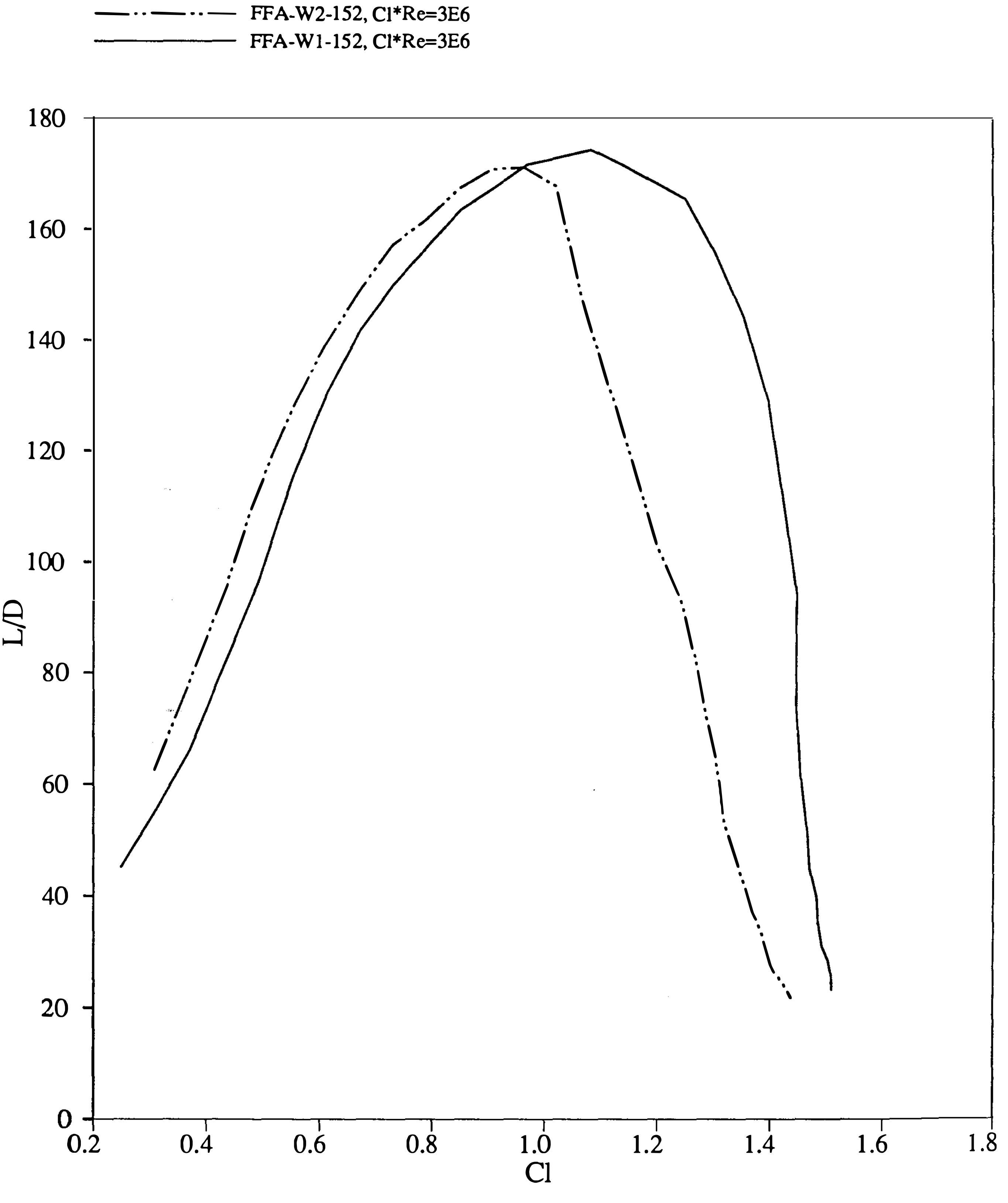


Fig.12 The lift/drag ratio versus C_l for the FFA-W1-152 and the FFA-W2-152 airfoil when calculations are made at $Re \times C_l$=3 million.

Fig.13 The FFA-W1-152 and FFA-W2-152 airfoils plotted superimposed on each other.

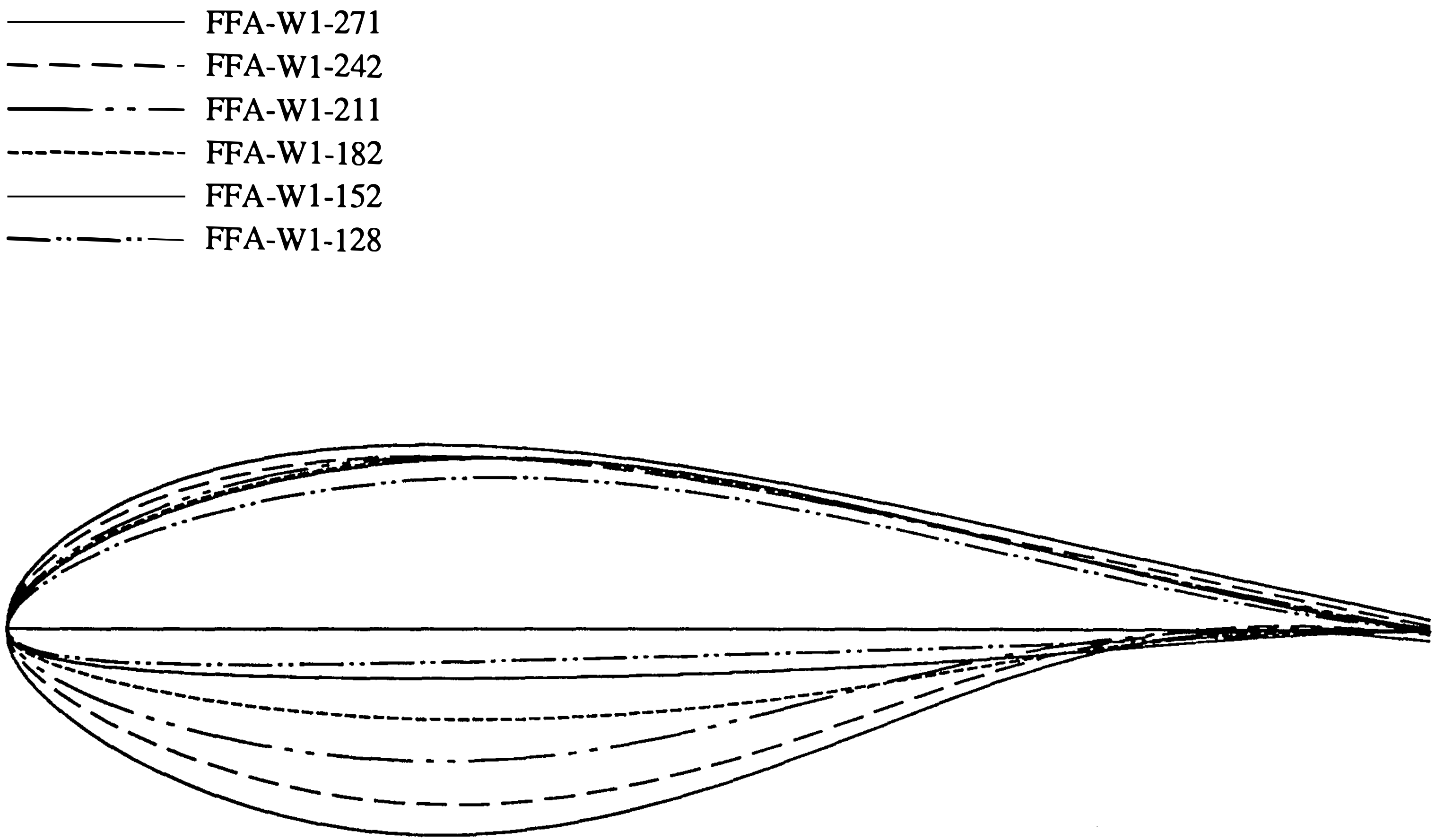


Fig.14 The FFA-W1-xxx airfoils plotted superimposed on each other.

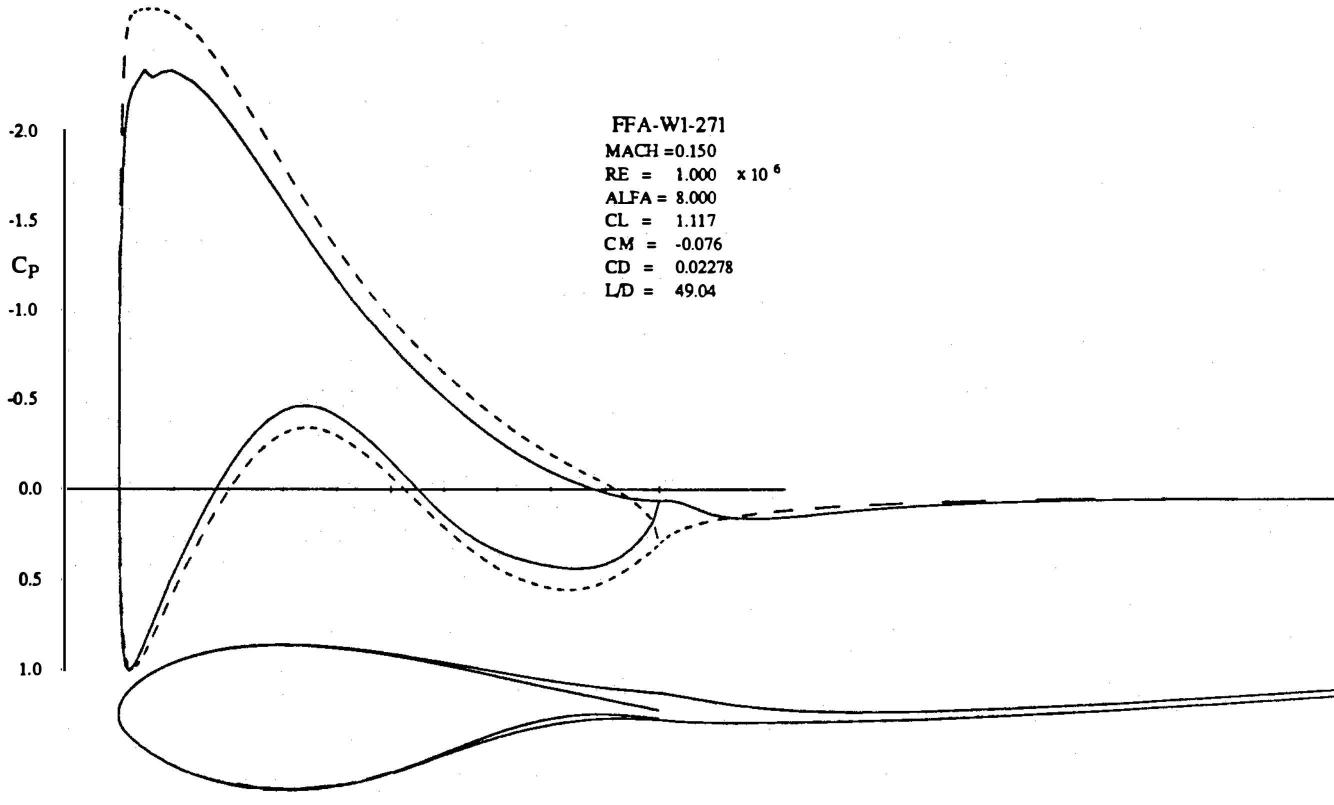


Fig.15a The FFA-W1-271 airfoil. Calculations at an angle of attack slightly above the design angle of attack. Re=1 million and the transition is forced at x/c=5% at both sides of the airfoil.

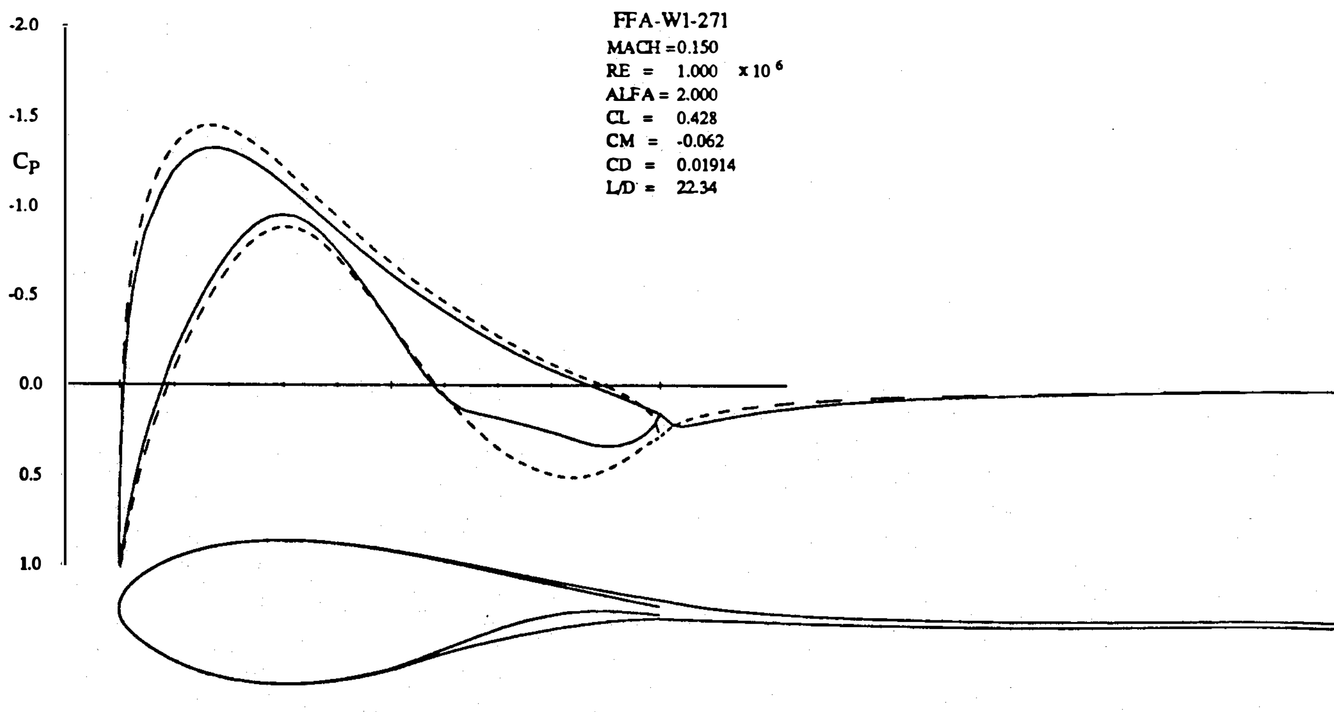


Fig.15b Calculations at a low angle of attack at Re=3 million and with forced transition at 5% x/c on both sides of the airfoil.

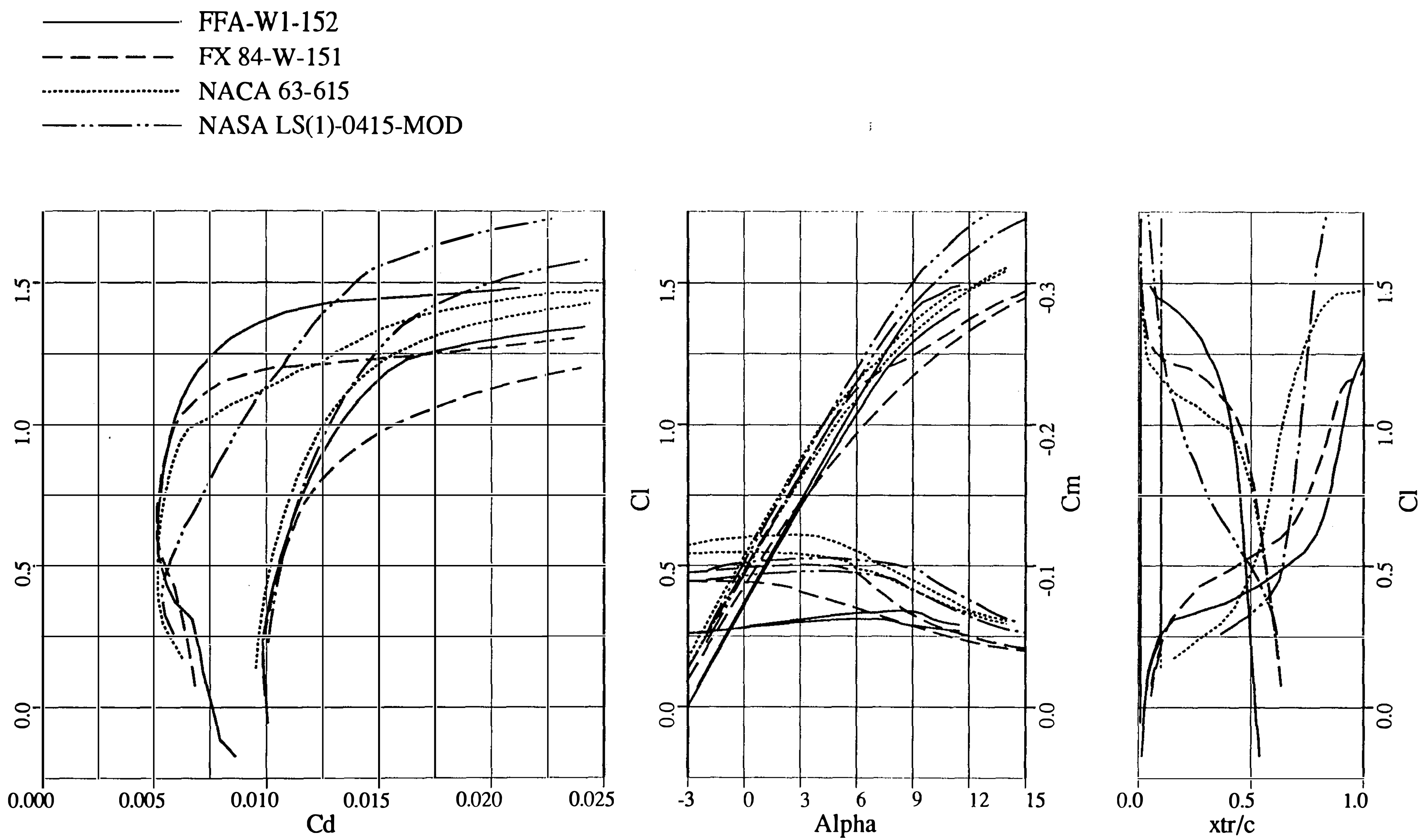


Fig.16 XFOIL calculations for four ≈15% thick airfoils. Re=3 million, and M=0.15.

1) Free transition and Acrit=9.
2) Forced transition at x/c=1% at the suction side and x/c=10% at the pressure side.

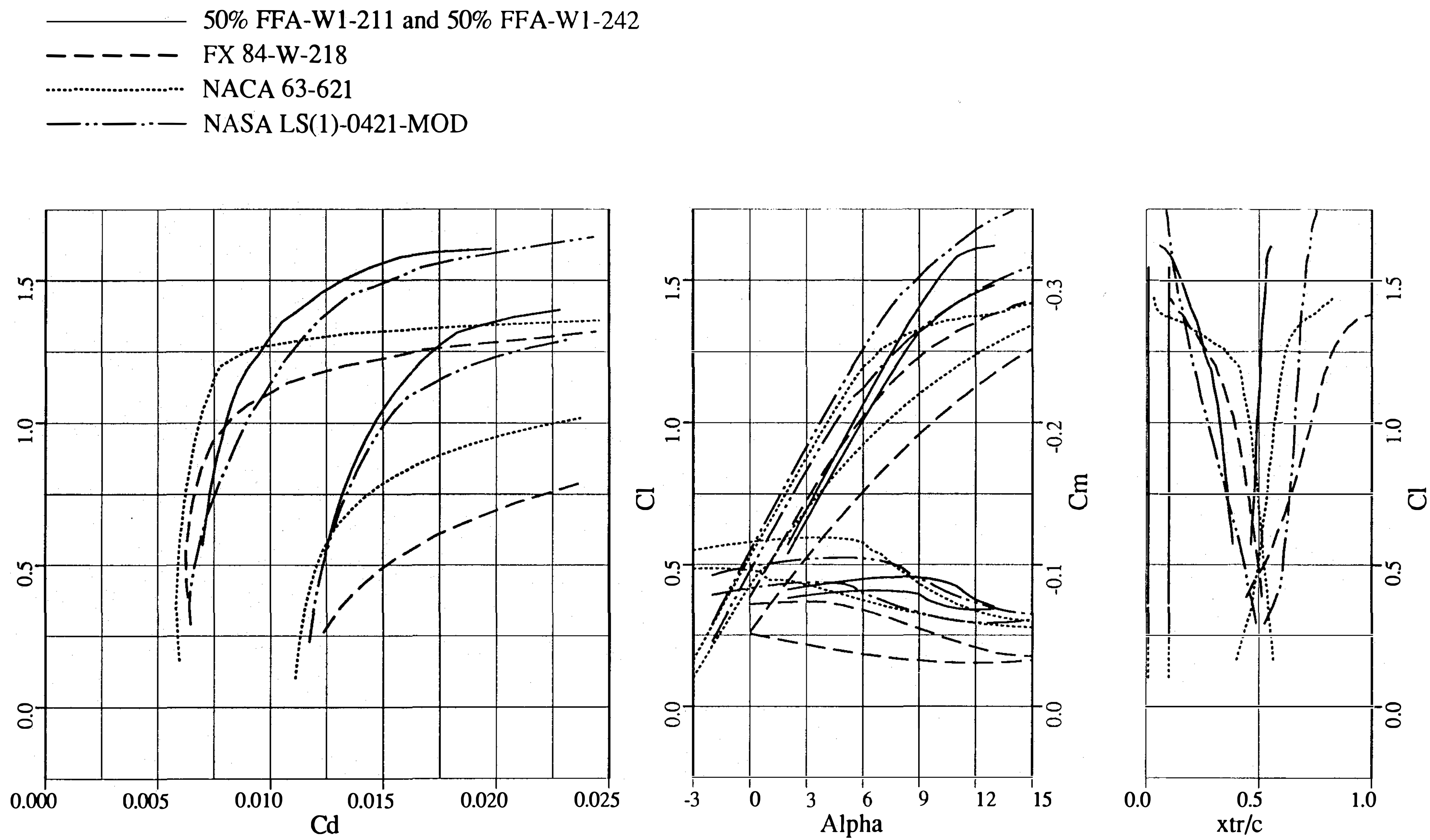
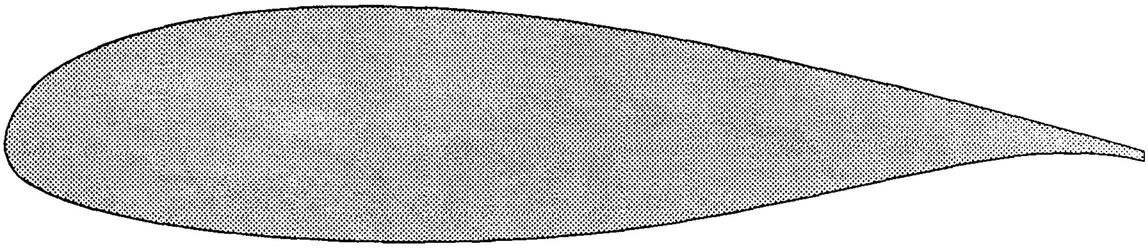
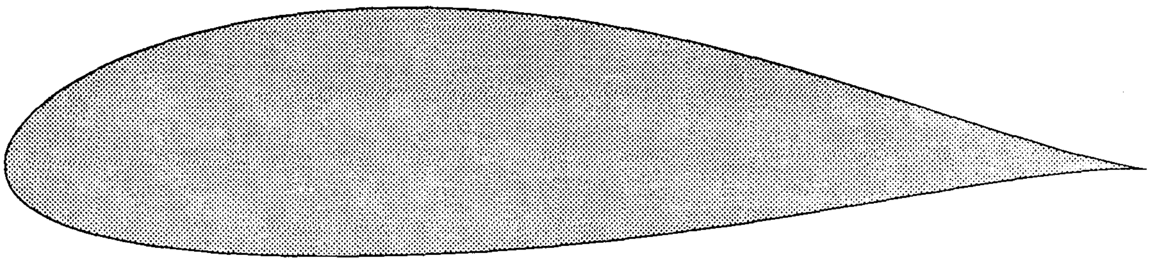


Fig.17 XFOIL calculations for four ≈21% thick airfoils. Re=3 million, M=0.15.

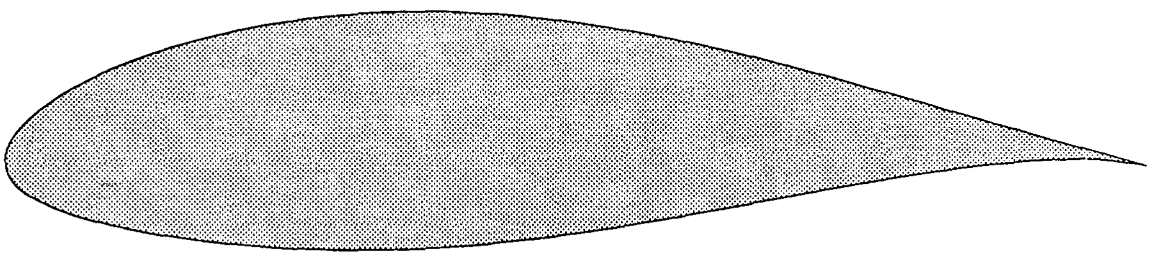
1) Free transition and Acrit=9.
2) Forced transition at x/c=1% at the suction side and at x/c=10% at the pressure side.



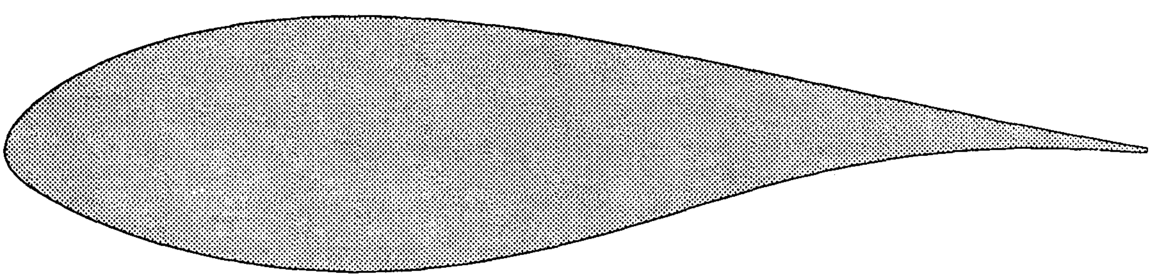
LS(1)-0421 MOD



FX 84-W-218



NACA 63-621



Interpolated airfoil, 50% FFA-W1-211 and 50% FFA-W1-242

Fig.18 The four airfoils used in the calculations presented in Fig.17.

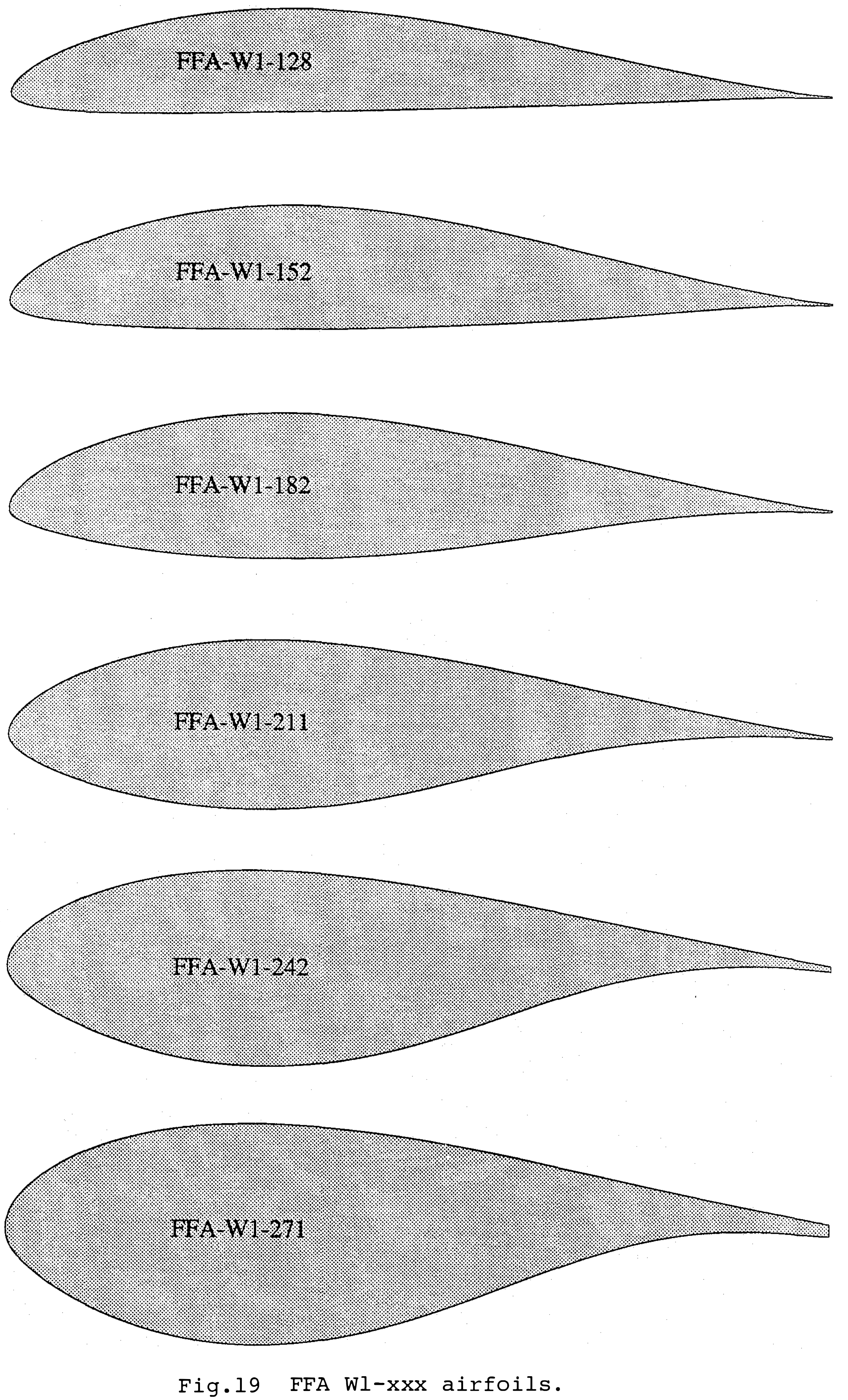


Fig.19 FFA W1-xxx airfoils.

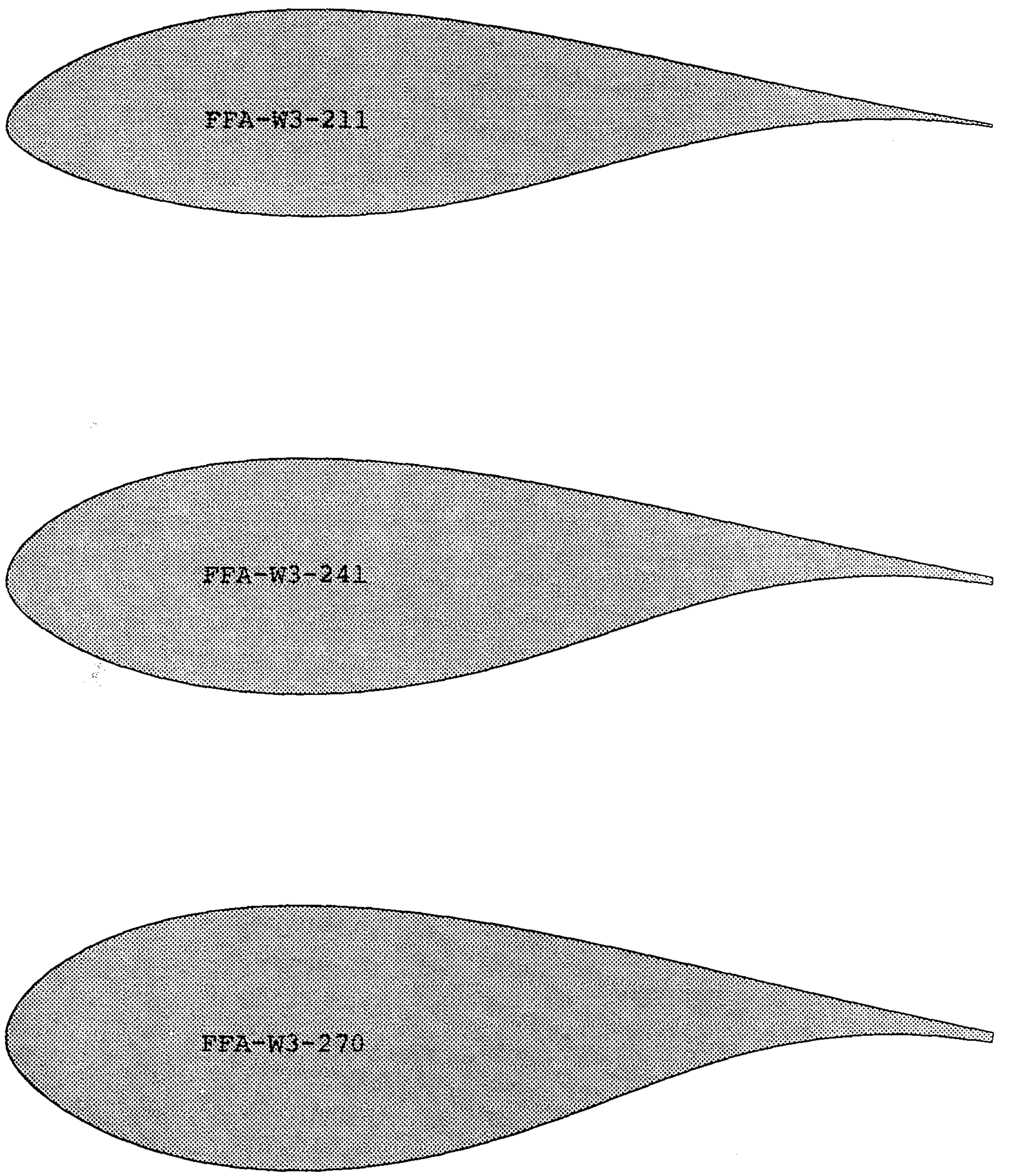


Fig.20 FFA-W3-xxx airfoils.

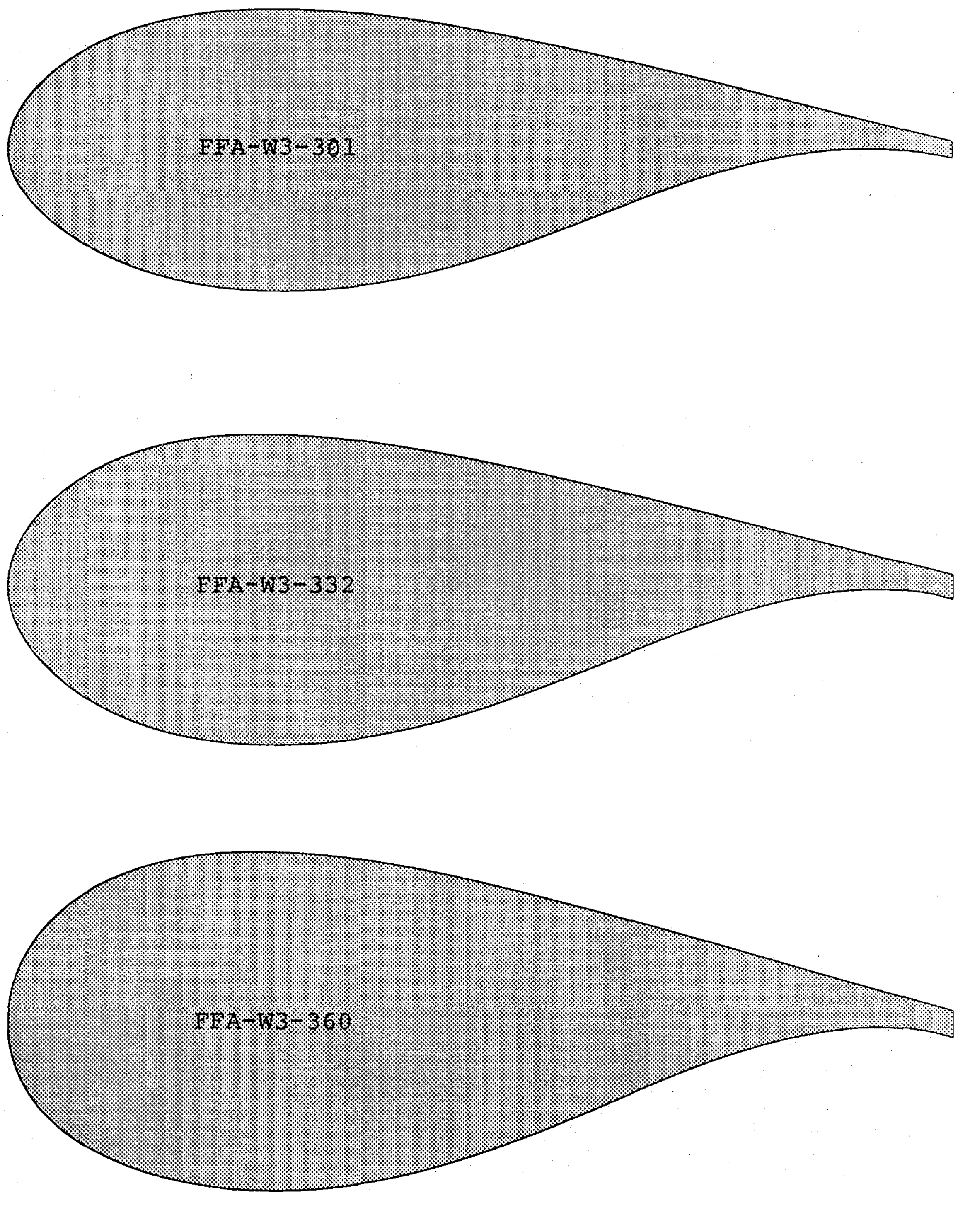


Fig.21 FFA-W3-xxx airfoils.

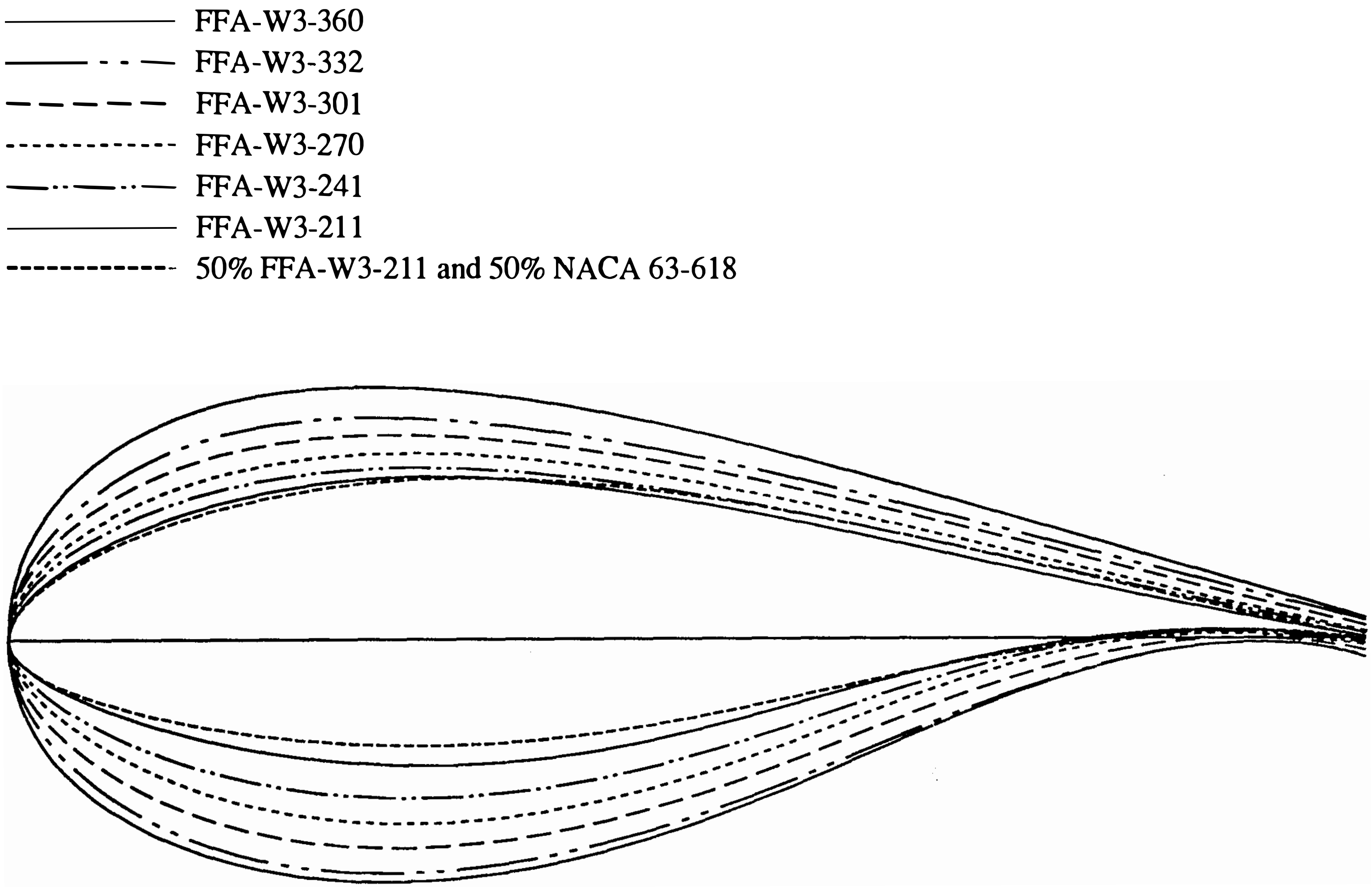


Fig.22 The FFA-W3-xxx plotted superimposed on each other.

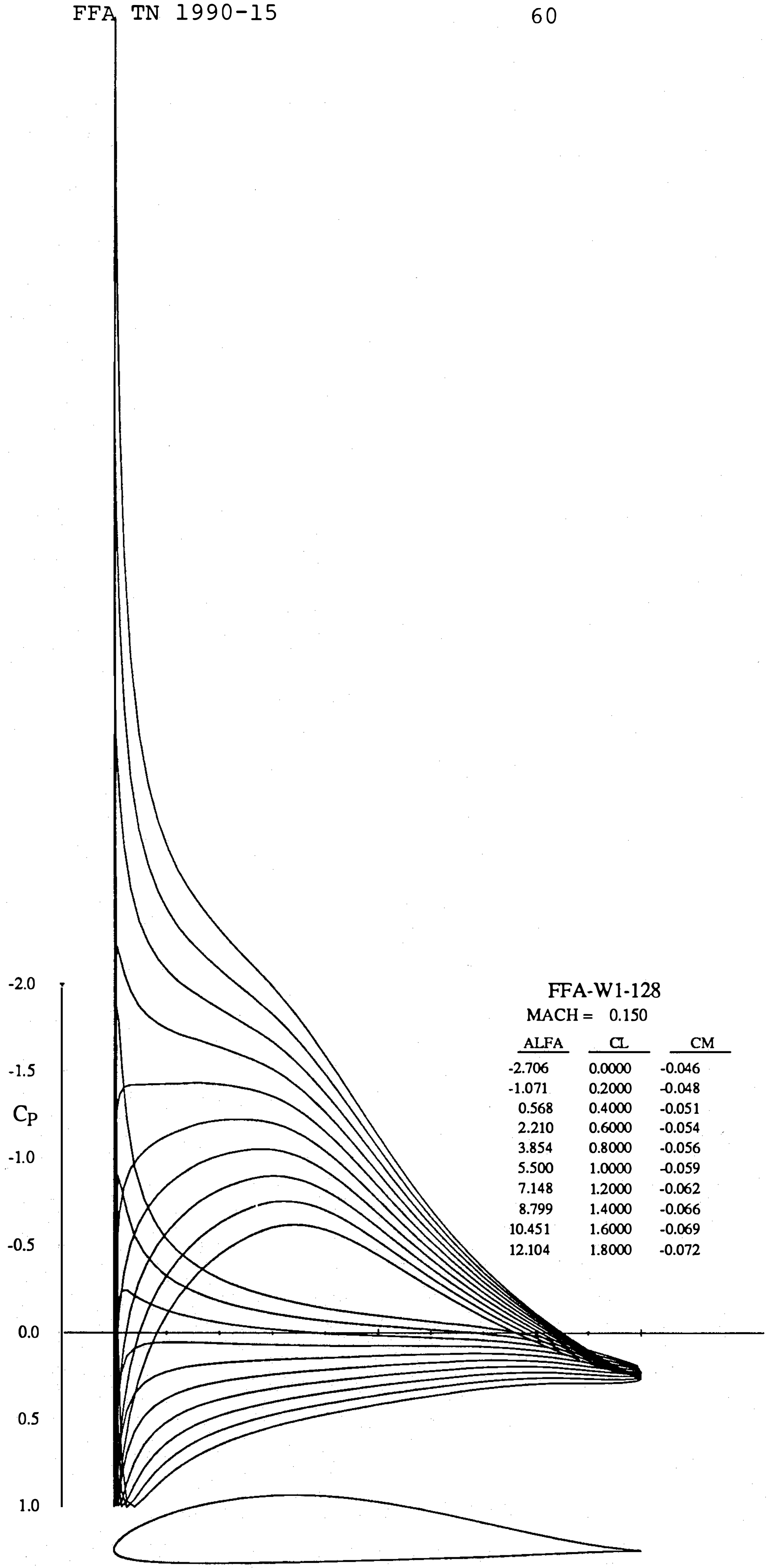

-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0
FFA-W1-128
MACH = 0.150
ALFA CL CM
-2.706 0.0000 -0.046
-1.071 0.2000 -0.048
0.568 0.4000 -0.051
2.210 0.6000 -0.054
3.854 0.8000 -0.056
5.500 1.0000 -0.059
7.148 1.2000 -0.062
8.799 1.4000 -0.066
10.451 1.6000 -0.069
12.104 1.8000 -0.072

Fig.23a

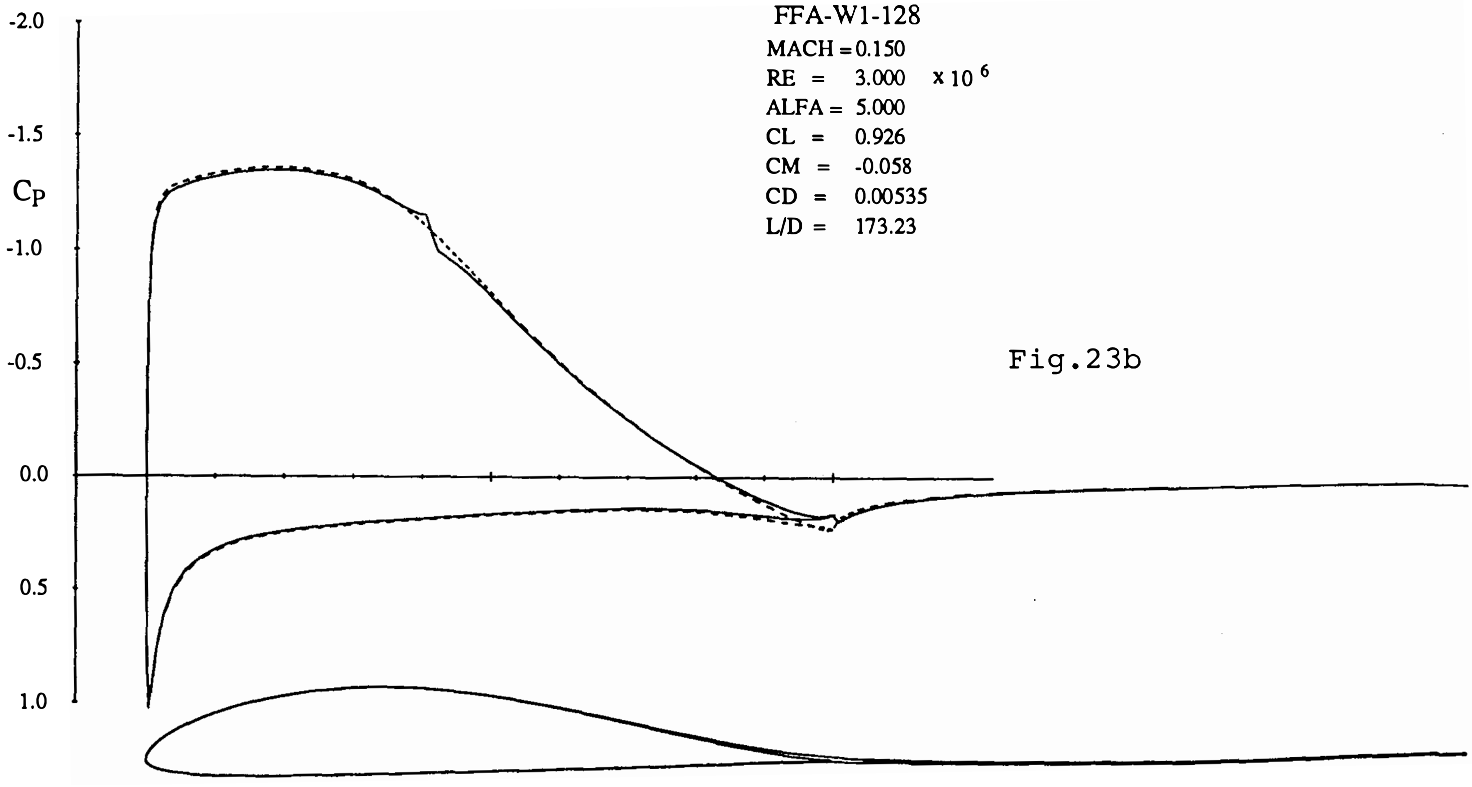

-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
C_P
FFA-W1-128
MACH = 0.150
RE = 3.000 x 10 6
ALFA = 5.000
CL = 0.926
CM = -0.058
CD = 0.00535
L/D = 173.23

Fig.23b

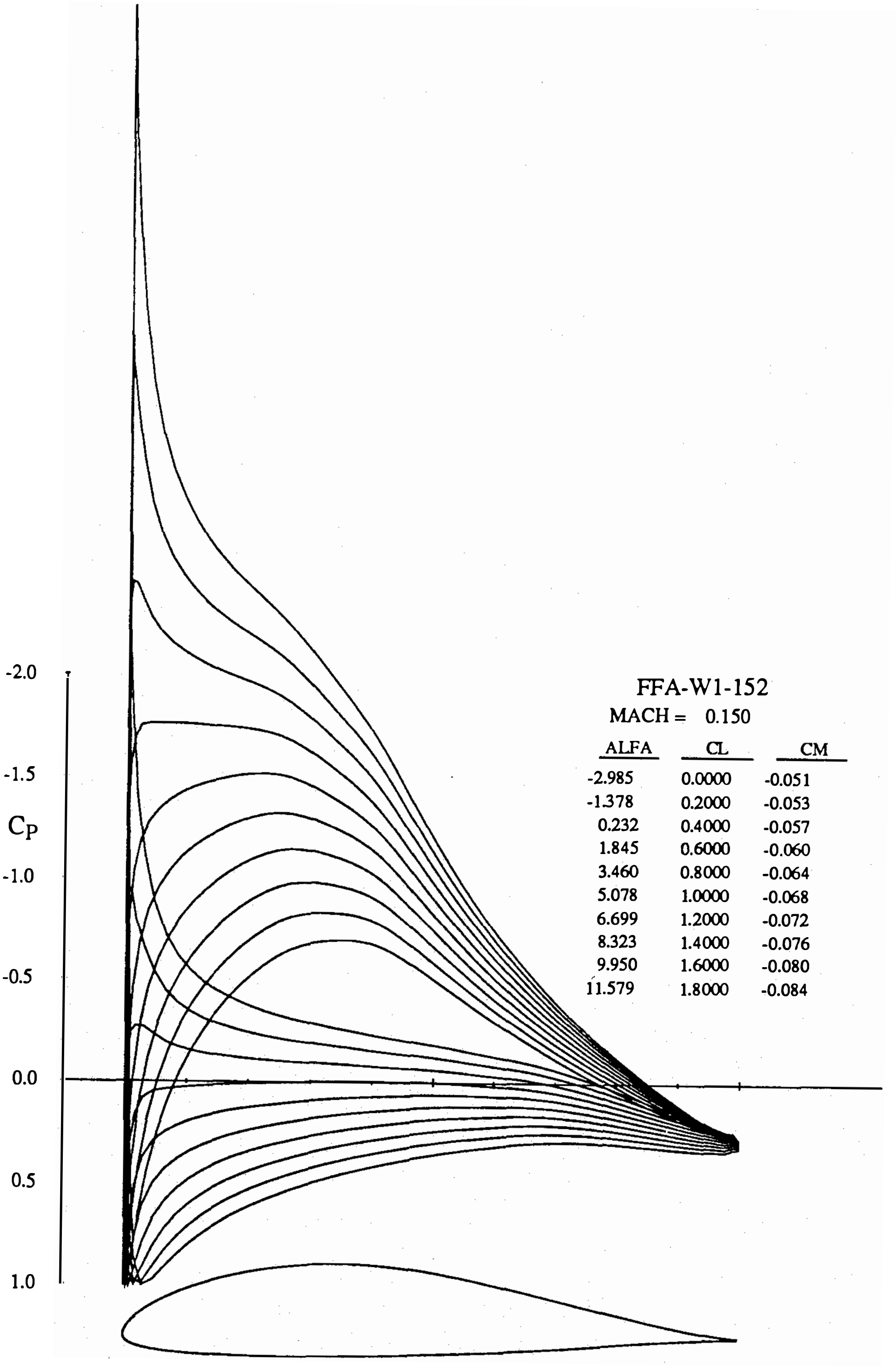

-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0
FFA-W1-152
MACH = 0.150
ALFA CL CM
-2.985 0.0000 -0.051
-1.378 0.2000 -0.053
0.232 0.4000 -0.057
1.845 0.6000 -0.060
3.460 0.8000 -0.064
5.078 1.0000 -0.068
6.699 1.2000 -0.072
8.323 1.4000 -0.076
9.950 1.6000 -0.080
11.579 1.8000 -0.084

Fig.23c

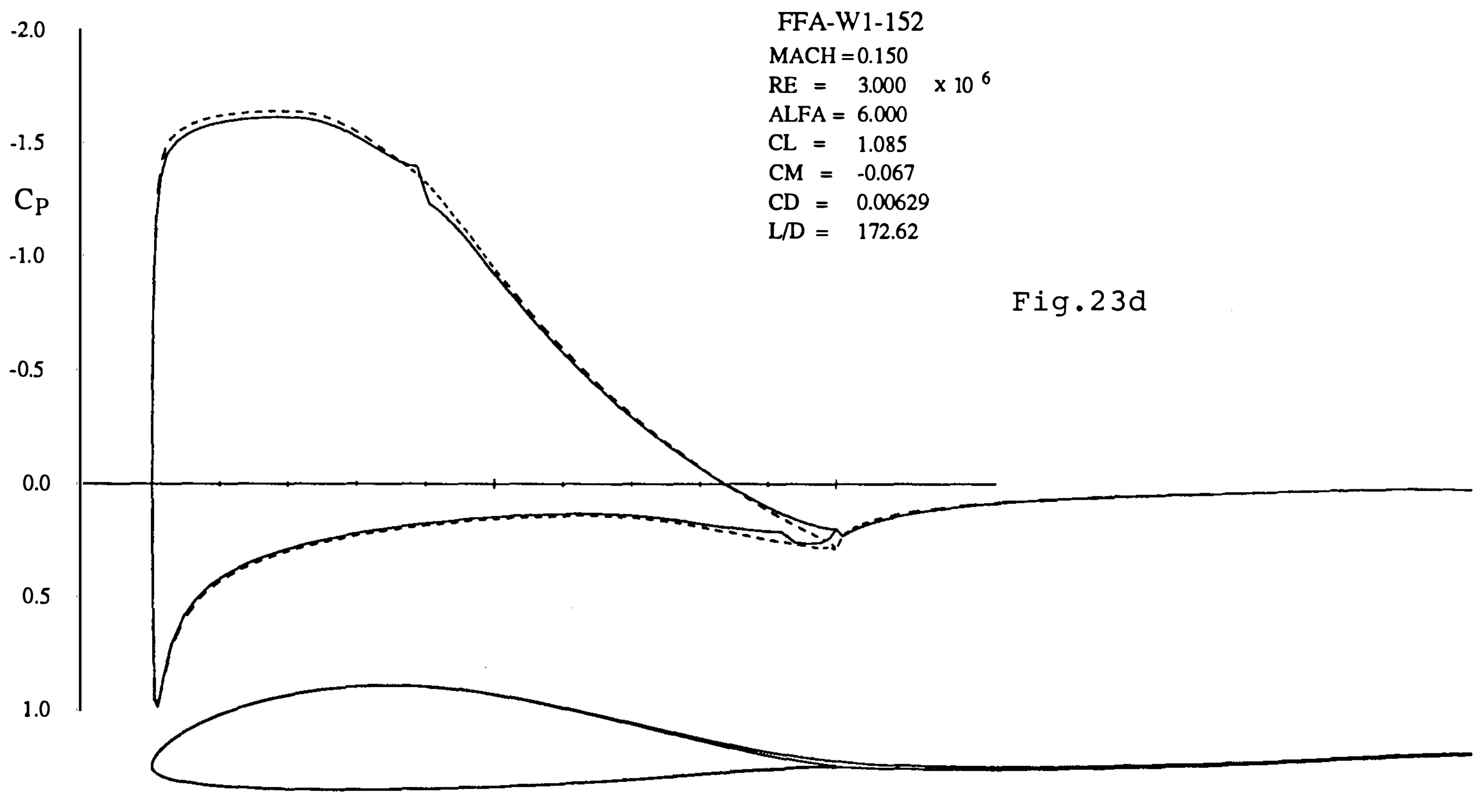

FFA-W1-152
MACH = 0.150
RE = 3.000 x 10 6
ALFA = 6.000
CL = 1.085
CM = -0.067
CD = 0.00629
L/D = 172.62
-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0

Fig.23d

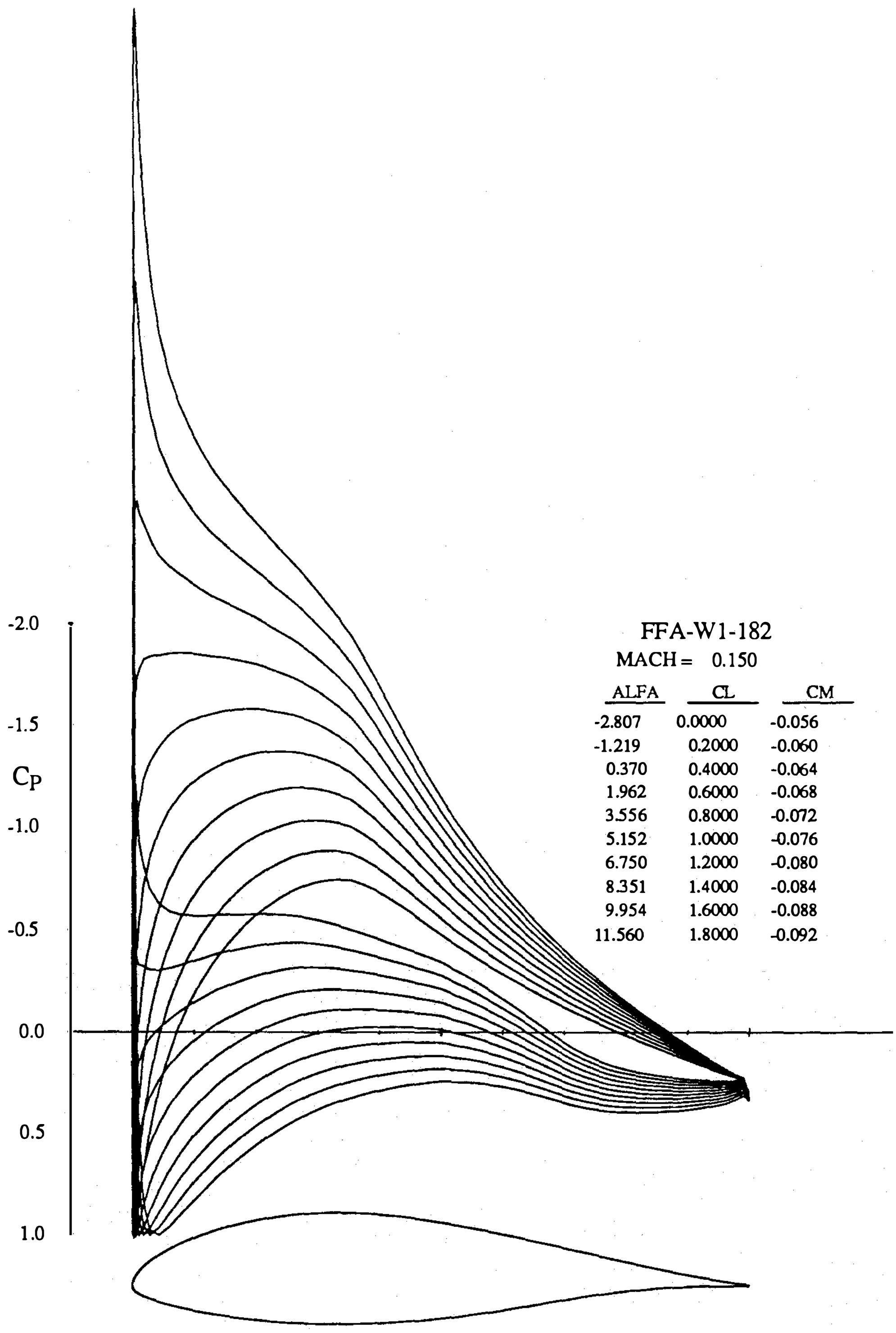

-2.0
-1.5
C_P
-1.0
-0.5
0.0
0.5
1.0
FFA-W1-182
MACH = 0.150
ALFA CL CM
-2.807 0.0000 -0.056
-1.219 0.2000 -0.060
0.370 0.4000 -0.064
1.962 0.6000 -0.068
3.556 0.8000 -0.072
5.152 1.0000 -0.076
6.750 1.2000 -0.080
8.351 1.4000 -0.084
9.954 1.6000 -0.088
11.560 1.8000 -0.092

Fig.23e

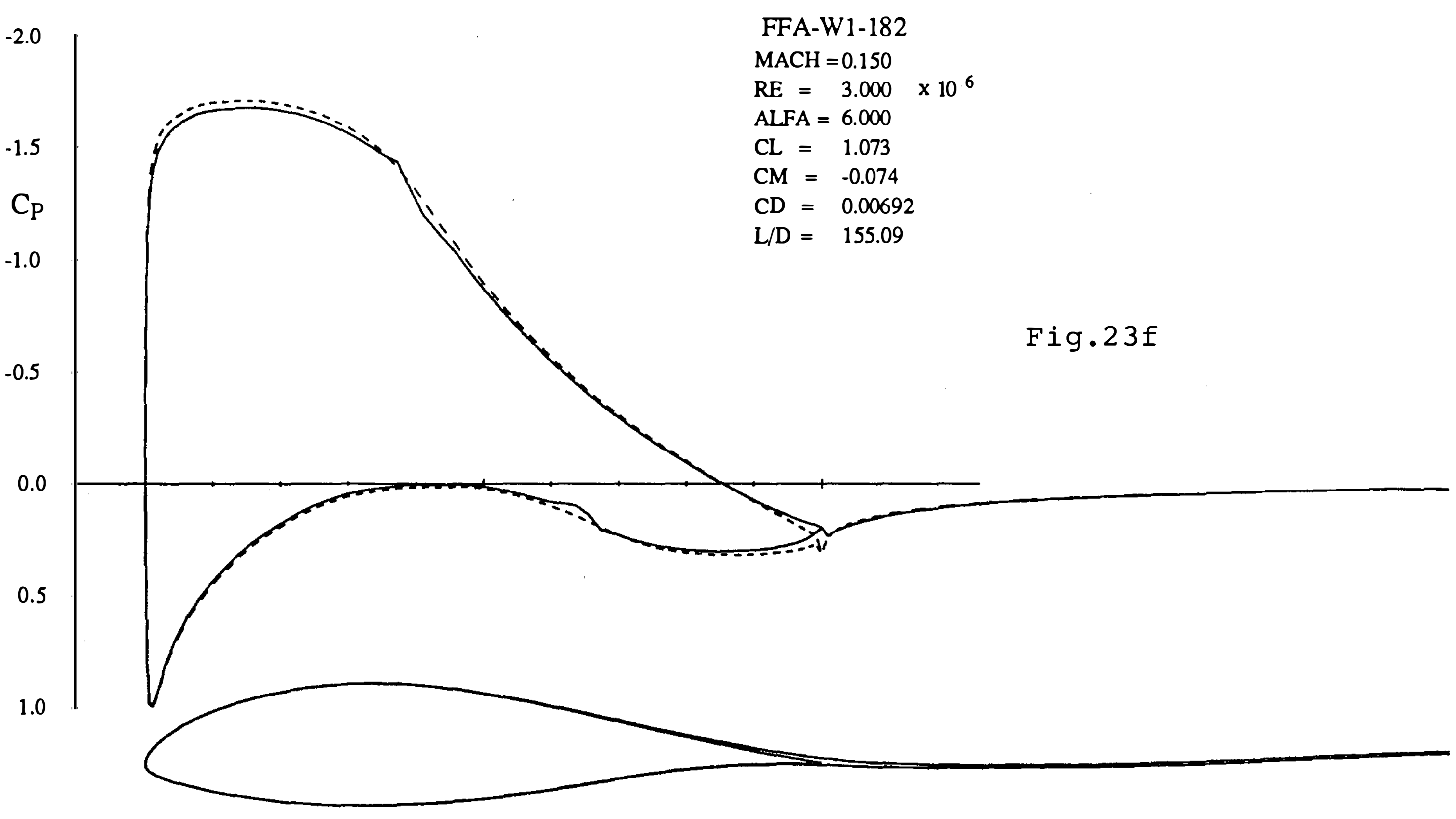

FFA-W1-182
MACH =0.150
RE = 3.000 x 10 6
ALFA = 6.000
CL = 1.073
CM = -0.074
CD = 0.00692
L/D = 155.09
-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0

Fig.23f

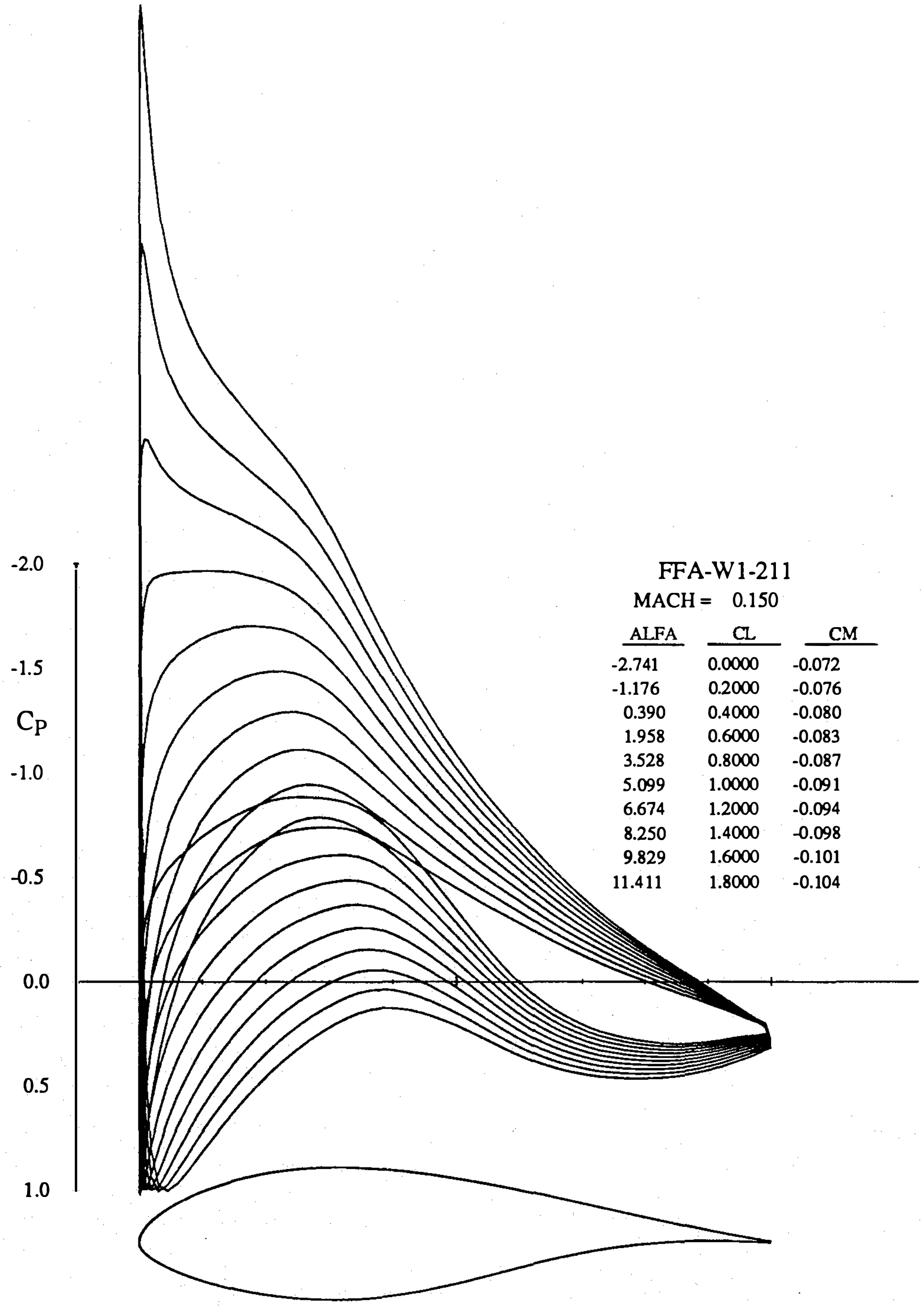

-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0
FFA-W1-211
MACH = 0.150
ALFA CL CM
-2.741 0.0000 -0.072
-1.176 0.2000 -0.076
0.390 0.4000 -0.080
1.958 0.6000 -0.083
3.528 0.8000 -0.087
5.099 1.0000 -0.091
6.674 1.2000 -0.094
8.250 1.4000 -0.098
9.829 1.6000 -0.101
11.411 1.8000 -0.104

Fig.23g

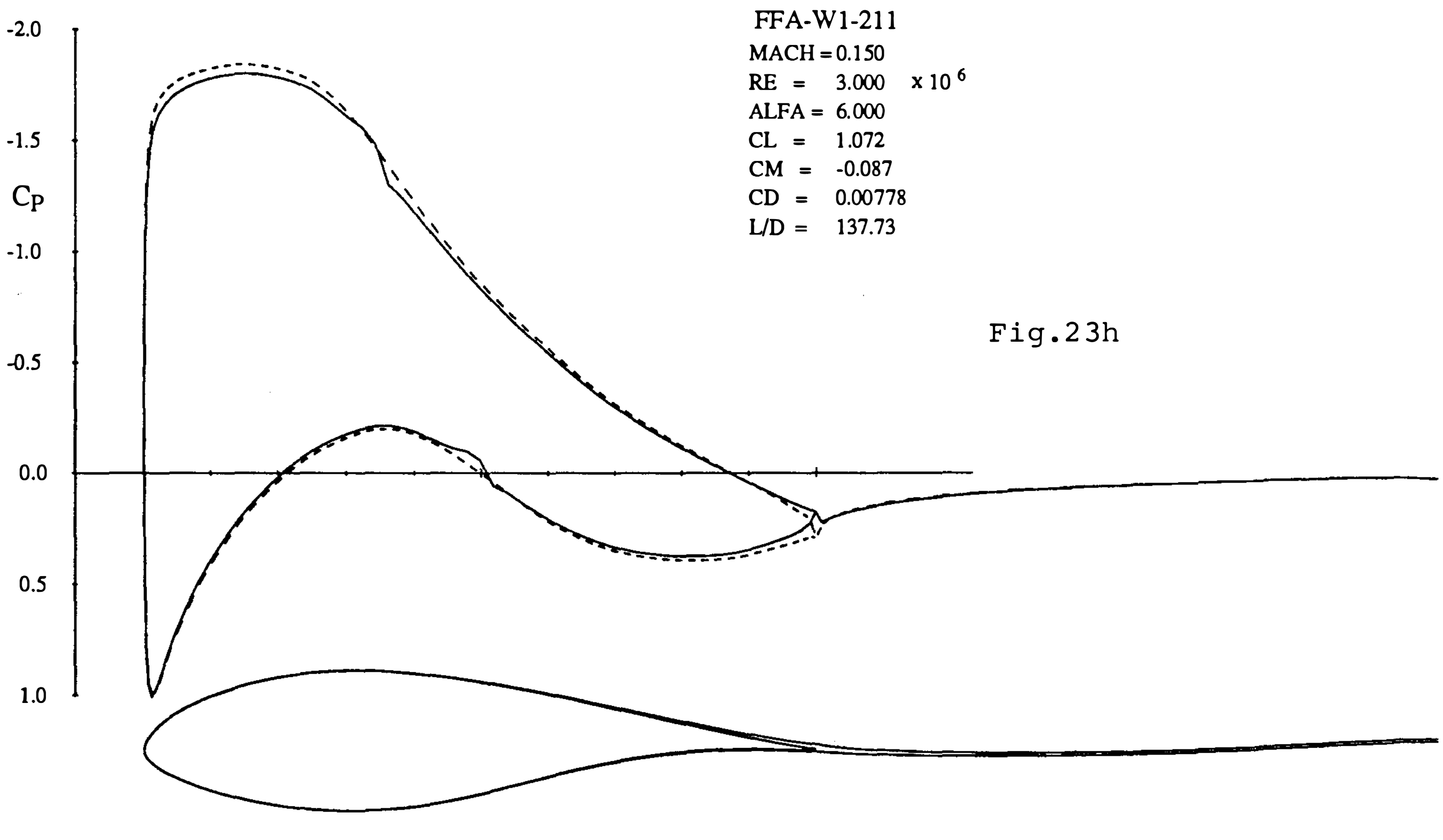

-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0
FFA-W1-211
MACH = 0.150
RE = 3.000 x 10 6
ALFA = 6.000
CL = 1.072
CM = -0.087
CD = 0.00778
L/D = 137.73

Fig.23h

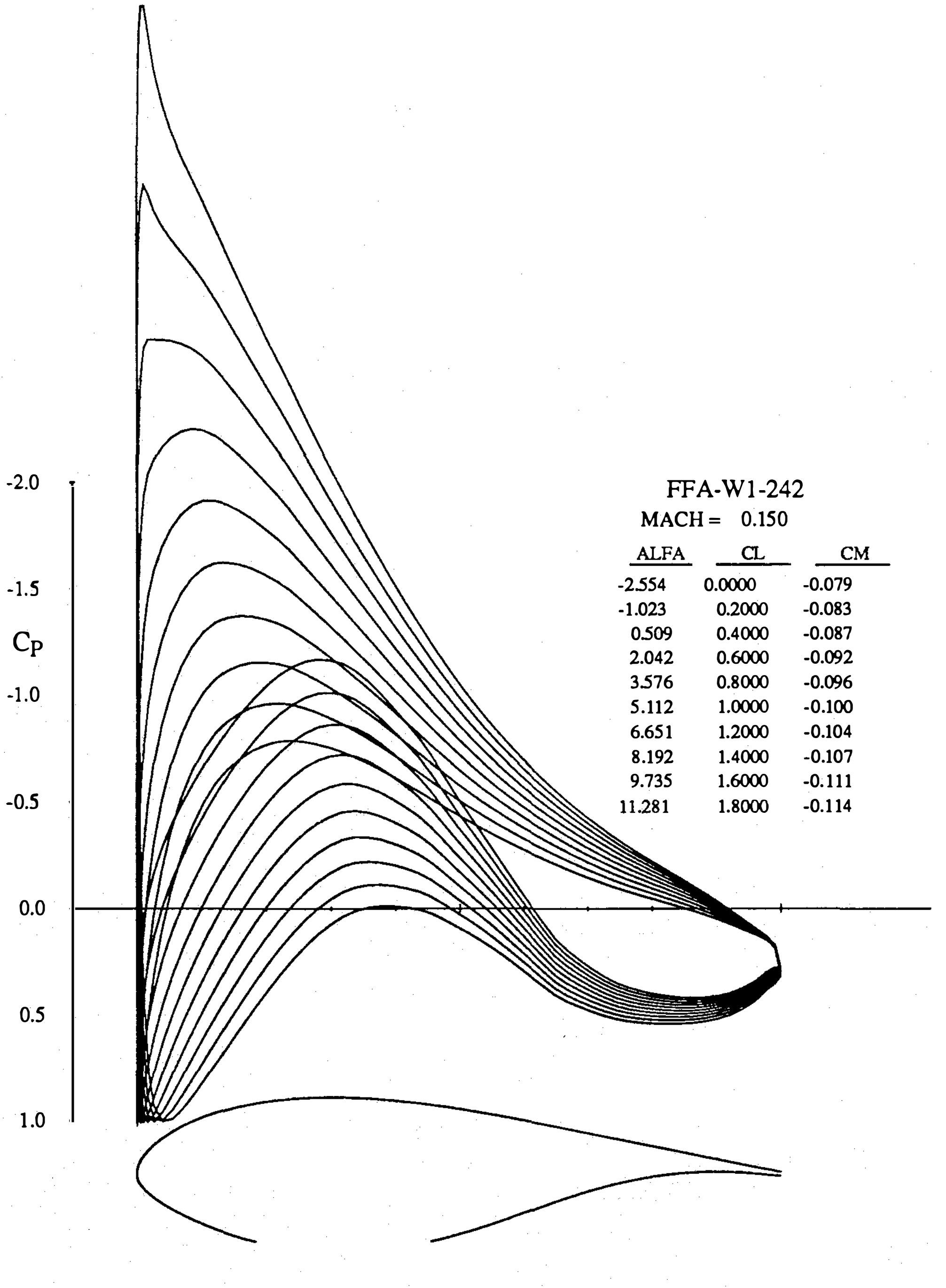

FFA-W1-242
MACH = 0.150
ALFA CL CM
-2.554 0.0000 -0.079
-1.023 0.2000 -0.083
0.509 0.4000 -0.087
2.042 0.6000 -0.092
3.576 0.8000 -0.096
5.112 1.0000 -0.100
6.651 1.2000 -0.104
8.192 1.4000 -0.107
9.735 1.6000 -0.111
11.281 1.8000 -0.114
-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0

Fig.23i

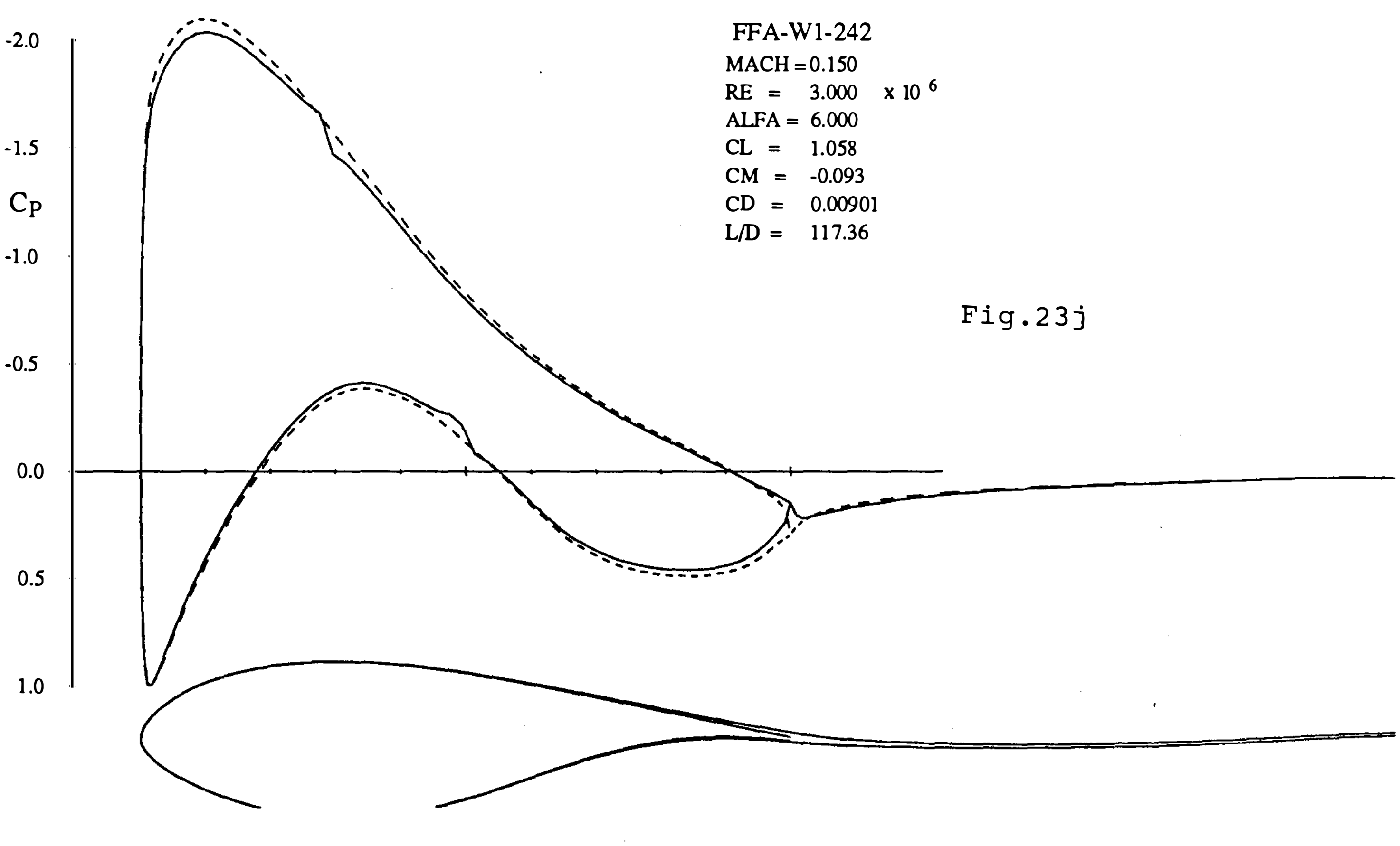

FFA-W1-242
MACH = 0.150
RE = 3.000 x 10 6
ALFA = 6.000
CL = 1.058
CM = -0.093
CD = 0.00901
L/D = 117.36
-2.0
-1.5
CP
-1.0
-0.5
0.0
0.5
1.0

Fig.23j

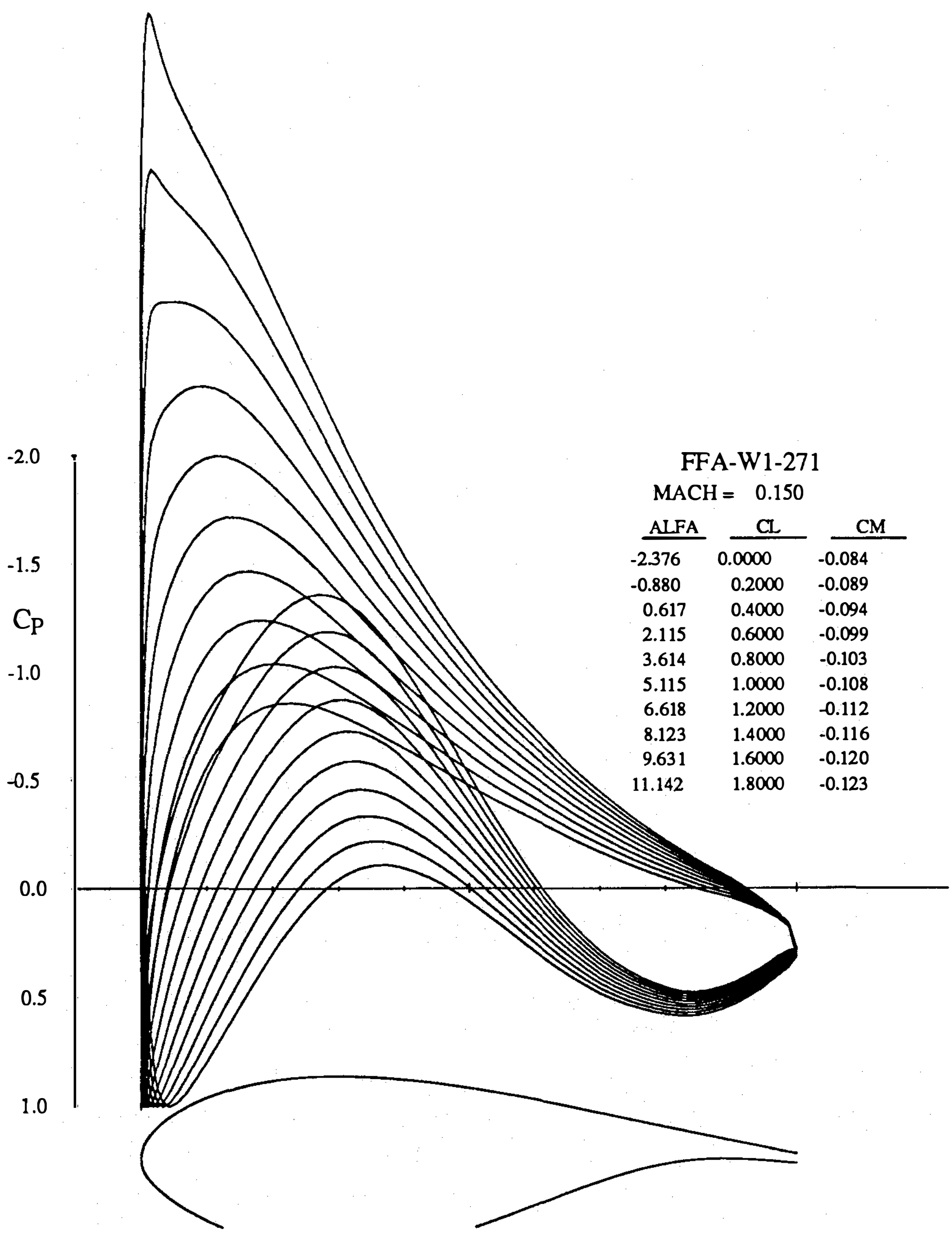

-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0
FFA-W1-271
MACH = 0.150
ALFA CL CM
-2.376 0.0000 -0.084
-0.880 0.2000 -0.089
0.617 0.4000 -0.094
2.115 0.6000 -0.099
3.614 0.8000 -0.103
5.115 1.0000 -0.108
6.618 1.2000 -0.112
8.123 1.4000 -0.116
9.631 1.6000 -0.120
11.142 1.8000 -0.123

Fig 23k

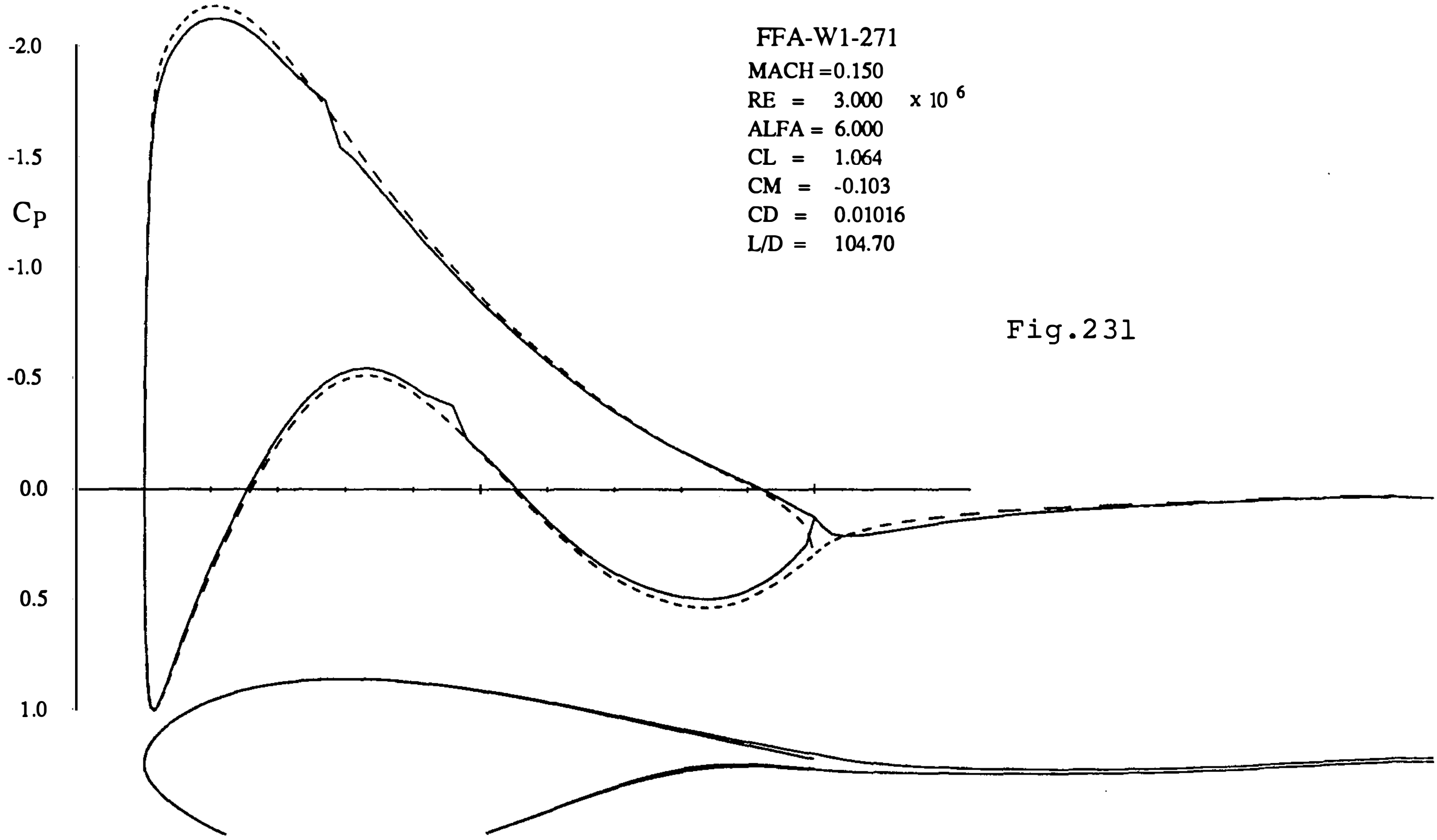

-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0
FFA-W1-271
MACH =0.150
RE = 3.000 x 10 6
ALFA = 6.000
CL = 1.064
CM = -0.103
CD = 0.01016
L/D = 104.70

Fig.231

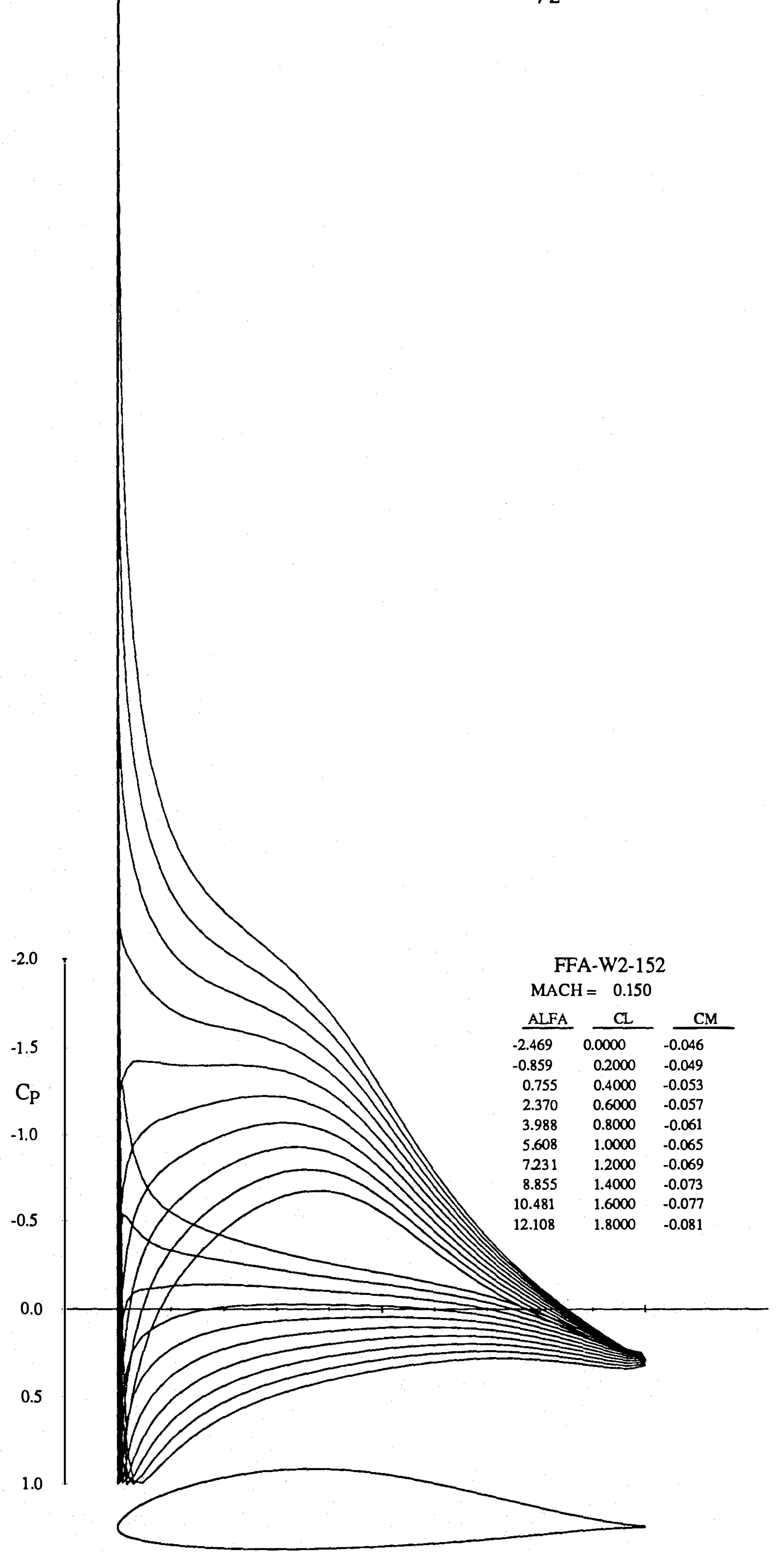

FFA-W2-152
MACH = 0.150
ALFA CL CM
-2.469 0.0000 -0.046
-0.859 0.2000 -0.049
0.755 0.4000 -0.053
2.370 0.6000 -0.057
3.988 0.8000 -0.061
5.608 1.0000 -0.065
7.231 1.2000 -0.069
8.855 1.4000 -0.073
10.481 1.6000 -0.077
12.108 1.8000 -0.081
-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0

Fig.24a

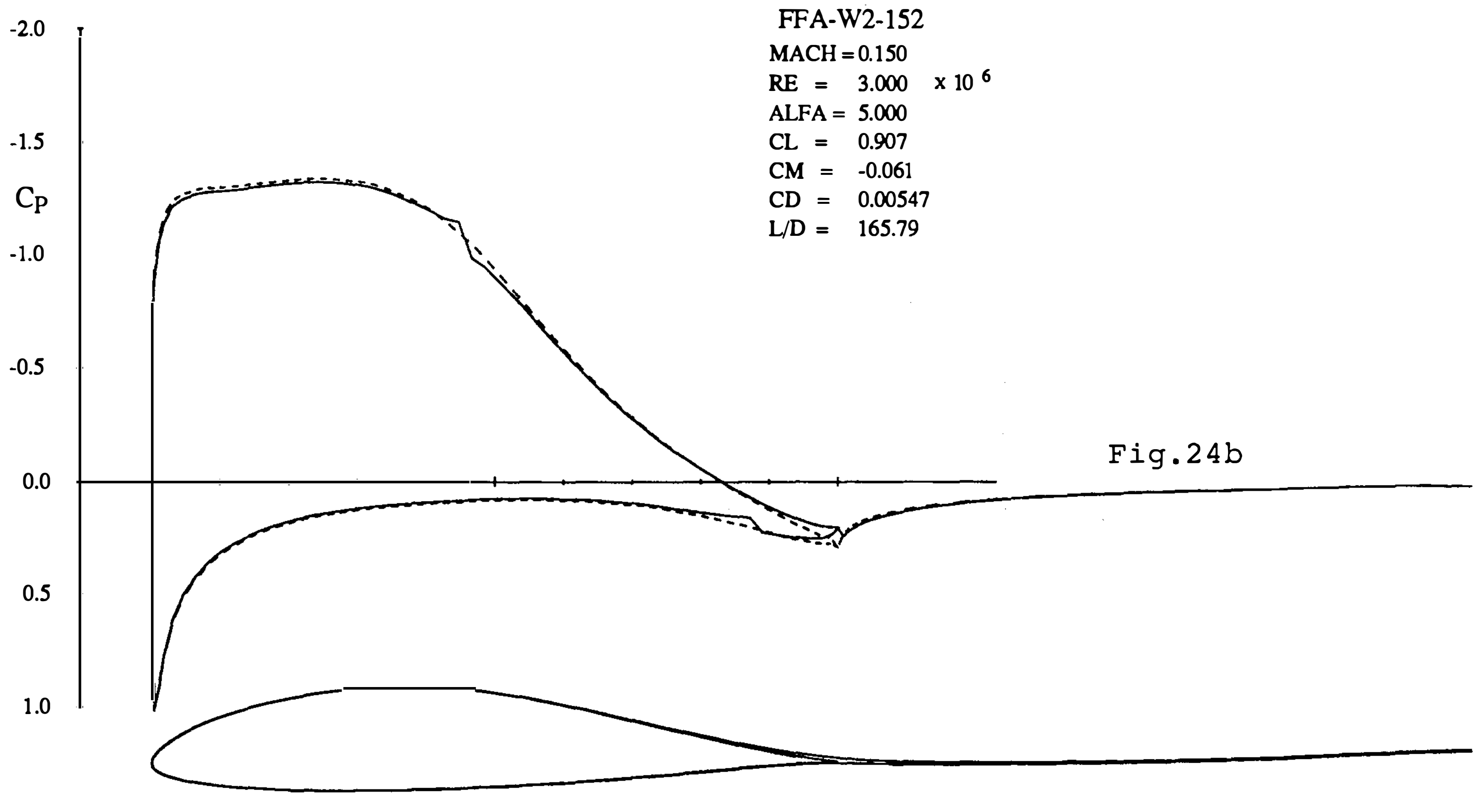

FFA-W2-152
MACH = 0.150
RE = 3.000 x 10 6
ALFA = 5.000
CL = 0.907
CM = -0.061
CD = 0.00547
L/D = 165.79
-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0

Fig.24b

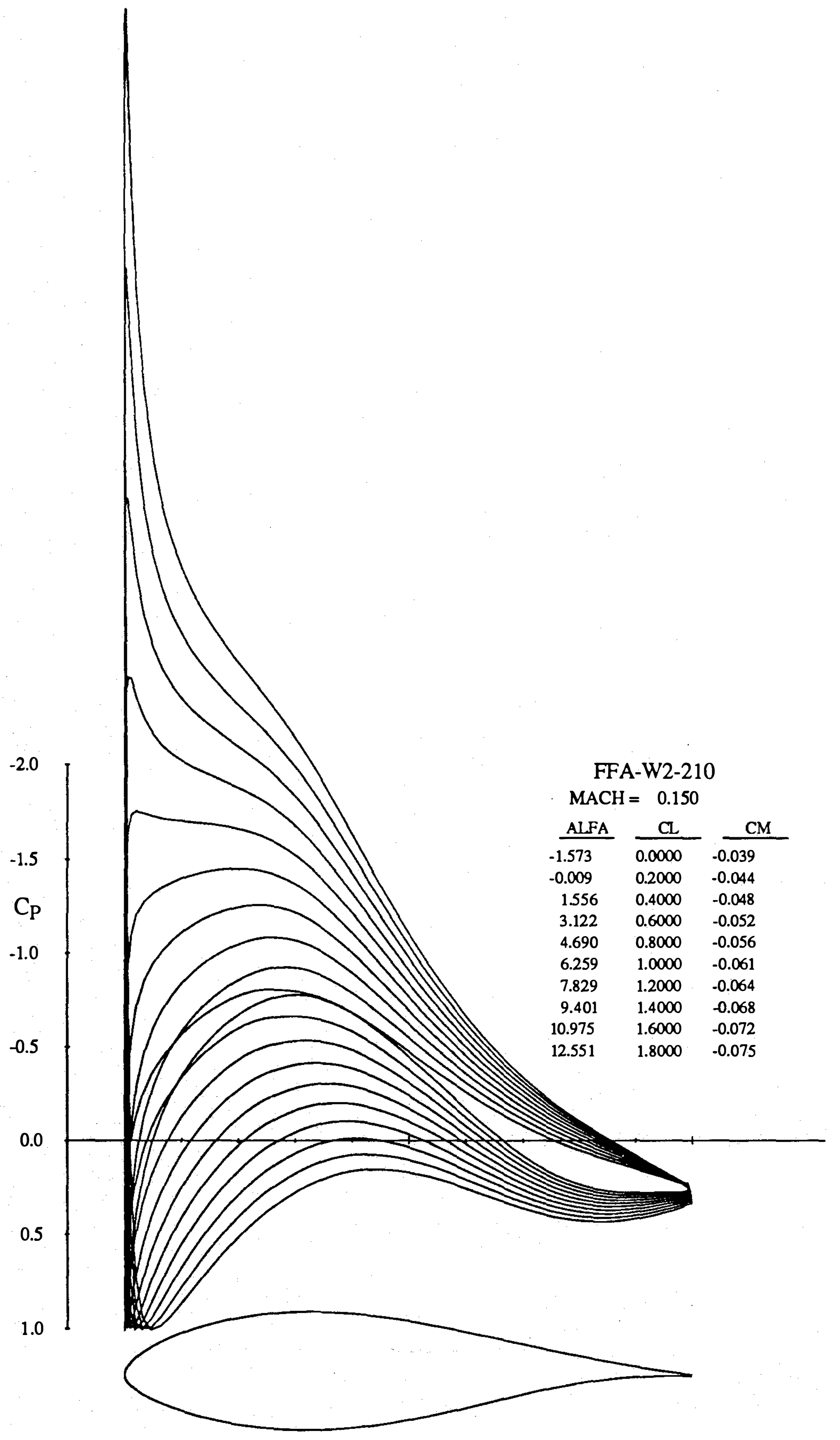

-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0
FFA-W2-210
MACH = 0.150
ALFA CL CM
-1.573 0.0000 -0.039
-0.009 0.2000 -0.044
1.556 0.4000 -0.048
3.122 0.6000 -0.052
4.690 0.8000 -0.056
6.259 1.0000 -0.061
7.829 1.2000 -0.064
9.401 1.4000 -0.068
10.975 1.6000 -0.072
12.551 1.8000 -0.075

Fig.24c

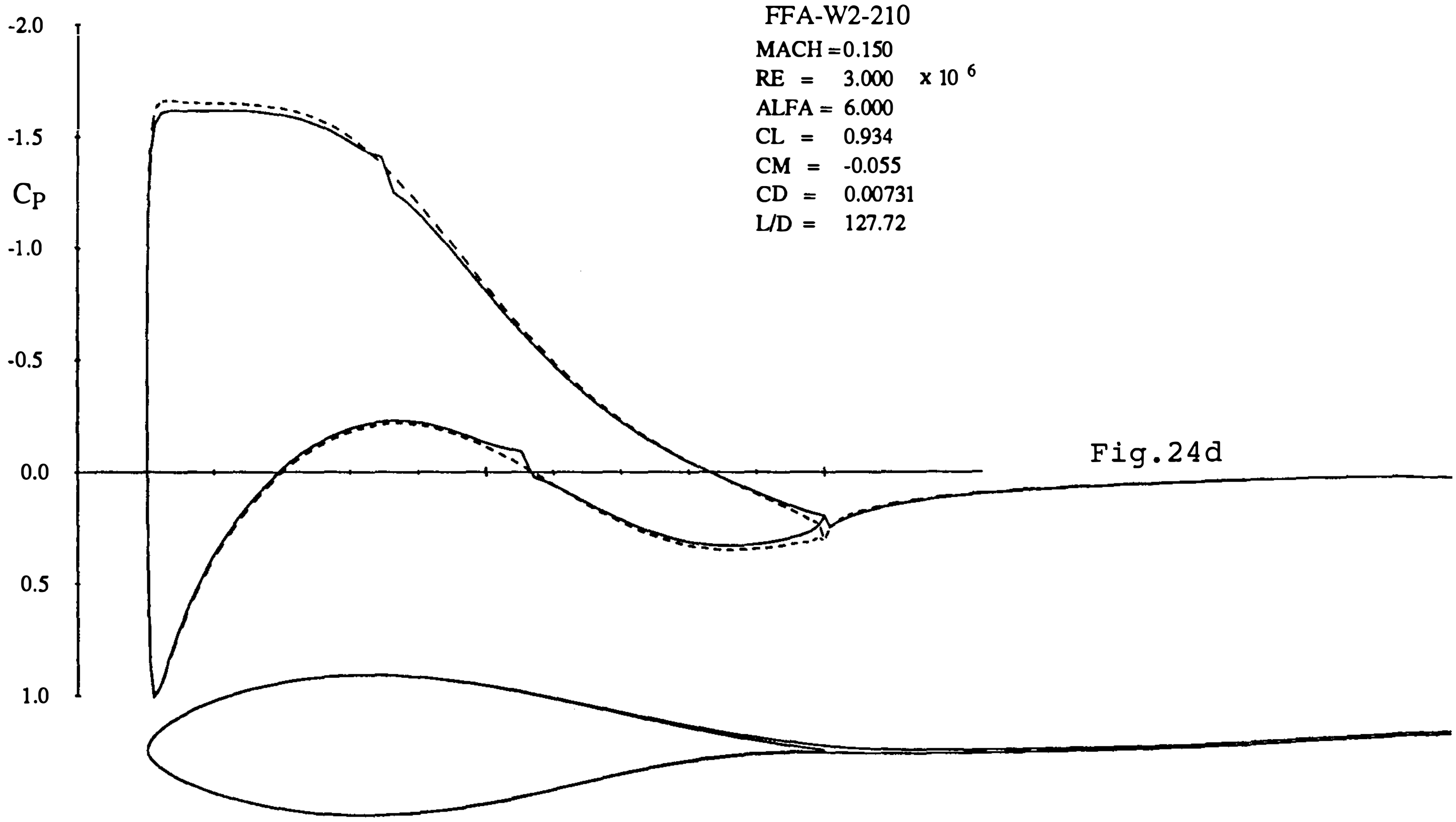

FFA-W2-210
MACH =0.150
RE = 3.000 x 10 6
ALFA = 6.000
CL = 0.934
CM = -0.055
CD = 0.00731
L/D = 127.72
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0

Fig.24d

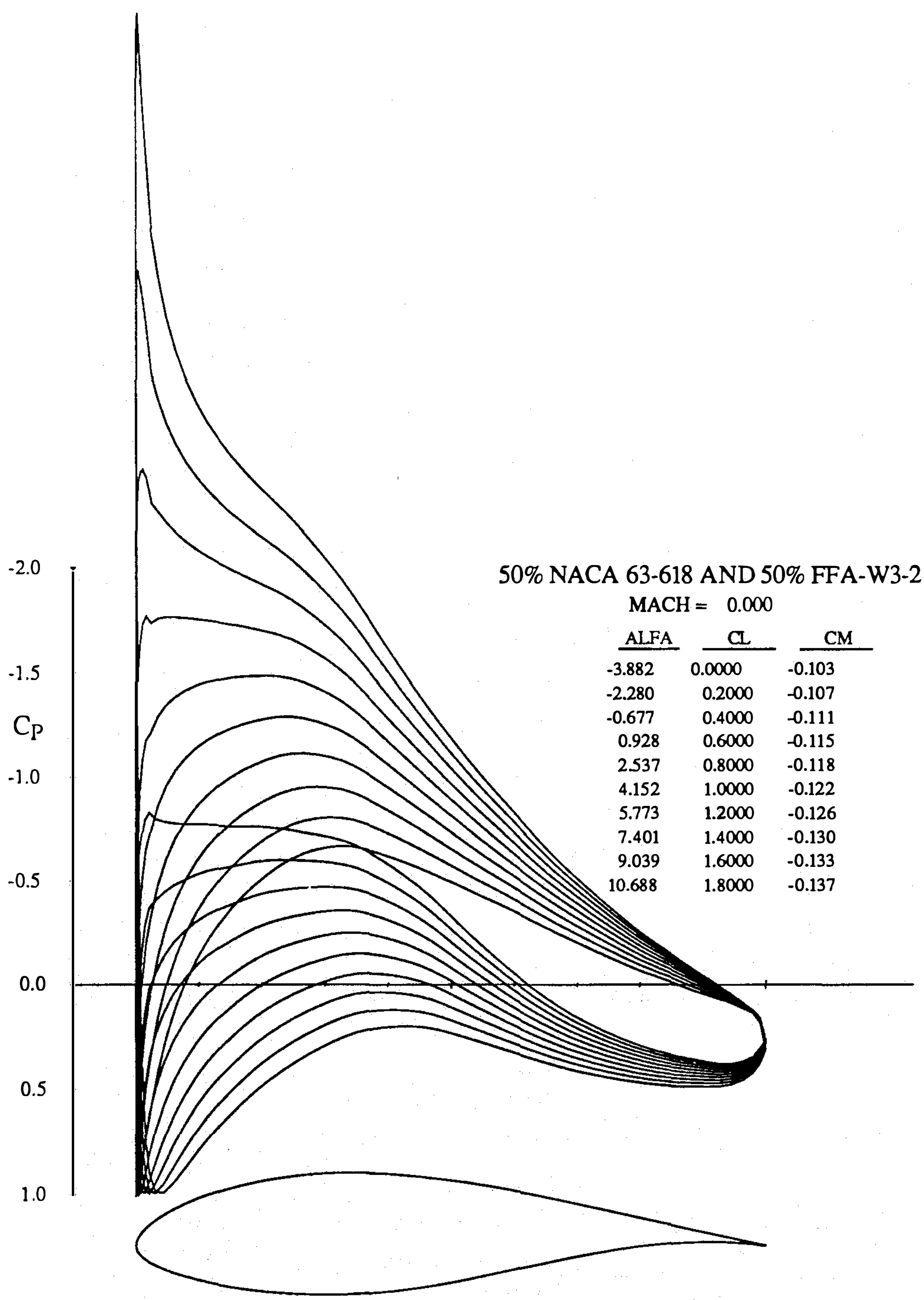

50% NACA 63-618 AND 50% FFA-W3-2
MACH = 0.000
ALFA CL CM
-3.882 0.0000 -0.103
-2.280 0.2000 -0.107
-0.677 0.4000 -0.111
0.928 0.6000 -0.115
2.537 0.8000 -0.118
4.152 1.0000 -0.122
5.773 1.2000 -0.126
7.401 1.4000 -0.130
9.039 1.6000 -0.133
10.688 1.8000 -0.137
-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0

Fig.25a

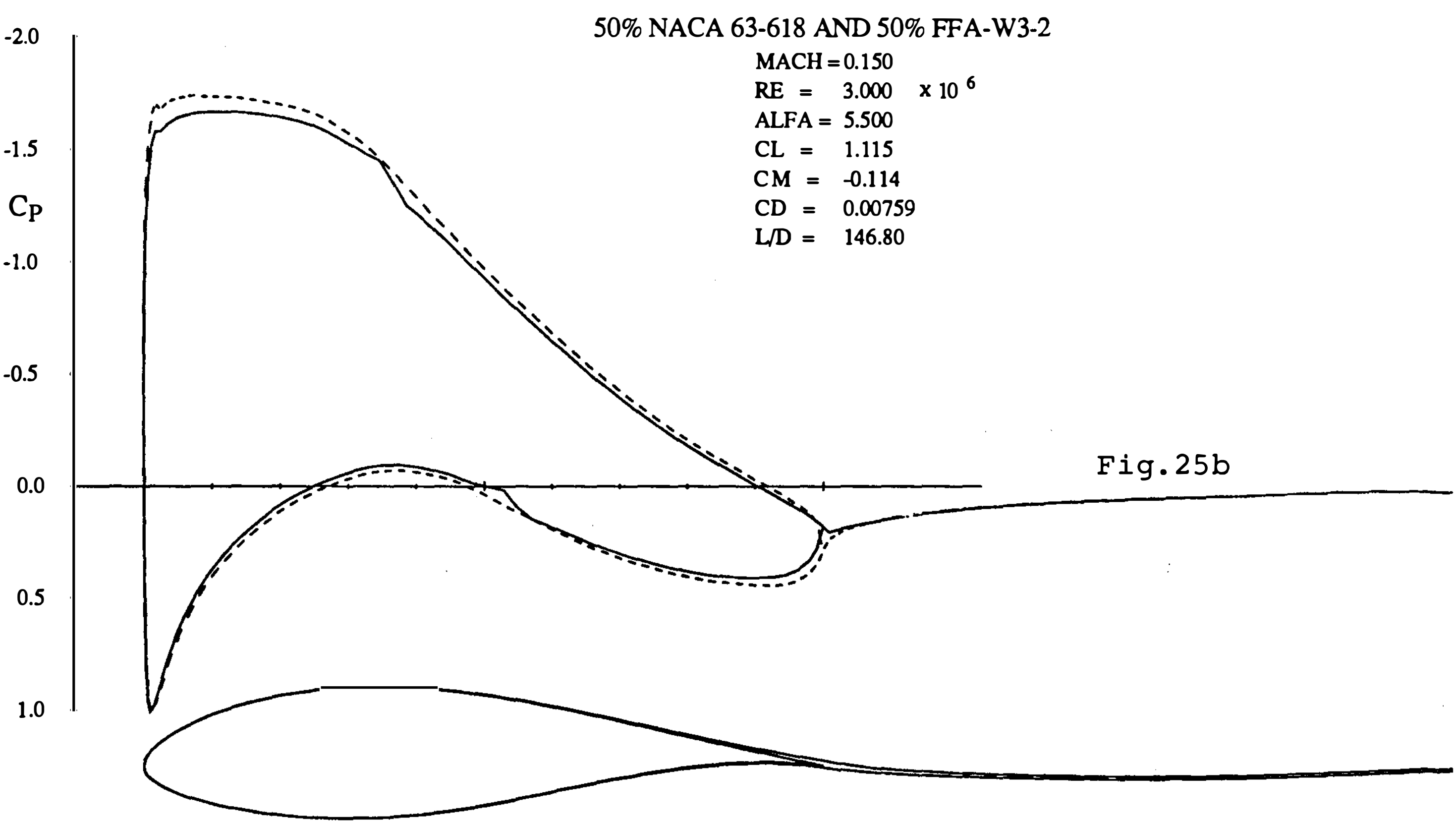

50% NACA 63-618 AND 50% FFA-W3-2
MACH = 0.150
RE = 3.000 x 10 6
ALFA = 5.500
CL = 1.115
CM = -0.114
CD = 0.00759
L/D = 146.80
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0

Fig.25b

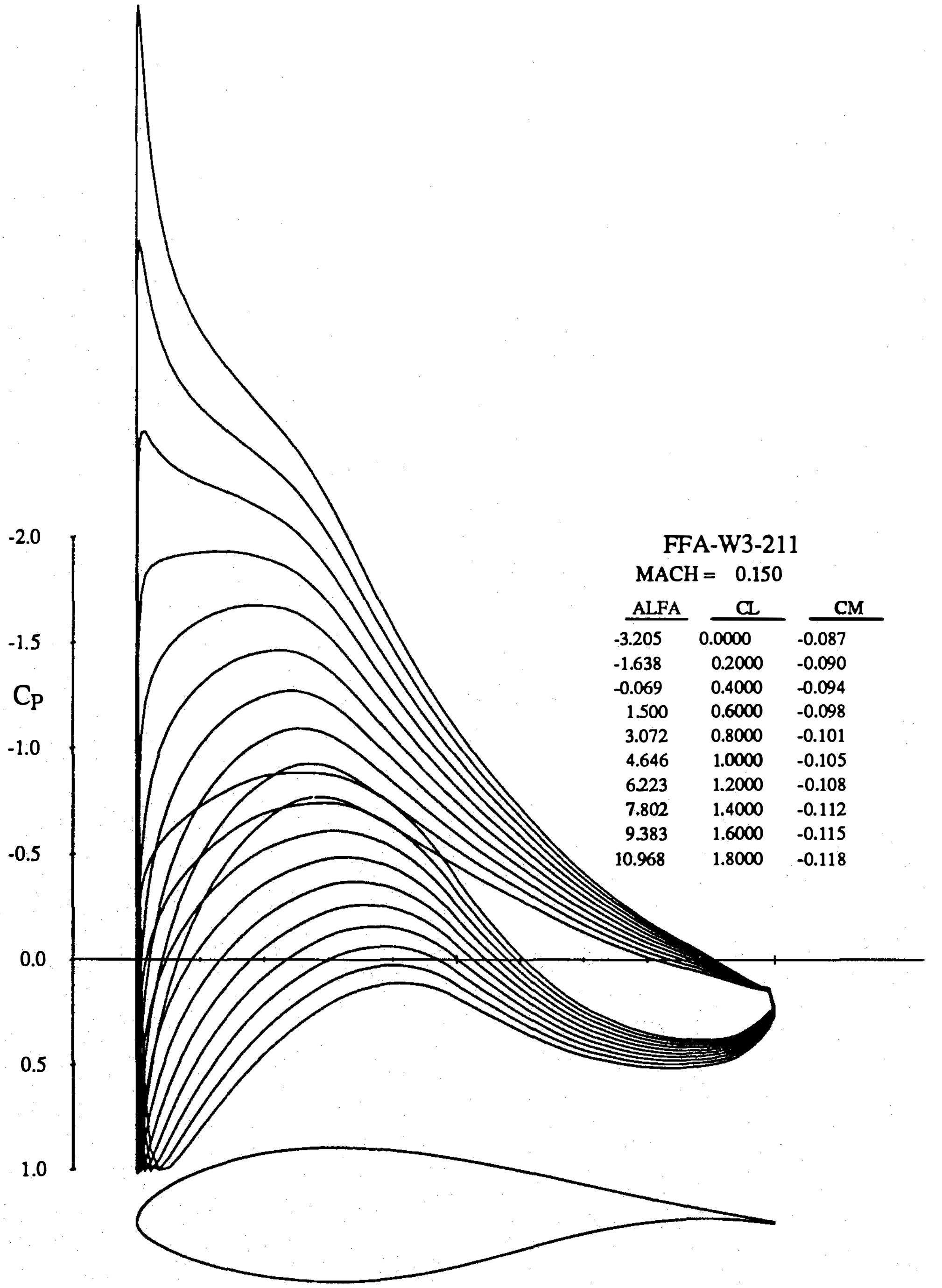
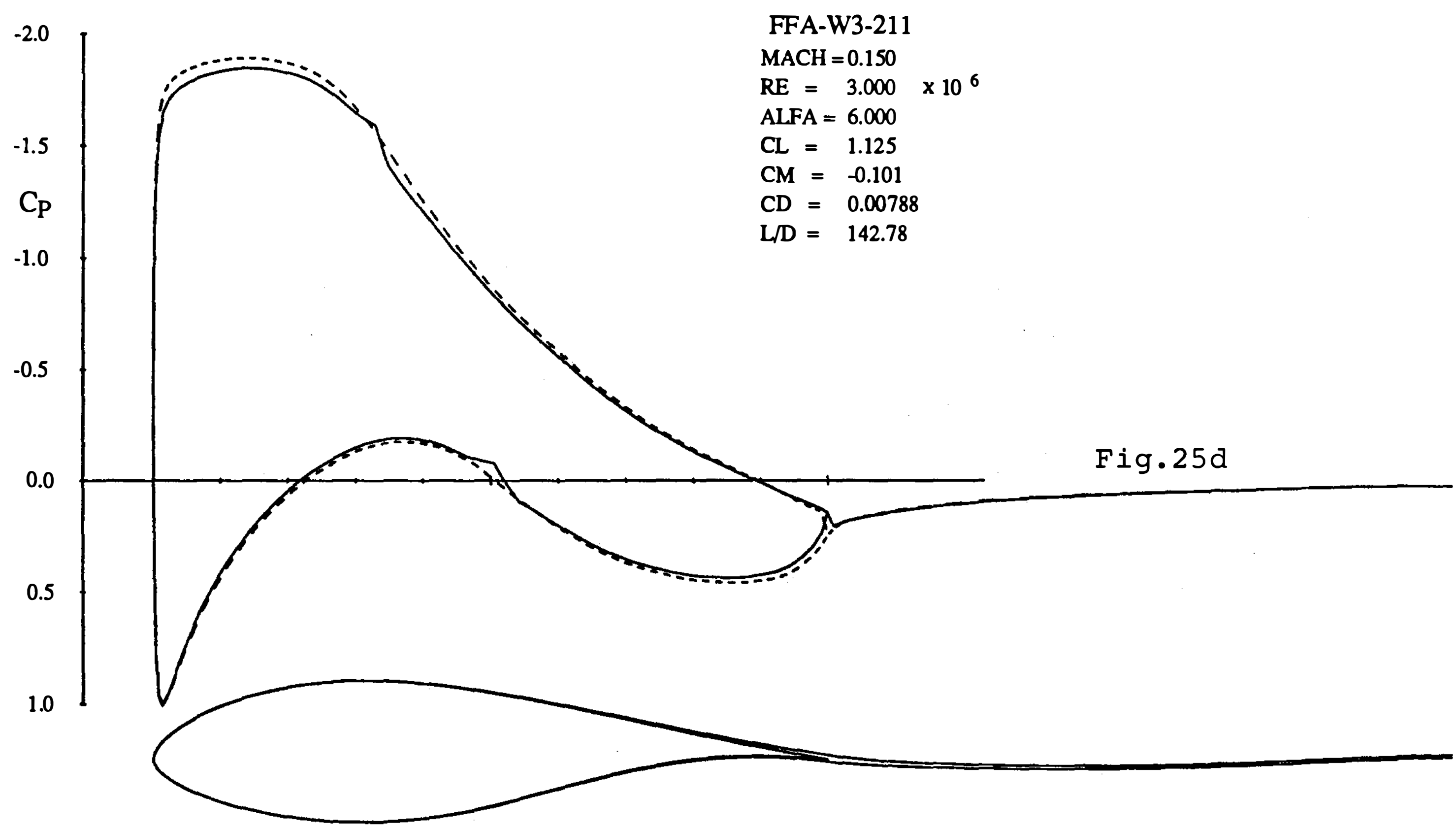

-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0
FFA-W3-211
MACH = 0.150
ALFA CL CM
-3.205 0.0000 -0.087
-1.638 0.2000 -0.090
-0.069 0.4000 -0.094
1.500 0.6000 -0.098
3.072 0.8000 -0.101
4.646 1.0000 -0.105
6.223 1.2000 -0.108
7.802 1.4000 -0.112
9.383 1.6000 -0.115
10.968 1.8000 -0.118

Fig.25c



FFA-W3-211
MACH = 0.150
RE = 3.000 x 10 6
ALFA = 6.000
CL = 1.125
CM = -0.101
CD = 0.00788
L/D = 142.78
-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0

Fig.25d

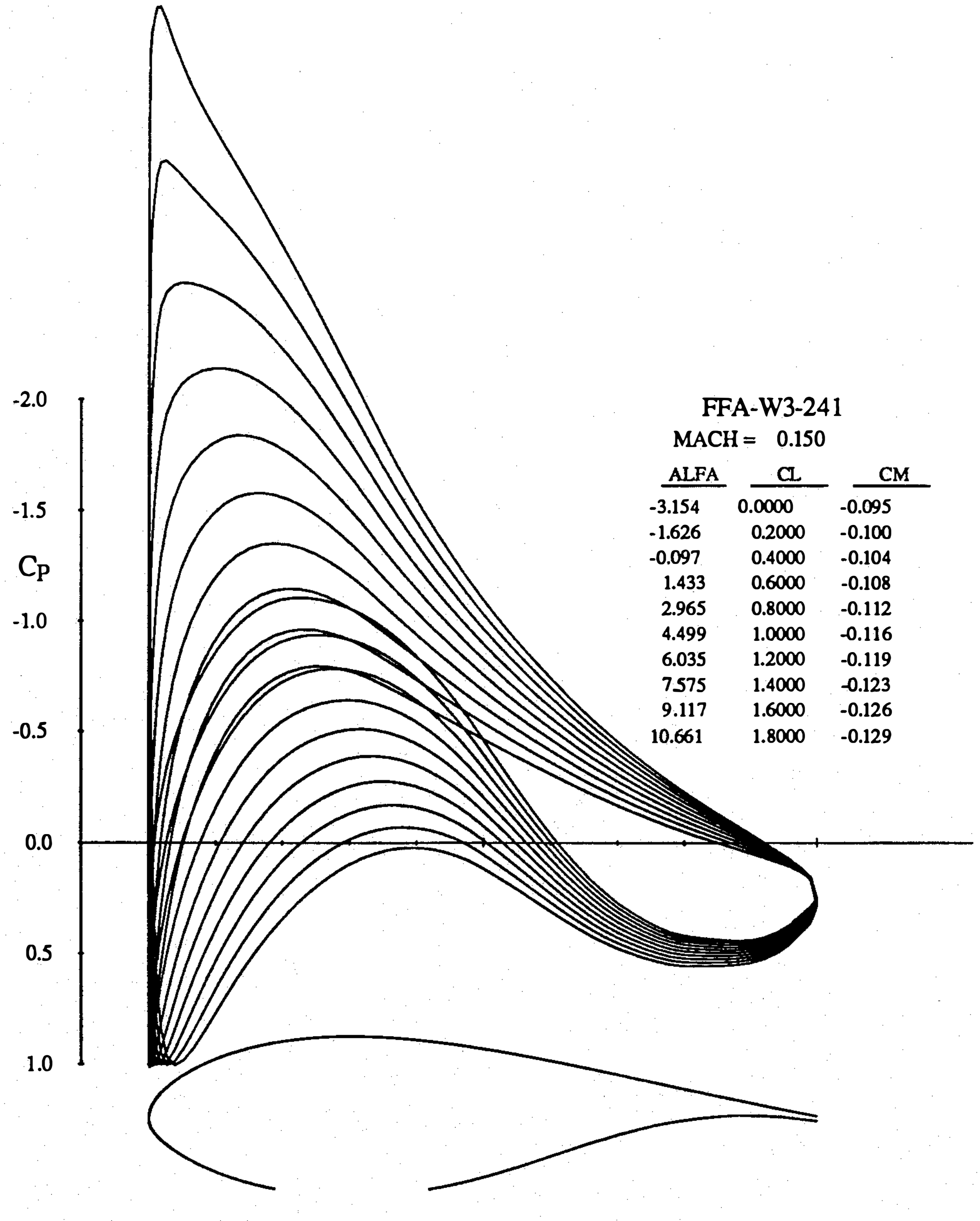

FFA-W3-241
MACH = 0.150
ALFA CL CM
-3.154 0.0000 -0.095
-1.626 0.2000 -0.100
-0.097 0.4000 -0.104
1.433 0.6000 -0.108
2.965 0.8000 -0.112
4.499 1.0000 -0.116
6.035 1.2000 -0.119
7.575 1.4000 -0.123
9.117 1.6000 -0.126
10.661 1.8000 -0.129
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0

Fig.25e

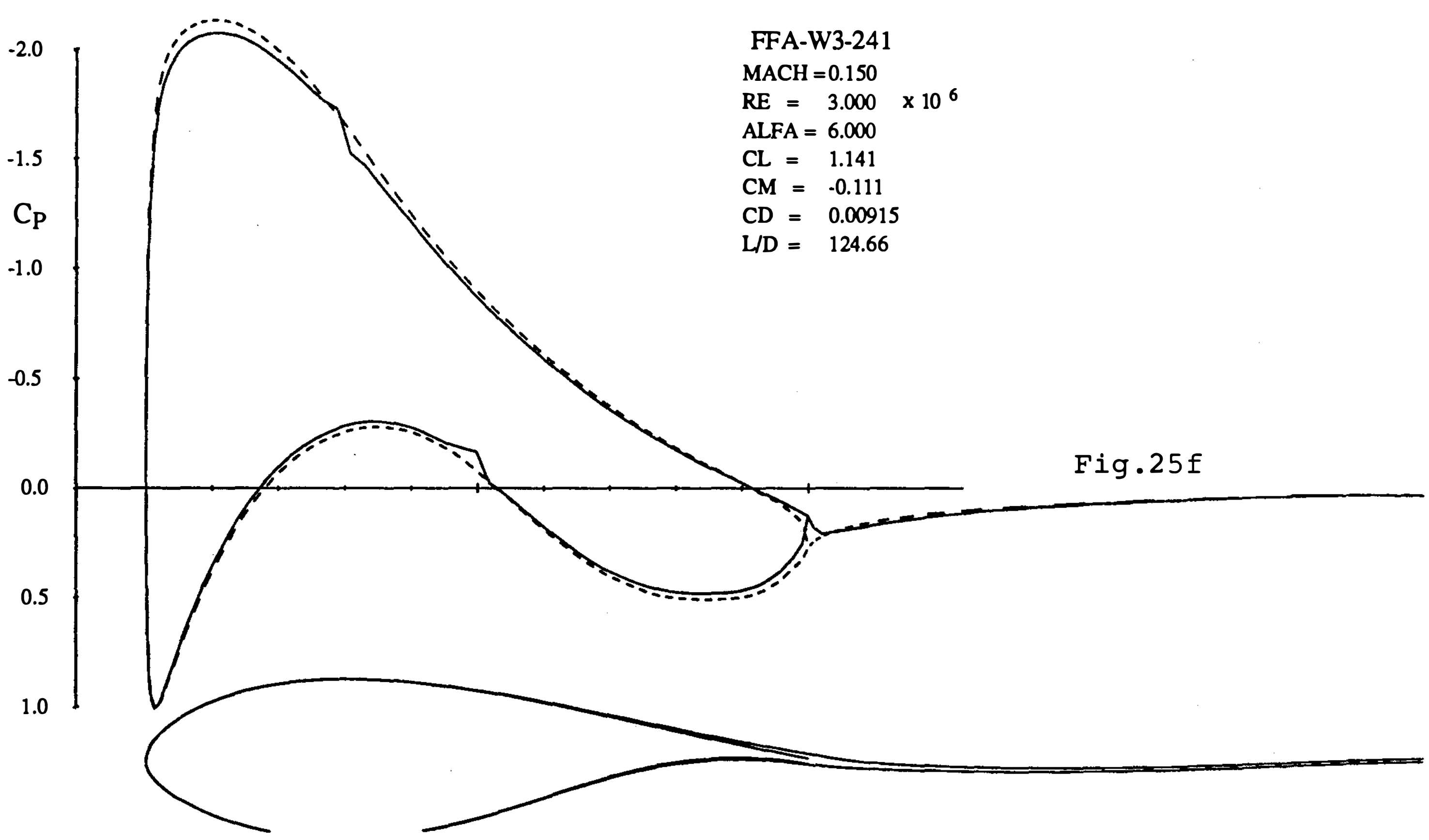

FFA-W3-241
MACH = 0.150
RE = 3.000 x 10 6
ALFA = 6.000
CL = 1.141
CM = -0.111
CD = 0.00915
L/D = 124.66
-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0

Fig.25f

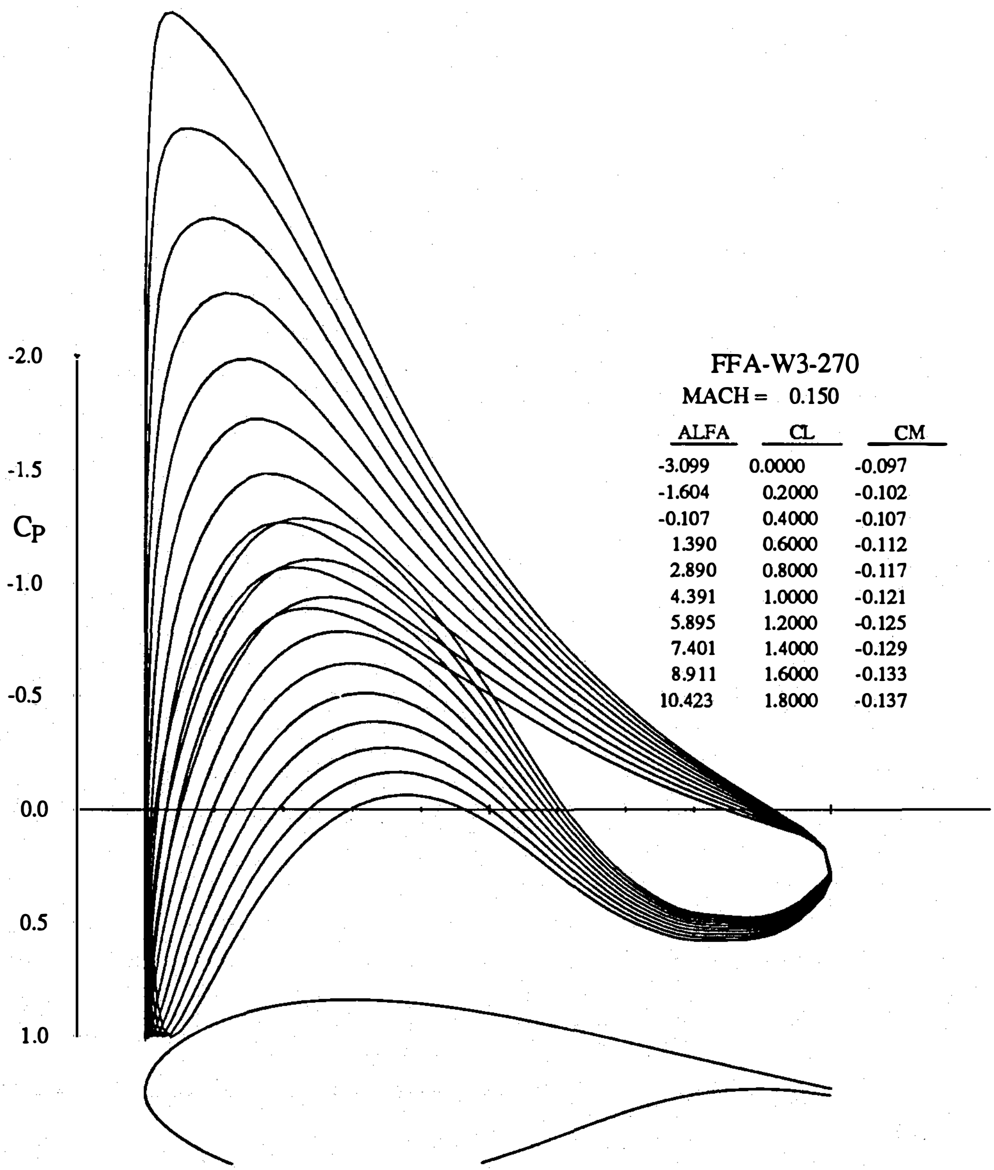

FFA-W3-270
MACH = 0.150
ALFA CL CM
-3.099 0.0000 -0.097
-1.604 0.2000 -0.102
-0.107 0.4000 -0.107
1.390 0.6000 -0.112
2.890 0.8000 -0.117
4.391 1.0000 -0.121
5.895 1.2000 -0.125
7.401 1.4000 -0.129
8.911 1.6000 -0.133
10.423 1.8000 -0.137
-2.0
-1.5
C_P
-1.0
-0.5
0.0
0.5
1.0

Fig.25g

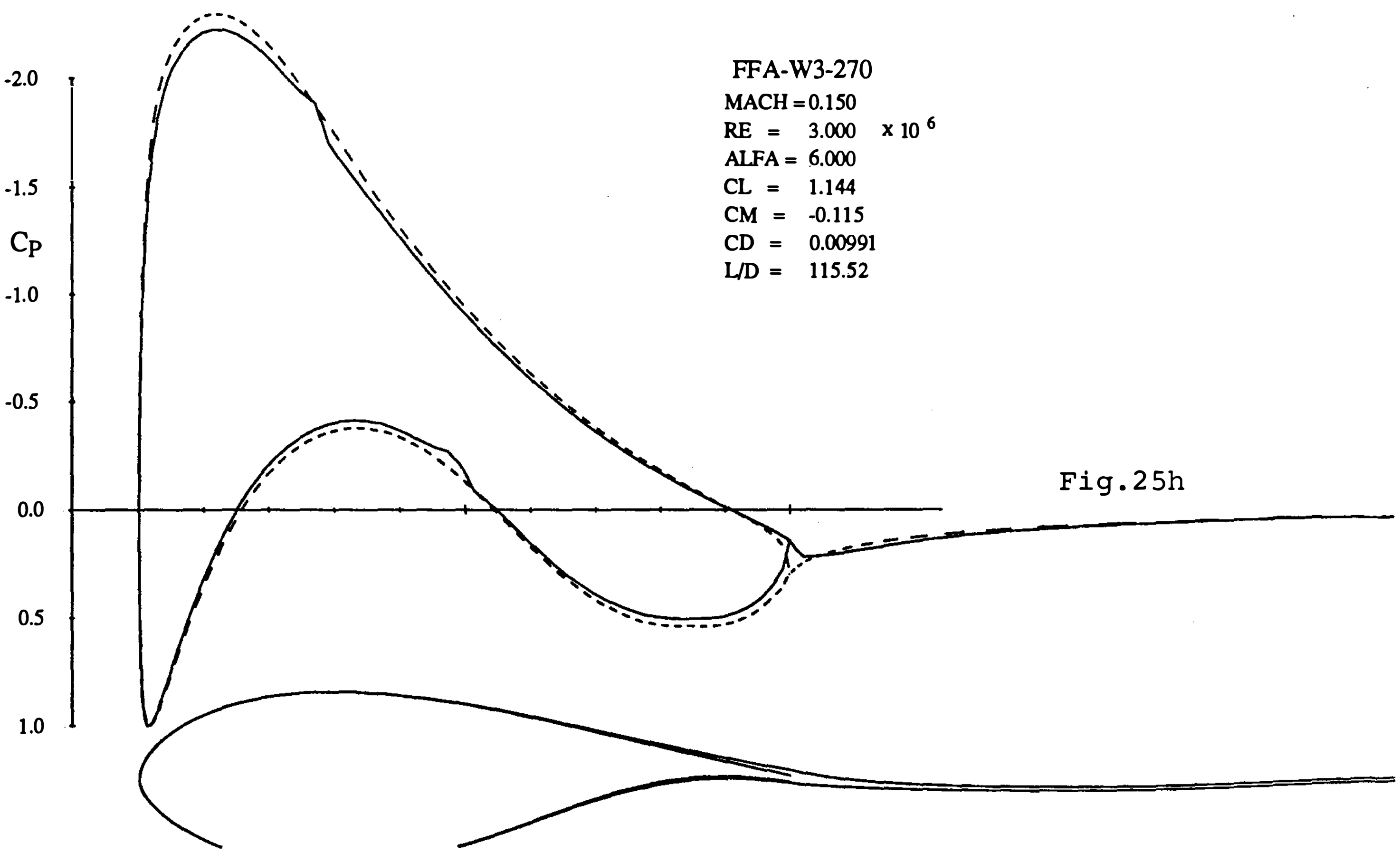

FFA-W3-270
MACH = 0.150
RE = 3.000 x 10 6
ALFA = 6.000
CL = 1.144
CM = -0.115
CD = 0.00991
L/D = 115.52
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0

Fig.25h

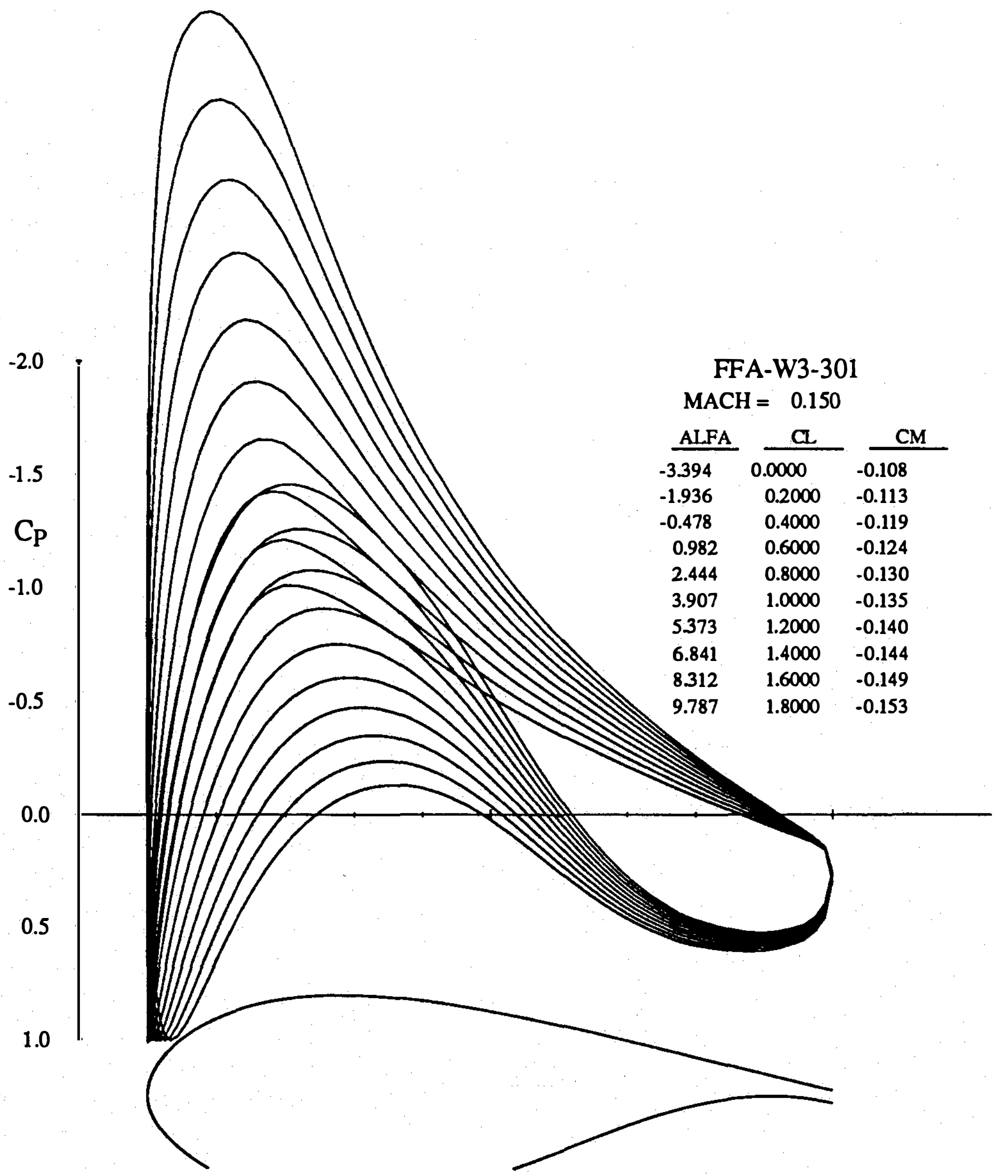

FFA-W3-301
MACH = 0.150
ALFA CL CM
-3.394 0.0000 -0.108
-1.936 0.2000 -0.113
-0.478 0.4000 -0.119
0.982 0.6000 -0.124
2.444 0.8000 -0.130
3.907 1.0000 -0.135
5.373 1.2000 -0.140
6.841 1.4000 -0.144
8.312 1.6000 -0.149
9.787 1.8000 -0.153
-2.0
-1.5
C_P
-1.0
-0.5
0.0
0.5
1.0

Fig.25i

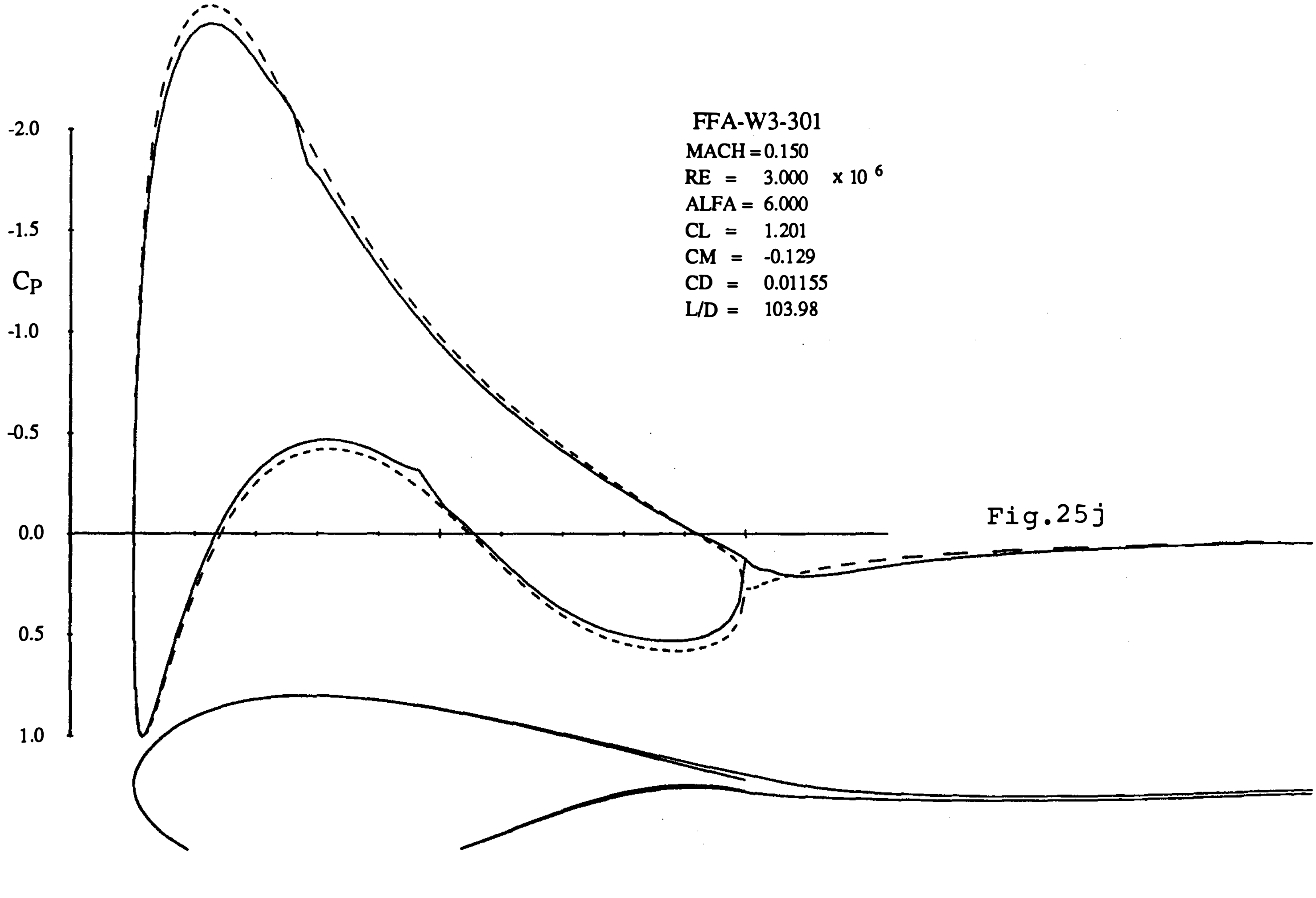

FFA-W3-301
MACH = 0.150
RE = 3.000 x 10 6
ALFA = 6.000
CL = 1.201
CM = -0.129
CD = 0.01155
L/D = 103.98
-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0

Fig.25j

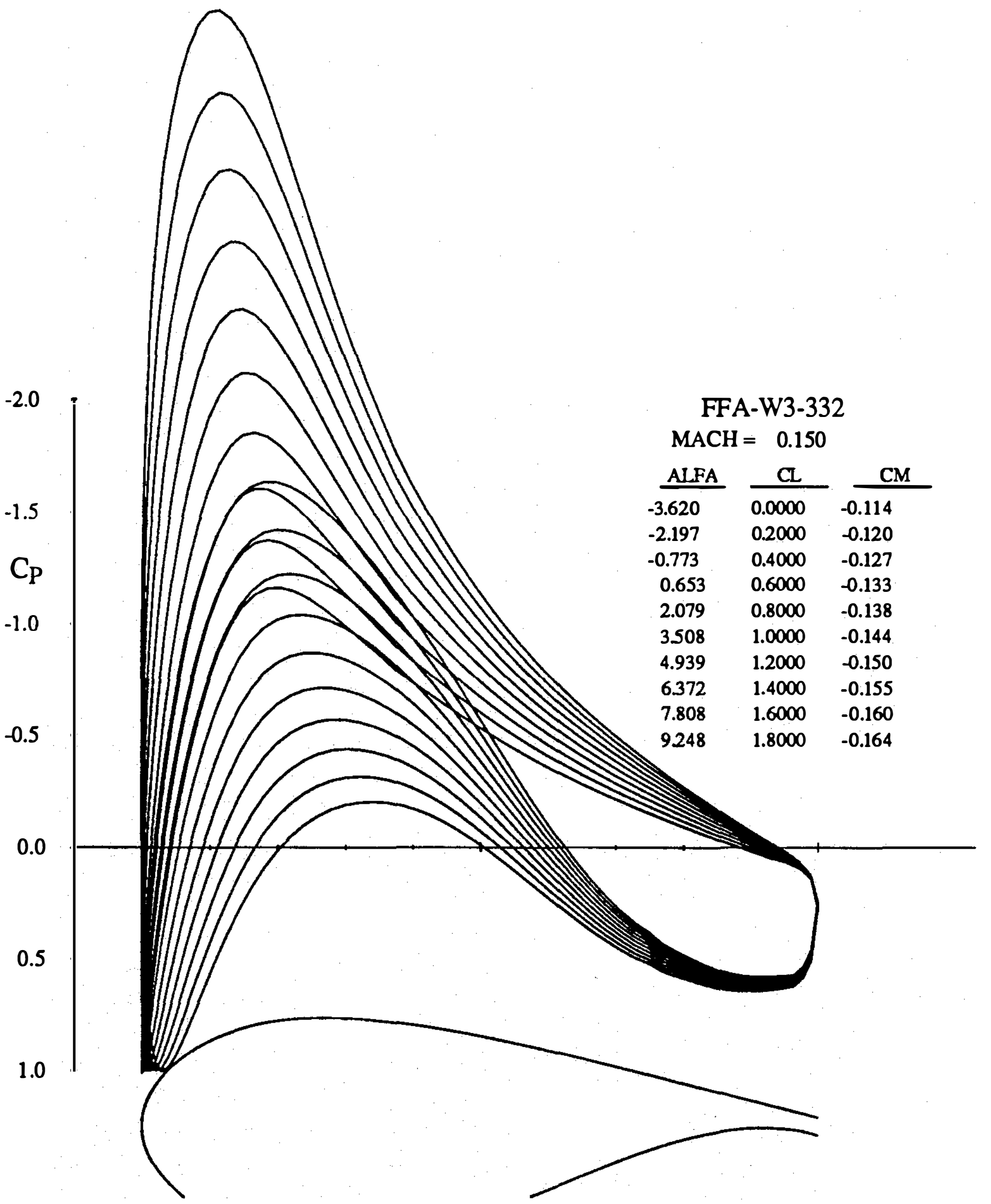

-2.0
-1.5
CP
-1.0
-0.5
0.0
0.5
1.0
FFA-W3-332
MACH = 0.150
ALFA CL CM
-3.620 0.0000 -0.114
-2.197 0.2000 -0.120
-0.773 0.4000 -0.127
0.653 0.6000 -0.133
2.079 0.8000 -0.138
3.508 1.0000 -0.144
4.939 1.2000 -0.150
6.372 1.4000 -0.155
7.808 1.6000 -0.160
9.248 1.8000 -0.164

Fig.25k

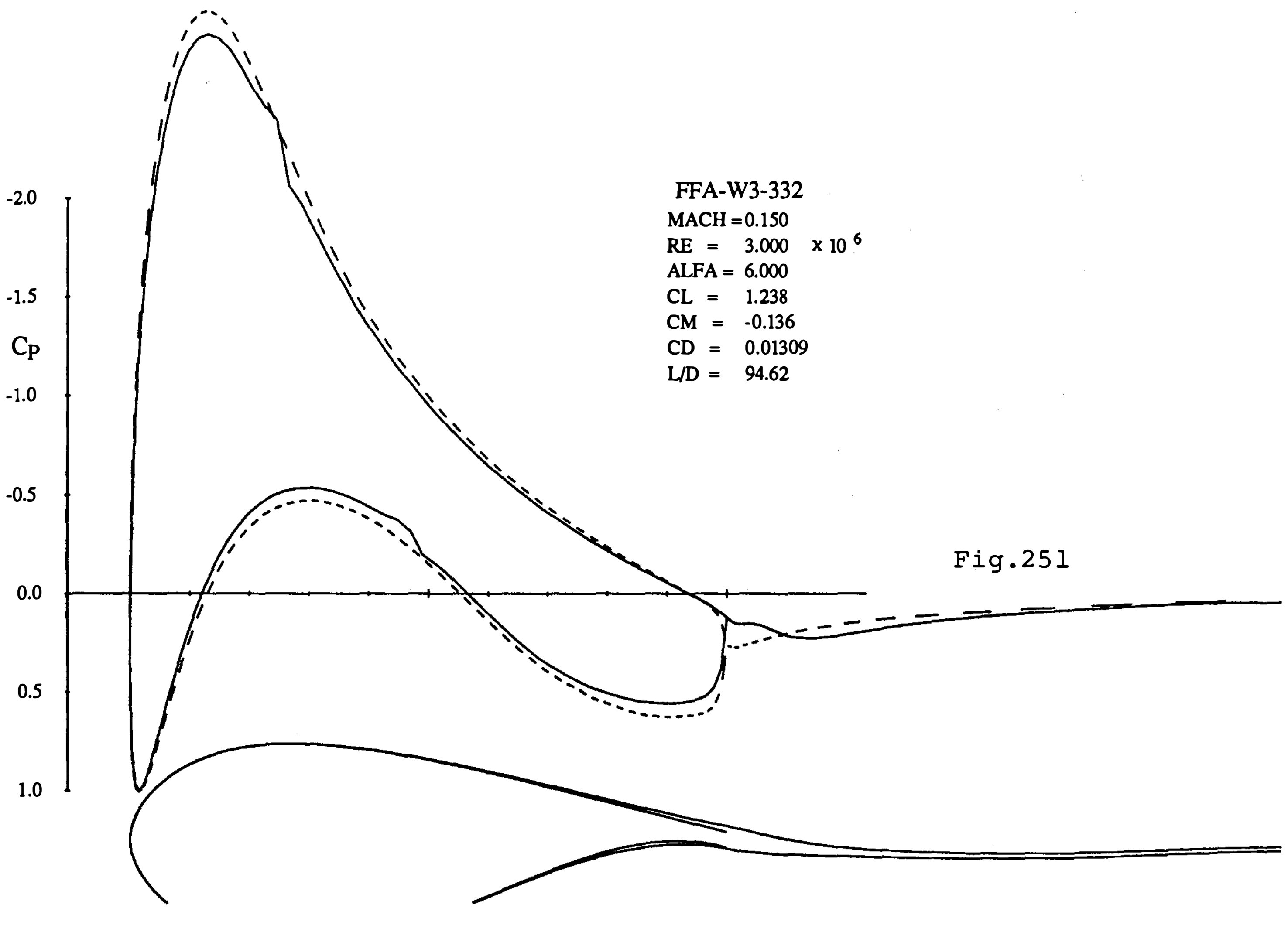

FFA-W3-332
MACH =0.150
RE = 3.000 x 10 6
ALFA = 6.000
CL = 1.238
CM = -0.136
CD = 0.01309
L/D = 94.62
-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0

Fig.251

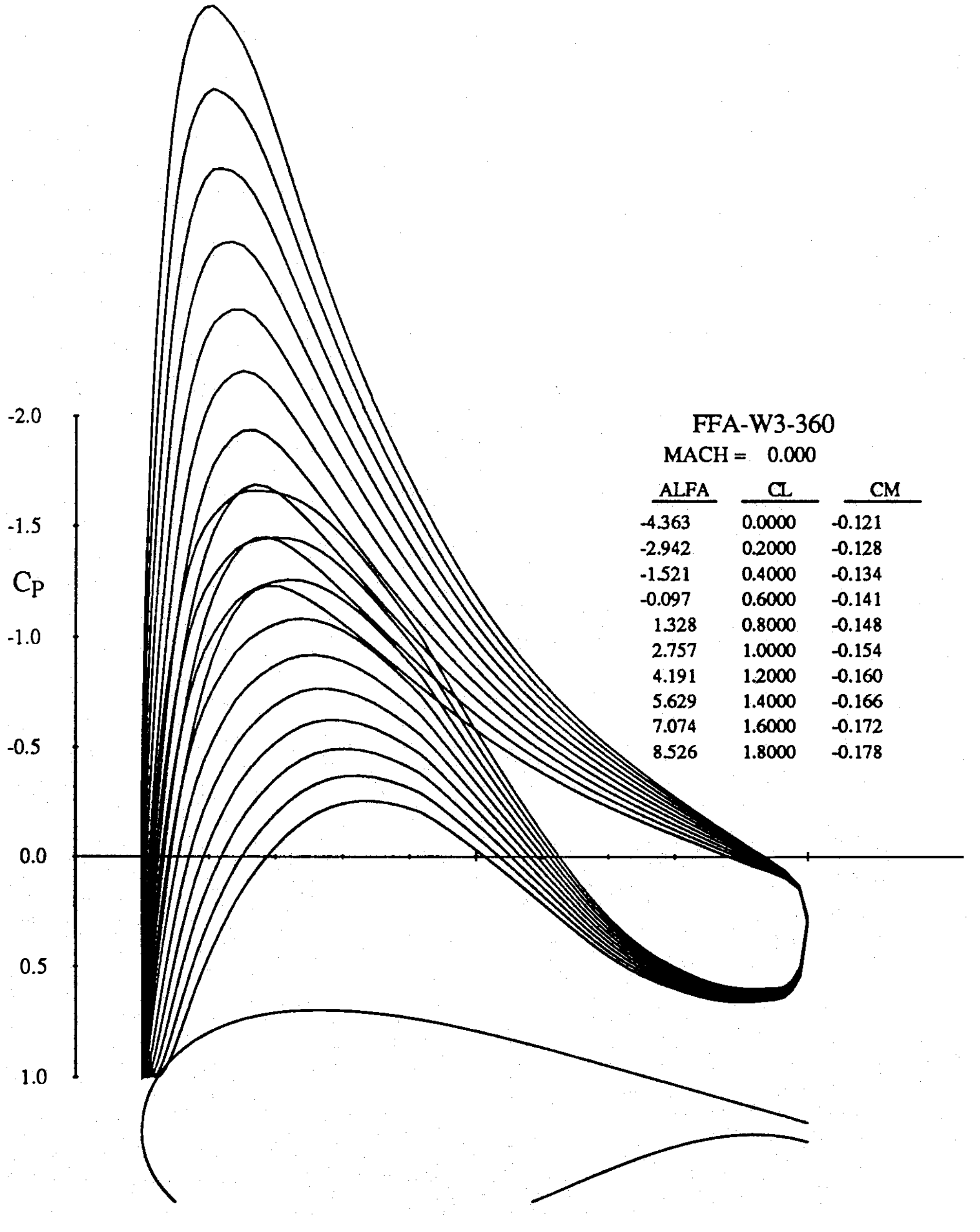

FFA-W3-360
MACH = 0.000
ALFA CL CM
-4.363 0.0000 -0.121
-2.942 0.2000 -0.128
-1.521 0.4000 -0.134
-0.097 0.6000 -0.141
1.328 0.8000 -0.148
2.757 1.0000 -0.154
4.191 1.2000 -0.160
5.629 1.4000 -0.166
7.074 1.6000 -0.172
8.526 1.8000 -0.178
Cp
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0

Fig.25m

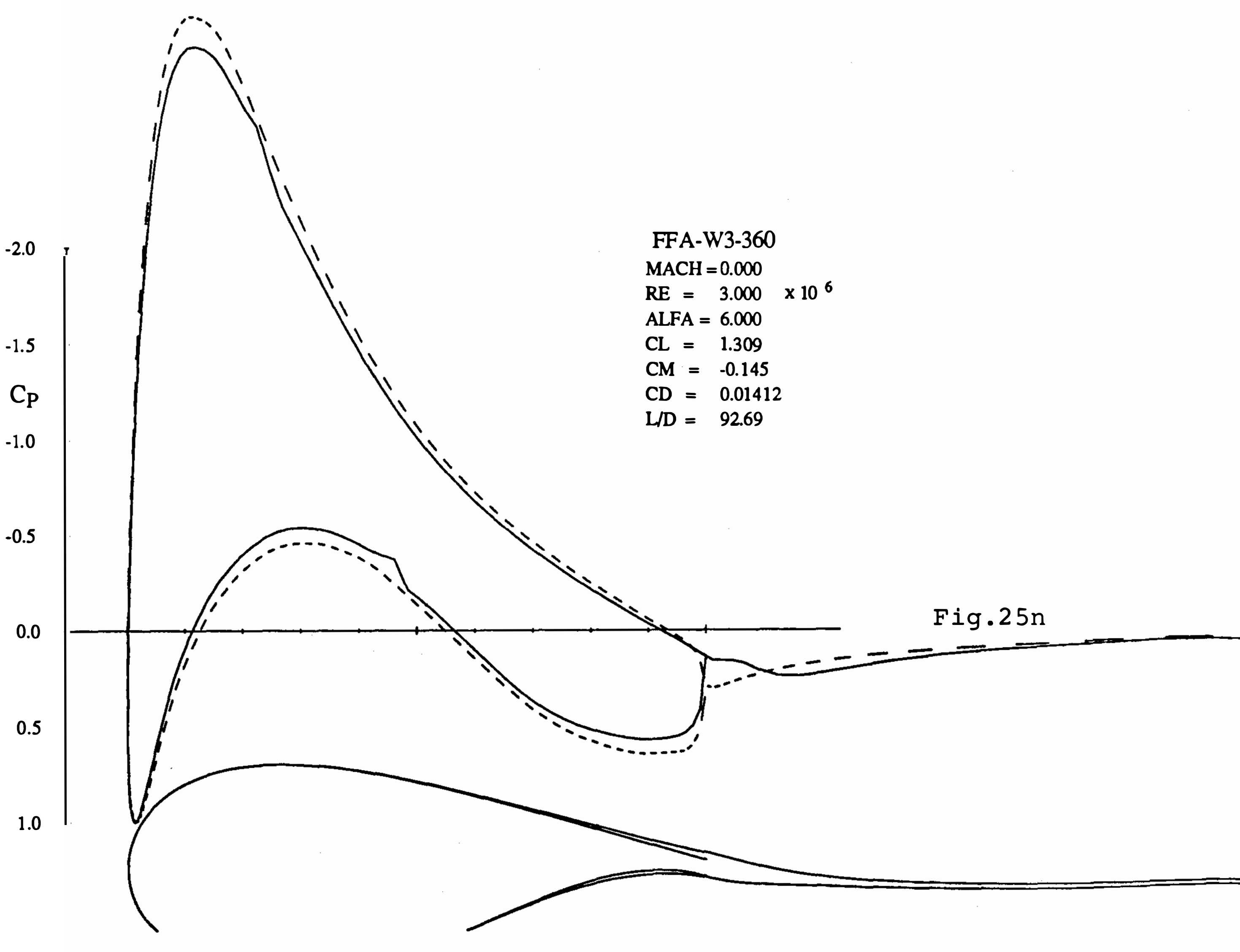

FFA-W3-360
MACH = 0.000
RE = 3.000 x 10 6
ALFA = 6.000
CL = 1.309
CM = -0.145
CD = 0.01412
L/D = 92.69
-2.0
-1.5
Cp
-1.0
-0.5
0.0
0.5
1.0

Fig.25n

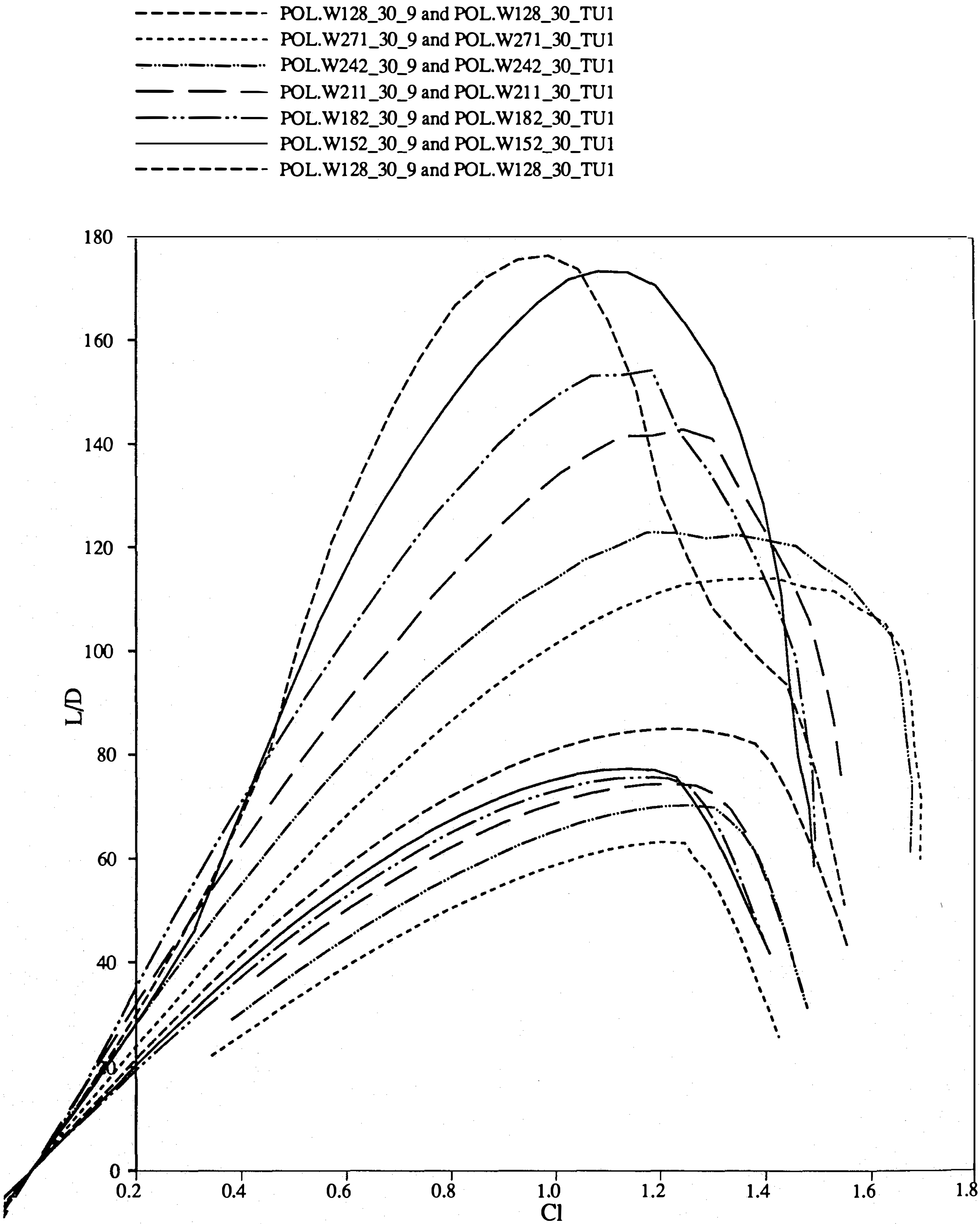


Fig.26 The lift/drag ratio versus C_l. XFOIL calculations for the FFA-W1-xxx airfoils at Re=3 million and M=0.15.

1) Free transition with Acrit=9.
2) Forced transition at x/c=1% at the suction side and x/c=10% at the pressure side.

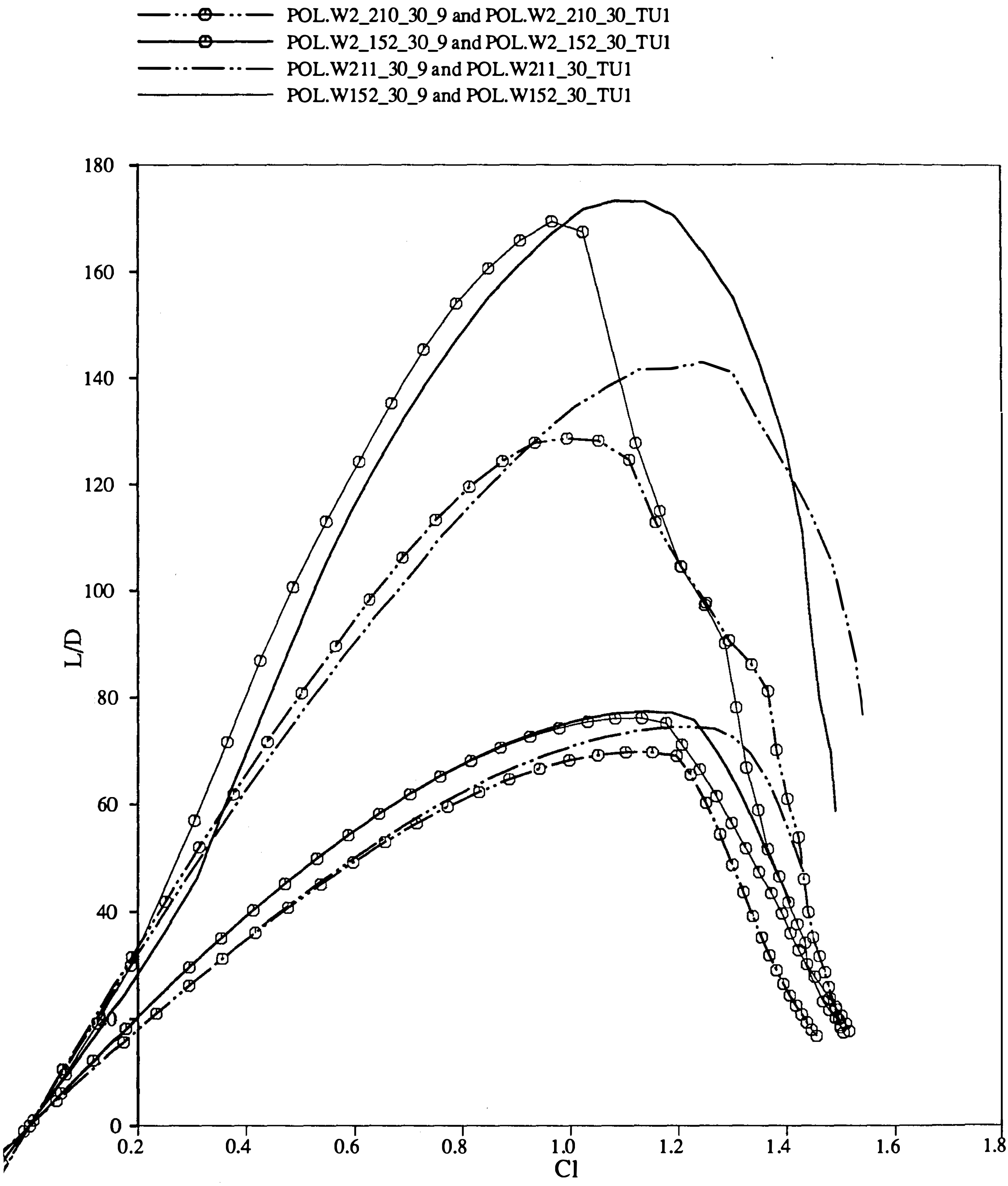


Fig.27 The lift/drag ratio versus C_l. Comparison of FFA-W1 airfoils and FFA-W2 airfoils.

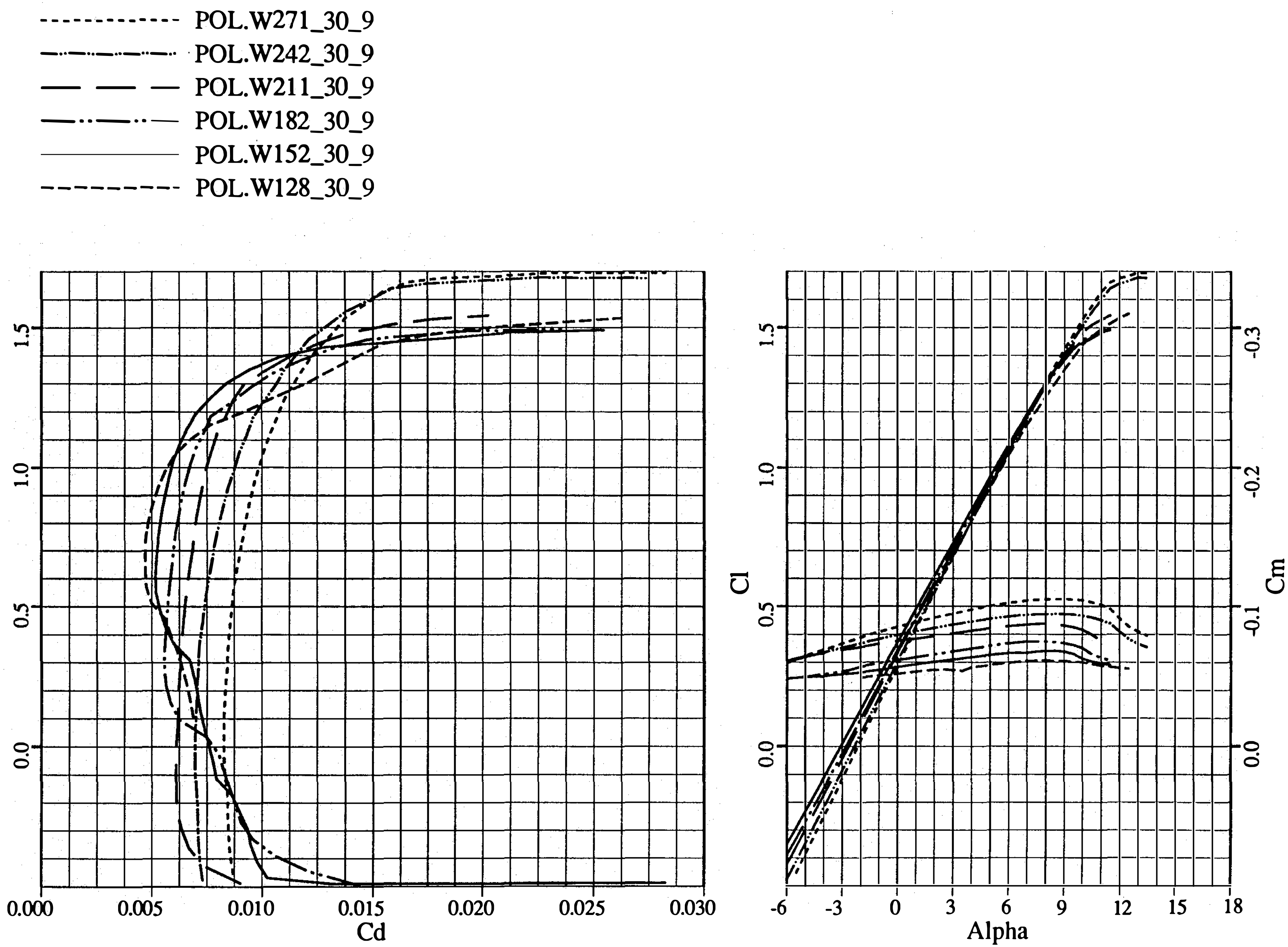


Fig.28 FFA-W1-xxx airfoils. XFOIL calculations at Re=3 million, M=0.15, free transition and Acrit=9.

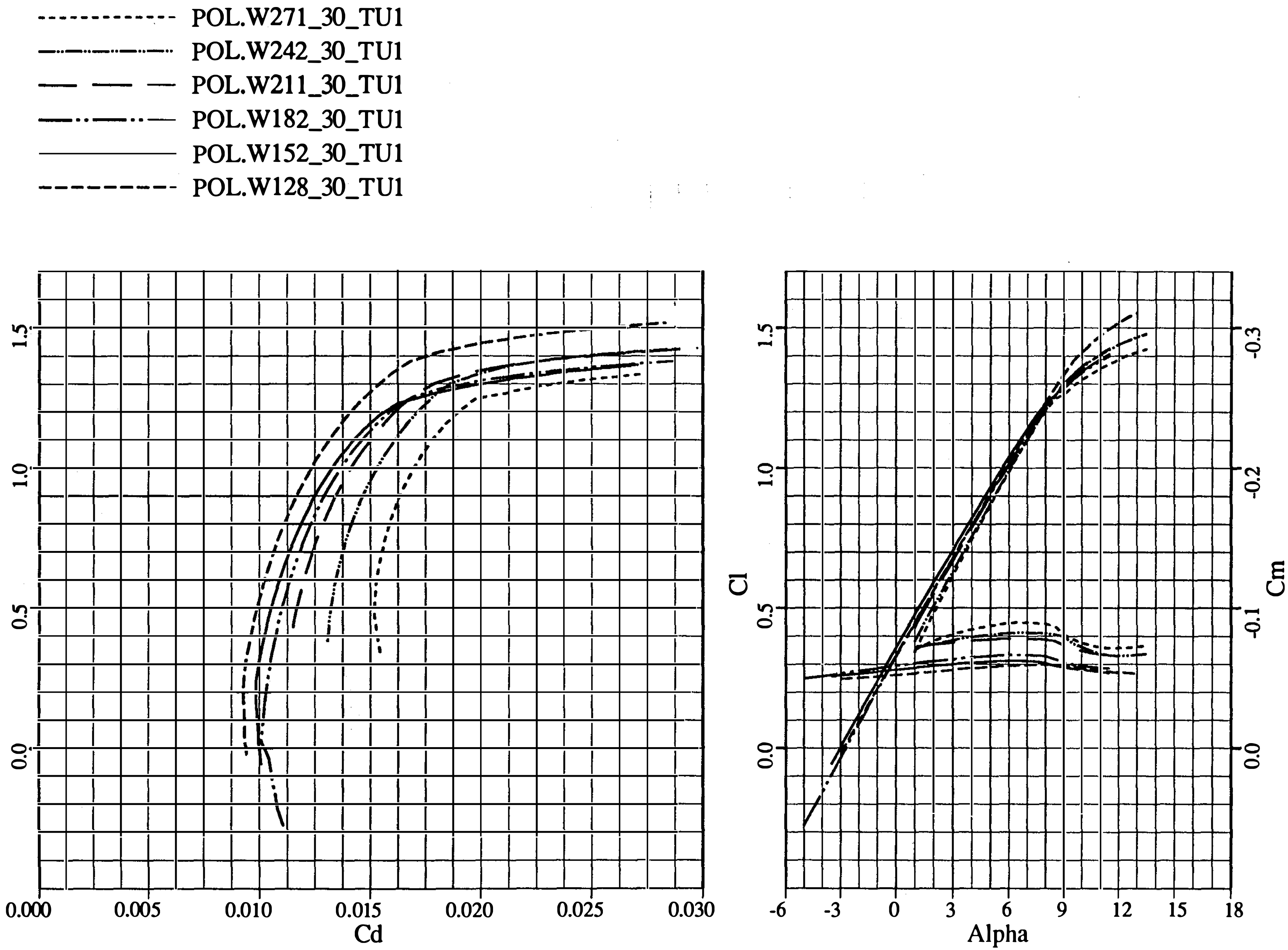


Fig.29 FFA-W1-xxx airfoils. XFOIL calculations at Re=3 million, M=0.15, forced transition at x/c=1% at the suction side and x/c=10% at the pressure side.

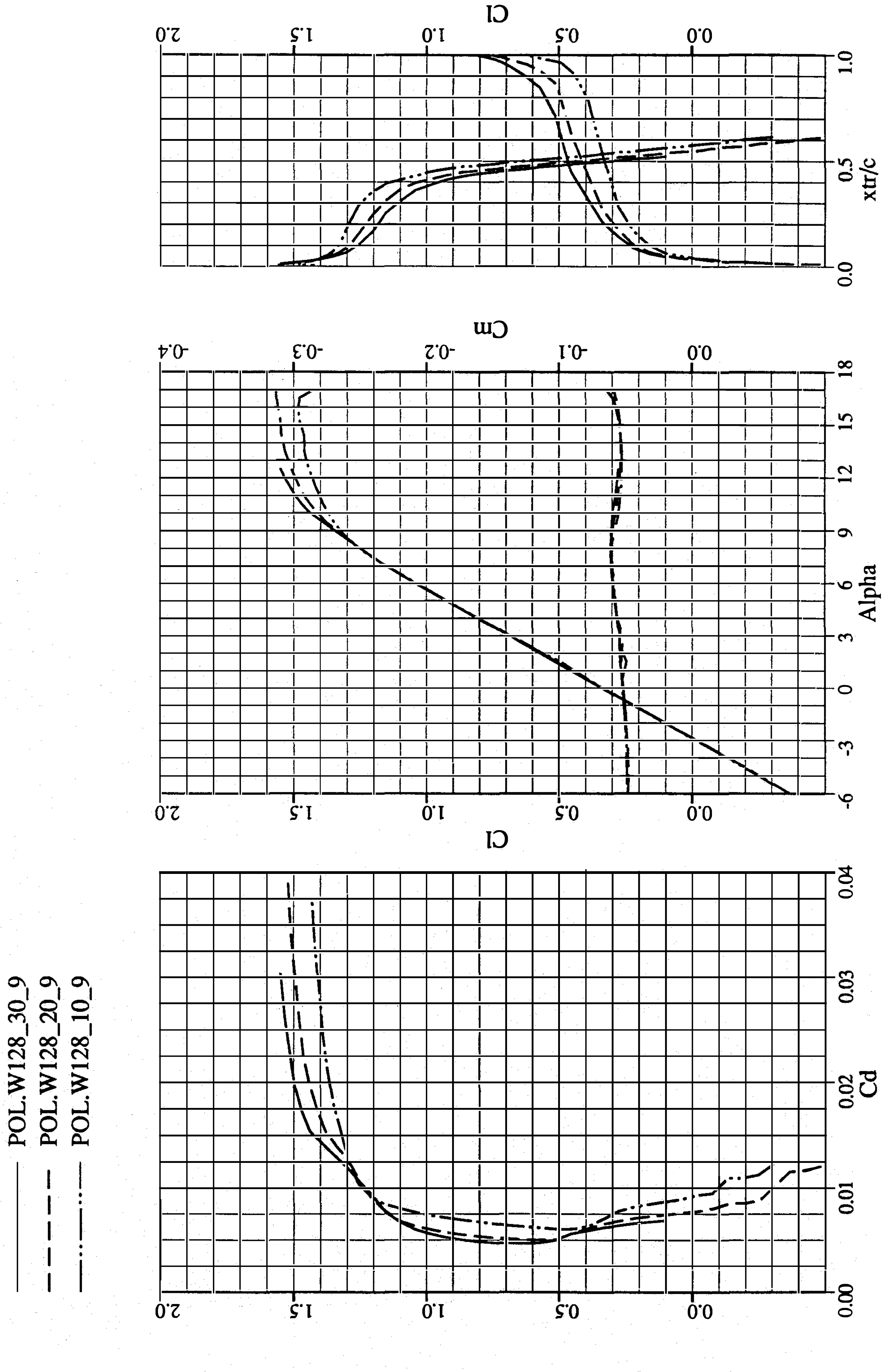

Cl
2.0
1.5
1.0
0.5
0.0
1.0
0.5
xtr/c
0.0
Cm
-0.4
-0.3
-0.2
-0.1
0.0
18
15
12
9
6
Alpha
3
0
-3
-6
Cl
0.04
0.03
0.02
Cd
0.01
0.00
POL.W128_30_9
POL.W128_20_9
POL.W128_10_9

Fig.30a

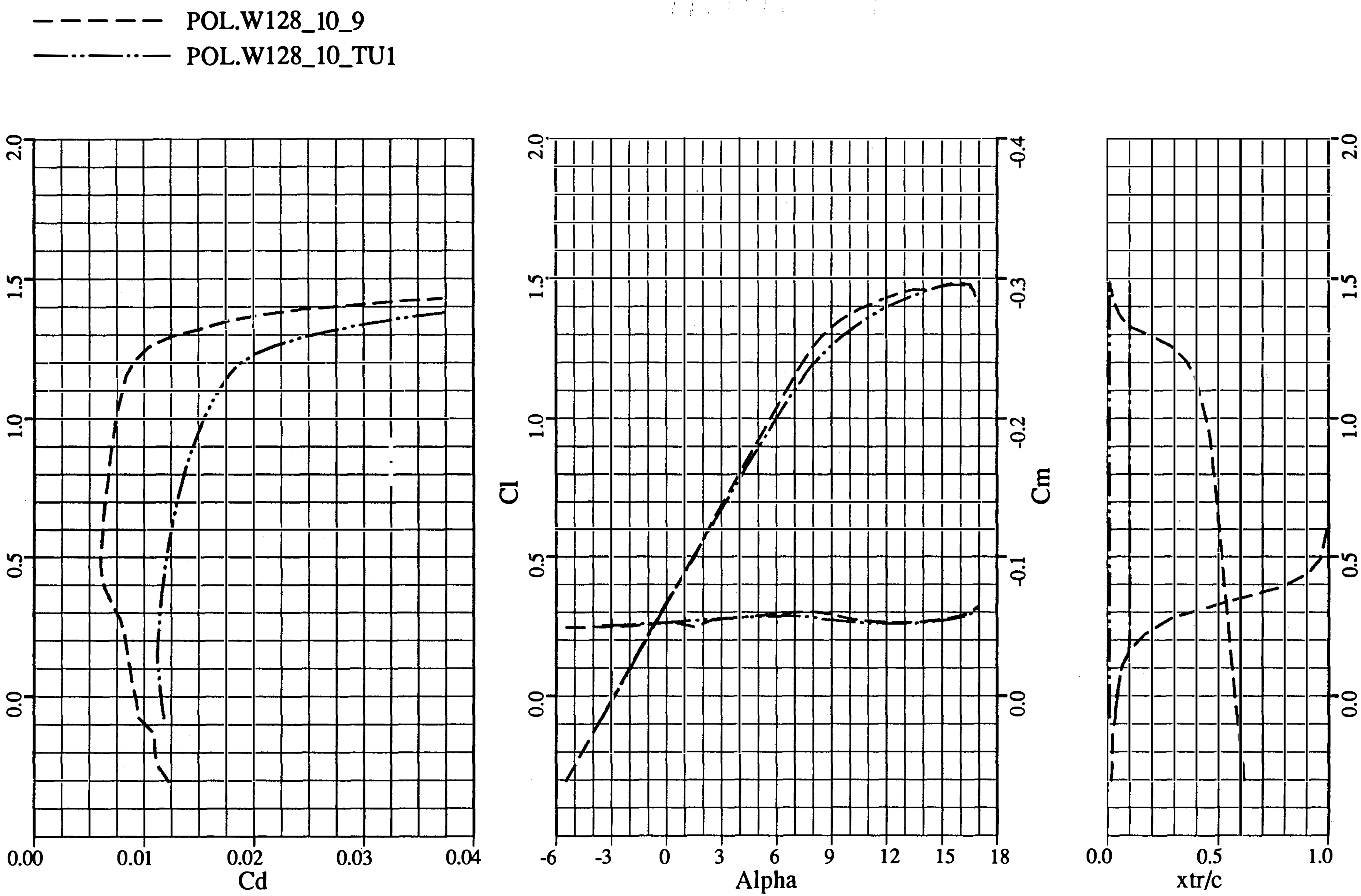

POL.W128_10_9
POL.W128_10_TU1
Cl
Cd
Alpha
Cm
xtr/c
0.00
0.01
0.02
0.03
0.04
-6
-3
0
3
6
9
12
15
18
0.0
0.5
1.0
1.5
2.0
-0.4
-0.3
-0.2
-0.1

Fig.30b

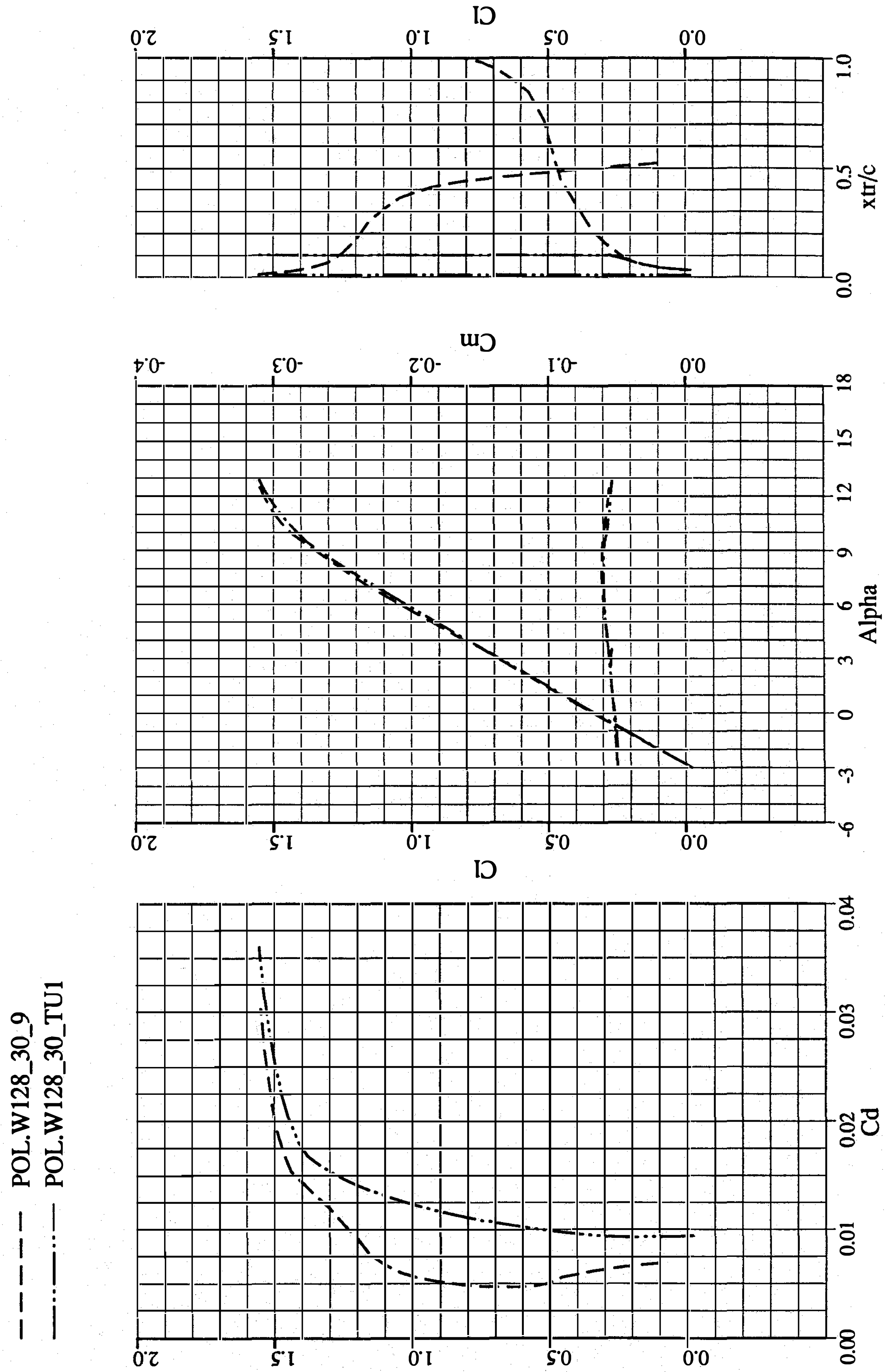

POL.W128_30_9
POL.W128_30_TU1
Cl
2.0
1.5
1.0
0.5
0.0
xtr/c
0.0
0.5
1.0
Cm
-0.4
-0.3
-0.2
-0.1
0.0
Alpha
-6
-3
0
3
6
9
12
15
18
Cd
0.00
0.01
0.02
0.03
0.04

Fig.30c

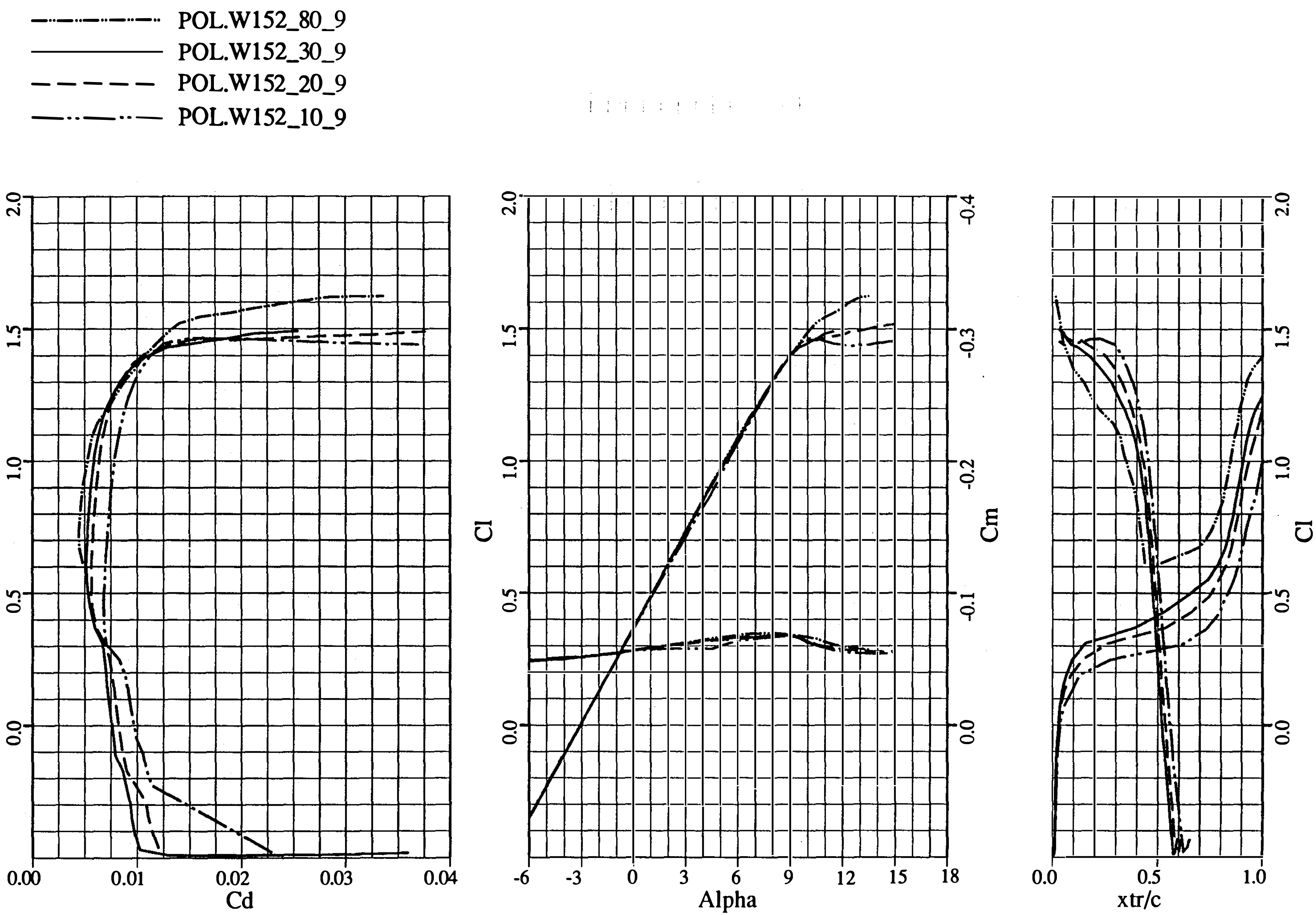

POL.W152_80_9
POL.W152_30_9
POL.W152_20_9
POL.W152_10_9
Cl
Cd
0.00
0.01
0.02
0.03
0.04
0.0
0.5
1.0
1.5
2.0
Alpha
-6
-3
0
3
6
9
12
15
18
Cm
0.0
-0.1
-0.2
-0.3
-0.4
xtr/c
0.0
0.5
1.0

Fig.31a

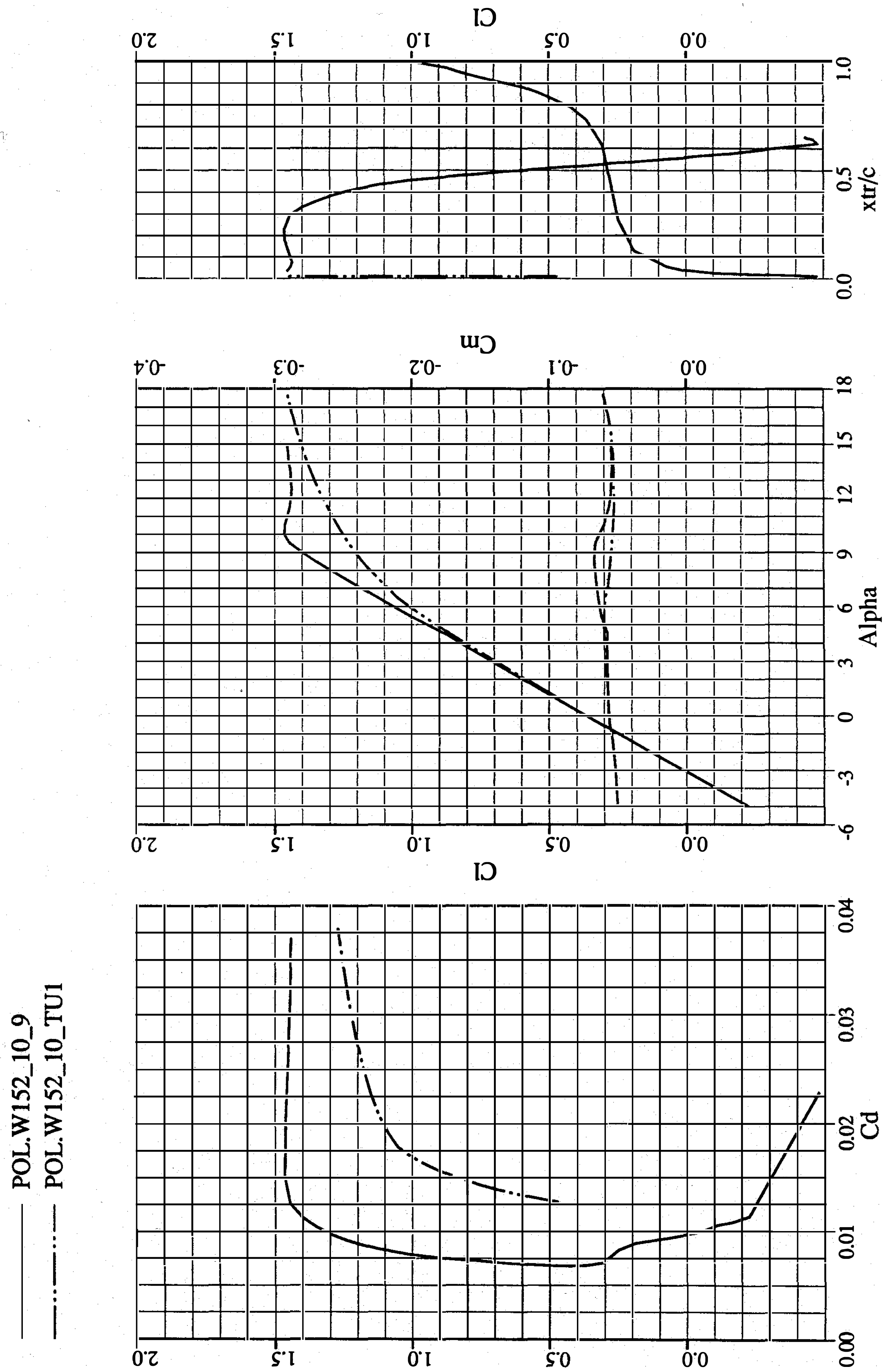


Fig.31b

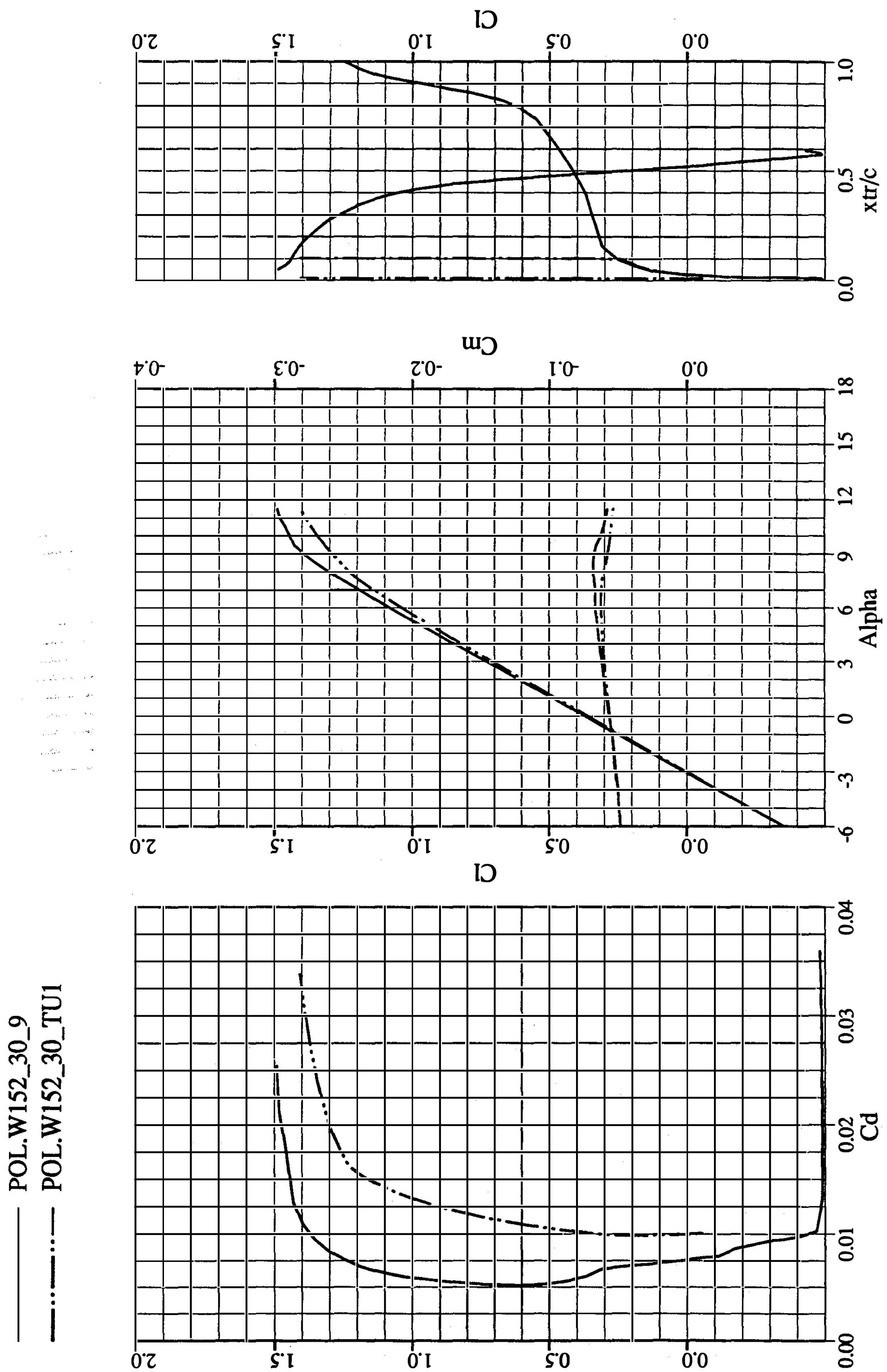


Fig.31c

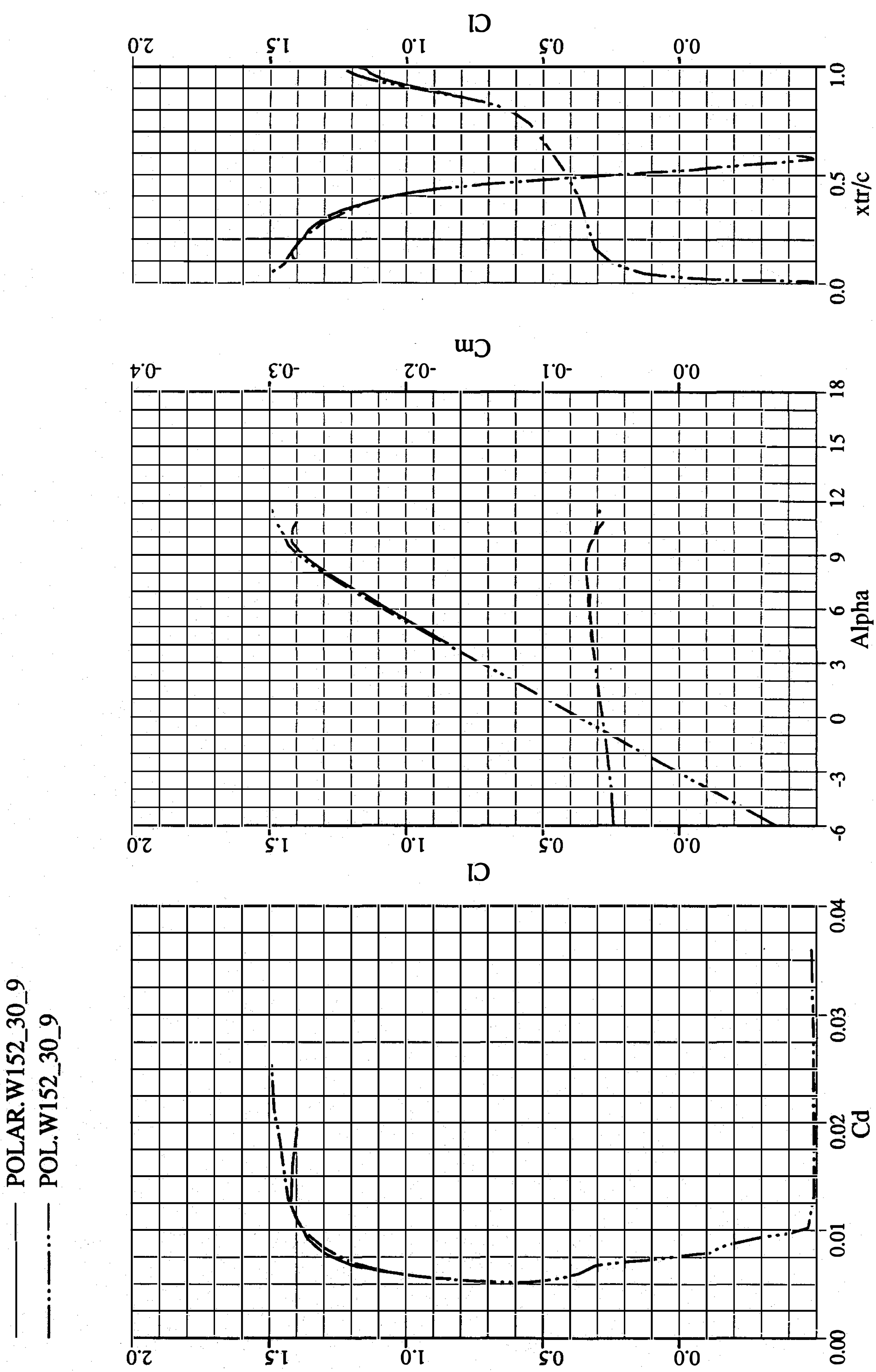

POLAR.W152_30_9
POL.W152_30_9
Cl
Cm
xtr/c
Alpha
Cd
2.0
1.5
1.0
0.5
0.0
-0.4
-0.3
-0.2
-0.1
0.0
1.0
0.5
0.0
18
15
12
9
6
3
0
-3
-6
0.04
0.03
0.02
0.01
0.00

Fig.31d

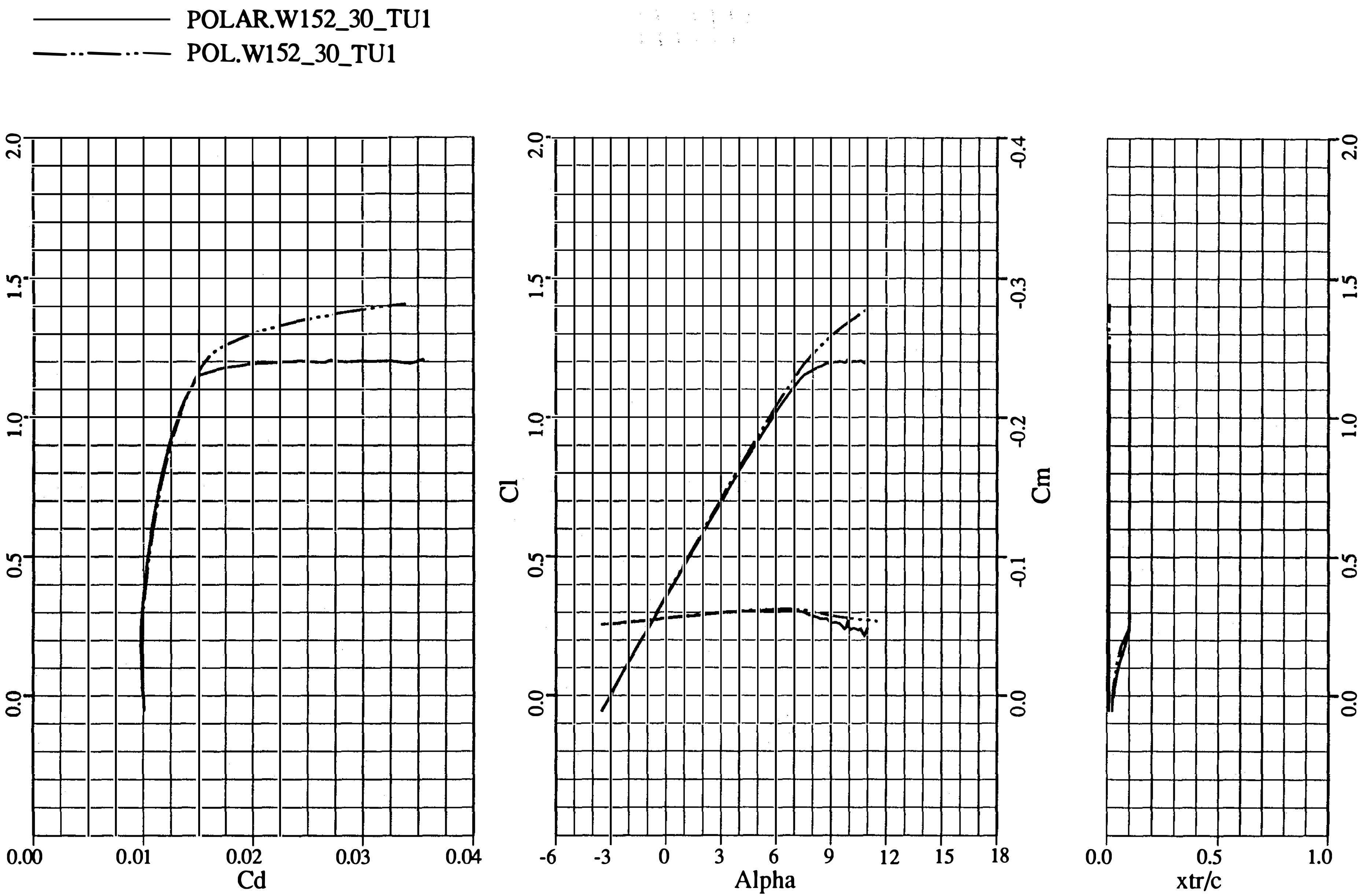

POLAR.W152_30_TU1
POL.W152_30_TU1
Cl
Cd
Alpha
Cm
xtr/c
0.00
0.01
0.02
0.03
0.04
-6
-3
0
3
6
9
12
15
18
0.0
0.5
1.0
1.5
2.0
-0.4
-0.3
-0.2
-0.1

Fig.3le

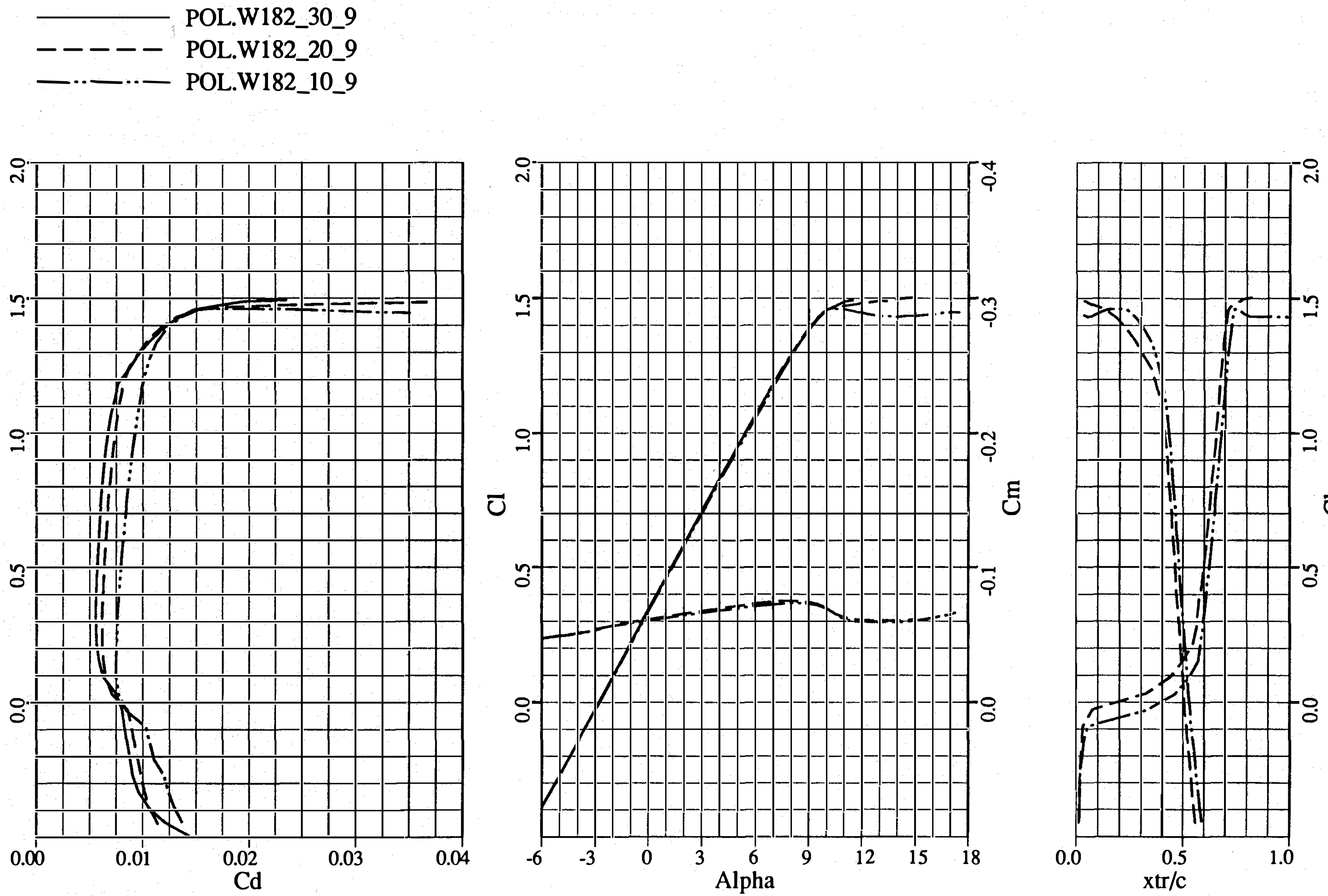

POL.W182_30_9
POL.W182_20_9
POL.W182_10_9
Cl
Cd
Alpha
Cm
xtr/c
0.00
0.01
0.02
0.03
0.04
-6
-3
0
3
6
9
12
15
18
0.0
0.5
1.0
1.5
2.0
-0.4
-0.3
-0.2
-0.1

Fig.32a

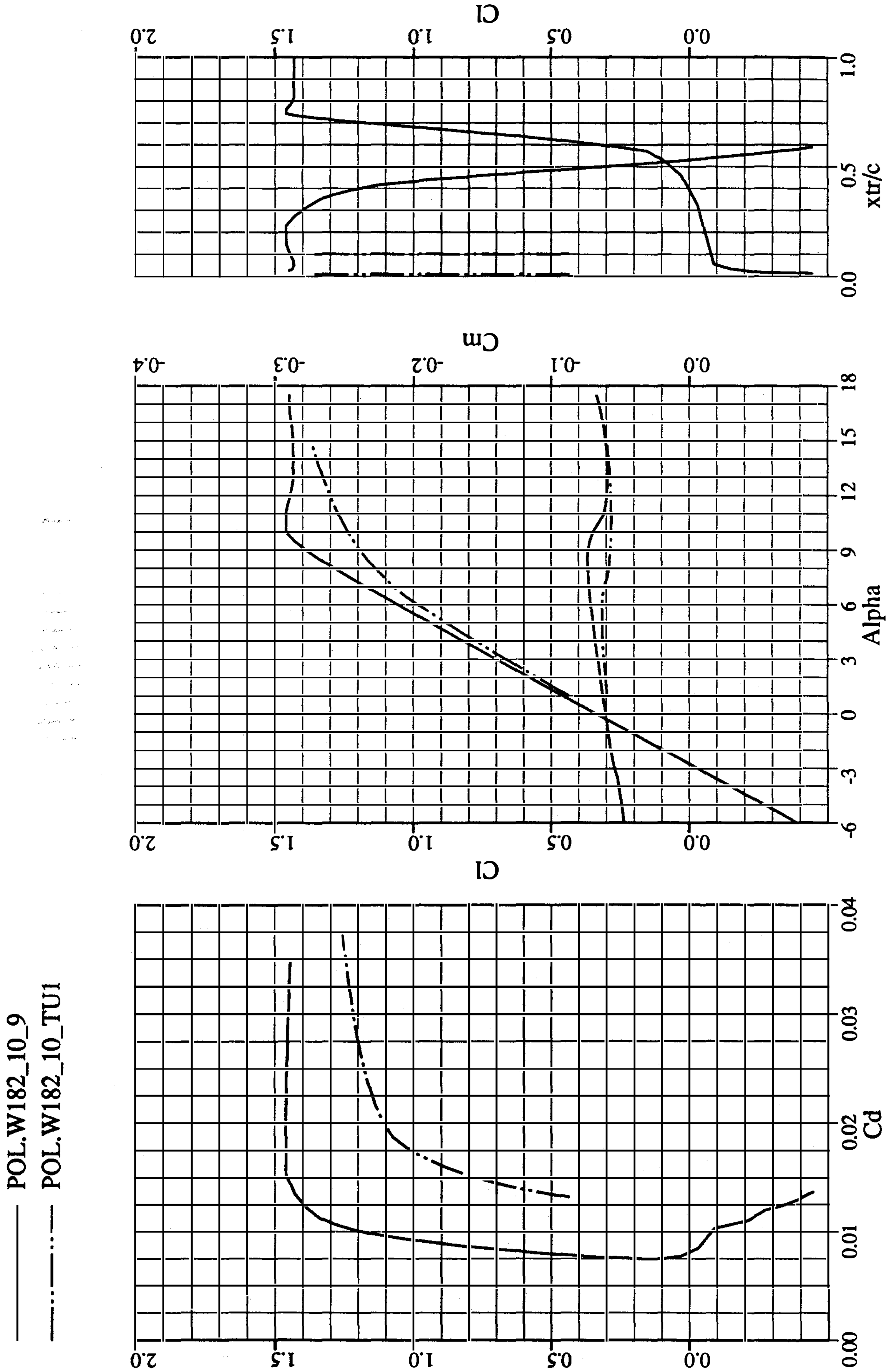


Fig.32b

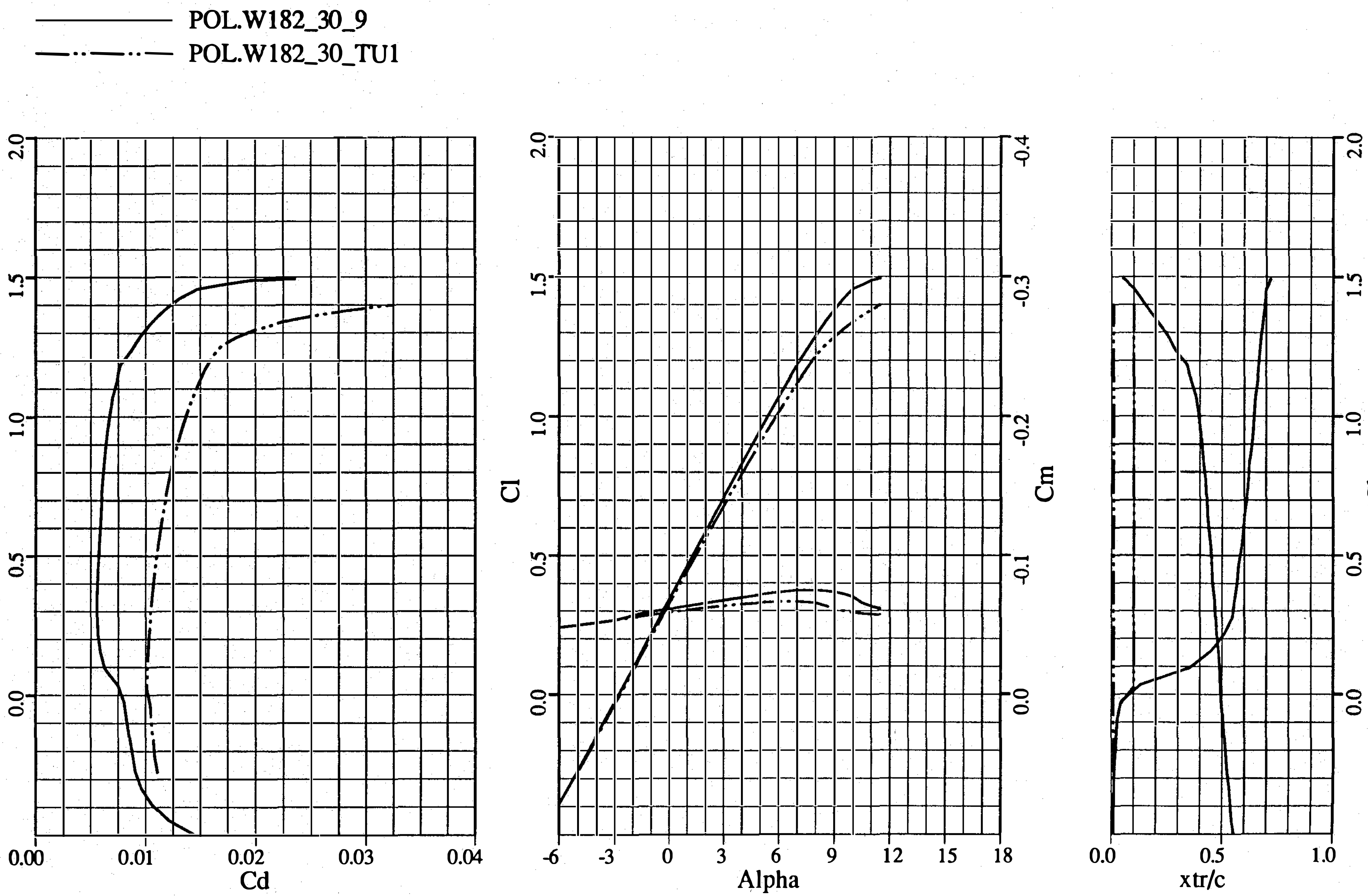

POL.W182_30_9
POL.W182_30_TU1
Cl
Cd
0.00
0.01
0.02
0.03
0.04
0.0
0.5
1.0
1.5
2.0
Alpha
-6
-3
0
3
6
9
12
15
18
Cm
0.0
-0.1
-0.2
-0.3
-0.4
xtr/c
0.0
0.5
1.0

Fig.32c

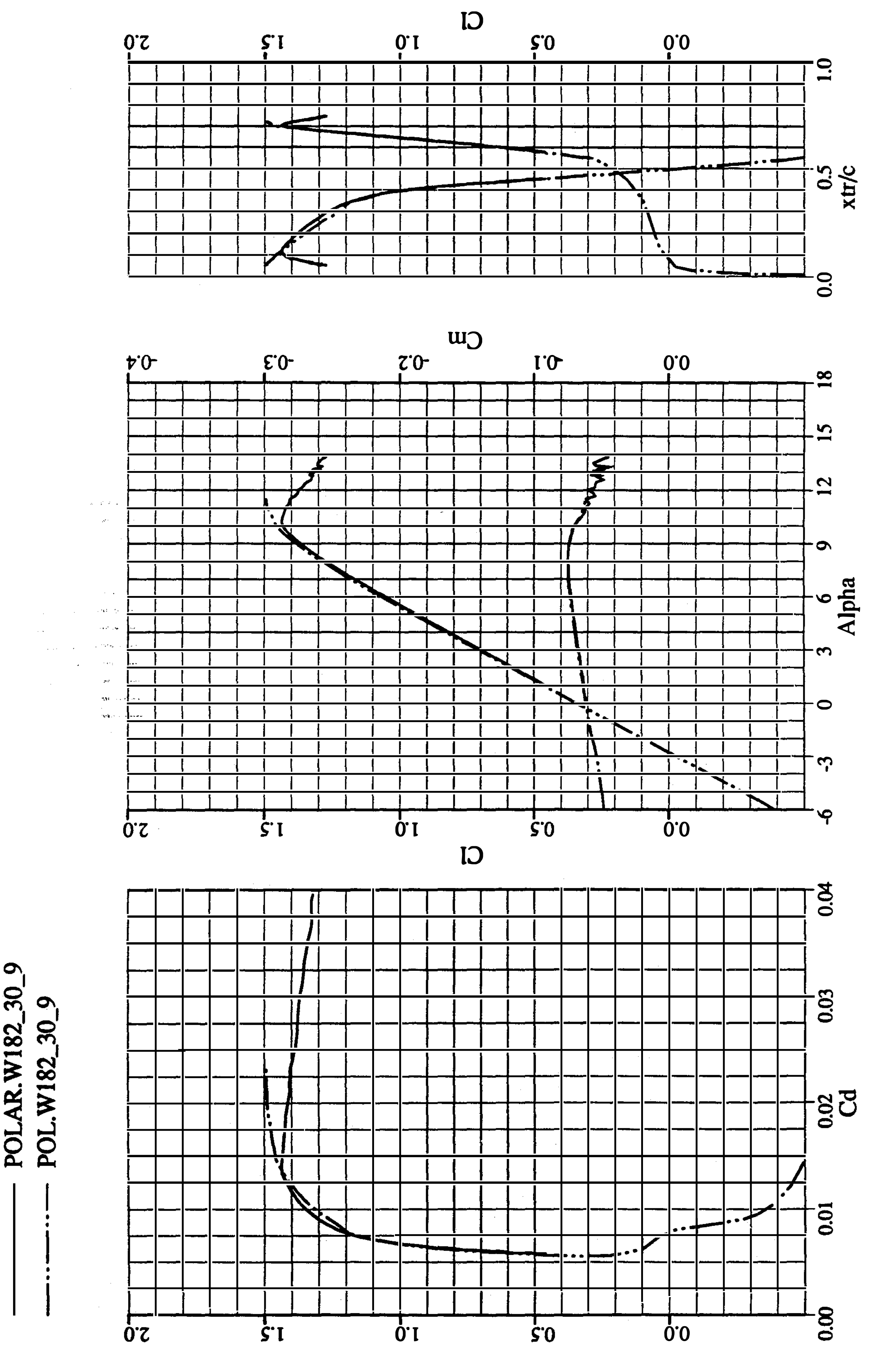


Fig.32d

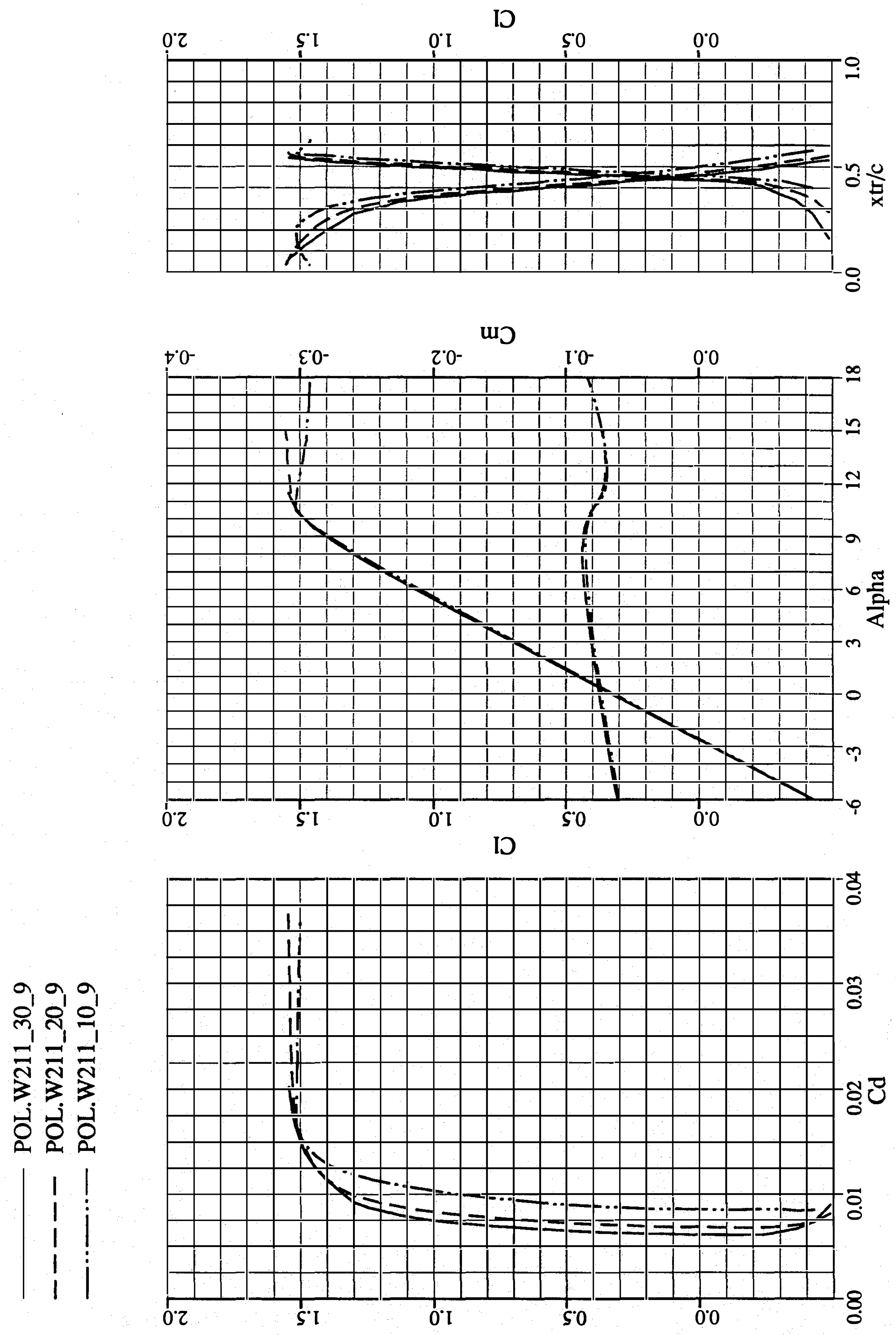


Fig.33a

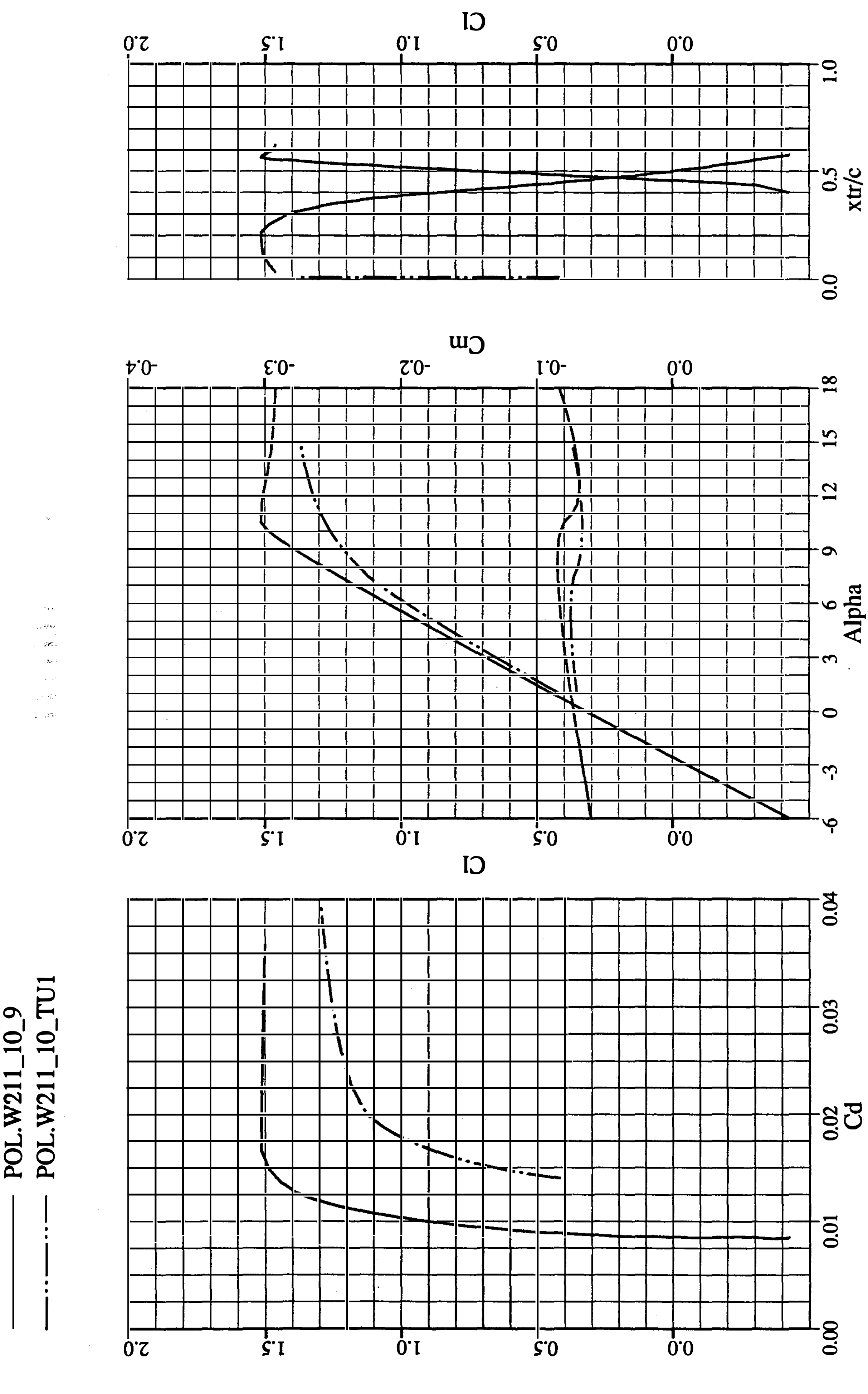

POL.W211_10_9
POL.W211_10_TU1
Cl
Cd
Cm
Alpha
xtr/c
0.0
0.5
1.0
1.5
2.0
-0.1
-0.2
-0.3
-0.4
0.00
0.01
0.02
0.03
0.04
-6
-3
0
3
6
9
12
15
18

Fig.33b

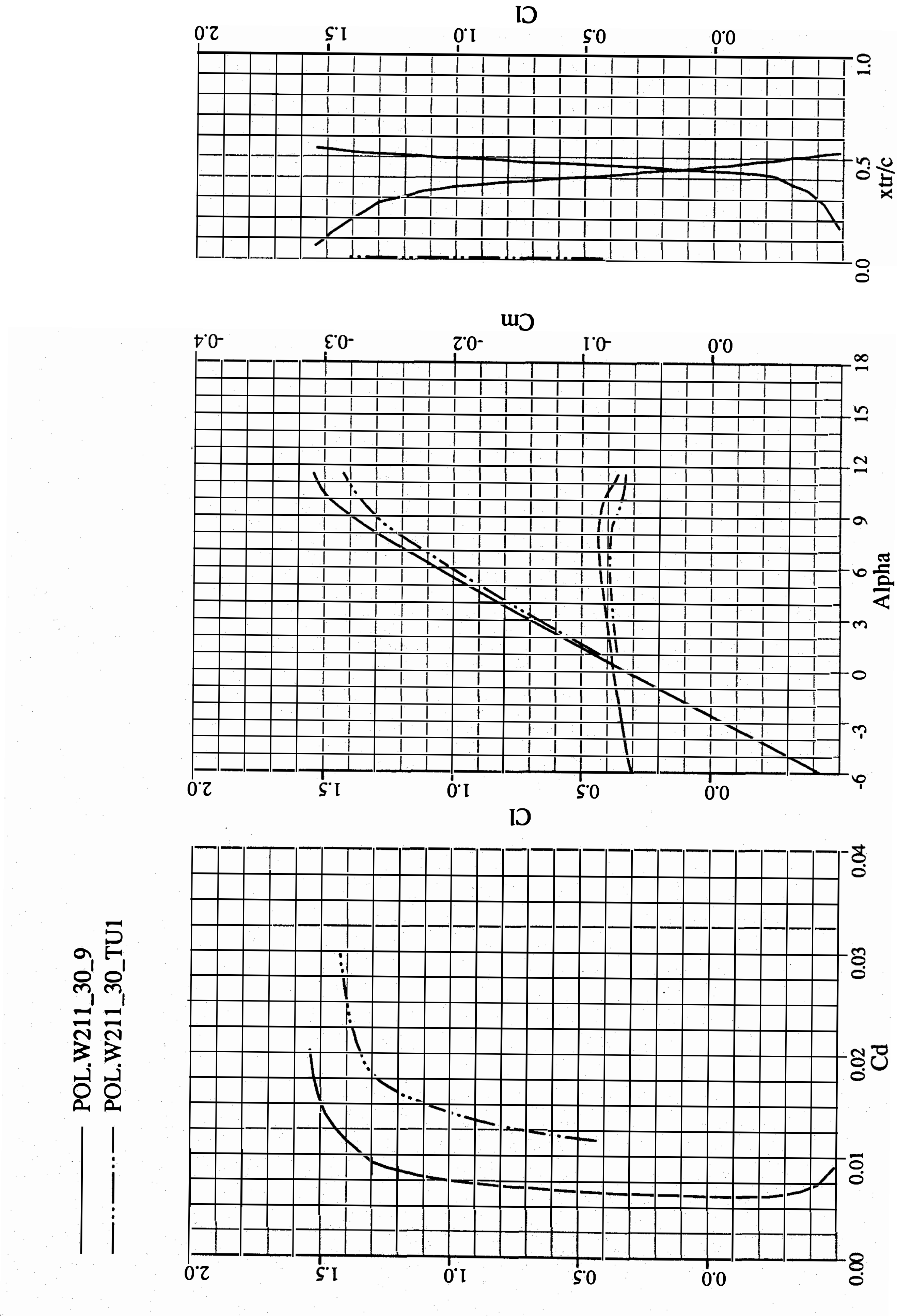


Fig.33c

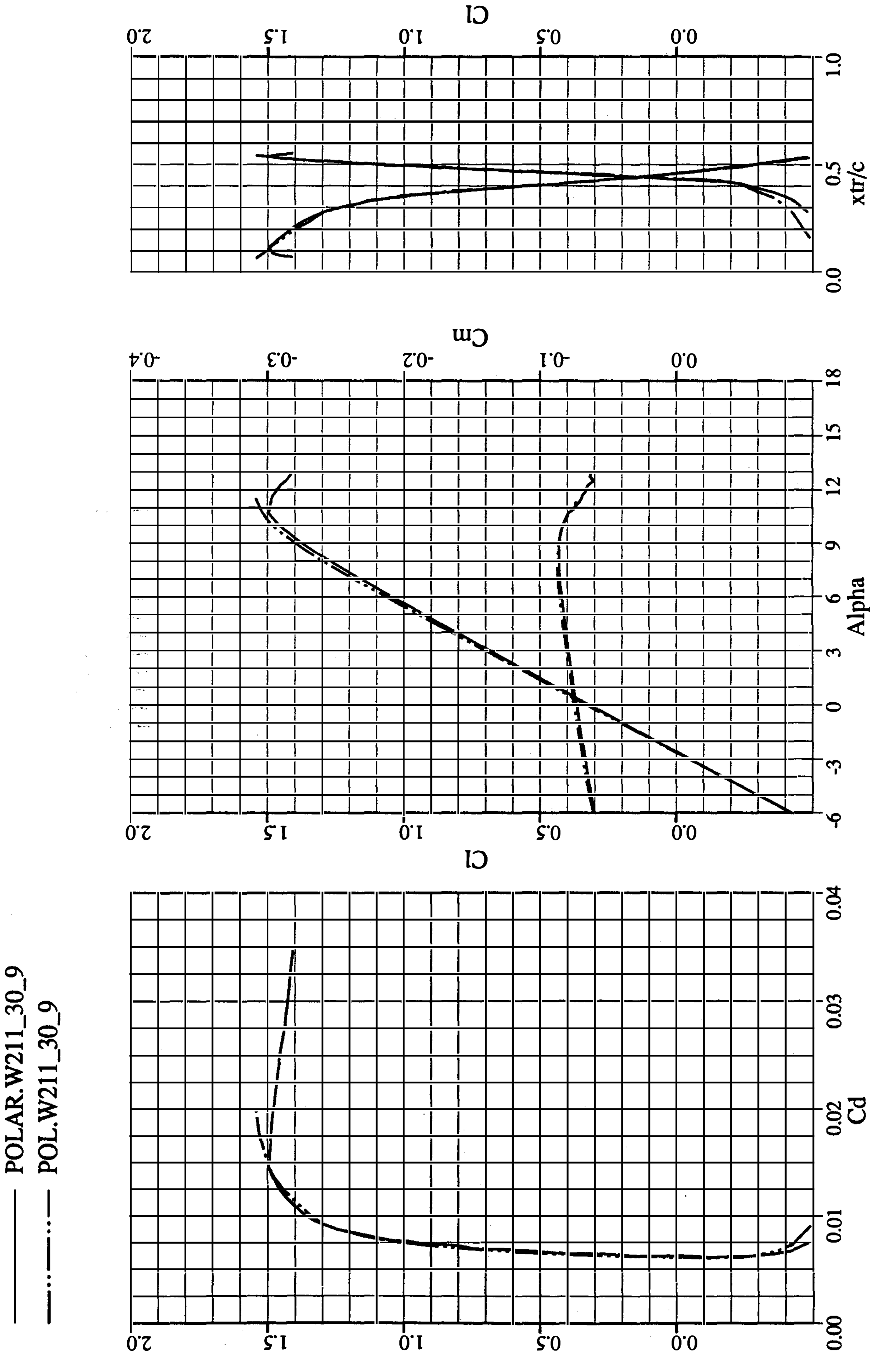


Fig.33d

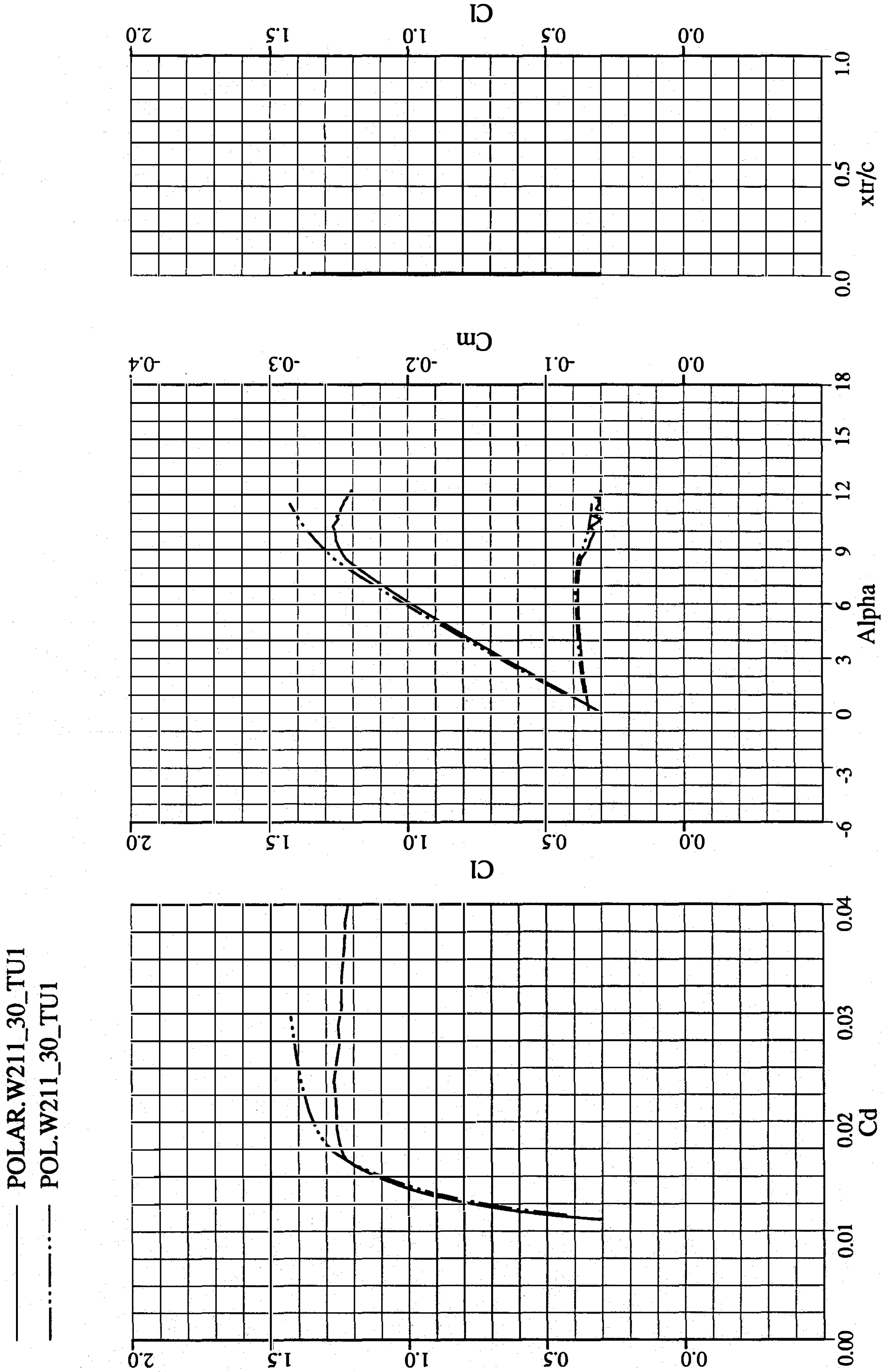

POLAR.W211_30_TU1
POL.W211_30_TU1
Cl
Cd
Cm
Alpha
xtr/c
0.00
0.01
0.02
0.03
0.04
-6
-3
0
3
6
9
12
15
18
0.0
0.5
1.0
1.5
2.0
-0.1
-0.2
-0.3
-0.4

Fig.33e

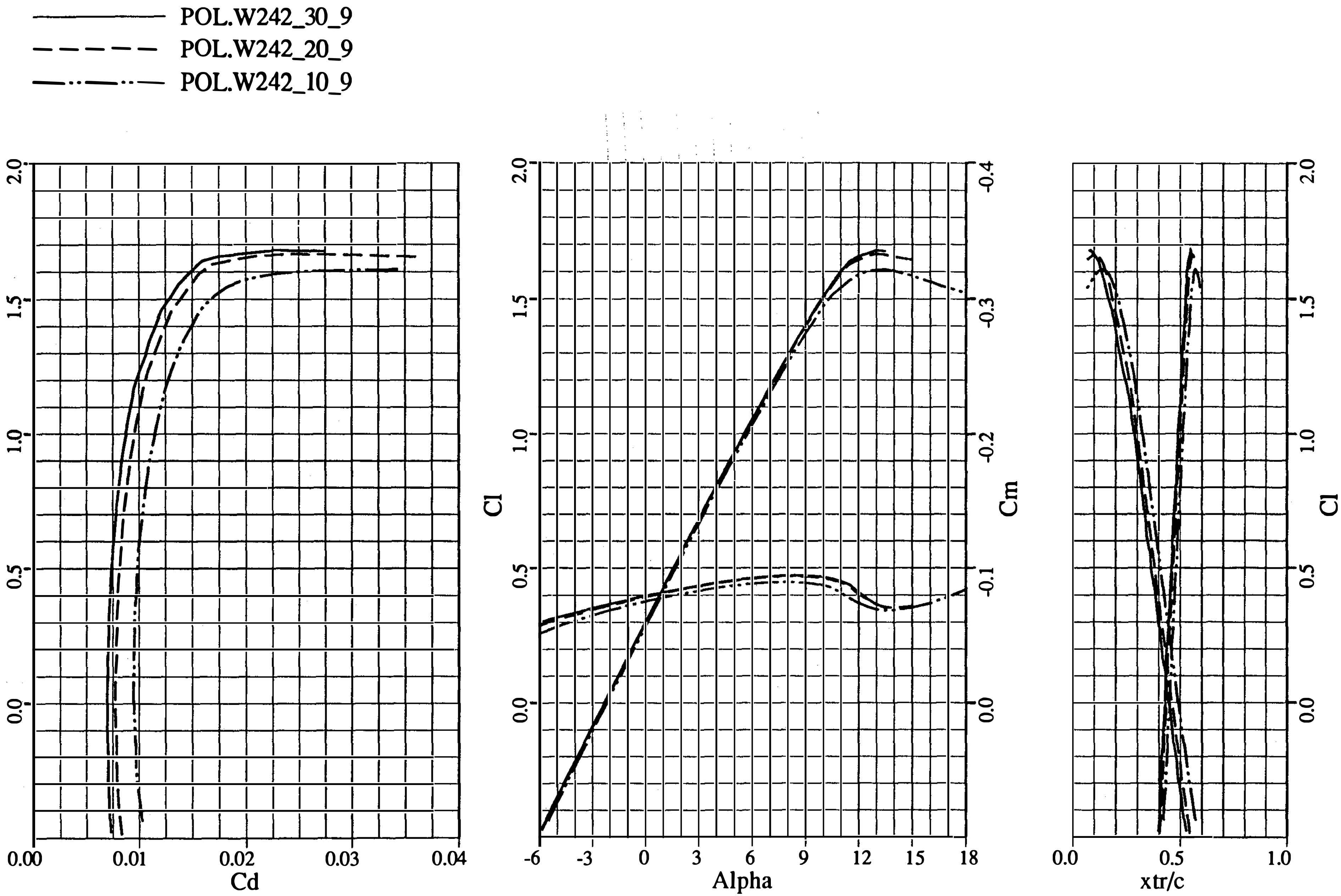

POL.W242_30_9
POL.W242_20_9
POL.W242_10_9
Cd
0.00
0.01
0.02
0.03
0.04
0.0
0.5
1.0
1.5
2.0
Cl
Alpha
-6
-3
0
3
6
9
12
15
18
Cm
0.0
-0.1
-0.2
-0.3
-0.4
xtr/c
0.0
0.5
1.0
Cl

Fig.34a

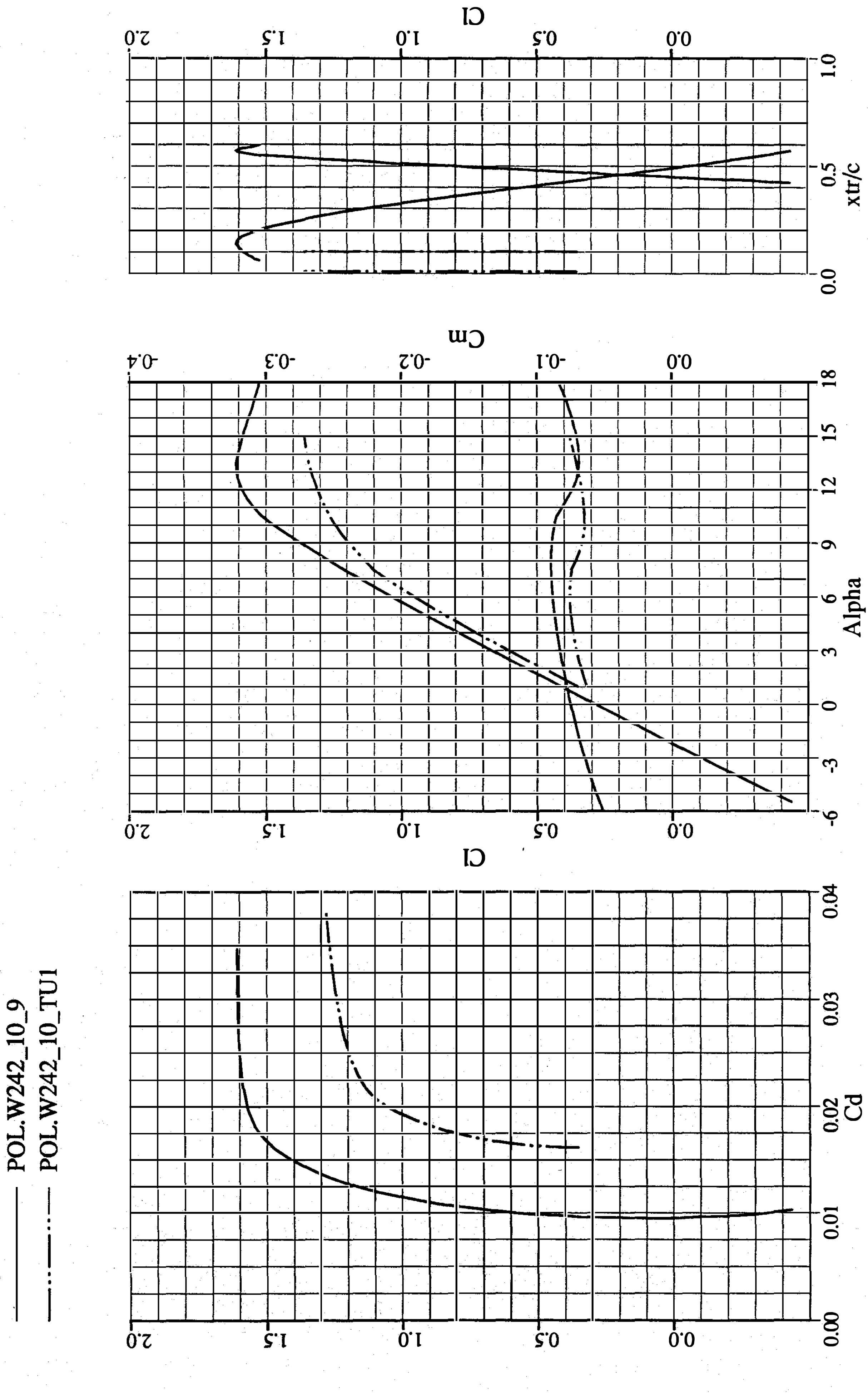


Fig.34b

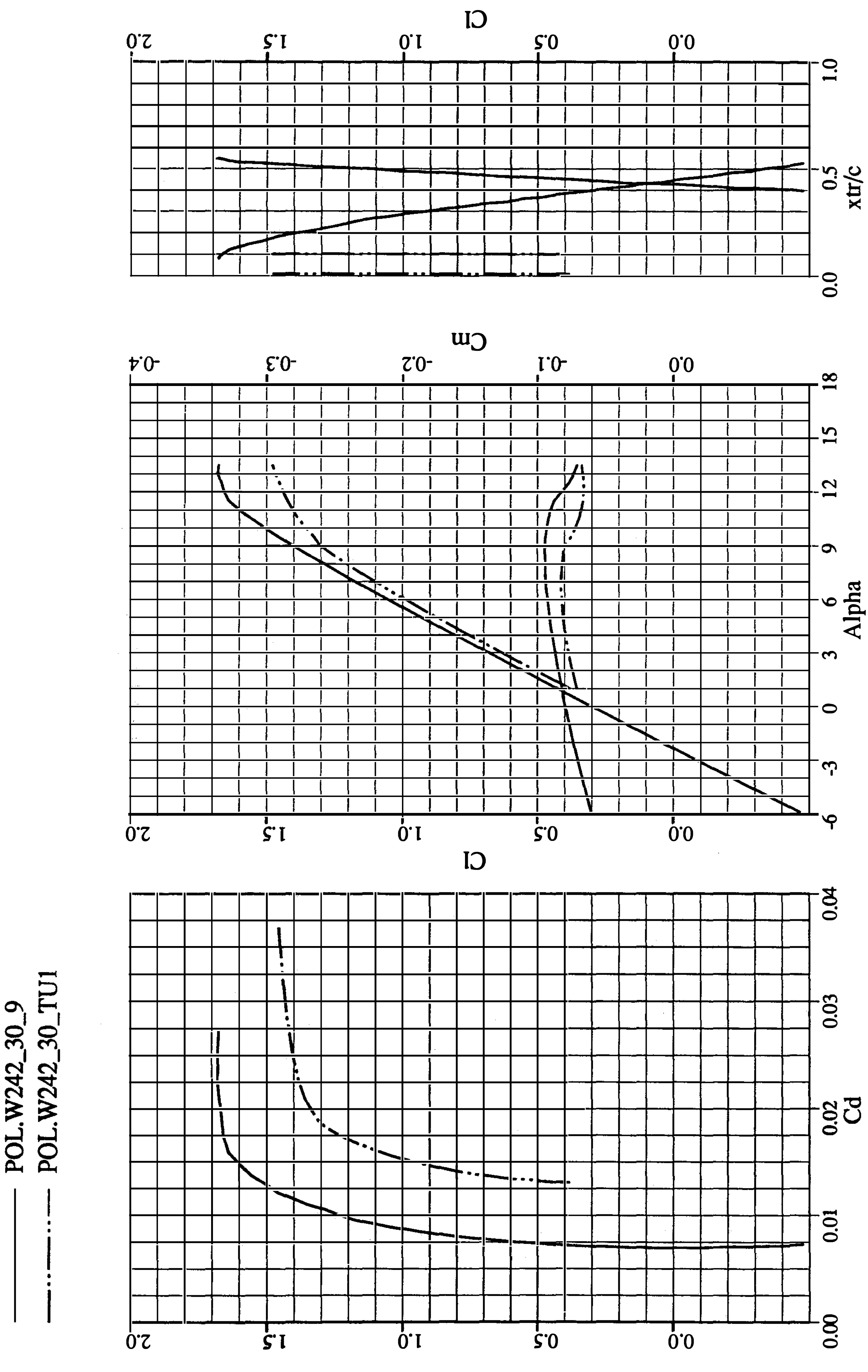


Fig.34c

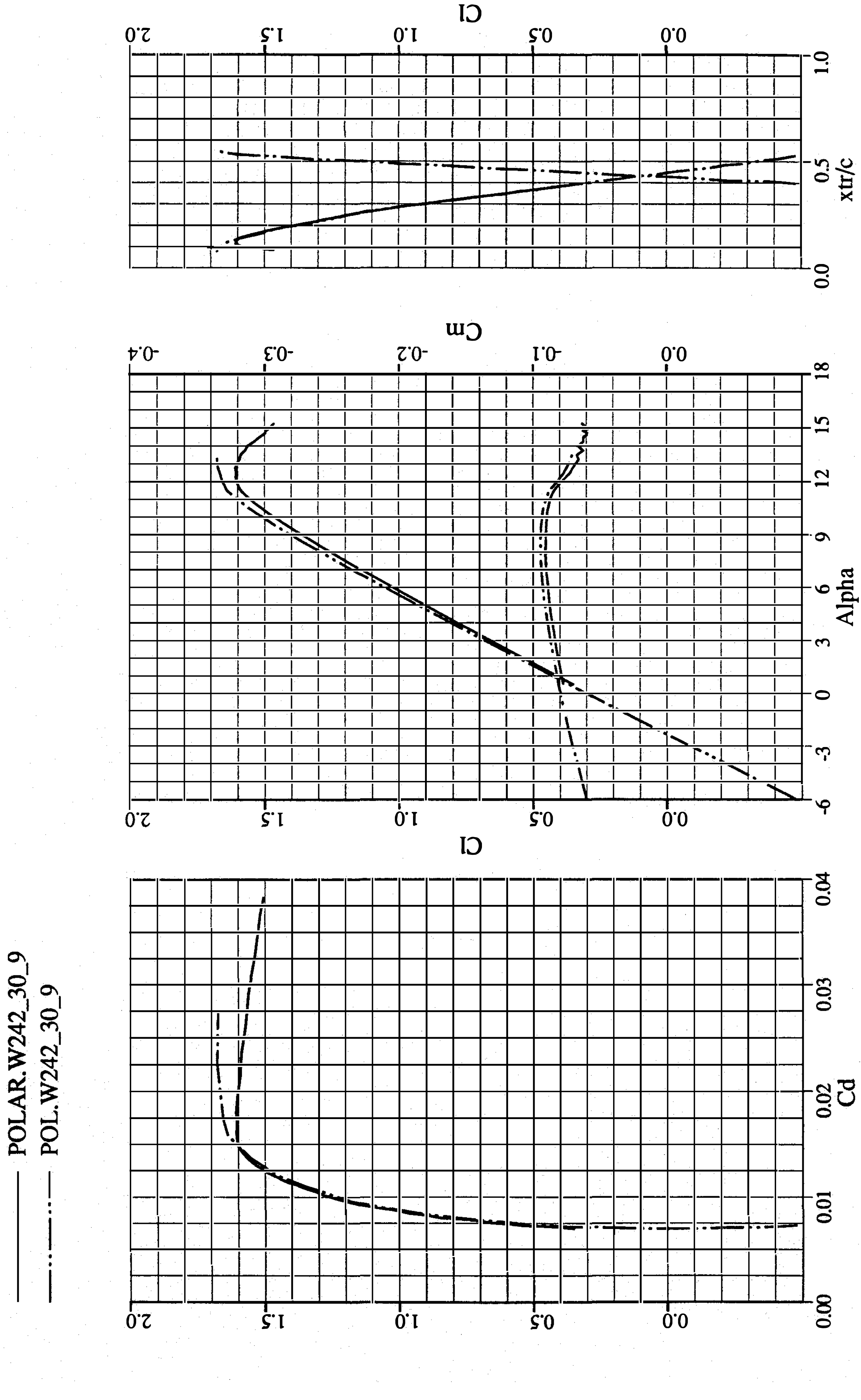


Fig.34d

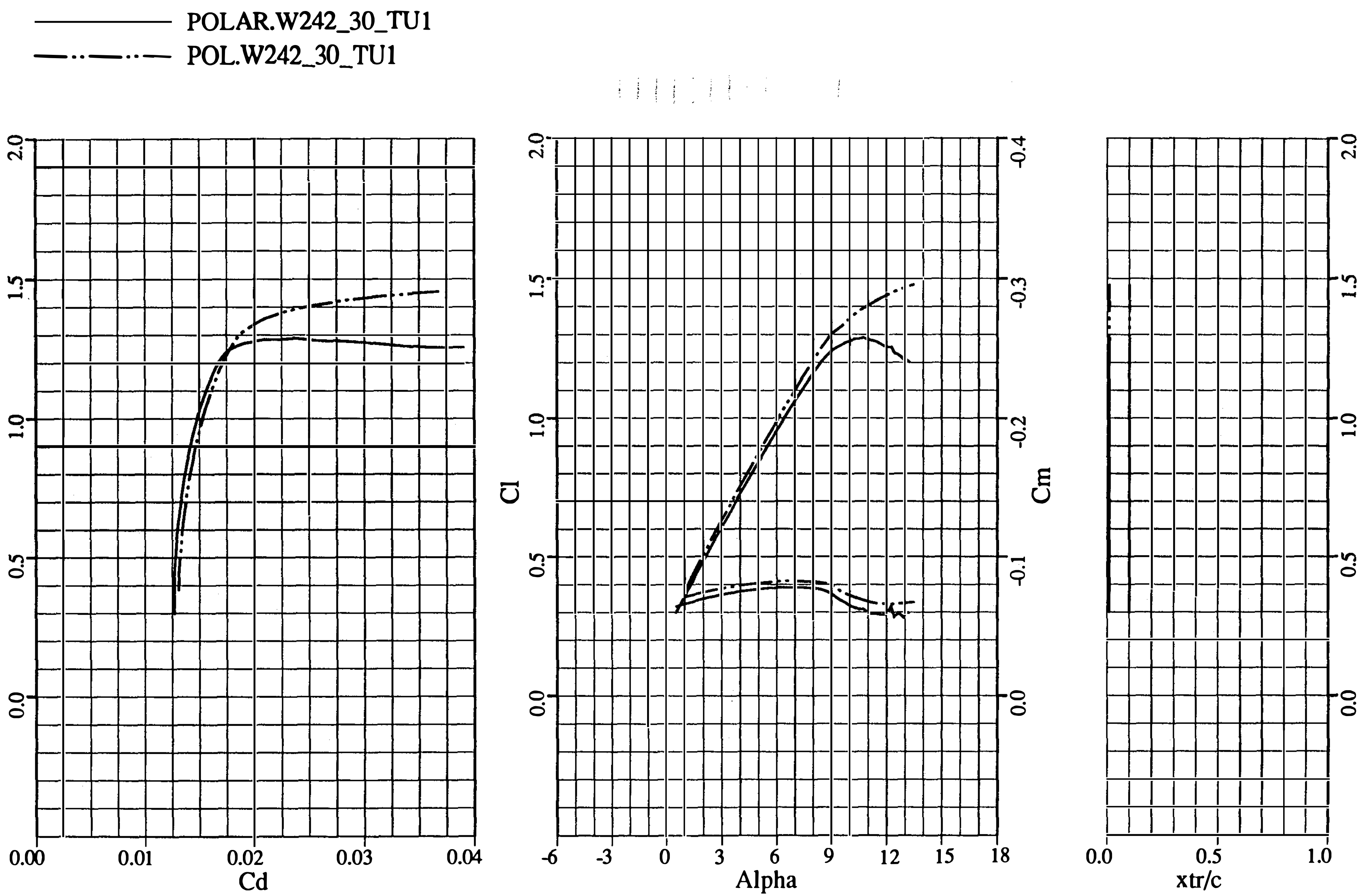

POLAR.W242_30_TU1
POL.W242_30_TU1
Cd
0.00
0.01
0.02
0.03
0.04
Alpha
-6
-3
0
3
6
9
12
15
18
xtr/c
0.0
0.5
1.0
Cl
Cm
0.0
0.5
1.0
1.5
2.0
-0.1
-0.2
-0.3
-0.4

Fig.34e

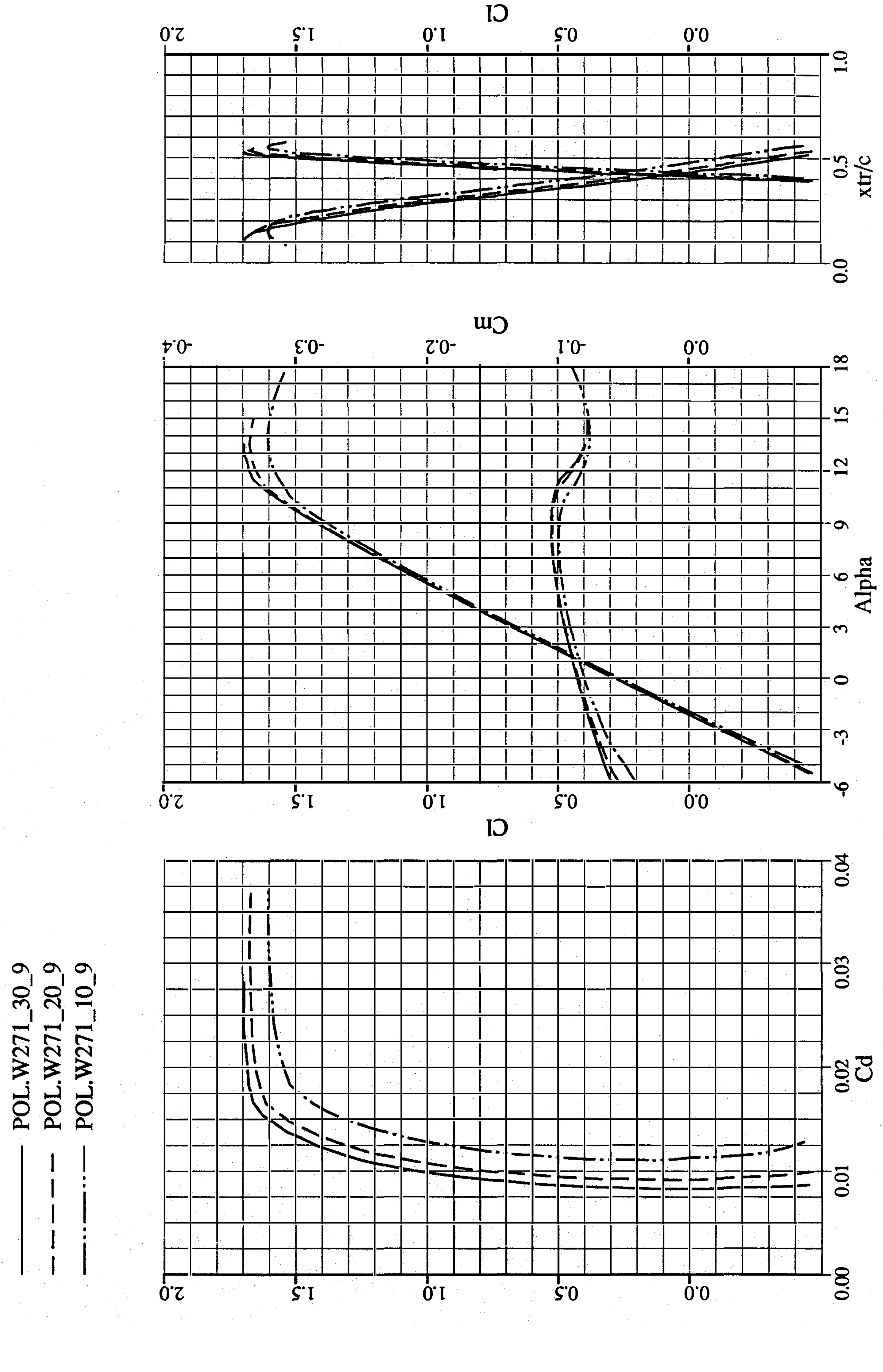


Fig.35a

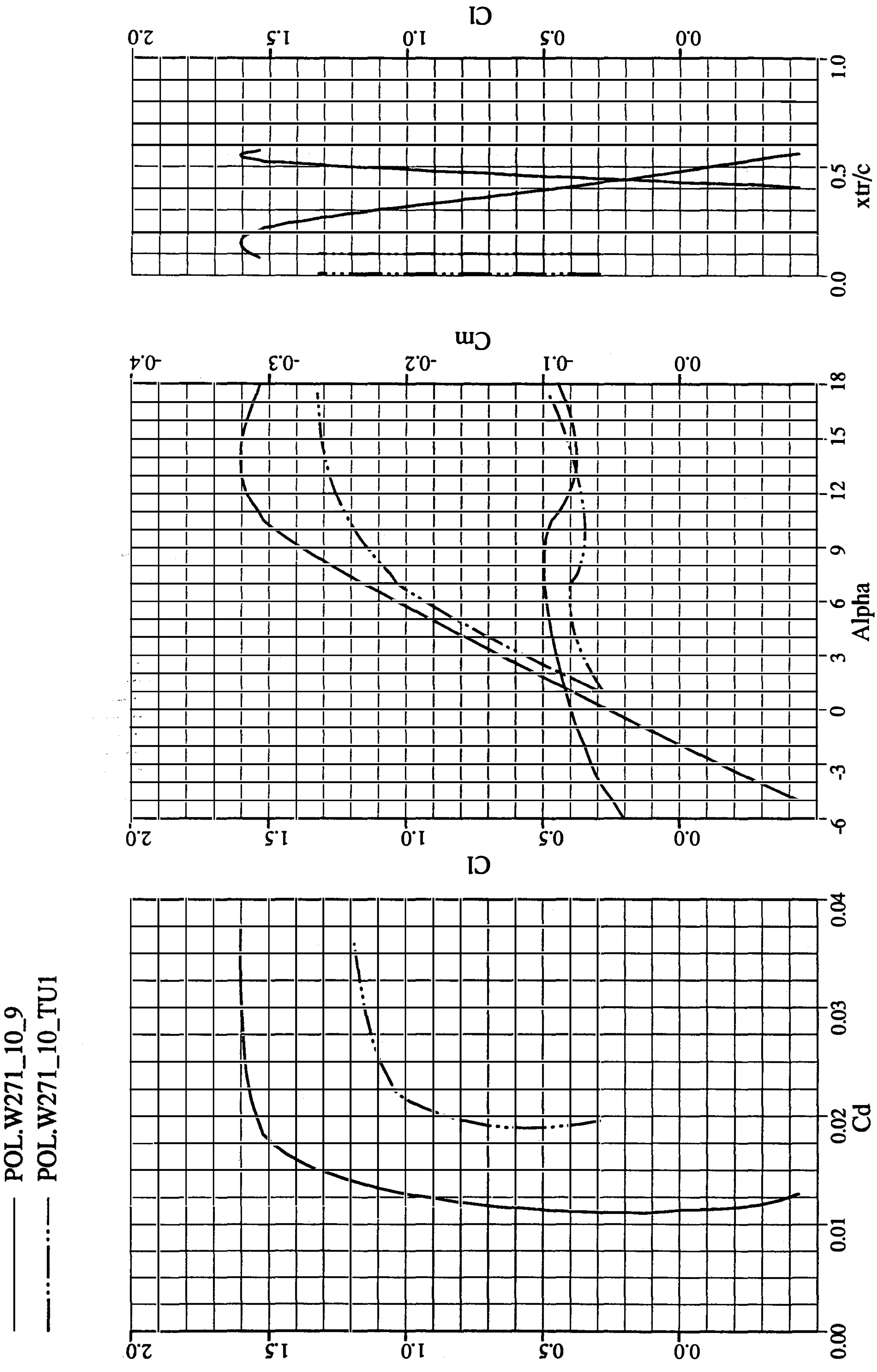

POL.W271_10_9
POL.W271_10_TU1
Cl
2.0
1.5
1.0
0.5
0.0
xtr/c
0.0
0.5
1.0
Cm
-0.4
-0.3
-0.2
-0.1
0.0
Alpha
-6
-3
0
3
6
9
12
15
18
Cd
0.00
0.01
0.02
0.03
0.04

Fig.35b

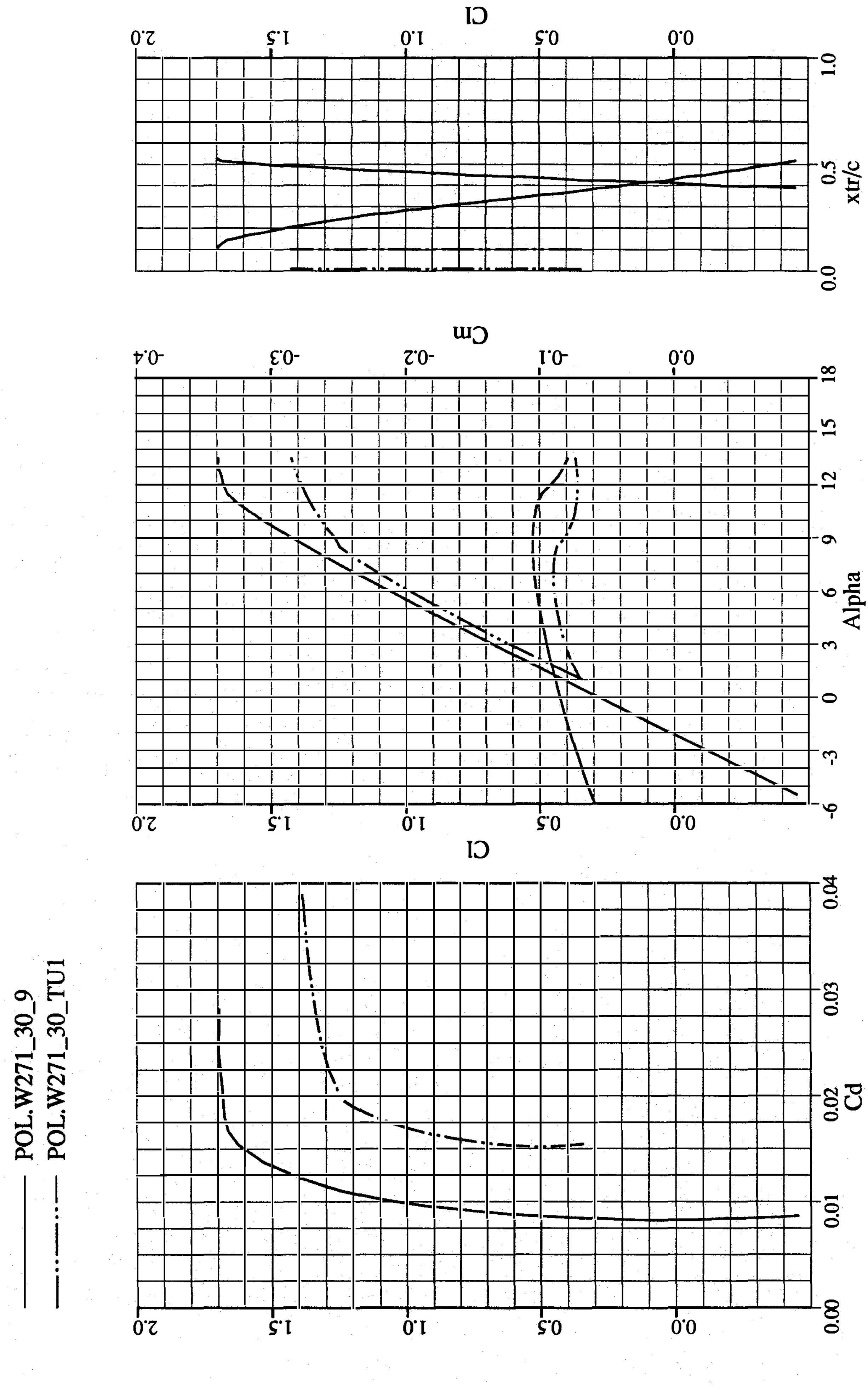

POL.W271_30_9
POL.W271_30_TU1
Cl
Cd
0.00
0.01
0.02
0.03
0.04
Cm
Alpha
-6
-3
0
3
6
9
12
15
18
xtr/c
0.0
0.5
1.0
2.0
1.5
0.5
0.0
-0.4
-0.3
-0.2
-0.1

Fig.35c

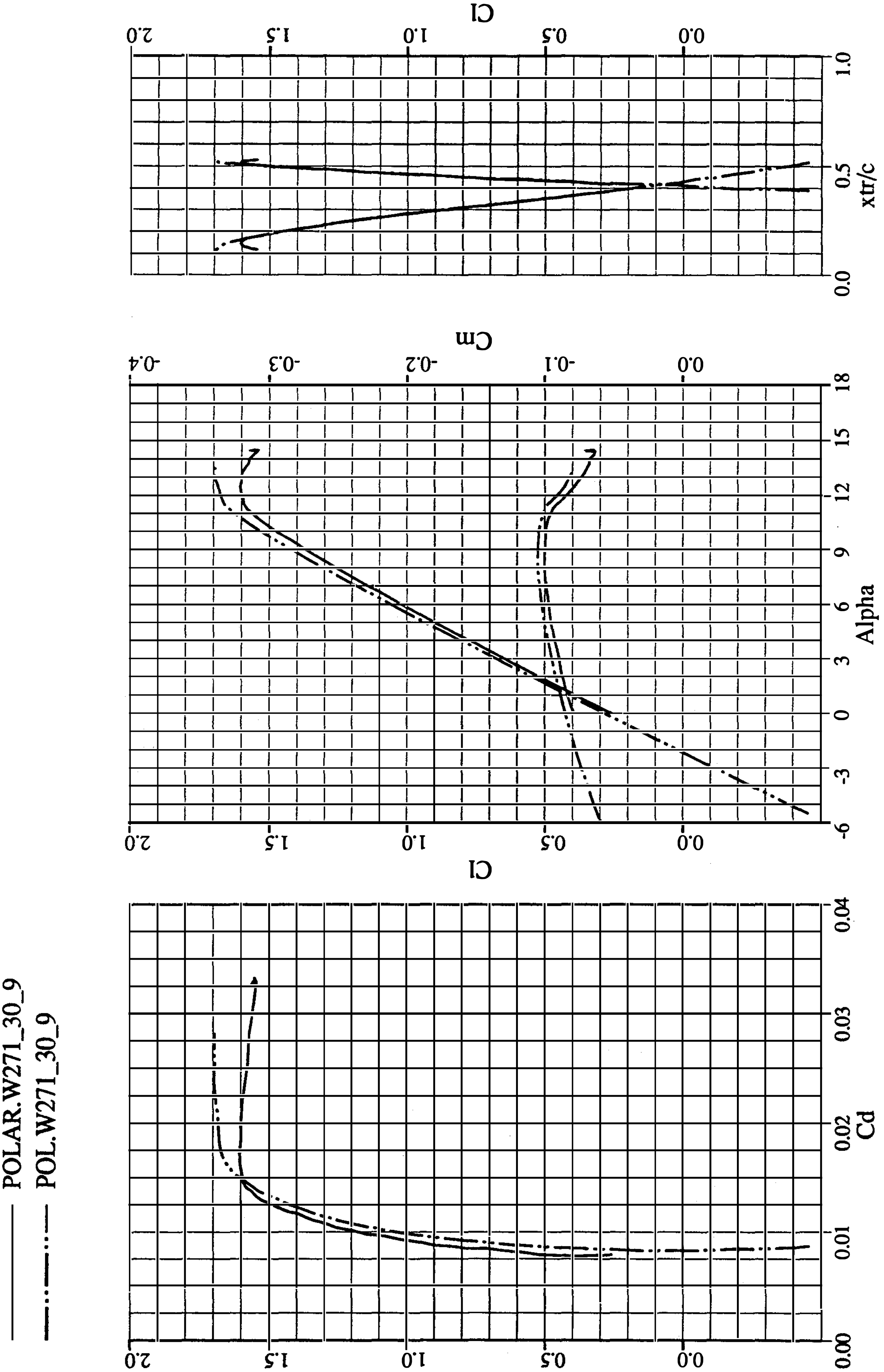

POLAR.W271_30_9
POL.W271_30_9
Cl
Cd
0.00
0.01
0.02
0.03
0.04
0.0
0.5
1.0
1.5
2.0
Cm
Alpha
-6
-3
0
3
6
9
12
15
18
-0.4
-0.3
-0.2
-0.1
0.0
xtr/c
0.0
0.5
1.0

Fig.35d

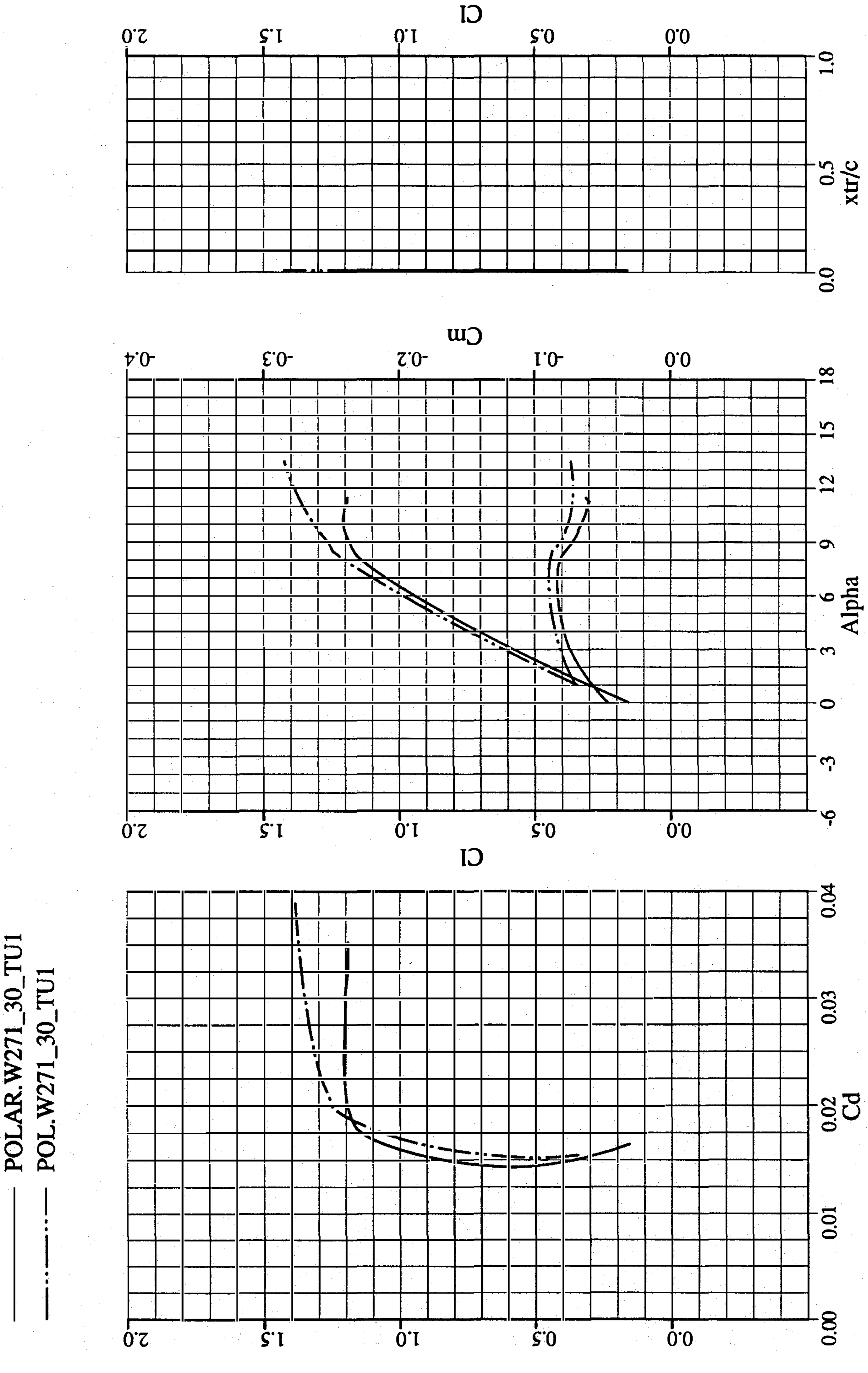


Fig.35e

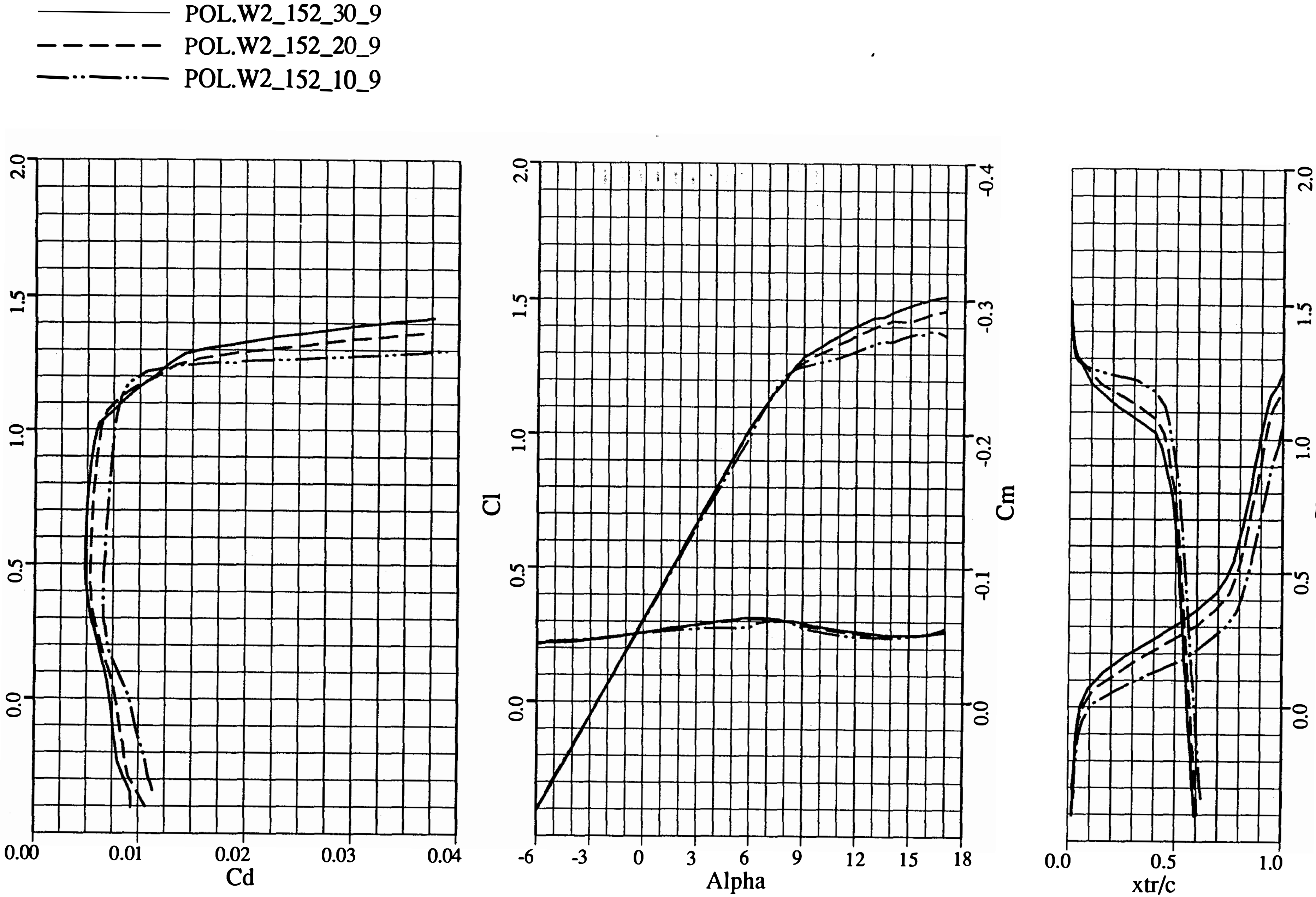

POL.W2_152_30_9
POL.W2_152_20_9
POL.W2_152_10_9
Cd
Cl
Alpha
Cm
xtr/c
0.00
0.01
0.02
0.03
0.04
-6
-3
0
3
6
9
12
15
18
0.0
0.5
1.0
1.5
2.0
-0.4
-0.3
-0.2
-0.1

Fig.36a

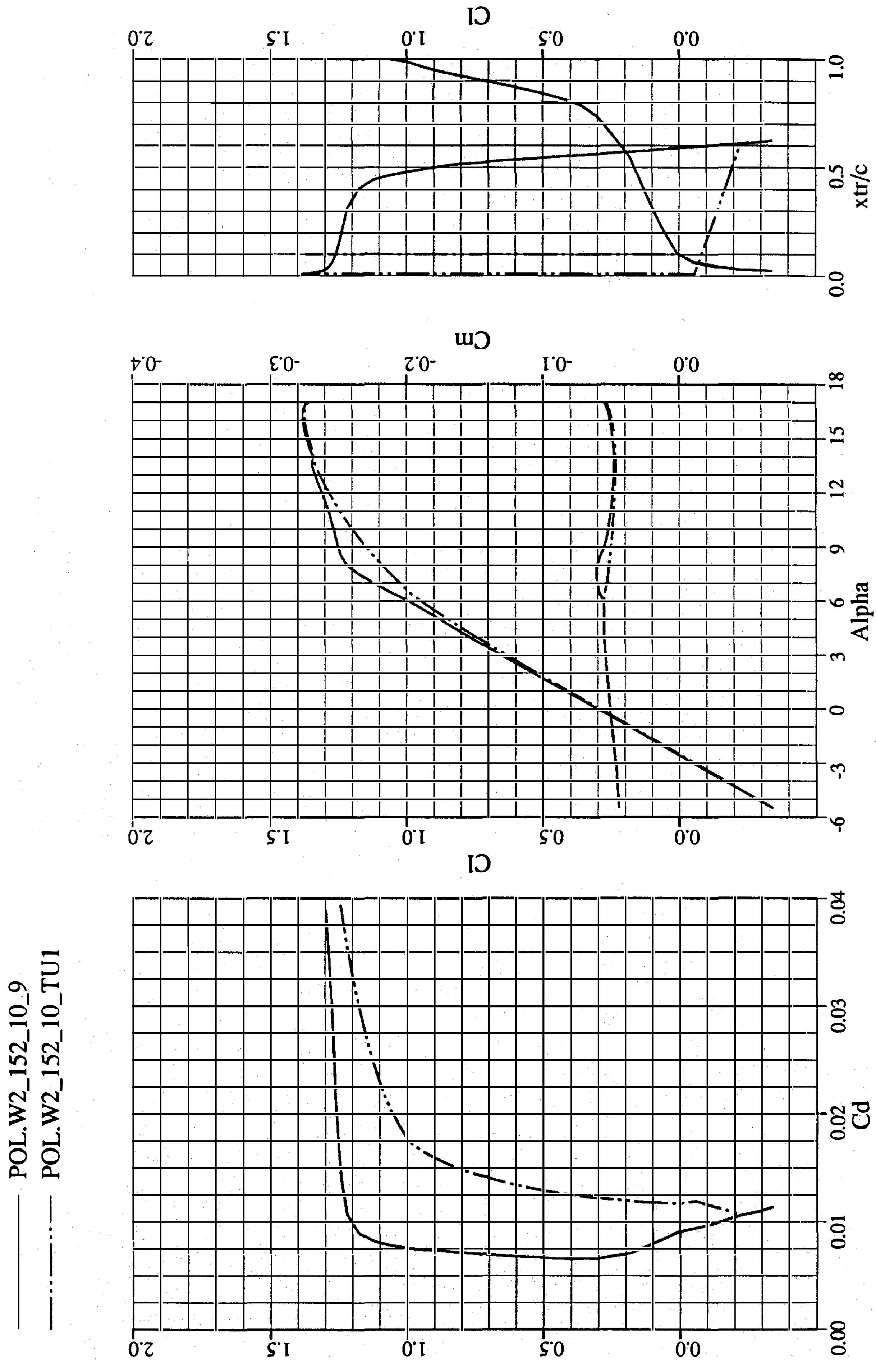

POL.W2_152_10_9
POL.W2_152_10_TU1
Cl
Cd
0.00
0.01
0.02
0.03
0.04
0.0
0.5
1.0
1.5
2.0
Cm
Alpha
-6
-3
0
3
6
9
12
15
18
-0.4
-0.3
-0.2
-0.1
xtr/c

Fig.36b

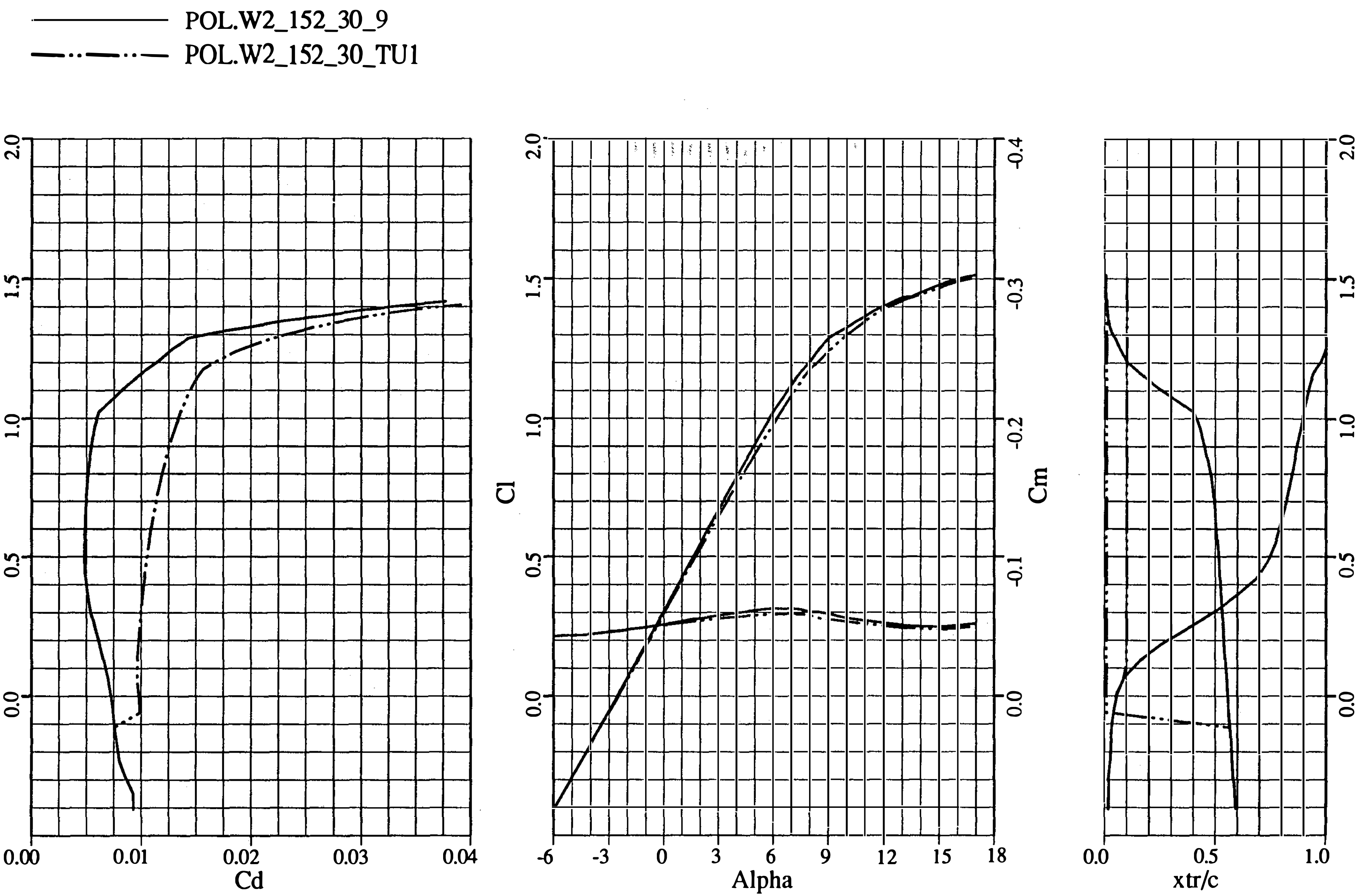

POL.W2_152_30_9
POL.W2_152_30_TU1
Cl
Cd
Alpha
Cm
xtr/c
0.00
0.01
0.02
0.03
0.04
-6
-3
0
3
6
9
12
15
18
0.0
0.5
1.0
1.5
2.0
-0.1
-0.2
-0.3
-0.4

Fig.36c

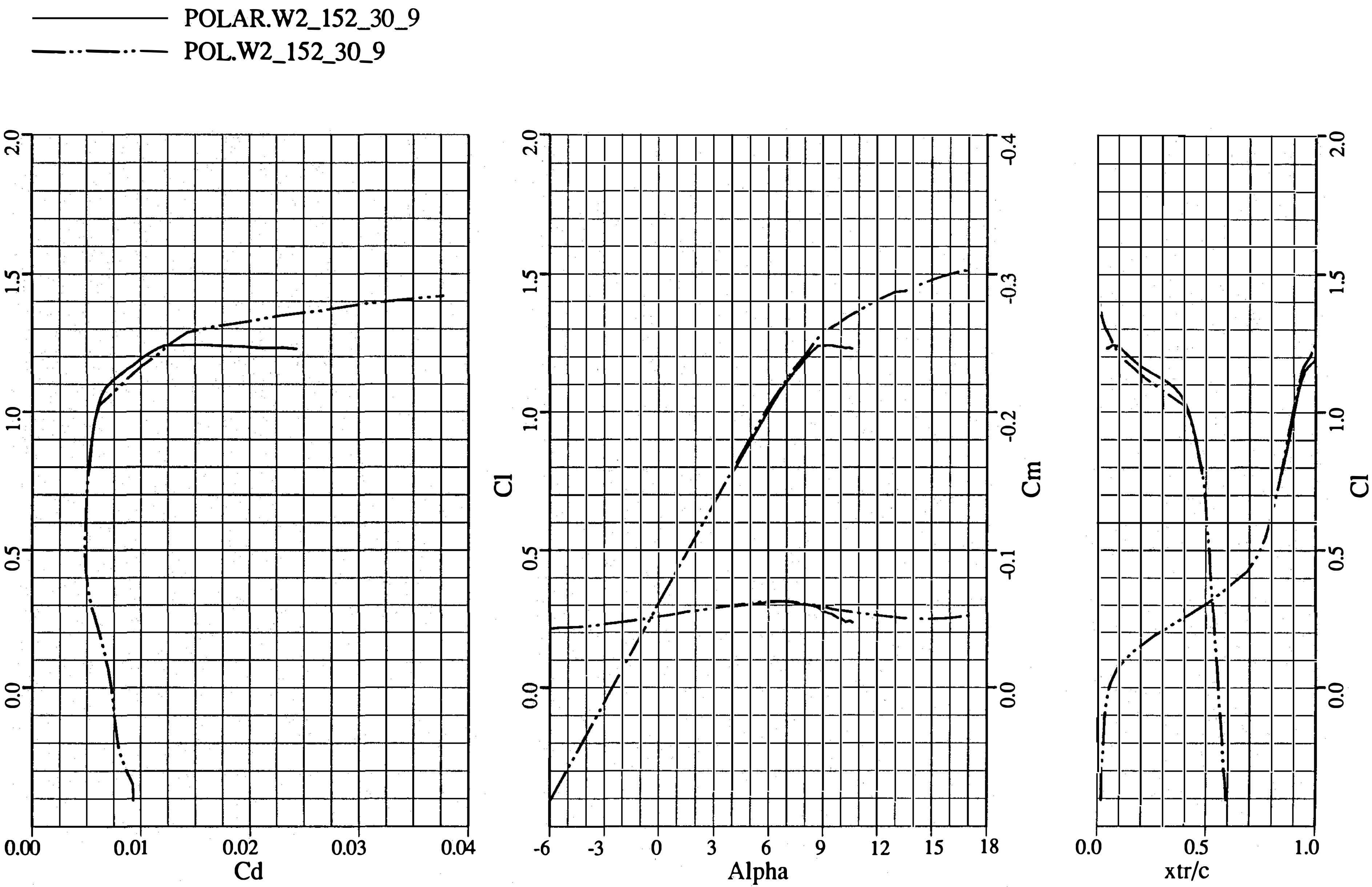

POLAR.W2_152_30_9
POL.W2_152_30_9
Cd
Alpha
xtr/c
Cl
Cm
0.00
0.01
0.02
0.03
0.04
-6
-3
0
3
6
9
12
15
18
0.0
0.5
1.0
1.5
2.0
-0.4
-0.3
-0.2
-0.1

Fig.36d

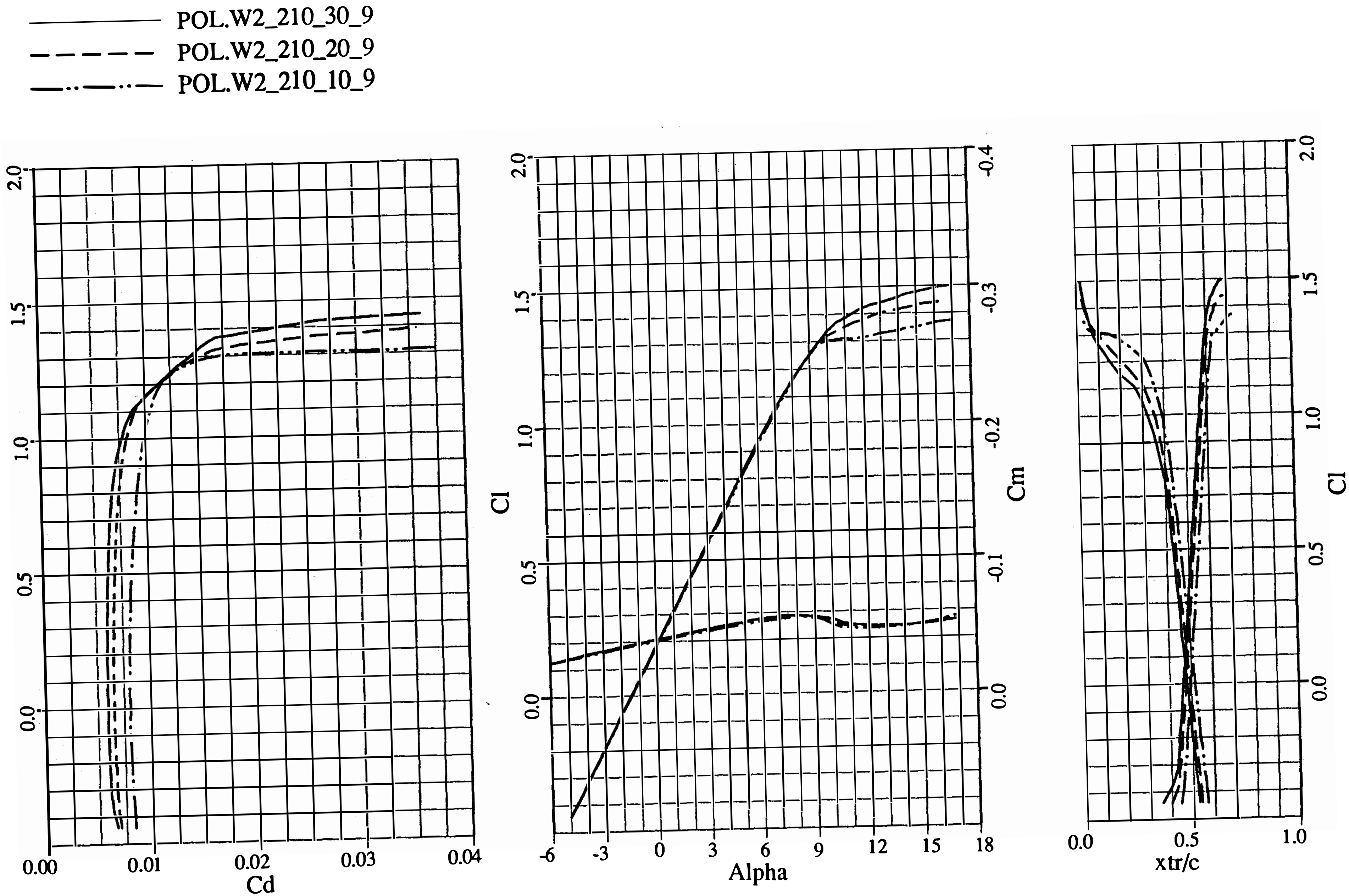

POL.W2_210_30_9
POL.W2_210_20_9
POL.W2_210_10_9
Cd
Cl
Alpha
Cm
xtr/c
Cl

Fig.37a

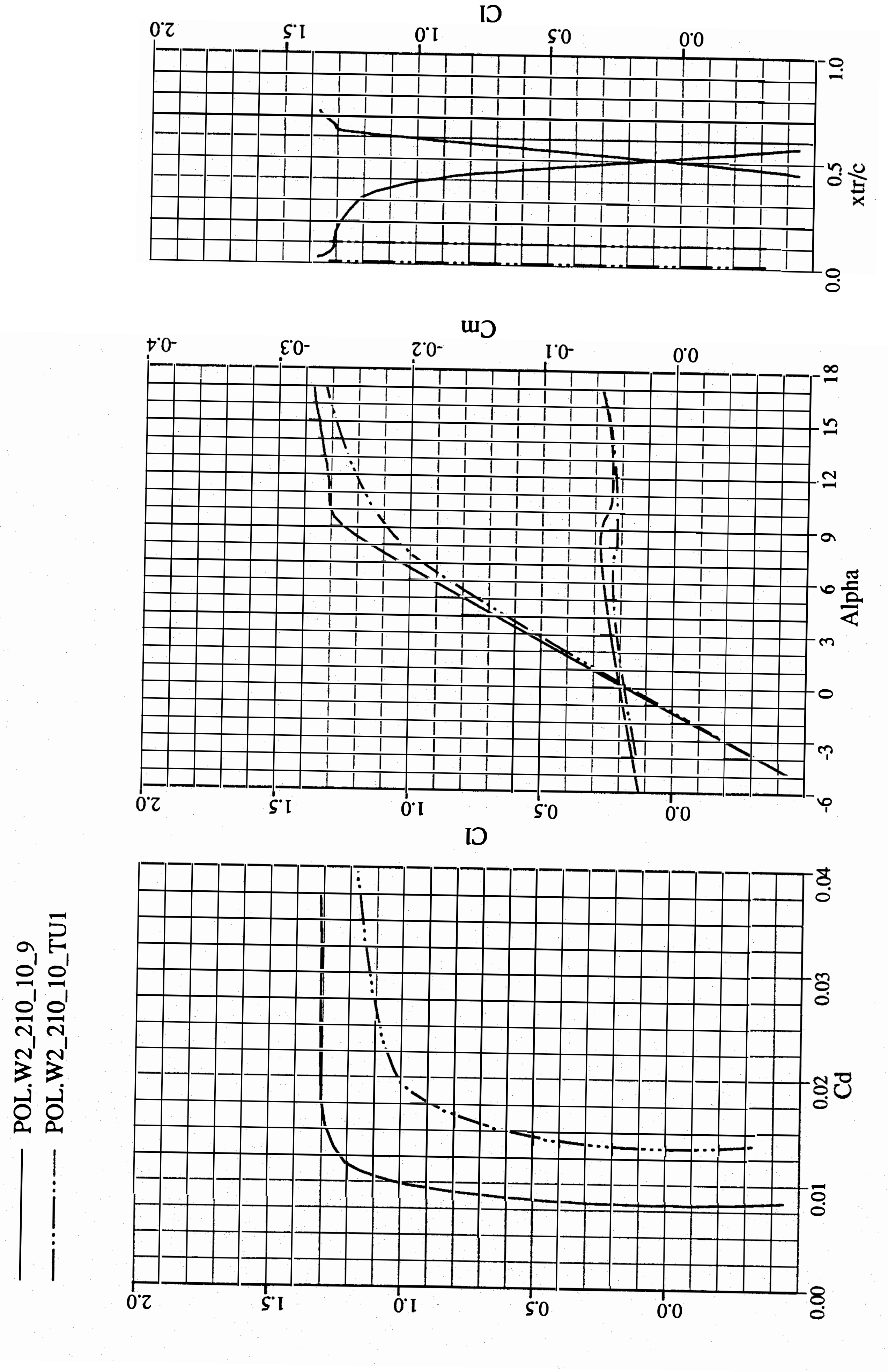

POL.W2_210_10_9
POL.W2_210_10_TU1
Cl
Cd
0.00
0.01
0.02
0.03
0.04
0.0
0.5
1.0
1.5
2.0
Cm
Alpha
-6
-3
0
3
6
9
12
15
18
-0.4
-0.3
-0.2
-0.1
0.0
xtr/c
0.0
0.5
1.0

Fig.37b

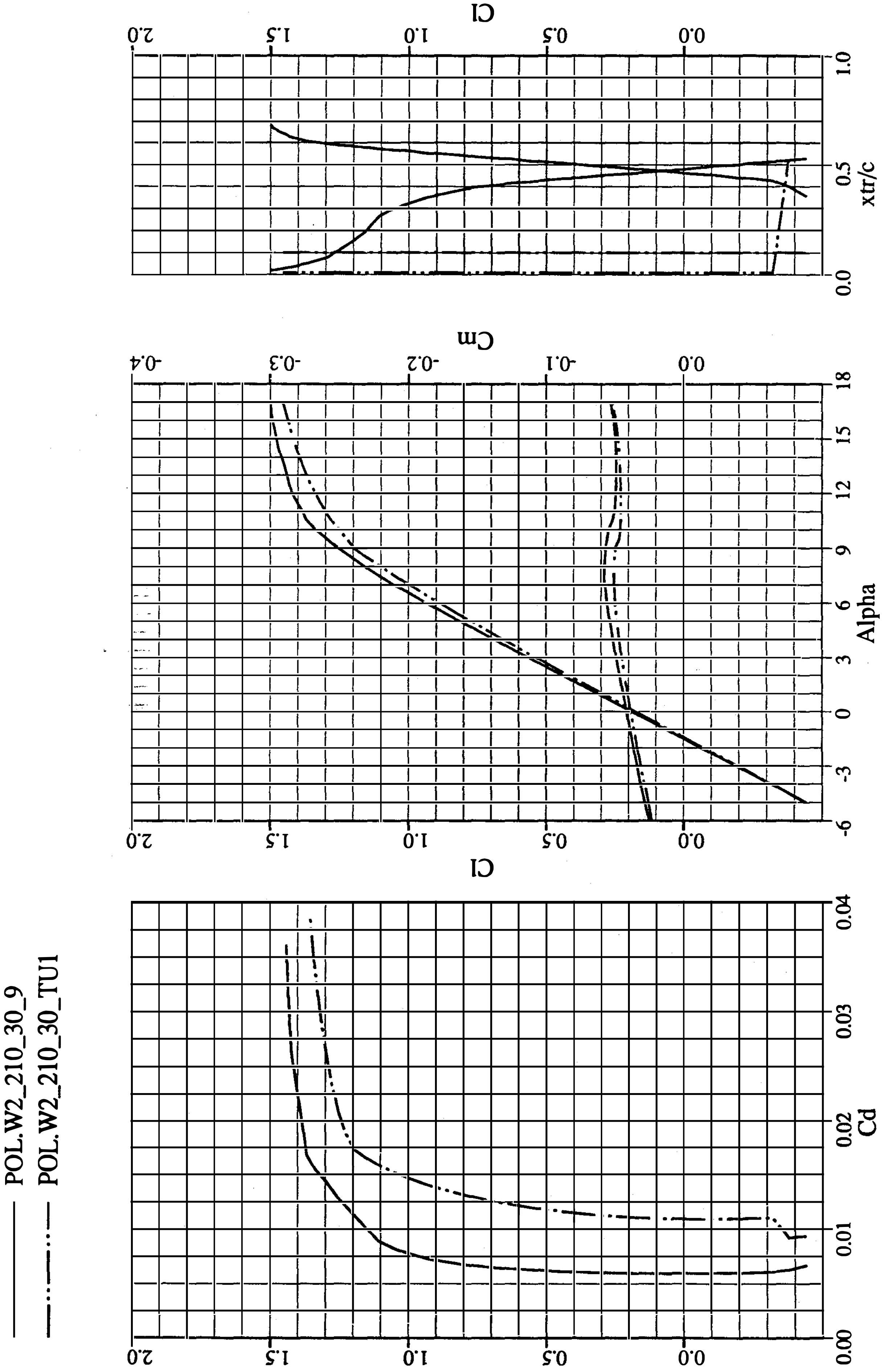


Fig.37c

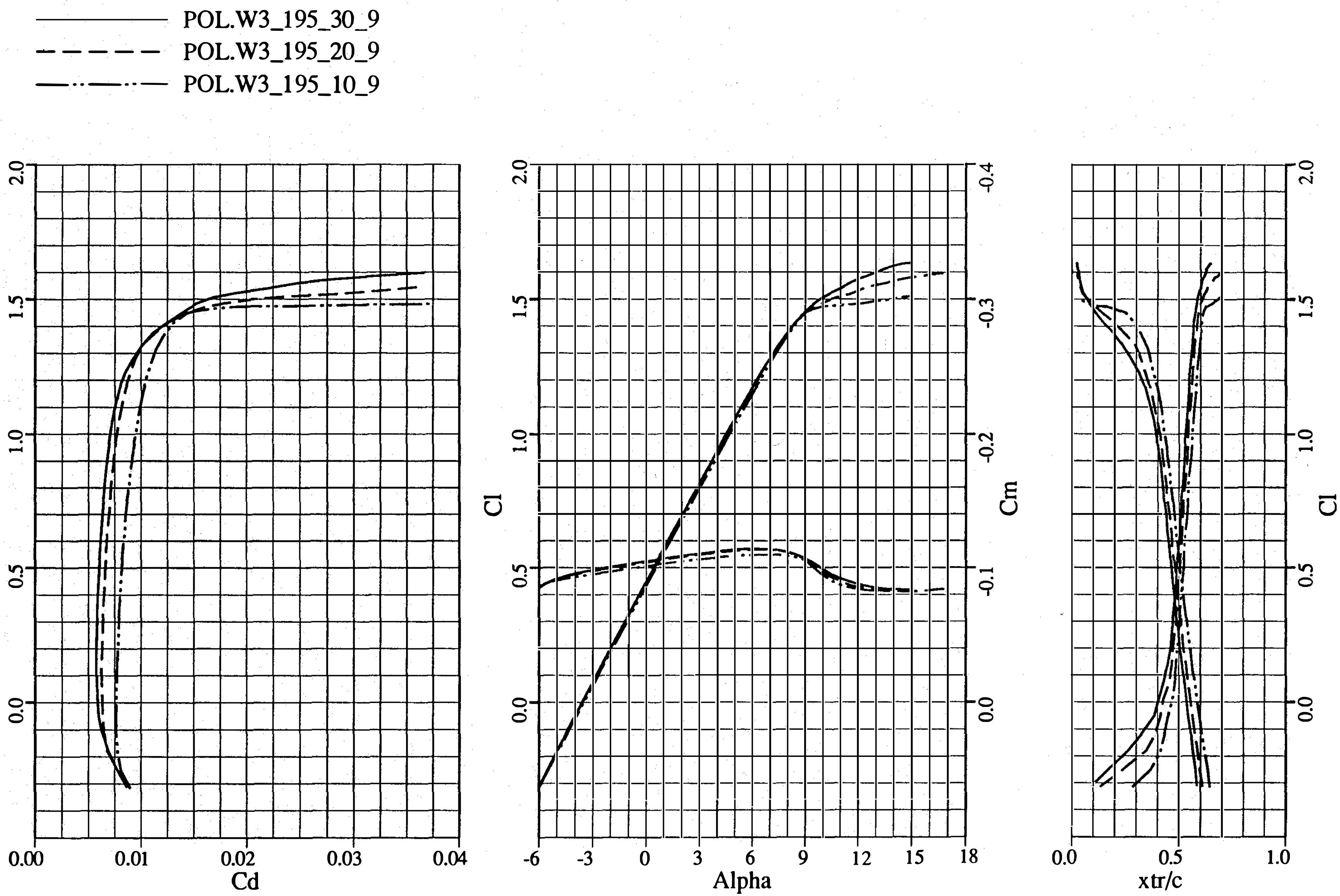

POL.W3_195_30_9
POL.W3_195_20_9
POL.W3_195_10_9
Cl
Cd
0.00
0.01
0.02
0.03
0.04
0.0
0.5
1.0
1.5
2.0
Alpha
-6
-3
0
3
6
9
12
15
18
Cm
-0.4
-0.3
-0.2
-0.1
0.0
xtr/c
0.0
0.5
1.0

Fig.38a

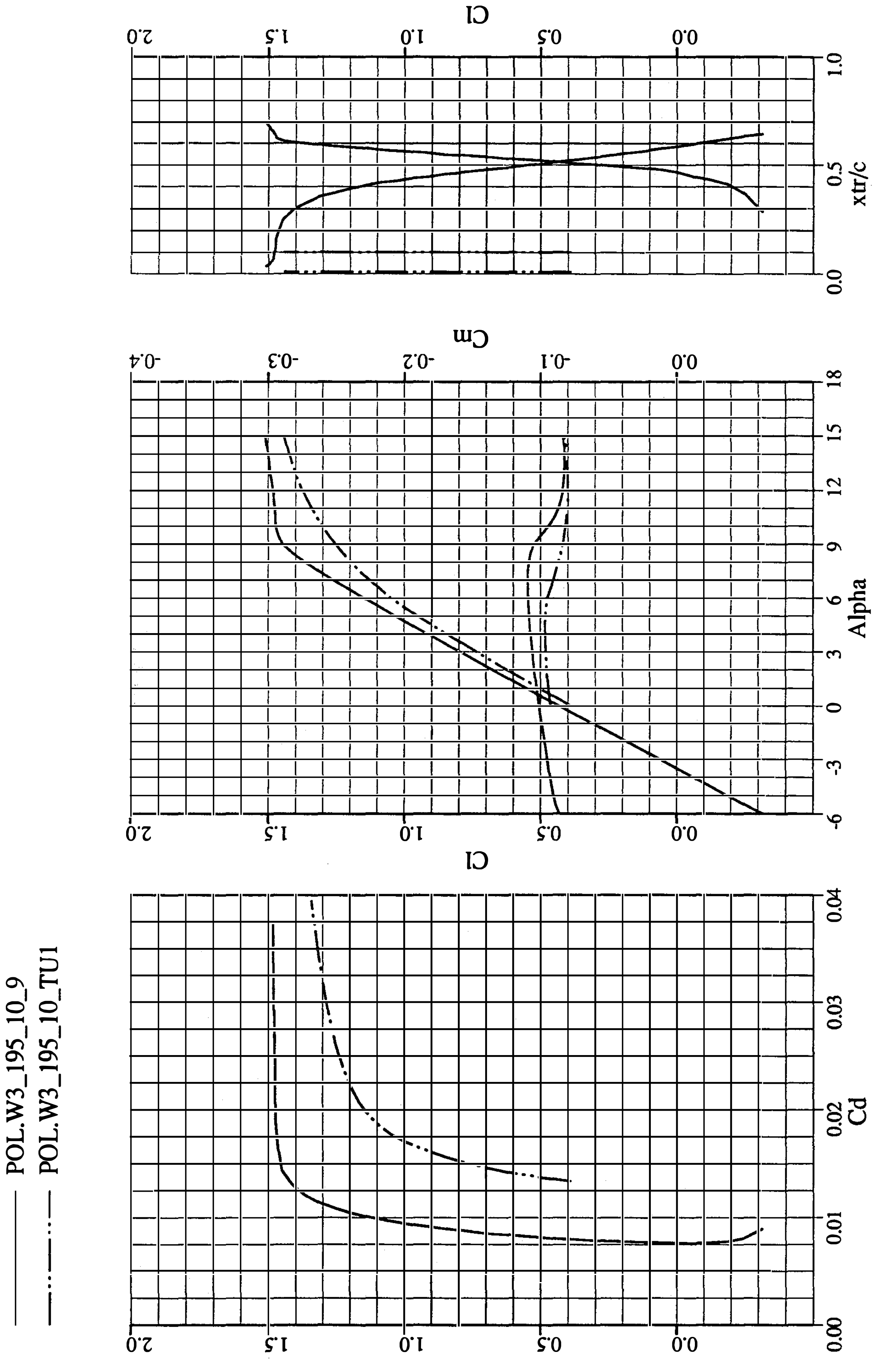


Fig.38b

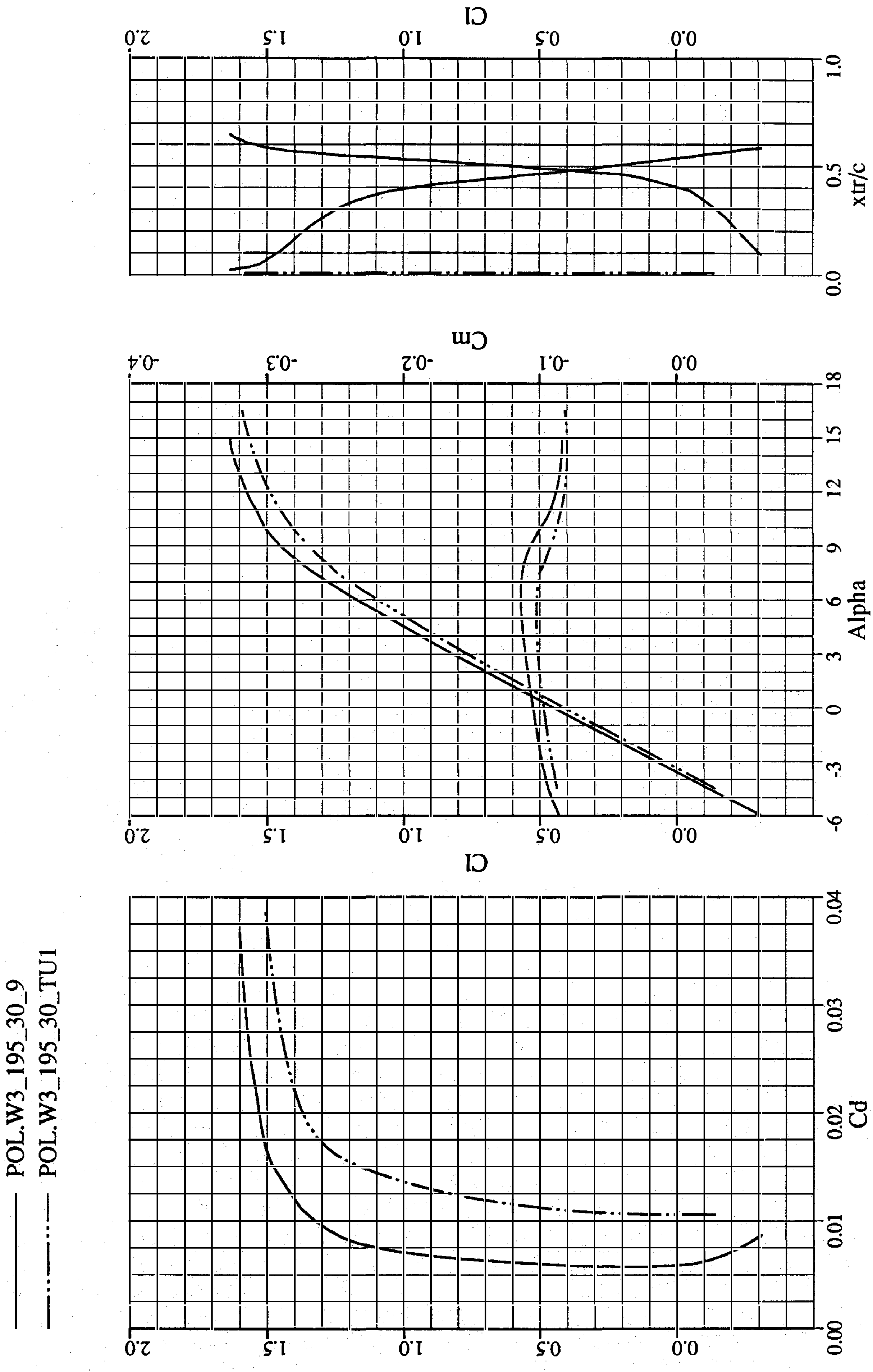


Fig.38c

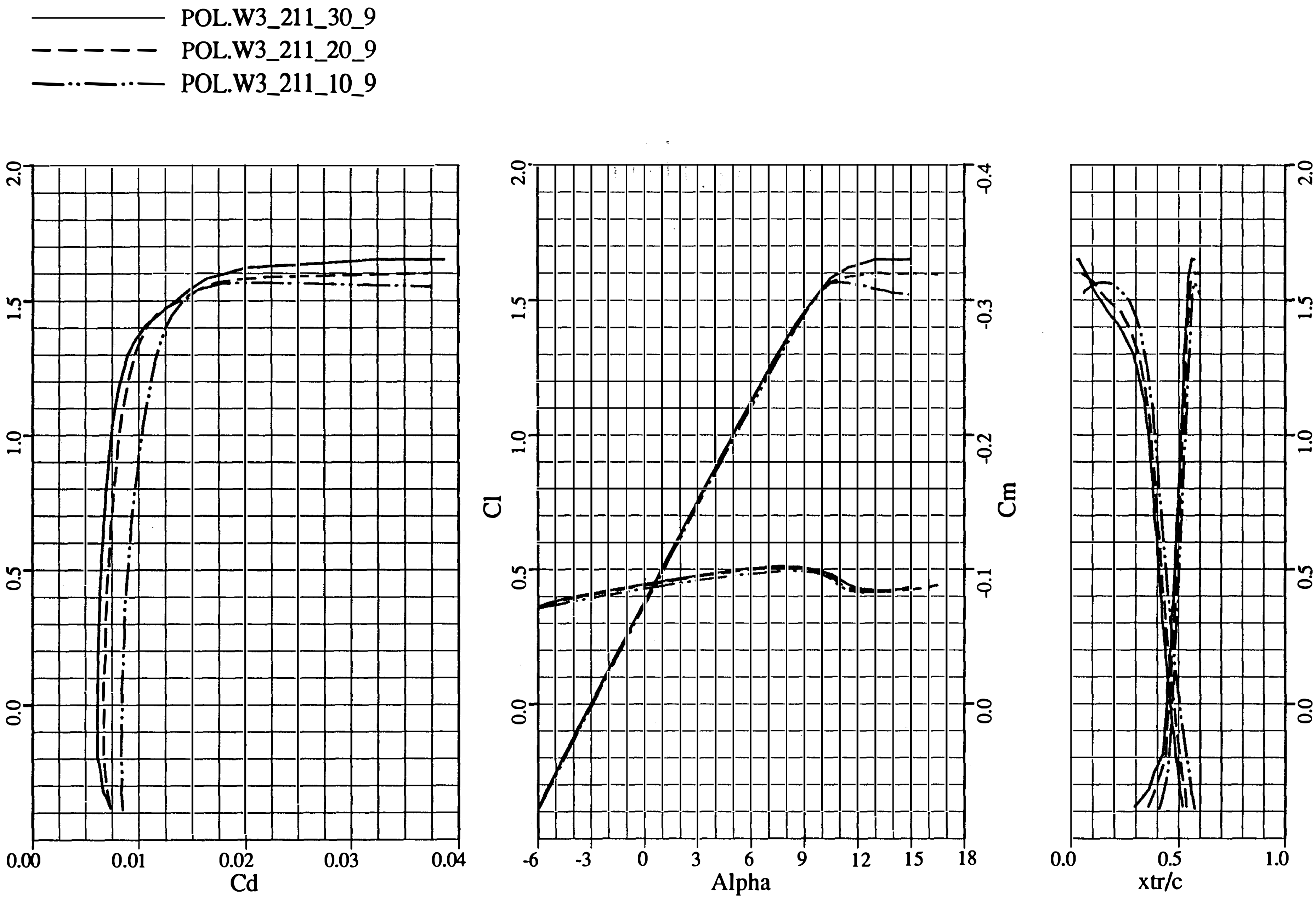

POL.W3_211_30_9
POL.W3_211_20_9
POL.W3_211_10_9
Cl
Cd
0.00
0.01
0.02
0.03
0.04
-0.5
0.0
0.5
1.0
1.5
2.0
Alpha
-6
-3
0
3
6
9
12
15
18
Cm
-0.4
-0.3
-0.2
-0.1
0.0
xtr/c
0.0
0.5
1.0

Fig.39a

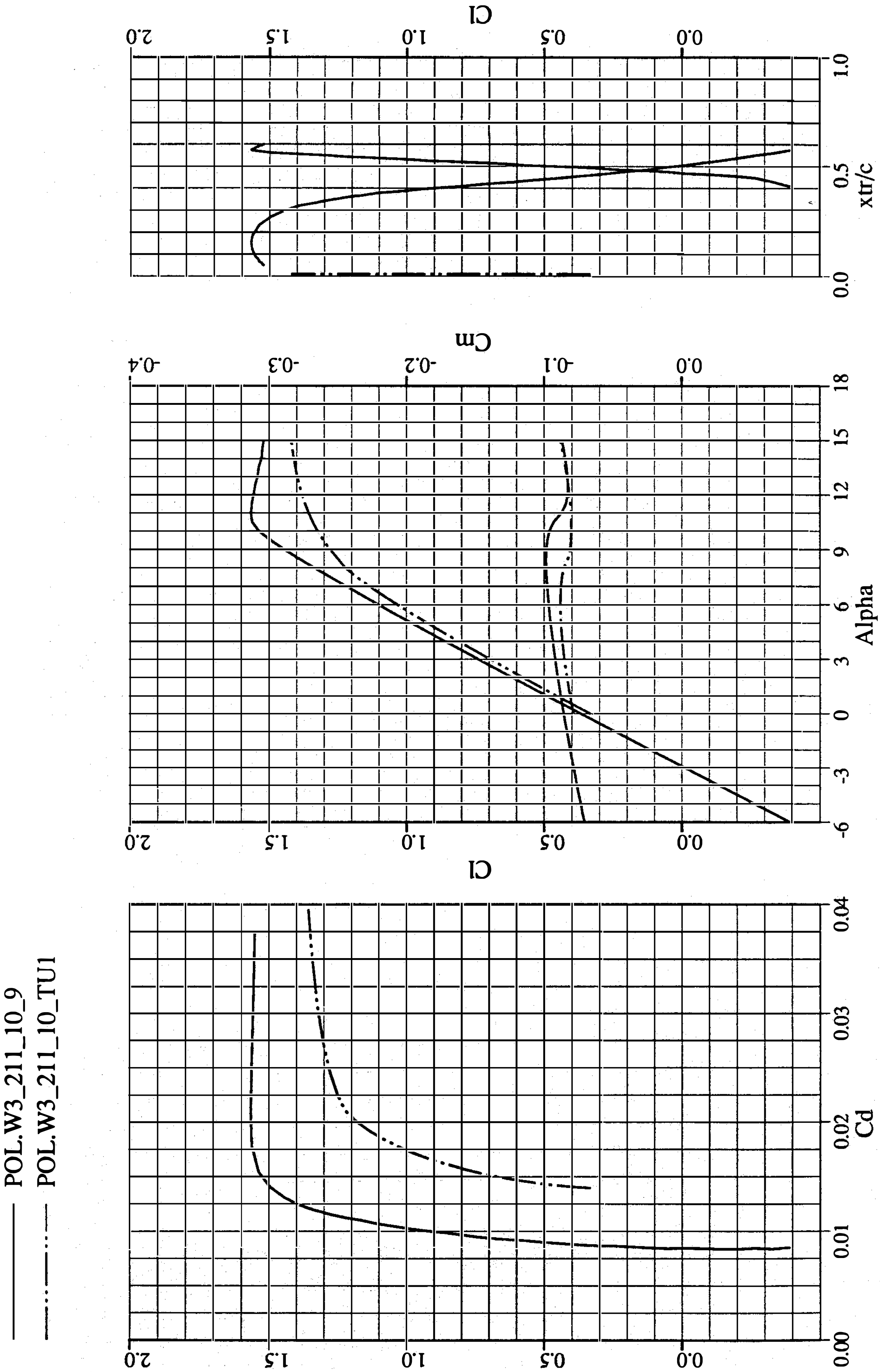


Fig.39b

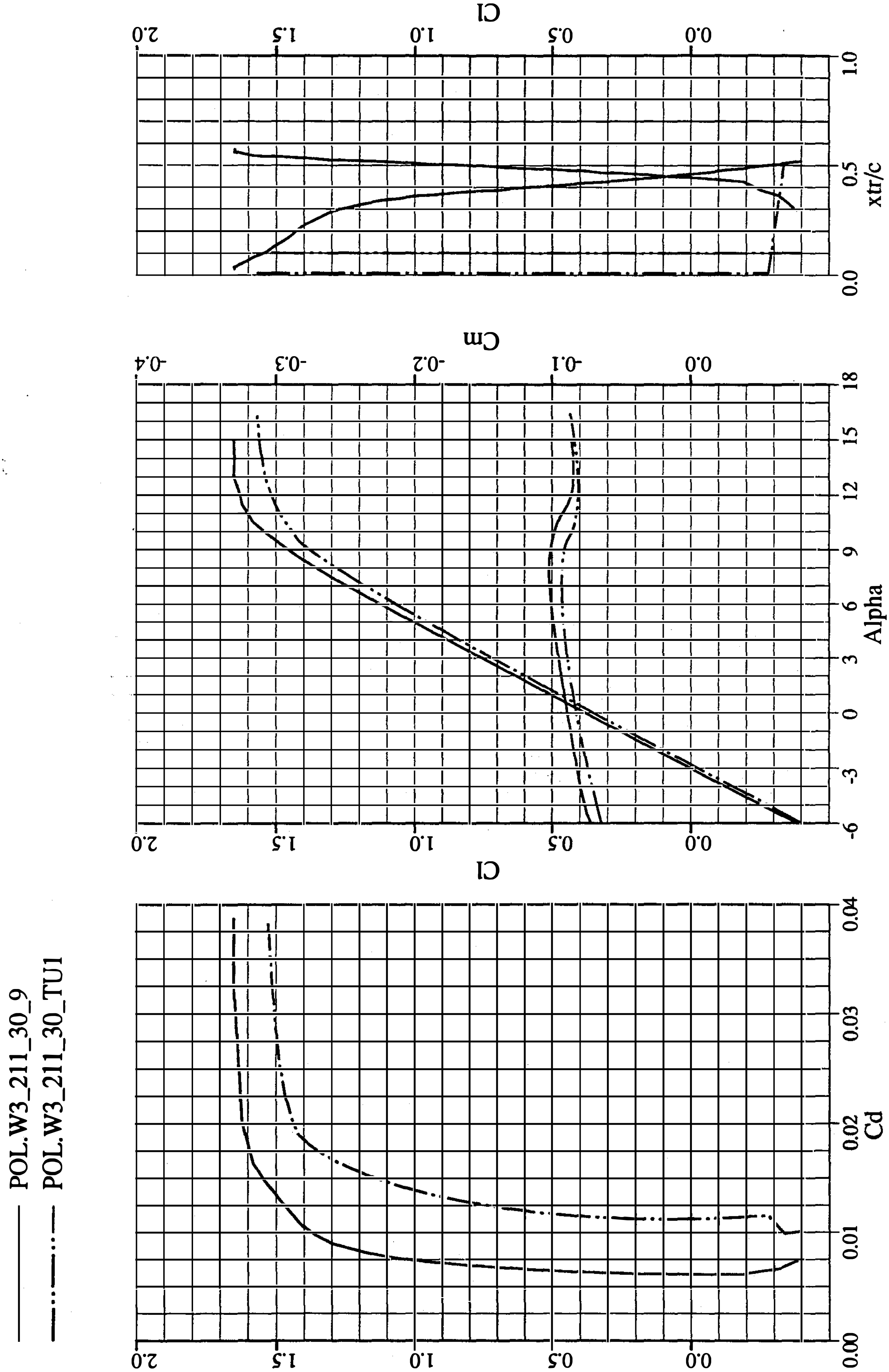


Fig.39c

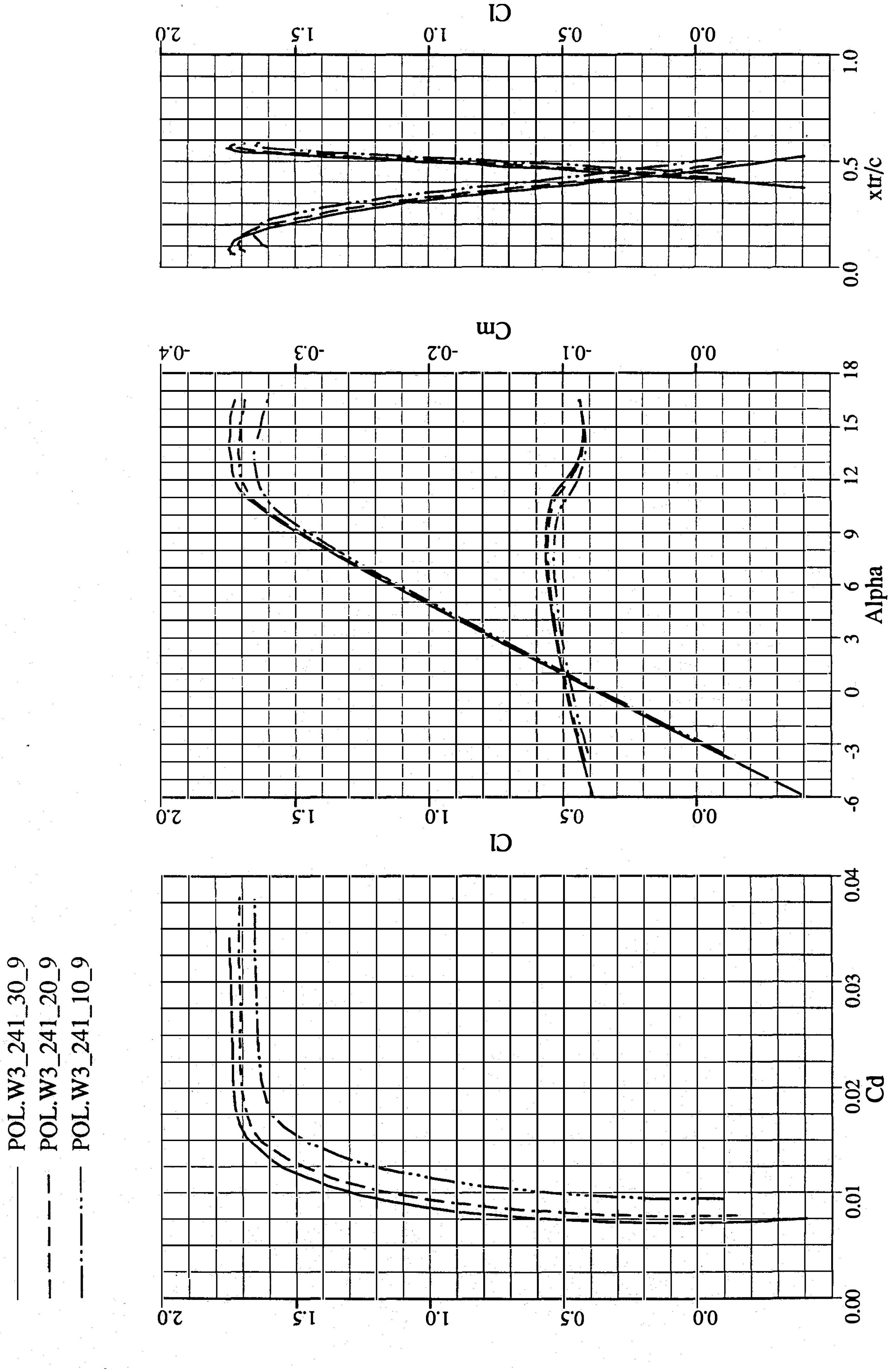


Fig.40a

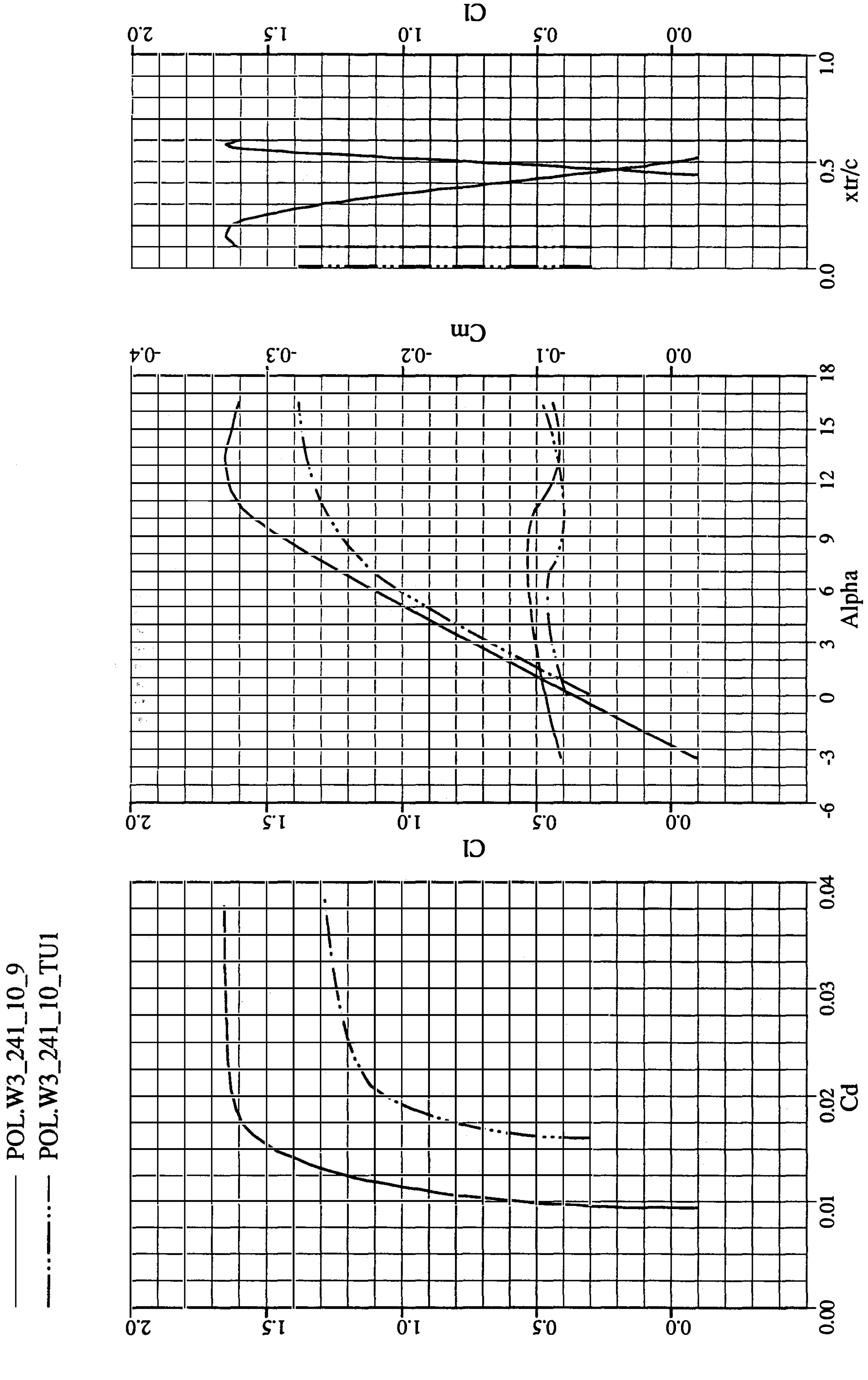


Fig.40b

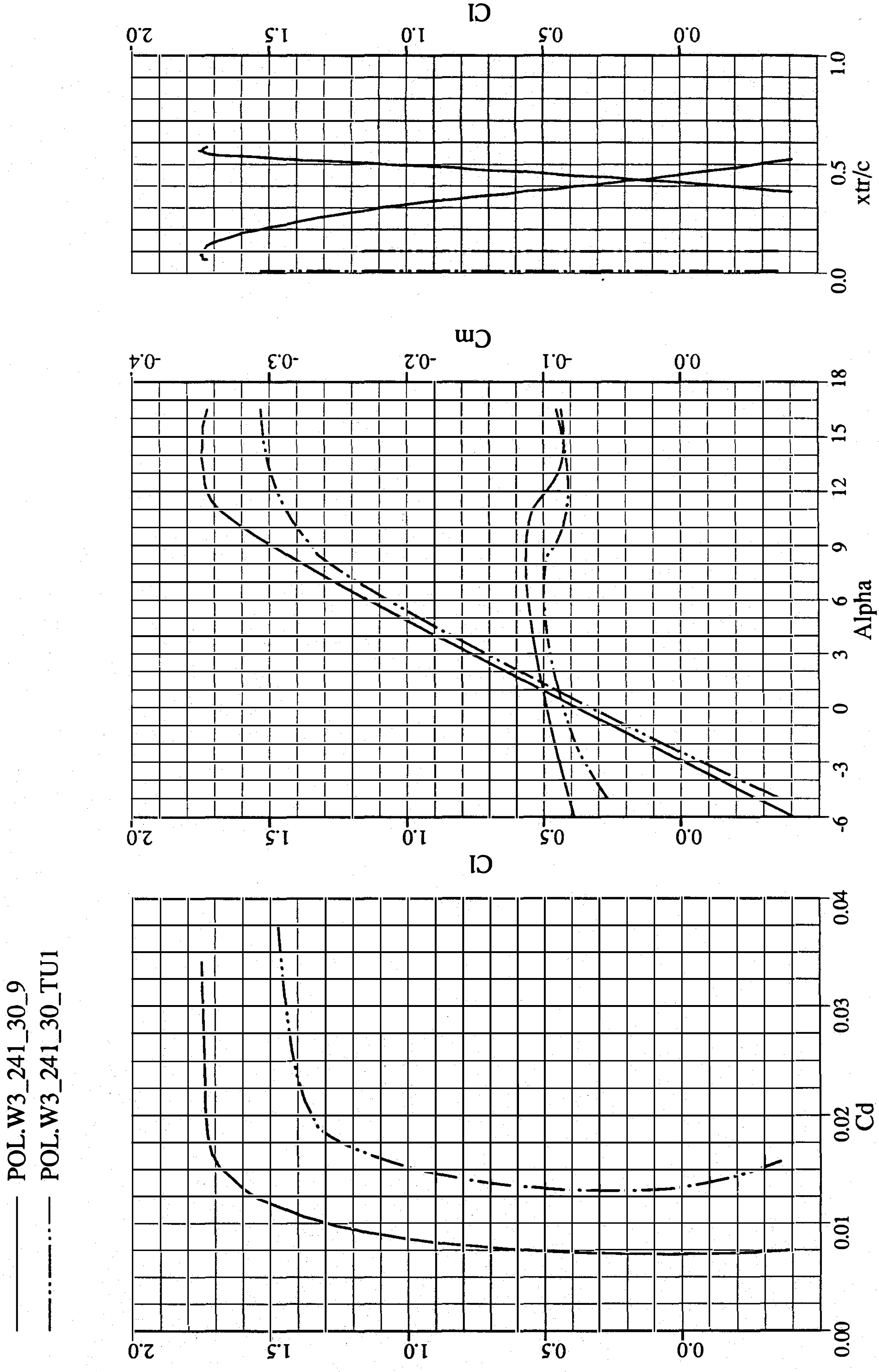

POL.W3_241_30_9
POL.W3_241_30_TU1
Cl
Cd
0.00
0.01
0.02
0.03
0.04
0.0
0.5
1.0
1.5
2.0
Alpha
-6
-3
0
3
6
9
12
15
18
Cm
-0.4
-0.3
-0.2
-0.1
0.0
xtr/c

Fig.40c

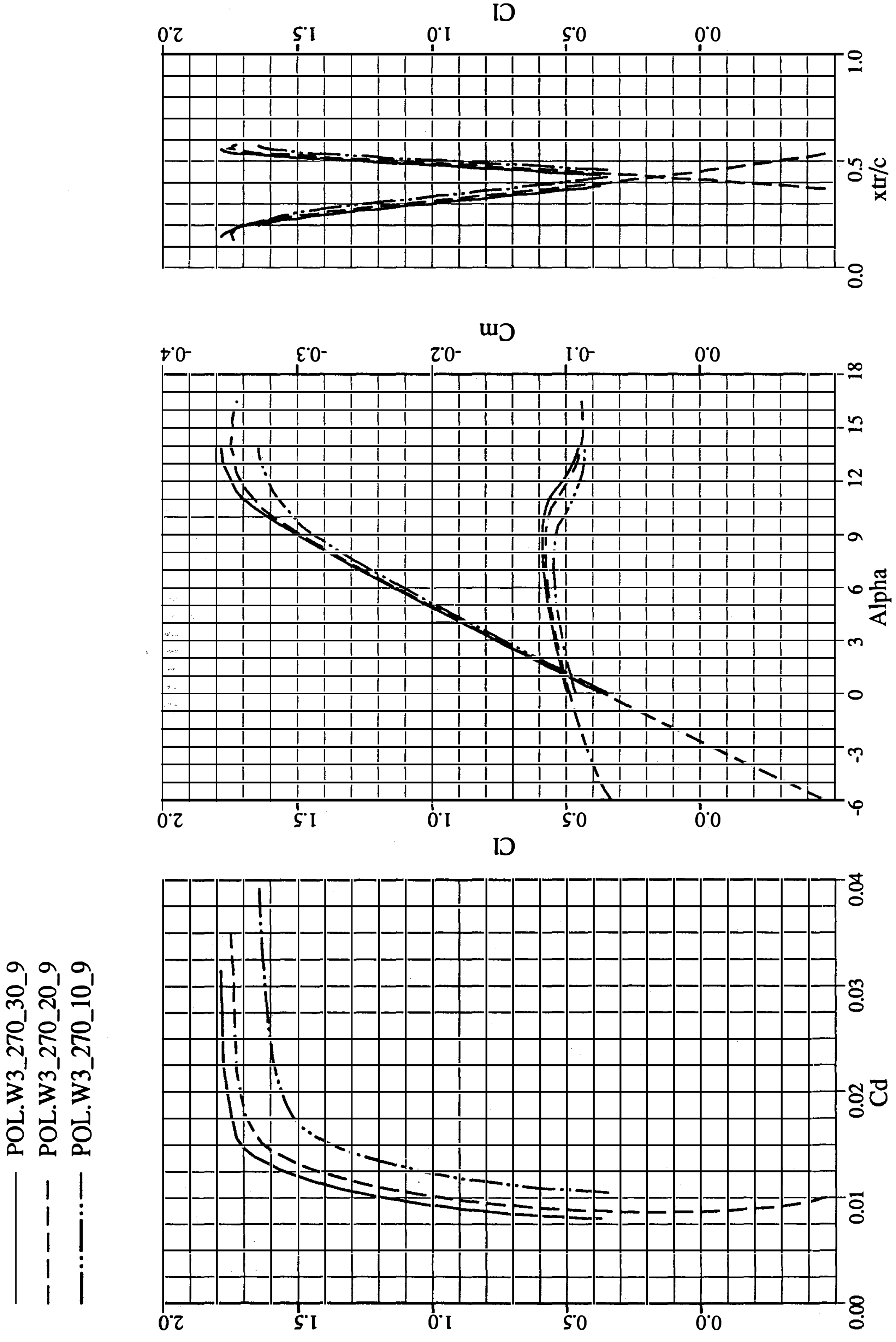

Cl
xtr/c
Cm
Alpha
Cl
Cd
POL.W3_270_30_9
POL.W3_270_20_9
POL.W3_270_10_9

Fig.41a

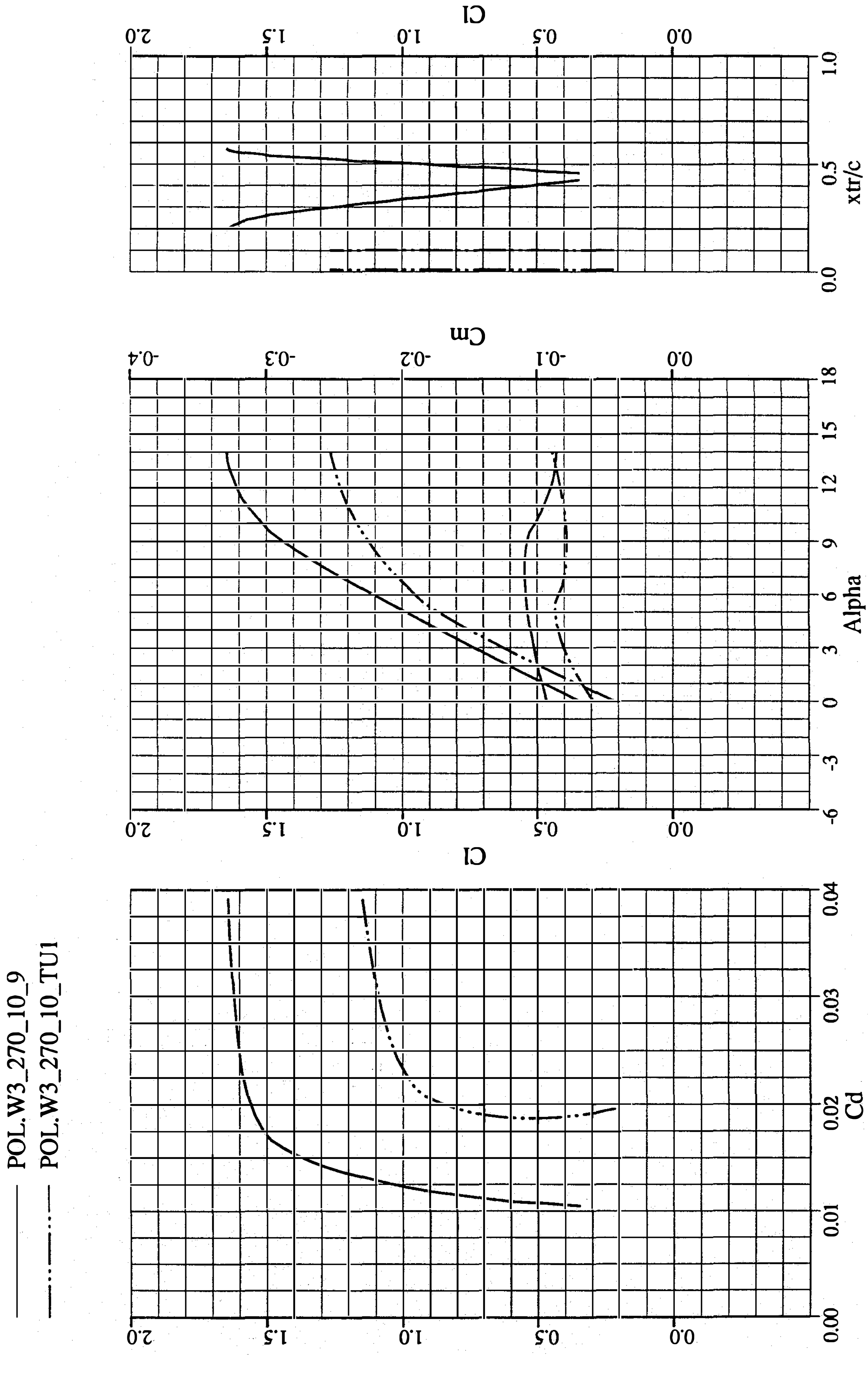


Fig.41b

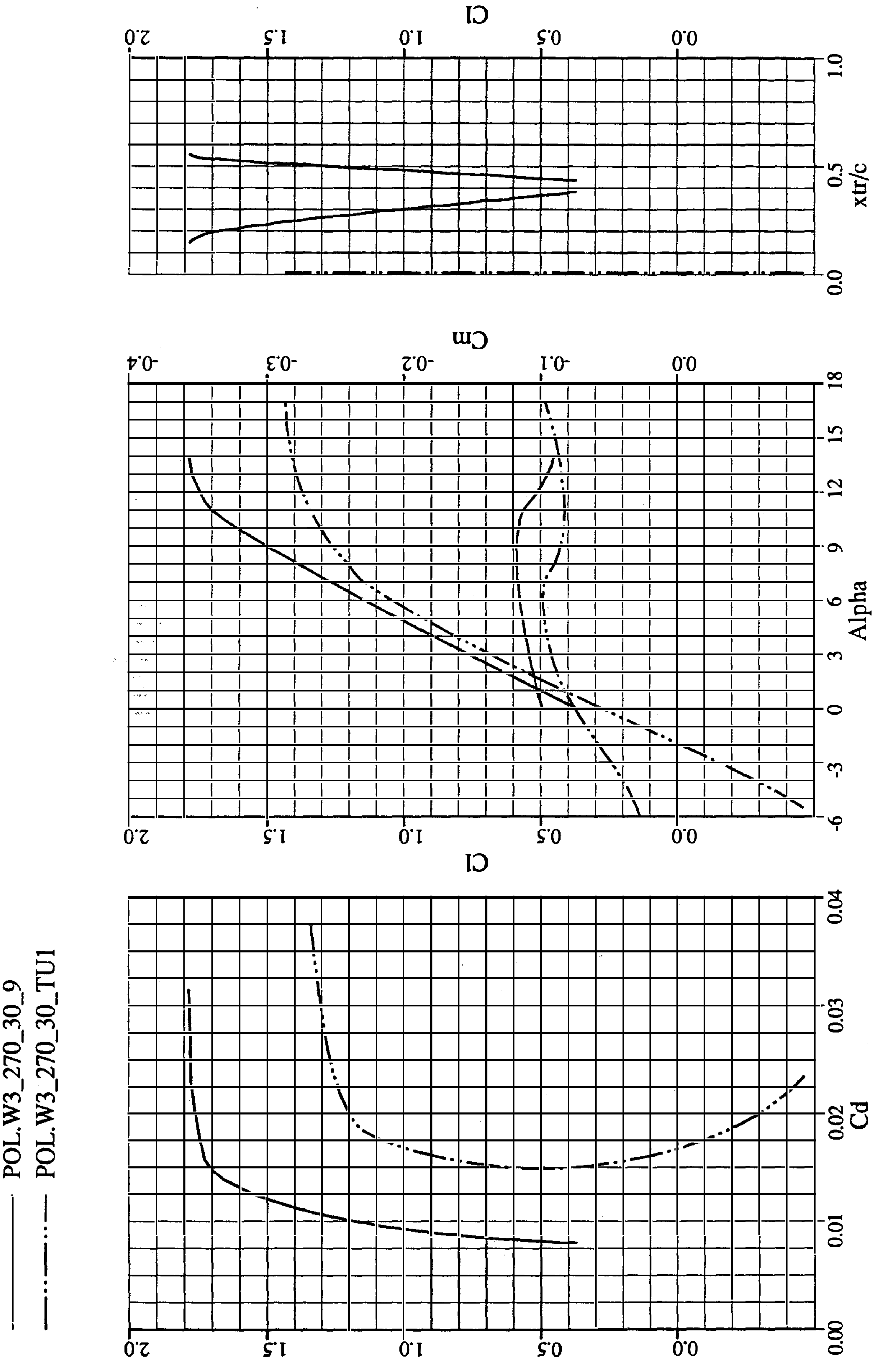


Fig.41c

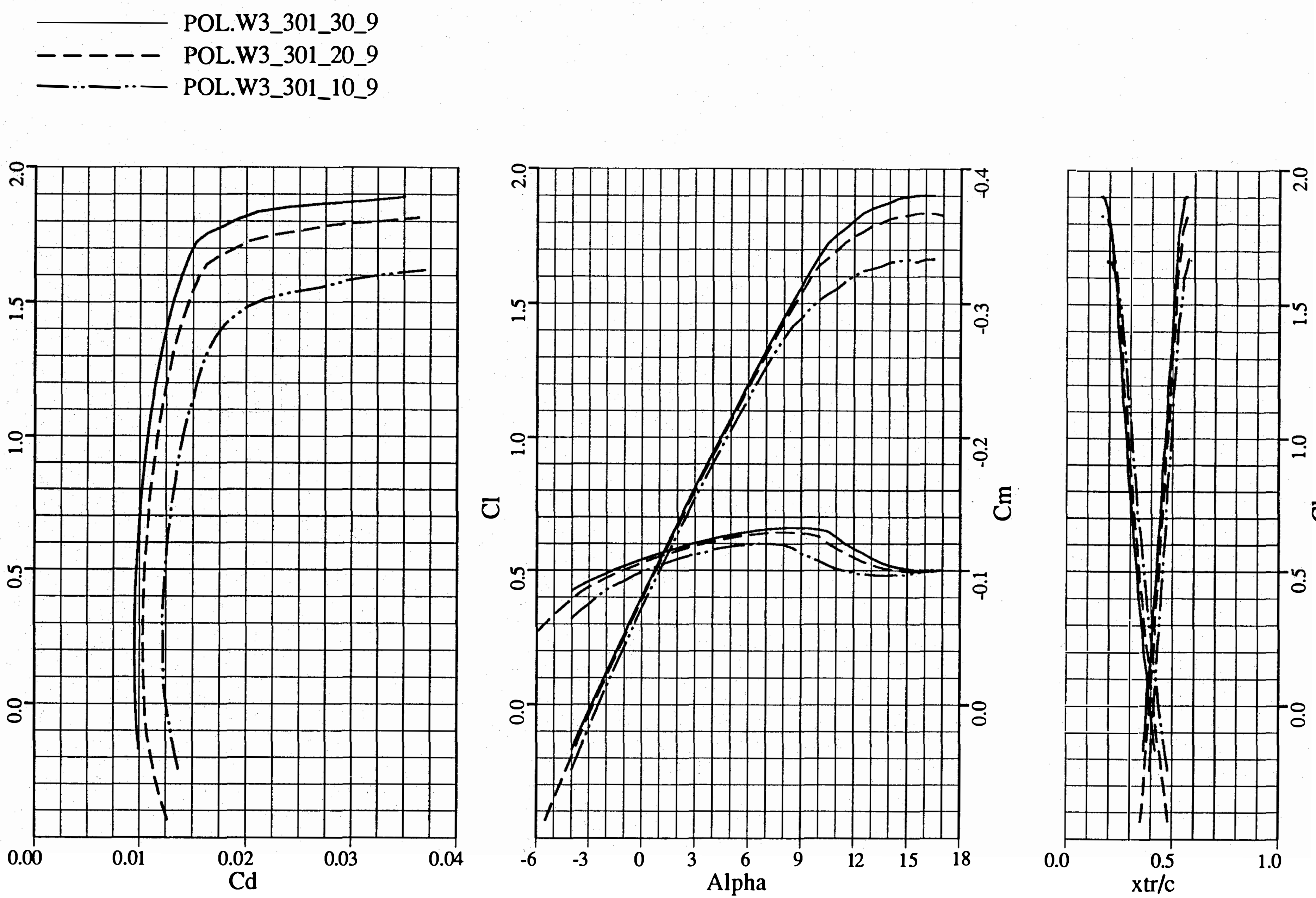

POL.W3_301_30_9
POL.W3_301_20_9
POL.W3_301_10_9
Cl
Cd
Alpha
Cm
xtr/c
0.00
0.01
0.02
0.03
0.04
-6
-3
0
3
6
9
12
15
18
0.0
0.5
1.0
1.5
2.0
-0.4
-0.3
-0.2
-0.1

Fig.42a

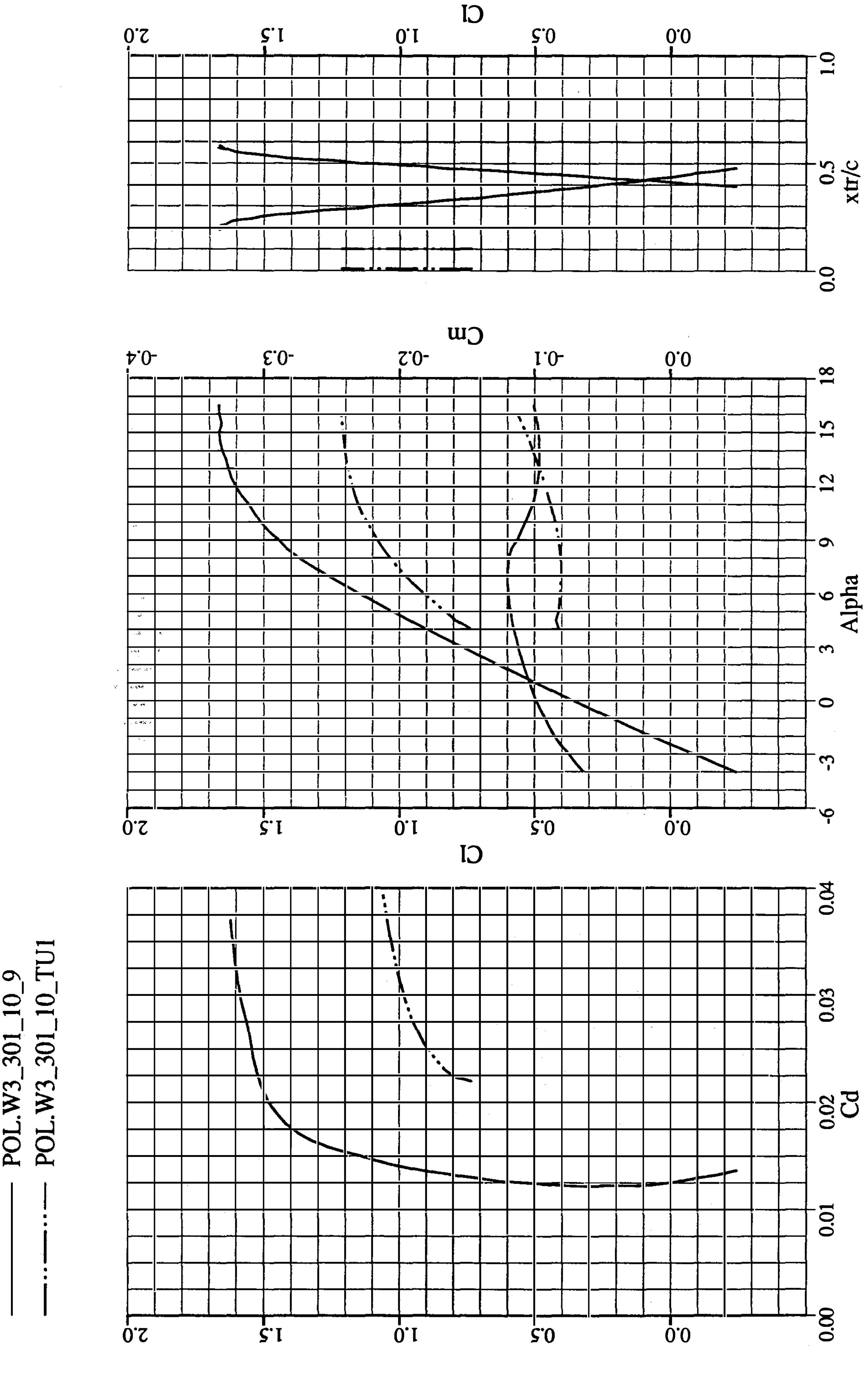

POL.W3_301_10_9
POL.W3_301_10_TU1
Cl
0.0
0.5
1.0
1.5
2.0
xtr/c
0.0
0.5
1.0
Cm
0.0
-0.1
-0.2
-0.3
-0.4
Alpha
-6
-3
0
3
6
9
12
15
18
Cd
0.00
0.01
0.02
0.03
0.04

Fig.42b

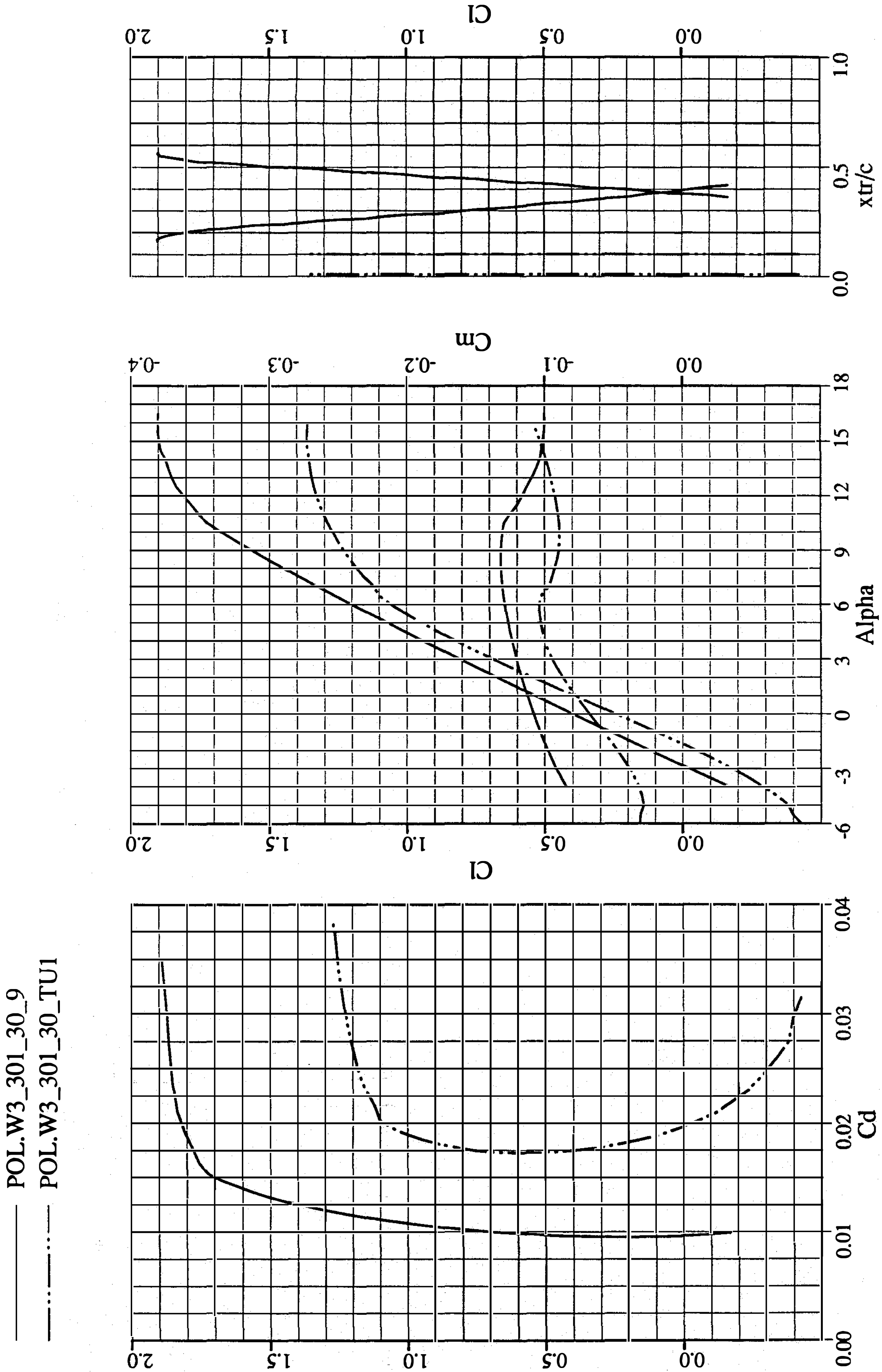


Fig.42c

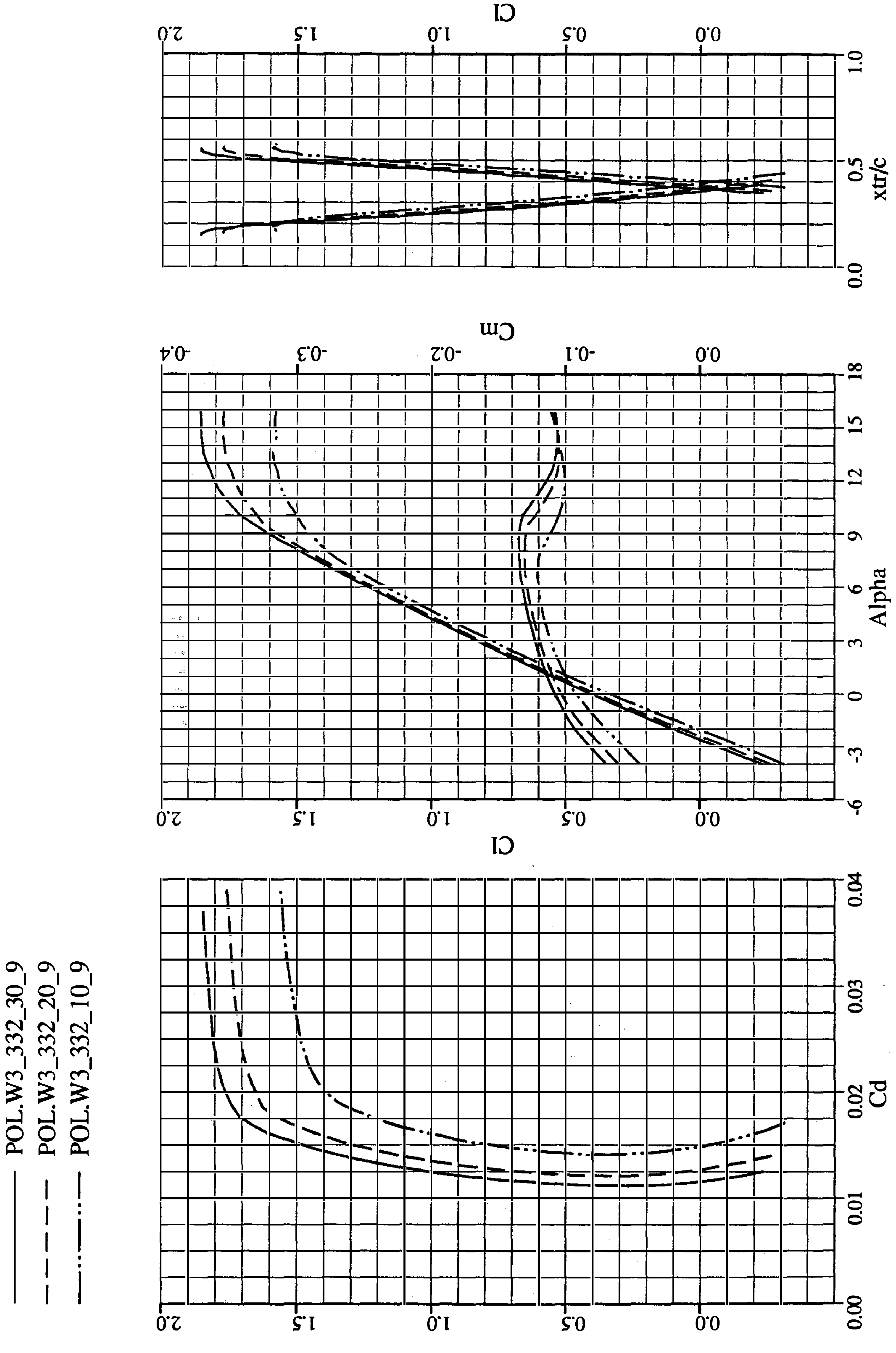


Fig.43a

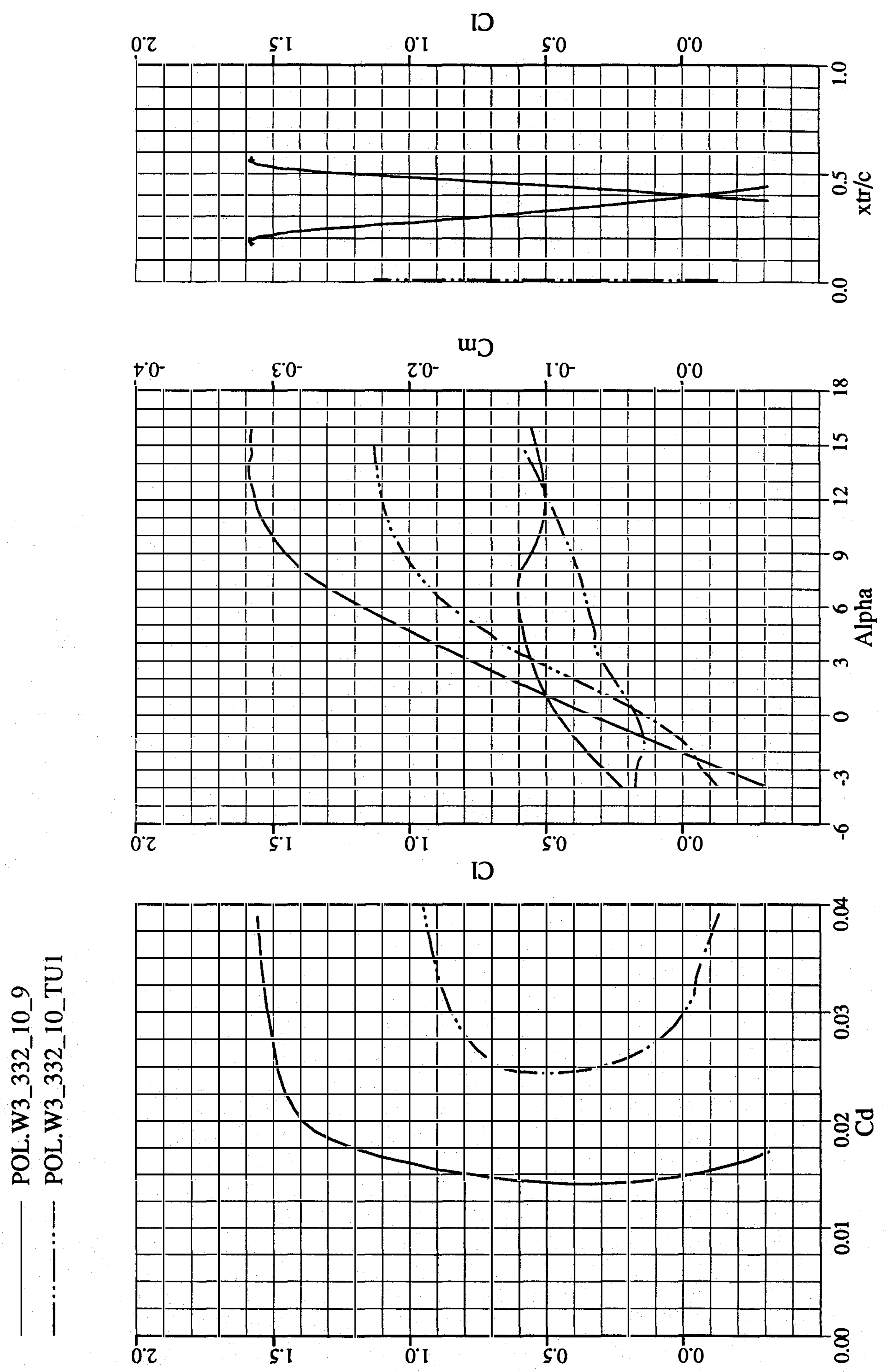


Fig.43b

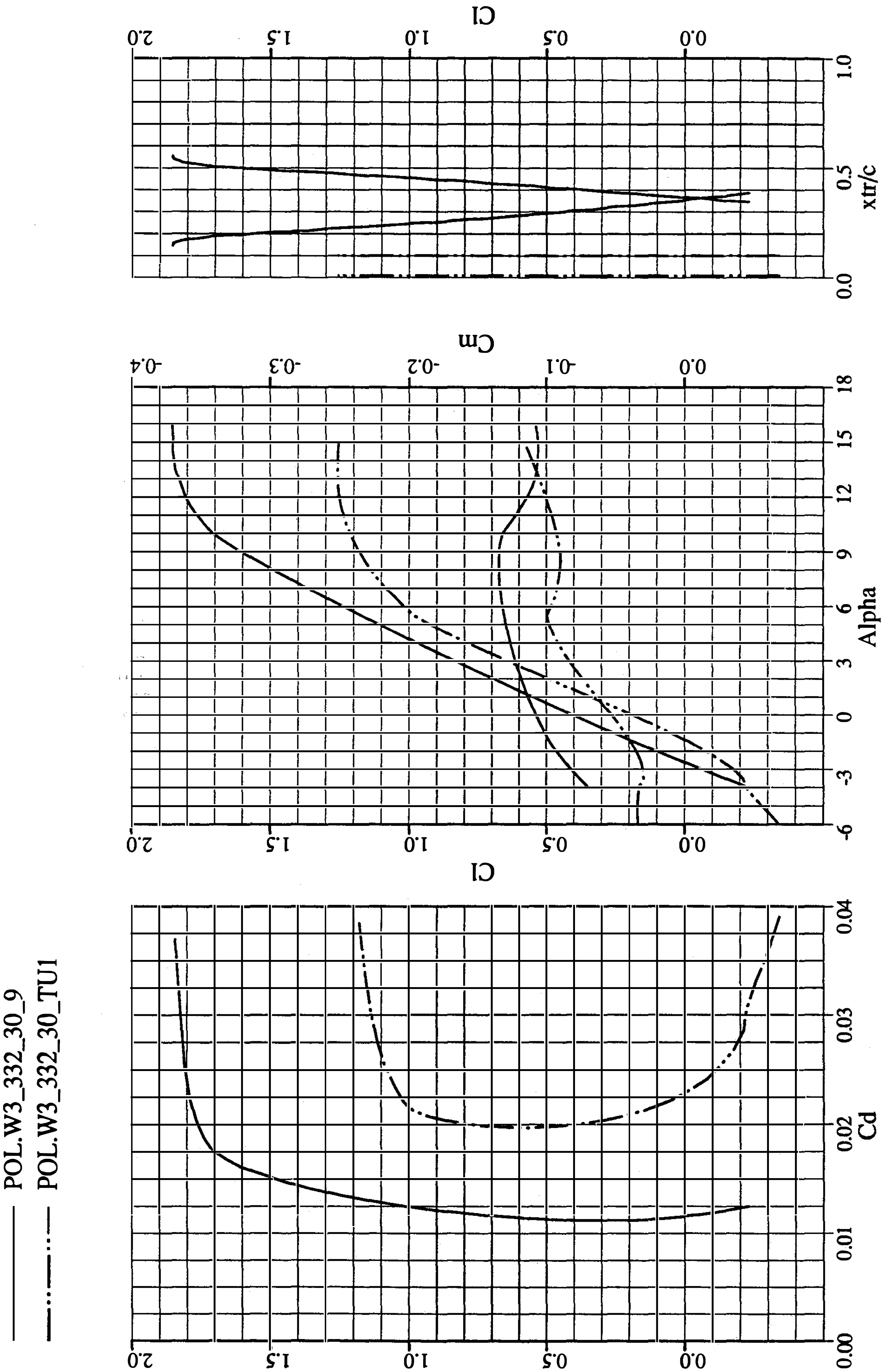

POL.W3_332_30_9
POL.W3_332_30_TU1
Cl
Cd
Alpha
Cm
xtr/c
0.00
0.01
0.02
0.03
0.04
-6
-3
0
3
6
9
12
15
18
0.0
0.5
1.0
1.5
2.0
-0.1
-0.2
-0.3
-0.4

Fig.43c

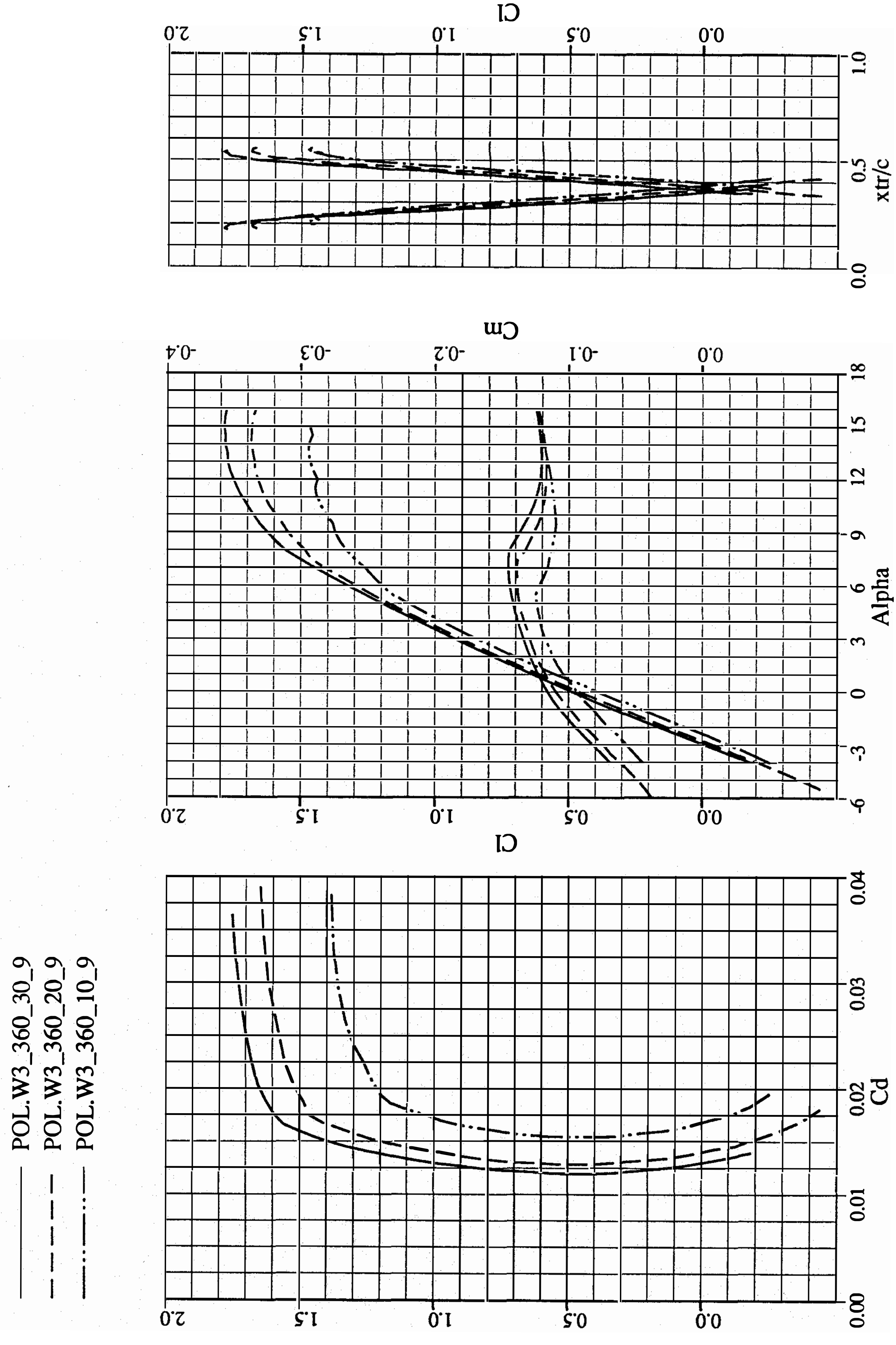


Fig.44a

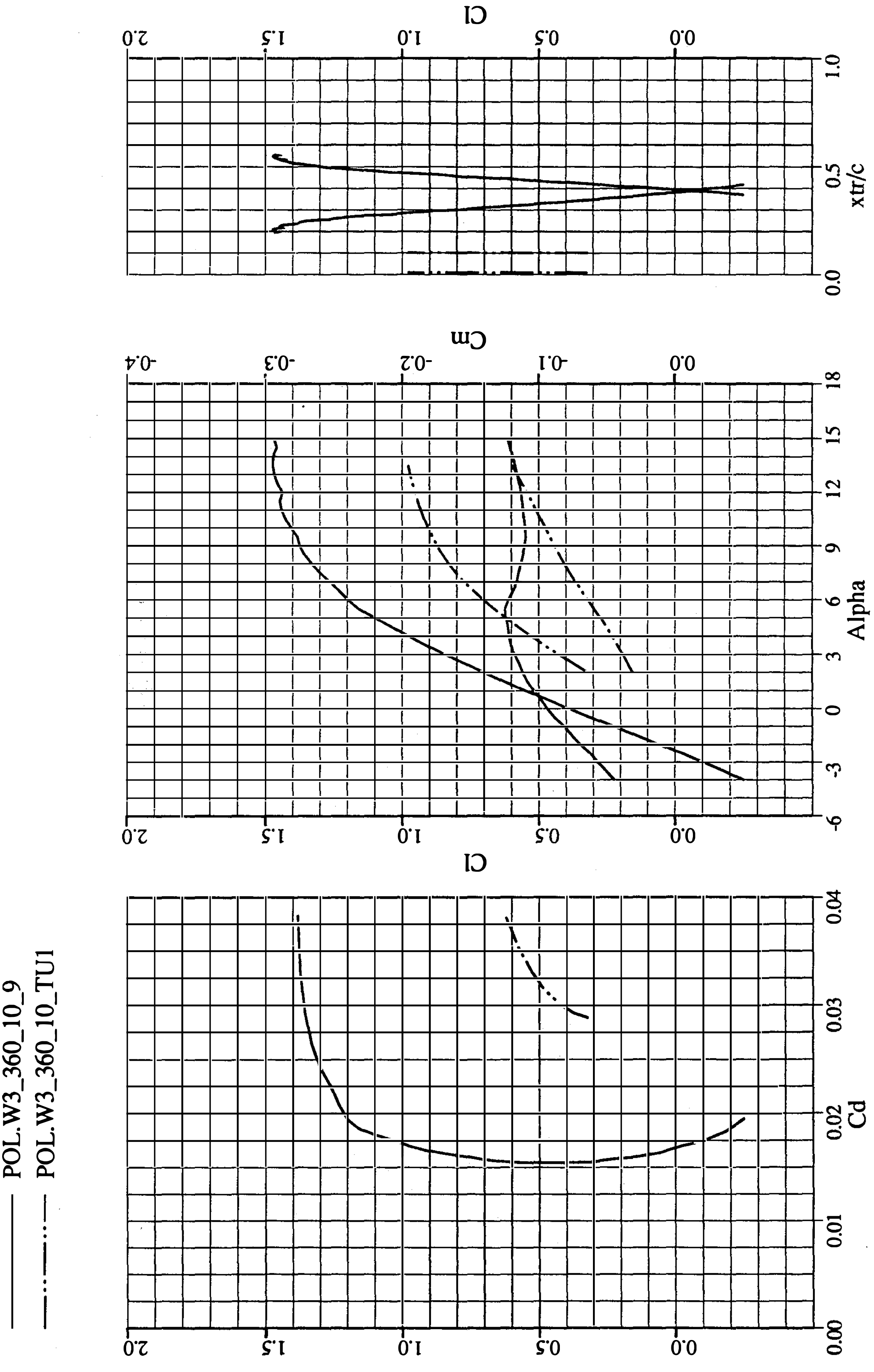


Fig.44b

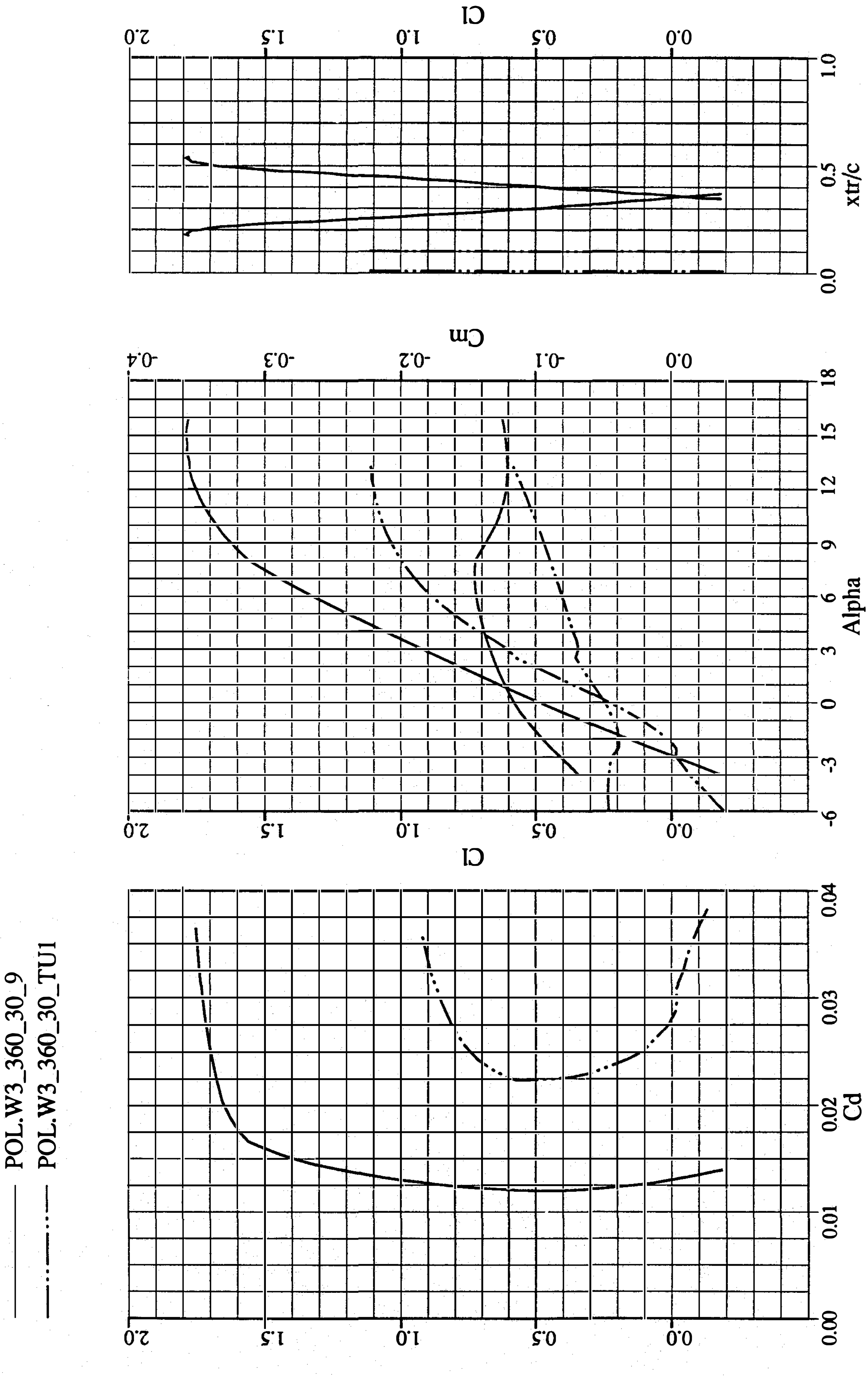

POL.W3_360_30_9
POL.W3_360_30_TU1
Cl
Cd
Cm
Alpha
xtr/c
0.0
0.5
1.0
1.5
2.0
0.00
0.01
0.02
0.03
0.04
-6
-3
0
3
6
9
12
15
18
-0.1
-0.2
-0.3
-0.4

Fig.44c

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Title

COORDINATES AND CALCULATIONS FOR THE FFA-W1-xxx, FFA-W2-xxx AND FFA-W3-xxx SERIES OF AIRFOILS FOR HORIZONTAL AXIS WIND TURBINES

Author(s): Anders Björck

Work performed by: Anders Björck

Checked by: Björn Montgomerie

Approved by: Anders Gustafsson
Head, Aerodynamics A2

Abstracts

Airfoils for use on horizontal axis wind turbines have been designed. The airfoils are devided into three different series:

The first series, FFA-W1-xxx, constitutes airfoils with thickness to chord ratios from 12.8% to 27.1%. The design lift coefficients for the FFA-W1-xxx series range from 0.9 for the 12.8% airfoil, 1.05 for a 15.2% airfoil to 1.2 for the 27.1% airfoil.

Two airfoils in a second series FFA-W2-xxx are designed with design lift coefficients approximately 0.15 units lower than that for the FFA-W1-xxx series.

The third series of airfoils constitutes airfoils with thickness to chord ratios ranging from 19.5% to 36%. The 21.1% and 19.5% thick airfoils are designed to conform to thinner NACA 63-600 airfoils which are then to be used for the outer parts of the wind turbine blade. The thicker airfoils are designed to offer better aerodynamic performance for a given thickness to chord ratio than thick NACA 63-600 airfoils.

Key words

AIRFOILS, HORIZONTAL AXIS WIND TURBINES

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APPENDIX

COORDINATES AND CALCULATIONS FOR THE FFA-W1-*xxx*, FFA-W2-*xxx* AND FFA-W3-*xxx* SERIES OF AIRFOILS FOR HORIZONTAL AXIS WIND TURBINES

by

Anders Björck

Stockholm 1990

FLYGTEKNISKA FÖRSÖKSANSTALTEN

The Aeronautical Research
Institute of Sweden

Aerodynamics Department

FFA TN 1990-15

APPENDIX

COORDINATES AND CALCULATIONS FOR THE FFA-W1-xxx, FFA-W2-xxx AND FFA-W3-xxx SERIES OF AIRFOILS FOR HORIZONTAL AXIS WIND TURBINES

by

Anders Björck

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ISES polar driver Version 1.3

Calculated polar for: Airfoil FFA-W1-152

Vortex + doublet far field
Entropy conserved

Mach = 0.150 Re = 3.000 e 6 Acrit = 9.000

alpha	CL	CD	CDi	CM	S xtr	P xtr
4.000	0.8363	0.00554	0.00000	-0.0627	0.4405	0.8719
4.500	0.8949	0.00556	0.00000	-0.0635	0.4337	0.8877
5.000	0.9526	0.00572	0.00000	-0.0643	0.4233	0.9018
5.500	1.0097	0.00594	0.00000	-0.0649	0.4103	0.9173
6.000	1.0658	0.00614	0.00000	-0.0654	0.3960	0.9353
6.250	1.0936	0.00621	0.00000	-0.0656	0.3880	0.9455
6.500	1.1195	0.00635	0.00000	-0.0655	0.3779	0.9595
6.667	1.1363	0.00644	0.00000	-0.0652	0.3706	0.9709
6.778	1.1471	0.00650	0.00000	-0.0650	0.3671	0.9842
6.852	1.1565	0.00653	0.00000	-0.0653	0.3652	0.9902
6.901	1.1640	0.00656	0.00000	-0.0658	0.3615	0.9946
6.951	1.1723	0.00660	0.00000	-0.0665	0.3594	0.9977
7.000	1.1793	0.00664	0.00000	-0.0669	0.3577	1.0000
7.250	1.2064	0.00686	0.00000	-0.0672	0.3464	1.0000
7.500	1.2337	0.00712	0.00000	-0.0675	0.3361	1.0000
7.750	1.2604	0.00741	0.00000	-0.0678	0.3211	1.0000
8.000	1.2868	0.00772	0.00000	-0.0681	0.3076	1.0000
8.250	1.3118	0.00811	0.00000	-0.0682	0.2867	1.0000
8.500	1.3353	0.00862	0.00000	-0.0682	0.2655	1.0000
8.750	1.3578	0.00916	0.00000	-0.0681	0.2418	1.0000
9.000	1.3750	0.01001	0.00000	-0.0674	0.2085	1.0000
9.250	1.3930	0.01073	0.00000	-0.0668	0.1841	1.0000
9.417	1.4026	0.01129	0.00000	-0.0661	0.1692	1.0000
9.583	1.4127	0.01180	0.00000	-0.0656	0.1545	1.0000
9.750	1.4201	0.01236	0.00000	-0.0646	0.1423	1.0000
10.000	1.4179	0.01346	0.00000	-0.0618	0.1301	1.0000
10.250	1.4165	0.01485	0.00000	-0.0603	0.1192	1.0000
10.500	1.4132	0.01653	0.00000	-0.0590	0.1097	1.0000
10.750	1.4026	0.01846	0.00000	-0.0561	0.1026	1.0000
10.861	1.3989	0.01943	0.00000	-0.0554	0.0994	1.0000

ISES polar driver Version 1.3

Calculated polar for: Airfoil FFA-W1-152

Vortex + doublet far field
Entropy conserved

Mach = 0.150 Re = 3.000 e 6 Acrit = 9.000

alpha	CL	CD	CDi	CM	S xtr	P xtr
4.000	0.8068	0.01175	0.00000	-0.0603	0.0100	0.1000
4.500	0.8605	0.01208	0.00000	-0.0607	0.0100	0.1000
5.000	0.9137	0.01245	0.00000	-0.0611	0.0100	0.1000
5.500	0.9657	0.01286	0.00000	-0.0613	0.0100	0.1000
6.000	1.0164	0.01331	0.00000	-0.0614	0.0100	0.1000
6.250	1.0404	0.01356	0.00000	-0.0614	0.0100	0.1000
6.500	1.0644	0.01383	0.00000	-0.0613	0.0100	0.1000
6.750	1.0879	0.01411	0.00000	-0.0611	0.0100	0.1000
7.000	1.1104	0.01442	0.00000	-0.0609	0.0100	0.1000
7.250	1.1325	0.01476	0.00000	-0.0607	0.0100	0.1000
7.500	1.1517	0.01519	0.00000	-0.0601	0.0100	0.1000
7.750	1.1598	0.01587	0.00000	-0.0584	0.0100	0.1000
8.000	1.1697	0.01668	0.00000	-0.0575	0.0100	0.1000
8.250	1.1785	0.01762	0.00000	-0.0565	0.0100	0.1000
8.500	1.1836	0.01871	0.00000	-0.0551	0.0100	0.1000
8.750	1.1936	0.01990	0.00000	-0.0555	0.0100	0.1000
9.000	1.1937	0.02128	0.00000	-0.0524	0.0100	0.1000
9.250	1.1988	0.02278	0.00000	-0.0529	0.0100	0.1000
9.500	1.2008	0.02438	0.00000	-0.0519	0.0100	0.1000
9.750	1.1982	0.02602	0.00000	-0.0481	0.0100	0.1000
9.824	1.1986	0.02662	0.00000	-0.0491	0.0100	0.1000
9.898	1.2062	0.02711	0.00000	-0.0528	0.0100	0.1000
9.972	1.2008	0.02759	0.00000	-0.0493	0.0100	0.1000
10.083	1.1985	0.02848	0.00000	-0.0478	0.0100	0.1000
10.250	1.2024	0.02971	0.00000	-0.0489	0.0100	0.1000
10.500	1.2007	0.03160	0.00000	-0.0474	0.0100	0.1000
10.611	1.2031	0.03248	0.00000	-0.0484	0.0100	0.1000
10.722	1.1995	0.03326	0.00000	-0.0453	0.0100	0.1000
10.833	1.1947	0.03426	0.00000	-0.0430	0.0100	0.1000
11.000	1.2080	0.03550	0.00000	-0.0487	0.0100	0.1000
3.500	0.7515	0.01145	0.00000	-0.0599	0.0100	0.1000
3.000	0.6951	0.01117	0.00000	-0.0595	0.0100	0.1000
2.500	0.6398	0.01092	0.00000	-0.0587	0.0100	0.1000
2.000	0.5833	0.01070	0.00000	-0.0581	0.0100	0.1000
1.500	0.5257	0.01050	0.00000	-0.0575	0.0100	0.1000
1.000	0.4684	0.01031	0.00000	-0.0569	0.0100	0.1000
0.500	0.4113	0.01014	0.00000	-0.0561	0.0100	0.1000
0.000	0.3535	0.01000	0.00000	-0.0555	0.0100	0.1000
-0.500	0.2955	0.00987	0.00000	-0.0548	0.0100	0.1000
-1.000	0.2372	0.00976	0.00000	-0.0541	0.0100	0.1000
-1.500	0.1791	0.00975	0.00000	-0.0534	0.0100	0.0829
-2.000	0.1210	0.00980	0.00000	-0.0527	0.0100	0.0588
-2.500	0.0629	0.00987	0.00000	-0.0521	0.0100	0.0432
-3.000	0.0048	0.00994	0.00000	-0.0515	0.0100	0.0327
-3.066	-0.0029	0.00996	0.00000	-0.0514	0.0100	0.0307
-3.095	-0.0062	0.00999	0.00000	-0.0514	0.0100	0.0298

ISES polar driver Version 1.3

Calculated polar for: AIRFOIL FFA-W1-182

Vortex + doublet far field
Entropy conserved

Mach = 0.150 Re = 3.000 e 6 Acrit = 9.000

alpha	CL	CD	CDi	CM	S xtr	P xtr
1.000	0.4565	0.00583	0.00000	-0.0626	0.4524	0.5781
1.500	0.5172	0.00585	0.00000	-0.0636	0.4487	0.5865
2.000	0.5777	0.00588	0.00000	-0.0647	0.4435	0.5958
2.500	0.6375	0.00604	0.00000	-0.0657	0.4366	0.6027
3.000	0.6978	0.00608	0.00000	-0.0667	0.4322	0.6105
3.500	0.7579	0.00615	0.00000	-0.0677	0.4265	0.6177
4.000	0.8168	0.00635	0.00000	-0.0687	0.4191	0.6252
4.500	0.8770	0.00640	0.00000	-0.0697	0.4138	0.6307
5.000	0.9357	0.00657	0.00000	-0.0706	0.4047	0.6383
5.500	0.9949	0.00671	0.00000	-0.0716	0.3969	0.6435
6.000	1.0525	0.00695	0.00000	-0.0724	0.3854	0.6513
6.500	1.1104	0.00719	0.00000	-0.0732	0.3708	0.6571
7.000	1.1669	0.00749	0.00000	-0.0739	0.3503	0.6633
7.500	1.2216	0.00796	0.00000	-0.0744	0.3266	0.6699
7.750	1.2475	0.00831	0.00000	-0.0745	0.3093	0.6724
8.000	1.2737	0.00863	0.00000	-0.0747	0.2946	0.6749
8.250	1.2982	0.00902	0.00000	-0.0746	0.2725	0.6784
8.500	1.3219	0.00949	0.00000	-0.0745	0.2525	0.6821
8.750	1.3448	0.00997	0.00000	-0.0743	0.2326	0.6850
9.000	1.3663	0.01053	0.00000	-0.0739	0.2111	0.6875
9.167	1.3796	0.01095	0.00000	-0.0735	0.1967	0.6892
9.333	1.3922	0.01140	0.00000	-0.0731	0.1818	0.6912
9.500	1.4036	0.01184	0.00000	-0.0724	0.1676	0.6936
9.750	1.4196	0.01255	0.00000	-0.0715	0.1482	0.6968
10.000	1.4333	0.01328	0.00000	-0.0702	0.1310	0.6999
10.250	1.4371	0.01411	0.00000	-0.0675	0.1184	0.7027
10.500	1.4334	0.01525	0.00000	-0.0646	0.1083	0.7057
10.750	1.4275	0.01676	0.00000	-0.0622	0.0998	0.7095
11.000	1.4229	0.01856	0.00000	-0.0613	0.0927	0.7129
11.250	1.4085	0.02092	0.00000	-0.0587	0.0870	0.7158
11.500	1.4049	0.02331	0.00000	-0.0609	0.0820	0.7181
11.667	1.3850	0.02519	0.00000	-0.0541	0.0795	0.7200
11.833	1.3768	0.02726	0.00000	-0.0556	0.0772	0.7223
12.000	1.3739	0.02926	0.00000	-0.0590	0.0747	0.7244
12.167	1.3601	0.03121	0.00000	-0.0543	0.0720	0.7264
12.333	1.3521	0.03332	0.00000	-0.0544	0.0696	0.7283
12.444	1.3431	0.03481	0.00000	-0.0535	0.0685	0.7293
12.583	1.3265	0.03666	0.00000	-0.0484	0.0666	0.7311
12.750	1.3269	0.03896	0.00000	-0.0533	0.0642	0.7326
12.799	1.3208	0.03961	0.00000	-0.0507	0.0636	0.7333
12.849	1.3322	0.04014	0.00000	-0.0571	0.0627	0.7341
12.898	1.3306	0.04067	0.00000	-0.0566	0.0619	0.7348
12.972	1.3163	0.04152	0.00000	-0.0504	0.0611	0.7357
13.083	1.3154	0.04304	0.00000	-0.0539	0.0602	0.7371
13.105	1.3105	0.04329	0.00000	-0.0514	0.0599	0.7374
13.128	1.3140	0.04355	0.00000	-0.0537	0.0594	0.7377
13.150	1.3010	0.04377	0.00000	-0.0468	0.0591	0.7380
13.185	1.3109	0.04422	0.00000	-0.0529	0.0586	0.7385
13.241	1.2963	0.04502	0.00000	-0.0463	0.0580	0.7391
13.302	1.2868	0.04591	0.00000	-0.0432	0.0574	0.7398
13.389	1.3036	0.04711	0.00000	-0.0542	0.0561	0.7409
13.500	1.3030	0.04857	0.00000	-0.0554	0.0544	0.7423
13.750	1.2845	0.05161	0.00000	-0.0484	0.0516	0.7450
13.861	1.2710	0.05298	0.00000	-0.0440	0.0508	0.7459

ISES polar driver Version 1.3

Calculated polar for: AIRFOIL FFA-W1-211

Vortex + doublet far field
Entropy conserved

Mach = 0.150 Re = 3.000 e 6 Acrit = 9.000

alpha	CL	CD	CDi	CM	S xtr	P xtr
-14.96300	-0.71900	0.07880	0.00000	-0.01710	0.77950	0.01480
-14.88900	-0.71970	0.07748	0.00000	-0.01870	0.77670	0.01480
-14.82700	-0.77710	0.07666	0.00000	0.00180	0.77430	0.01490
-14.77100	-0.76030	0.07580	0.00000	-0.00620	0.77240	0.01490
-14.72200	-0.78240	0.07482	0.00000	0.00080	0.77040	0.01500
-14.67300	-0.73430	0.07400	0.00000	-0.01810	0.76890	0.01500
-14.62300	-0.74000	0.07296	0.00000	-0.01720	0.76760	0.01500
-14.57400	-0.73950	0.07207	0.00000	-0.01890	0.76590	0.01500
-14.50000	-0.75650	0.07084	0.00000	-0.01380	0.76330	0.01510
-14.33300	-0.75260	0.06768	0.00000	-0.01920	0.75710	0.01520
-14.24100	-0.75370	0.06618	0.00000	-0.02060	0.75450	0.01530
-14.15700	-0.78130	0.06498	0.00000	-0.01180	0.75200	0.01540
-14.08300	-0.80980	0.06388	0.00000	-0.00260	0.74690	0.01550
-13.97200	-0.76330	0.06199	0.00000	-0.02260	0.74310	0.01560
-13.86100	-0.76400	0.06040	0.00000	-0.02370	0.74000	0.01580
-13.75000	-0.79360	0.05883	0.00000	-0.01420	0.73660	0.01590
-13.58300	-0.79920	0.05652	0.00000	-0.01410	0.73060	0.01610
-13.47200	-0.77800	0.05553	0.00000	-0.02190	0.72730	0.01640
-13.36100	-0.78210	0.05440	0.00000	-0.02130	0.72420	0.01660
-13.25000	-0.79040	0.05308	0.00000	-0.01960	0.72080	0.01670
-13.00000	-0.79170	0.05053	0.00000	-0.01970	0.71320	0.01730
-12.75000	-0.79140	0.04819	0.00000	-0.01990	0.70510	0.01780
-12.50000	-0.79380	0.04610	0.00000	-0.01940	0.69830	0.01840
-12.25000	-0.79030	0.04420	0.00000	-0.01990	0.69050	0.01900
-12.00000	-0.78460	0.04207	0.00000	-0.02160	0.68210	0.01960
-11.83300	-0.78730	0.04103	0.00000	-0.01990	0.67690	0.02000
-11.66700	-0.78500	0.03986	0.00000	-0.02040	0.67280	0.02040
-11.50000	-0.78450	0.03860	0.00000	-0.01980	0.66840	0.02080
-11.25000	-0.78480	0.03684	0.00000	-0.01950	0.66110	0.02120
-11.00000	-0.77450	0.03524	0.00000	-0.02170	0.65510	0.02200
-10.75000	-0.77620	0.03338	0.00000	-0.02080	0.64790	0.02270
-10.50000	-0.77280	0.03181	0.00000	-0.02070	0.64200	0.02340
-10.25000	-0.77770	0.02998	0.00000	-0.01960	0.63540	0.02400
-10.00000	-0.77100	0.02837	0.00000	-0.02080	0.62920	0.02490
-9.75000	-0.77440	0.02659	0.00000	-0.01980	0.62070	0.02550
-9.50000	-0.78030	0.02466	0.00000	-0.01940	0.61510	0.02630
-9.25000	-0.78950	0.02259	0.00000	-0.01890	0.60890	0.02750
-9.00000	-0.77870	0.02073	0.00000	-0.02110	0.60220	0.03080
-8.75000	-0.76080	0.01854	0.00000	-0.02590	0.59540	0.03350
-8.50000	-0.73510	0.01653	0.00000	-0.03180	0.58840	0.03810
-8.25000	-0.70600	0.01481	0.00000	-0.03670	0.58150	0.04370
-8.00000	-0.67300	0.01318	0.00000	-0.04200	0.57420	0.05210
-7.75000	-0.63860	0.01173	0.00000	-0.04670	0.56780	0.06570
-7.50000	-0.60750	0.01109	0.00000	-0.04880	0.56040	0.08070
-7.16700	-0.56540	0.00991	0.00000	-0.05210	0.55140	0.12550
-6.94400	-0.53740	0.00901	0.00000	-0.05430	0.54540	0.17400
-6.72200	-0.50950	0.00806	0.00000	-0.05640	0.54090	0.23400
-6.50000	-0.48200	0.00745	0.00000	-0.05780	0.53550	0.28010
-6.00000	-0.42020	0.00674	0.00000	-0.05980	0.52460	0.34190
-5.50000	-0.35870	0.00645	0.00000	-0.06120	0.51340	0.37120
-5.00000	-0.29700	0.00630	0.00000	-0.06250	0.50170	0.39020
-4.50000	-0.23530	0.00624	0.00000	-0.06350	0.49280	0.41200
-4.00000	-0.17350	0.00618	0.00000	-0.06470	0.48310	0.42010
-3.50000	-0.11160	0.00616	0.00000	-0.06580	0.47520	0.42530
-3.00000	-0.05020	0.00625	0.00000	-0.06670	0.46580	0.42910
-2.50000	0.01170	0.00630	0.00000	-0.06780	0.45750	0.43240
-2.00000	0.07350	0.00624	0.00000	-0.06900	0.45030	0.43720
-1.50000	0.13510	0.00625	0.00000	-0.07010	0.44260	0.44180
-1.00000	0.19660	0.00634	0.00000	-0.07110	0.43520	0.44550
-0.50000	0.25800	0.00644	0.00000	-0.07200	0.42870	0.44780
0.00000	0.31920	0.00653	0.00000	-0.07310	0.42100	0.45190
0.500	0.3810	0.00646	0.00000	-0.0742	0.4160	0.4568

1.000	0.4422	0.00655	0.00000	-0.0752	0.4090	0.4606
1.500	0.5031	0.00667	0.00000	-0.0761	0.4030	0.4636
2.000	0.5641	0.00678	0.00000	-0.0770	0.3980	0.4659
2.500	0.6247	0.00686	0.00000	-0.0780	0.3905	0.4708
3.000	0.6857	0.00689	0.00000	-0.0790	0.3857	0.4751
3.500	0.7463	0.00698	0.00000	-0.0800	0.3797	0.4789
4.000	0.8057	0.00723	0.00000	-0.0807	0.3723	0.4818
4.500	0.8663	0.00732	0.00000	-0.0815	0.3676	0.4845
5.000	0.9264	0.00738	0.00000	-0.0824	0.3611	0.4895
5.500	0.9847	0.00763	0.00000	-0.0832	0.3528	0.4940
6.000	1.0448	0.00771	0.00000	-0.0840	0.3447	0.4979
6.500	1.1029	0.00798	0.00000	-0.0846	0.3342	0.5010
7.000	1.1599	0.00831	0.00000	-0.0851	0.3201	0.5047
7.500	1.2176	0.00857	0.00000	-0.0857	0.3057	0.5098
7.750	1.2451	0.00873	0.00000	-0.0858	0.2963	0.5121
8.000	1.2723	0.00897	0.00000	-0.0860	0.2873	0.5143
8.250	1.2993	0.00921	0.00000	-0.0860	0.2766	0.5164
8.500	1.3257	0.00950	0.00000	-0.0860	0.2645	0.5184
8.750	1.3508	0.00987	0.00000	-0.0859	0.2500	0.5199
9.000	1.3747	0.01032	0.00000	-0.0856	0.2337	0.5214
9.250	1.3966	0.01085	0.00000	-0.0851	0.2149	0.5230
9.324	1.4038	0.01097	0.00000	-0.0851	0.2097	0.5238
9.398	1.4098	0.01114	0.00000	-0.0848	0.2036	0.5246
9.472	1.4162	0.01130	0.00000	-0.0847	0.1983	0.5254
9.583	1.4251	0.01156	0.00000	-0.0844	0.1897	0.5265
9.750	1.4383	0.01195	0.00000	-0.0839	0.1771	0.5282
9.917	1.4507	0.01237	0.00000	-0.0833	0.1646	0.5298
10.083	1.4624	0.01283	0.00000	-0.0828	0.1525	0.5313
10.250	1.4725	0.01330	0.00000	-0.0818	0.1406	0.5327
10.500	1.4864	0.01403	0.00000	-0.0804	0.1252	0.5348
10.667	1.4933	0.01455	0.00000	-0.0792	0.1166	0.5361
10.741	1.4943	0.01476	0.00000	-0.0782	0.1129	0.5366
10.815	1.4920	0.01501	0.00000	-0.0764	0.1104	0.5371
10.889	1.4908	0.01530	0.00000	-0.0753	0.1082	0.5375
11.000	1.4916	0.01571	0.00000	-0.0746	0.1040	0.5380
11.250	1.4875	0.01691	0.00000	-0.0715	0.0966	0.5392
11.500	1.4847	0.01836	0.00000	-0.0695	0.0899	0.5403
11.667	1.4804	0.01960	0.00000	-0.0689	0.0864	0.5417
11.833	1.4714	0.02106	0.00000	-0.0672	0.0831	0.5435
12.000	1.4652	0.02252	0.00000	-0.0661	0.0803	0.5448
12.250	1.4543	0.02502	0.00000	-0.0650	0.0766	0.5466
12.500	1.4344	0.02801	0.00000	-0.0606	0.0732	0.5481
12.750	1.4204	0.03148	0.00000	-0.0635	0.0709	0.5493
13.000	1.4088	0.03463	0.00000	-0.0624	0.0680	0.5507

ISES polar driver Version 1.3

Calculated polar for: AIRFOIL FFA-W1-211

Vortex + doublet far field
Entropy conserved

Mach = 0.150 Re = 3.000 e 6 Acrit = 9.000

alpha	CL	CD	CDi	CM	S xtr	P xtr
0.000	0.2981	0.01107	0.00000	-0.0688	0.0100	0.1000
0.500	0.3567	0.01117	0.00000	-0.0696	0.0100	0.1000
1.000	0.4168	0.01129	0.00000	-0.0707	0.0100	0.1000
1.500	0.4764	0.01142	0.00000	-0.0715	0.0100	0.1000
2.000	0.5342	0.01158	0.00000	-0.0724	0.0100	0.1000
2.500	0.5926	0.01177	0.00000	-0.0732	0.0100	0.1000
3.000	0.6517	0.01197	0.00000	-0.0738	0.0100	0.1000
3.500	0.7093	0.01220	0.00000	-0.0745	0.0100	0.1000
4.000	0.7662	0.01246	0.00000	-0.0751	0.0100	0.1000
4.500	0.8226	0.01274	0.00000	-0.0756	0.0100	0.1000
5.000	0.8780	0.01305	0.00000	-0.0760	0.0100	0.1000
5.500	0.9328	0.01340	0.00000	-0.0763	0.0100	0.1000
6.000	0.9862	0.01378	0.00000	-0.0765	0.0100	0.1000
6.500	1.0388	0.01421	0.00000	-0.0765	0.0100	0.1000
7.000	1.0893	0.01467	0.00000	-0.0764	0.0100	0.1000
7.500	1.1385	0.01520	0.00000	-0.0760	0.0100	0.1000
7.750	1.1616	0.01549	0.00000	-0.0757	0.0100	0.1000
8.000	1.1824	0.01580	0.00000	-0.0752	0.0100	0.1000
8.250	1.2061	0.01613	0.00000	-0.0748	0.0100	0.1000
8.500	1.2257	0.01649	0.00000	-0.0741	0.0100	0.1000
8.750	1.2369	0.01693	0.00000	-0.0719	0.0100	0.1000
9.000	1.2458	0.01753	0.00000	-0.0700	0.0100	0.1000
9.250	1.2540	0.01831	0.00000	-0.0685	0.0100	0.1000
9.500	1.2613	0.01929	0.00000	-0.0677	0.0100	0.1000
9.750	1.2632	0.02050	0.00000	-0.0659	0.0100	0.1000
10.000	1.2653	0.02198	0.00000	-0.0647	0.0100	0.1000
10.250	1.2722	0.02365	0.00000	-0.0676	0.0100	0.1000
10.500	1.2608	0.02574	0.00000	-0.0622	0.0100	0.1000
10.667	1.2516	0.02727	0.00000	-0.0596	0.0100	0.1000
10.833	1.2577	0.02890	0.00000	-0.0647	0.0100	0.1000
11.000	1.2443	0.03059	0.00000	-0.0585	0.0100	0.1000
11.250	1.2431	0.03341	0.00000	-0.0633	0.0100	0.1000
11.500	1.2336	0.03625	0.00000	-0.0609	0.0100	0.1000
11.667	1.2304	0.03828	0.00000	-0.0616	0.0100	0.1000
11.778	1.2203	0.03981	0.00000	-0.0607	0.0100	0.1000
11.889	1.2215	0.04128	0.00000	-0.0636	0.0100	0.1000
12.000	1.2115	0.04253	0.00000	-0.0593	0.0100	0.1000
12.250	1.2048	0.04583	0.00000	-0.0602	0.0100	0.1000

ISES polar driver Version 1.3

Calculated polar for: AIRFOIL FFA-W1-242

Vortex + doublet far field
Entropy conserved

Mach = 0.150 Re = 3.000 e 6 Acrit = 9.000

alpha	CL	CD	CDi	CM	S xtr	P xtr
0.500	0.3478	0.00697	0.00000	-0.0774	0.3908	0.4470
1.000	0.4098	0.00714	0.00000	-0.0785	0.3799	0.4499
1.500	0.4725	0.00722	0.00000	-0.0798	0.3708	0.4539
2.000	0.5357	0.00726	0.00000	-0.0812	0.3604	0.4591
2.500	0.5977	0.00742	0.00000	-0.0824	0.3502	0.4638
3.000	0.6595	0.00754	0.00000	-0.0834	0.3407	0.4672
3.500	0.7203	0.00777	0.00000	-0.0843	0.3311	0.4696
4.000	0.7823	0.00788	0.00000	-0.0855	0.3201	0.4744
4.500	0.8436	0.00799	0.00000	-0.0867	0.3117	0.4797
5.000	0.9034	0.00828	0.00000	-0.0875	0.3001	0.4839
5.500	0.9640	0.00844	0.00000	-0.0882	0.2916	0.4868
6.000	1.0227	0.00876	0.00000	-0.0887	0.2809	0.4893
6.500	1.0819	0.00898	0.00000	-0.0895	0.2696	0.4939
7.000	1.1409	0.00922	0.00000	-0.0901	0.2587	0.4985
7.500	1.1980	0.00958	0.00000	-0.0905	0.2467	0.5027
7.750	1.2262	0.00976	0.00000	-0.0907	0.2396	0.5047
8.000	1.2536	0.01000	0.00000	-0.0907	0.2335	0.5065
8.250	1.2815	0.01018	0.00000	-0.0907	0.2266	0.5079
8.500	1.3082	0.01050	0.00000	-0.0906	0.2179	0.5094
8.750	1.3348	0.01071	0.00000	-0.0905	0.2128	0.5108
9.000	1.3616	0.01092	0.00000	-0.0905	0.2062	0.5129
9.250	1.3874	0.01121	0.00000	-0.0903	0.1996	0.5154
9.500	1.4142	0.01139	0.00000	-0.0902	0.1939	0.5178
9.750	1.4391	0.01169	0.00000	-0.0900	0.1872	0.5200
10.000	1.4635	0.01201	0.00000	-0.0896	0.1810	0.5222
10.250	1.4880	0.01228	0.00000	-0.0892	0.1747	0.5244
10.500	1.5100	0.01268	0.00000	-0.0885	0.1675	0.5263
10.750	1.5329	0.01300	0.00000	-0.0878	0.1613	0.5282
11.000	1.5534	0.01342	0.00000	-0.0869	0.1544	0.5296
11.250	1.5721	0.01388	0.00000	-0.0858	0.1480	0.5310
11.500	1.5876	0.01432	0.00000	-0.0840	0.1416	0.5324
11.750	1.5973	0.01487	0.00000	-0.0814	0.1349	0.5346
12.000	1.6042	0.01536	0.00000	-0.0786	0.1296	0.5374
12.250	1.6070	0.01606	0.00000	-0.0753	0.1242	0.5395
12.500	1.6055	0.01705	0.00000	-0.0721	0.1191	0.5414
12.750	1.6087	0.01806	0.00000	-0.0702	0.1140	0.5432
13.000	1.6034	0.01951	0.00000	-0.0681	0.1094	0.5448
13.250	1.5932	0.02149	0.00000	-0.0656	0.1054	0.5462
13.500	1.5888	0.02355	0.00000	-0.0670	0.1019	0.5472
13.750	1.5730	0.02617	0.00000	-0.0623	0.0984	0.5481
14.000	1.5623	0.02926	0.00000	-0.0660	0.0956	0.5488
14.250	1.5369	0.03286	0.00000	-0.0620	0.0931	0.5497
14.500	1.5169	0.03683	0.00000	-0.0620	0.0907	0.5512
14.611	1.5050	0.03833	0.00000	-0.0596	0.0899	0.5518
14.722	1.4968	0.04004	0.00000	-0.0592	0.0889	0.5524
14.833	1.4947	0.04175	0.00000	-0.0620	0.0878	0.5531
14.944	1.4823	0.04343	0.00000	-0.0586	0.0866	0.5537
15.083	1.4750	0.04588	0.00000	-0.0608	0.0853	0.5544
15.250	1.4643	0.04873	0.00000	-0.0633	0.0840	0.5556

ISES polar driver Version 1.3

Calculated polar for: AIRFOIL FFA-W1-242

Vortex + doublet far field
Entropy conserved

Mach = 0.150 Re = 3.000 e 6 Acrit = 9.000

alpha	CL	CD	CDi	CM	S xtr	P xtr
0.500	0.2962	0.01267	0.00000	-0.0640	0.0100	0.1000
1.000	0.3598	0.01266	0.00000	-0.0660	0.0100	0.1000
1.500	0.4226	0.01269	0.00000	-0.0678	0.0100	0.1000
2.000	0.4839	0.01276	0.00000	-0.0697	0.0100	0.1000
2.500	0.5464	0.01286	0.00000	-0.0711	0.0100	0.1000
3.000	0.6060	0.01300	0.00000	-0.0727	0.0100	0.1000
3.500	0.6661	0.01317	0.00000	-0.0739	0.0100	0.1000
4.000	0.7268	0.01337	0.00000	-0.0748	0.0100	0.1000
4.500	0.7856	0.01360	0.00000	-0.0757	0.0100	0.1000
5.000	0.8433	0.01386	0.00000	-0.0765	0.0100	0.1000
5.500	0.8997	0.01416	0.00000	-0.0772	0.0100	0.1000
6.000	0.9558	0.01449	0.00000	-0.0775	0.0100	0.1000
6.500	1.0098	0.01486	0.00000	-0.0778	0.0100	0.1000
7.000	1.0629	0.01528	0.00000	-0.0778	0.0100	0.1000
7.500	1.1134	0.01574	0.00000	-0.0776	0.0100	0.1000
7.750	1.1368	0.01599	0.00000	-0.0773	0.0100	0.1000
8.000	1.1605	0.01625	0.00000	-0.0771	0.0100	0.1000
8.250	1.1837	0.01654	0.00000	-0.0766	0.0100	0.1000
8.500	1.2069	0.01684	0.00000	-0.0759	0.0100	0.1000
8.750	1.2253	0.01718	0.00000	-0.0748	0.0100	0.1000
9.000	1.2420	0.01754	0.00000	-0.0732	0.0100	0.1000
9.250	1.2519	0.01795	0.00000	-0.0707	0.0100	0.1000
9.500	1.2617	0.01845	0.00000	-0.0686	0.0100	0.1000
9.750	1.2709	0.01908	0.00000	-0.0669	0.0100	0.1000
10.000	1.2767	0.01987	0.00000	-0.0647	0.0100	0.1000
10.250	1.2835	0.02087	0.00000	-0.0635	0.0100	0.1000
10.500	1.2842	0.02213	0.00000	-0.0619	0.0100	0.1000
10.750	1.2884	0.02363	0.00000	-0.0622	0.0100	0.1000
11.000	1.2809	0.02551	0.00000	-0.0593	0.0100	0.1000
11.250	1.2784	0.02760	0.00000	-0.0586	0.0100	0.1000
11.500	1.2731	0.03011	0.00000	-0.0586	0.0100	0.1000
11.750	1.2626	0.03290	0.00000	-0.0580	0.0100	0.1000
12.000	1.2555	0.03588	0.00000	-0.0586	0.0100	0.1000
12.250	1.2562	0.03908	0.00000	-0.0639	0.0100	0.1000
12.417	1.2329	0.04132	0.00000	-0.0565	0.0100	0.1000
12.583	1.2300	0.04356	0.00000	-0.0583	0.0100	0.1000
12.750	1.2223	0.04602	0.00000	-0.0583	0.0100	0.1000
12.917	1.2109	0.04848	0.00000	-0.0558	0.0100	0.1000
13.083	1.2088	0.05085	0.00000	-0.0598	0.0100	0.1000
13.250	1.2013	0.05319	0.00000	-0.0592	0.0100	0.1000

ISES polar driver Version 1.3

Calculated polar for: AIRFOIL FFA-W1-271

Vortex + doublet far field
Entropy conserved

Mach = 0.150 Re = 3.000 e 6 Acrit = 9.000

alpha	CL	CD	CDi	CM	S xtr	P xtr
0.000	0.2567	0.00791	0.00000	-0.0788	0.3874	0.4185
0.500	0.3244	0.00782	0.00000	-0.0815	0.3772	0.4225
1.000	0.3908	0.00776	0.00000	-0.0837	0.3675	0.4272
1.500	0.4556	0.00784	0.00000	-0.0856	0.3574	0.4316
2.000	0.5203	0.00789	0.00000	-0.0872	0.3495	0.4355
2.500	0.5831	0.00812	0.00000	-0.0885	0.3384	0.4384
3.000	0.6458	0.00828	0.00000	-0.0894	0.3312	0.4406
3.500	0.7093	0.00848	0.00000	-0.0910	0.3204	0.4440
4.000	0.7742	0.00846	0.00000	-0.0928	0.3132	0.4489
4.500	0.8368	0.00868	0.00000	-0.0941	0.3025	0.4536
5.000	0.8999	0.00878	0.00000	-0.0954	0.2953	0.4579
5.500	0.9602	0.00910	0.00000	-0.0962	0.2848	0.4614
6.000	1.0214	0.00931	0.00000	-0.0970	0.2775	0.4637
6.500	1.0813	0.00962	0.00000	-0.0979	0.2668	0.4673
7.000	1.1427	0.00979	0.00000	-0.0989	0.2578	0.4724
7.500	1.2011	0.01016	0.00000	-0.0996	0.2468	0.4779
7.750	1.2306	0.01028	0.00000	-0.0998	0.2419	0.4800
8.000	1.2597	0.01044	0.00000	-0.1000	0.2369	0.4822
8.250	1.2876	0.01070	0.00000	-0.1001	0.2302	0.4841
8.500	1.3143	0.01095	0.00000	-0.0999	0.2258	0.4854
8.750	1.3427	0.01113	0.00000	-0.1000	0.2196	0.4868
9.000	1.3700	0.01139	0.00000	-0.0999	0.2141	0.4882
9.250	1.3951	0.01169	0.00000	-0.0996	0.2091	0.4898
9.500	1.4231	0.01184	0.00000	-0.0996	0.2046	0.4924
9.750	1.4495	0.01206	0.00000	-0.0995	0.1990	0.4950
10.000	1.4739	0.01236	0.00000	-0.0992	0.1933	0.4974
10.250	1.4984	0.01267	0.00000	-0.0987	0.1883	0.4999
10.500	1.5224	0.01292	0.00000	-0.0981	0.1828	0.5022
10.750	1.5431	0.01330	0.00000	-0.0971	0.1769	0.5044
11.000	1.5602	0.01377	0.00000	-0.0955	0.1715	0.5064
11.250	1.5797	0.01409	0.00000	-0.0943	0.1663	0.5083
11.500	1.5912	0.01457	0.00000	-0.0917	0.1610	0.5095
11.750	1.5933	0.01515	0.00000	-0.0876	0.1564	0.5105
12.000	1.5989	0.01570	0.00000	-0.0846	0.1521	0.5116
12.250	1.6033	0.01640	0.00000	-0.0815	0.1473	0.5137
12.500	1.6036	0.01735	0.00000	-0.0785	0.1428	0.5158
12.750	1.5991	0.01859	0.00000	-0.0760	0.1388	0.5178
13.000	1.6002	0.01980	0.00000	-0.0734	0.1348	0.5198
13.250	1.5965	0.02143	0.00000	-0.0711	0.1307	0.5217
13.500	1.5860	0.02345	0.00000	-0.0696	0.1272	0.5232
13.750	1.5735	0.02595	0.00000	-0.0678	0.1239	0.5247
14.000	1.5699	0.02818	0.00000	-0.0678	0.1208	0.5262
14.250	1.5530	0.03085	0.00000	-0.0643	0.1175	0.5273
14.417	1.5435	0.03294	0.00000	-0.0636	0.1155	0.5278
14.450	1.5582	0.03284	0.00000	-0.0705	0.1170	0.5280
14.464	1.5529	0.03300	0.00000	-0.0668	0.1169	0.5280
14.479	1.5484	0.03316	0.00000	-0.0640	0.1165	0.5281
14.489	1.5511	0.03327	0.00000	-0.0661	0.1165	0.5281
14.494	1.5497	0.03334	0.00000	-0.0655	0.1164	0.5282

ISES polar driver Version 1.3

Calculated polar for: AIRFOIL FFA-W1-271

Vortex + doublet far field
Entropy conserved

Mach = 0.150 Re = 3.000 e 6 Acrit = 9.000

alpha	CL	CD	CDi	CM	S xtr	P xtr
0.000	0.1546	0.01645	0.00000	-0.0465	0.0100	0.1000
0.500	0.2280	0.01580	0.00000	-0.0525	0.0100	0.1000
1.000	0.3003	0.01531	0.00000	-0.0577	0.0100	0.1000
1.500	0.3718	0.01492	0.00000	-0.0625	0.0100	0.1000
2.000	0.4422	0.01462	0.00000	-0.0669	0.0100	0.1000
2.500	0.5114	0.01441	0.00000	-0.0707	0.0100	0.1000
3.000	0.5787	0.01431	0.00000	-0.0741	0.0100	0.1000
3.500	0.6432	0.01437	0.00000	-0.0765	0.0100	0.1000
4.000	0.7055	0.01451	0.00000	-0.0783	0.0100	0.1000
4.500	0.7664	0.01471	0.00000	-0.0799	0.0100	0.1000
5.000	0.8265	0.01494	0.00000	-0.0812	0.0100	0.1000
5.500	0.8852	0.01523	0.00000	-0.0823	0.0100	0.1000
6.000	0.9417	0.01555	0.00000	-0.0829	0.0100	0.1000
6.500	0.9963	0.01592	0.00000	-0.0834	0.0100	0.1000
7.000	1.0491	0.01636	0.00000	-0.0833	0.0100	0.1000
7.500	1.0979	0.01685	0.00000	-0.0829	0.0100	0.1000
7.750	1.1202	0.01714	0.00000	-0.0822	0.0100	0.1000
8.000	1.1405	0.01746	0.00000	-0.0814	0.0100	0.1000
8.250	1.1581	0.01780	0.00000	-0.0798	0.0100	0.1000
8.500	1.1687	0.01818	0.00000	-0.0774	0.0100	0.1000
8.750	1.1775	0.01864	0.00000	-0.0747	0.0100	0.1000
9.000	1.1842	0.01920	0.00000	-0.0721	0.0100	0.1000
9.250	1.1930	0.01990	0.00000	-0.0703	0.0100	0.1000
9.500	1.1992	0.02076	0.00000	-0.0684	0.0100	0.1000
9.750	1.2031	0.02179	0.00000	-0.0671	0.0100	0.1000
10.000	1.2059	0.02305	0.00000	-0.0654	0.0100	0.1000
10.250	1.2078	0.02453	0.00000	-0.0640	0.0100	0.1000
10.500	1.2050	0.02622	0.00000	-0.0626	0.0100	0.1000
10.750	1.2025	0.02823	0.00000	-0.0616	0.0100	0.1000
11.000	1.2019	0.03028	0.00000	-0.0614	0.0100	0.1000
11.250	1.1912	0.03275	0.00000	-0.0595	0.0100	0.1000
11.500	1.1910	0.03534	0.00000	-0.0622	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W1-128

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
0.000	0.3348	0.00701	1.00000	-0.0522	0.5354	0.5314
0.500	0.3902	0.00635	1.00000	-0.0523	0.5257	0.7880
1.000	0.4417	0.00607	1.00000	-0.0507	0.5215	0.9093
1.500	0.4931	0.00603	1.00000	-0.0491	0.5158	0.9643
2.500	0.6299	0.00627	1.00000	-0.0543	0.5000	1.0000
3.000	0.6880	0.00639	1.00000	-0.0548	0.4945	1.0000
3.500	0.7465	0.00656	1.00000	-0.0554	0.4859	1.0000
4.000	0.8049	0.00676	1.00000	-0.0560	0.4781	1.0000
4.500	0.8636	0.00692	1.00000	-0.0567	0.4722	1.0000
5.000	0.9221	0.00713	1.00000	-0.0573	0.4627	1.0000
5.500	0.9804	0.00734	1.00000	-0.0580	0.4502	1.0000
6.000	1.0384	0.00764	1.00000	-0.0586	0.4309	1.0000
6.500	1.0959	0.00799	1.00000	-0.0592	0.4111	1.0000
7.000	1.1530	0.00838	1.00000	-0.0598	0.3926	1.0000
7.500	1.2074	0.00914	1.00000	-0.0601	0.3527	1.0000
8.000	1.2571	0.01042	1.00000	-0.0602	0.2896	1.0000
8.500	1.2966	0.01269	1.00000	-0.0596	0.1957	1.0000
9.000	1.3256	0.01557	1.00000	-0.0582	0.1087	1.0000
9.500	1.3522	0.01796	1.00000	-0.0566	0.0727	1.0000
10.000	1.3717	0.02085	1.00000	-0.0551	0.0544	1.0000
10.500	1.3901	0.02423	1.00000	-0.0542	0.0440	1.0000
11.000	1.4038	0.02831	1.00000	-0.0536	0.0347	1.0000
11.500	1.4189	0.03251	1.00000	-0.0531	0.0295	1.0000
12.000	1.4310	0.03719	1.00000	-0.0527	0.0236	1.0000
12.500	1.4437	0.04194	1.00000	-0.0526	0.0215	1.0000
13.000	1.4538	0.04711	1.00000	-0.0526	0.0197	1.0000
13.500	1.4612	0.05280	1.00000	-0.0528	0.0169	1.0000
14.000	1.4605	0.05986	1.00000	-0.0531	0.0131	1.0000
14.500	1.4636	0.06656	1.00000	-0.0538	0.0117	1.0000
15.000	1.4746	0.07242	1.00000	-0.0547	0.0112	1.0000
15.500	1.4813	0.07907	1.00000	-0.0559	0.0104	1.0000
16.000	1.4842	0.08651	1.00000	-0.0573	0.0093	1.0000
16.500	1.4780	0.09575	1.00000	-0.0592	0.0078	1.0000
17.000	1.4215	0.11503	1.00000	-0.0642	0.0060	1.0000
-0.500	0.2776	0.00775	1.00000	-0.0516	0.5422	0.2846
-1.000	0.2193	0.00818	1.00000	-0.0510	0.5468	0.1677
-1.500	0.1606	0.00844	1.00000	-0.0504	0.5555	0.1004
-2.000	0.1016	0.00867	1.00000	-0.0500	0.5628	0.0647
-2.500	0.0426	0.00890	1.00000	-0.0496	0.5687	0.0520
-3.000	-0.0163	0.00931	1.00000	-0.0494	0.5770	0.0382
-3.500	-0.0752	0.00948	1.00000	-0.0490	0.5851	0.0348
-4.000	-0.1322	0.01095	1.00000	-0.0494	0.5907	0.0227
-4.500	-0.1909	0.01097	1.00000	-0.0490	0.6007	0.0217
-5.000	-0.2491	0.01127	1.00000	-0.0488	0.6088	0.0197
-5.500	-0.3057	0.01230	1.00000	-0.0490	0.6147	0.0158

XFOIL Version 5.0

Calculated polar for: FFA-W1-128

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
0.000	0.3285	0.01153	1.00000	-0.0525	0.0100	0.1000
0.500	0.3865	0.01169	1.00000	-0.0530	0.0100	0.1000
1.000	0.4443	0.01188	1.00000	-0.0535	0.0100	0.1000
1.500	0.5019	0.01209	1.00000	-0.0539	0.0100	0.1000
2.000	0.5592	0.01233	1.00000	-0.0544	0.0100	0.1000
2.500	0.6162	0.01260	1.00000	-0.0549	0.0100	0.1000
3.000	0.6728	0.01289	1.00000	-0.0553	0.0100	0.1000
3.500	0.7291	0.01323	1.00000	-0.0557	0.0100	0.1000
4.000	0.7848	0.01359	1.00000	-0.0561	0.0100	0.1000
4.500	0.8400	0.01400	1.00000	-0.0565	0.0100	0.1000
5.000	0.8945	0.01445	1.00000	-0.0567	0.0100	0.1000
5.500	0.9482	0.01495	1.00000	-0.0570	0.0100	0.1000
6.000	1.0009	0.01552	1.00000	-0.0571	0.0100	0.1000
6.500	1.0524	0.01615	1.00000	-0.0572	0.0100	0.1000
7.000	1.1024	0.01686	1.00000	-0.0571	0.0100	0.1000
7.500	1.1504	0.01768	1.00000	-0.0568	0.0100	0.1000
8.000	1.1957	0.01864	1.00000	-0.0564	0.0100	0.1000
8.500	1.2292	0.02002	1.00000	-0.0550	0.0100	0.1000
9.000	1.2608	0.02194	1.00000	-0.0542	0.0100	0.1000
9.500	1.2893	0.02428	1.00000	-0.0535	0.0100	0.1000
10.000	1.3152	0.02702	1.00000	-0.0529	0.0100	0.1000
10.500	1.3385	0.03014	1.00000	-0.0525	0.0100	0.1000
11.000	1.3600	0.03361	1.00000	-0.0521	0.0100	0.1000
11.500	1.3790	0.03744	1.00000	-0.0519	0.0100	0.1000
12.000	1.3969	0.04156	1.00000	-0.0517	0.0100	0.1000
12.500	1.4127	0.04599	1.00000	-0.0517	0.0100	0.1000
13.000	1.4271	0.05071	1.00000	-0.0518	0.0100	0.1000
13.500	1.4408	0.05568	1.00000	-0.0520	0.0100	0.1000
14.000	1.4531	0.06099	1.00000	-0.0524	0.0100	0.1000
14.500	1.4632	0.06674	1.00000	-0.0530	0.0100	0.1000
15.000	1.4719	0.07292	1.00000	-0.0538	0.0100	0.1000
15.500	1.4770	0.07984	1.00000	-0.0548	0.0100	0.1000
16.000	1.4776	0.08760	1.00000	-0.0562	0.0094	0.1000
16.500	1.4729	0.09663	1.00000	-0.0581	0.0080	0.1000
17.000	1.4147	0.11626	1.00000	-0.0632	0.0060	0.1000
-0.500	0.2703	0.01140	1.00000	-0.0521	0.0100	0.1000
-1.000	0.2120	0.01128	1.00000	-0.0517	0.0100	0.1000
-1.500	0.1536	0.01121	1.00000	-0.0513	0.0100	0.0979
-2.000	0.0955	0.01131	1.00000	-0.0509	0.0100	0.0626
-2.500	0.0373	0.01142	1.00000	-0.0506	0.0100	0.0512
-3.500	-0.0790	0.01182	1.00000	-0.0501	0.0100	0.0345

XFOIL Version 5.0

Calculated polar for: FFA-W1-128

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 2.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
2.000	0.5658	0.00501	2.00000	-0.0529	0.4836	0.9200
2.500	0.6207	0.00503	2.00000	-0.0523	0.4775	0.9550
3.500	0.7481	0.00523	2.00000	-0.0555	0.4610	1.0000
4.000	0.8073	0.00537	2.00000	-0.0562	0.4543	1.0000
4.500	0.8665	0.00554	2.00000	-0.0569	0.4468	1.0000
5.000	0.9255	0.00575	2.00000	-0.0577	0.4318	1.0000
5.500	0.9842	0.00601	2.00000	-0.0584	0.4131	1.0000
6.000	1.0424	0.00633	2.00000	-0.0591	0.3958	1.0000
6.500	1.0997	0.00680	2.00000	-0.0598	0.3650	1.0000
7.000	1.1541	0.00771	2.00000	-0.0603	0.3110	1.0000
7.500	1.2054	0.00898	2.00000	-0.0606	0.2422	1.0000
8.000	1.2523	0.01066	2.00000	-0.0605	0.1612	1.0000
8.500	1.2942	0.01262	2.00000	-0.0601	0.0927	1.0000
9.000	1.3393	0.01396	2.00000	-0.0597	0.0595	1.0000
9.500	1.3813	0.01539	2.00000	-0.0590	0.0406	1.0000
10.000	1.4114	0.01731	2.00000	-0.0575	0.0328	1.0000
10.500	1.4387	0.01952	2.00000	-0.0564	0.0262	1.0000
11.000	1.4623	0.02233	2.00000	-0.0555	0.0220	1.0000
11.500	1.4793	0.02597	2.00000	-0.0548	0.0201	1.0000
12.000	1.4946	0.03002	2.00000	-0.0542	0.0172	1.0000
12.500	1.5098	0.03424	2.00000	-0.0537	0.0139	1.0000
13.000	1.5212	0.03900	2.00000	-0.0534	0.0119	1.0000
13.500	1.5329	0.04392	2.00000	-0.0534	0.0113	1.0000
14.000	1.5393	0.04957	2.00000	-0.0534	0.0101	1.0000
14.500	1.5467	0.05534	2.00000	-0.0537	0.0084	1.0000
15.000	1.5490	0.06202	2.00000	-0.0541	0.0065	1.0000
15.500	1.5521	0.06872	2.00000	-0.0548	0.0059	1.0000
16.000	1.5596	0.07513	2.00000	-0.0558	0.0058	1.0000
16.500	1.5642	0.08215	2.00000	-0.0571	0.0055	1.0000
17.000	1.5670	0.08954	2.00000	-0.0585	0.0052	1.0000
1.500	0.5097	0.00506	2.00000	-0.0531	0.4924	0.8617
1.000	0.4540	0.00561	2.00000	-0.0532	0.4986	0.6316
0.500	0.3956	0.00608	2.00000	-0.0525	0.5023	0.4450
0.000	0.3370	0.00653	2.00000	-0.0518	0.5120	0.2721
-0.500	0.2779	0.00680	2.00000	-0.0511	0.5188	0.1763
-1.000	0.2188	0.00708	2.00000	-0.0505	0.5233	0.1027
-1.500	0.1595	0.00727	2.00000	-0.0499	0.5303	0.0689
-2.000	0.1000	0.00742	2.00000	-0.0494	0.5389	0.0524
-2.500	0.0404	0.00761	2.00000	-0.0490	0.5443	0.0390
-3.000	-0.0189	0.00775	2.00000	-0.0486	0.5481	0.0363
-3.500	-0.0782	0.00803	2.00000	-0.0483	0.5597	0.0270
-4.000	-0.1370	0.00852	2.00000	-0.0481	0.5660	0.0226
-4.500	-0.1963	0.00853	2.00000	-0.0477	0.5703	0.0216
-5.000	-0.2552	0.00883	2.00000	-0.0475	0.5822	0.0183
-6.000	-0.3676	0.01157	2.00000	-0.0482	0.5933	0.0122
-6.500	-0.4261	0.01170	2.00000	-0.0479	0.6050	0.0119
-7.000	-0.4839	0.01208	2.00000	-0.0479	0.6121	0.0112
-7.500	-0.5403	0.01287	2.00000	-0.0481	0.6215	0.0102

XFOIL Version 5.0

Calculated polar for: FFA-W1-128

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-2.000	0.0985	0.00689	3.00000	-0.0490	0.5229	0.0463
-1.500	0.1582	0.00675	3.00000	-0.0496	0.5150	0.0604
-1.000	0.2179	0.00659	3.00000	-0.0502	0.5110	0.0869
-0.500	0.2774	0.00638	3.00000	-0.0508	0.5032	0.1425
0.000	0.3370	0.00617	3.00000	-0.0515	0.4960	0.2120
0.500	0.3963	0.00587	3.00000	-0.0523	0.4913	0.3301
1.000	0.4557	0.00560	3.00000	-0.0531	0.4841	0.4494
1.500	0.5140	0.00497	3.00000	-0.0539	0.4766	0.7159
2.000	0.5723	0.00473	3.00000	-0.0543	0.4713	0.8466
2.500	0.6301	0.00469	3.00000	-0.0546	0.4613	0.9024
3.000	0.6859	0.00468	3.00000	-0.0543	0.4559	0.9463
3.500	0.7404	0.00473	3.00000	-0.0537	0.4481	0.9799
4.000	0.8081	0.00485	3.00000	-0.0563	0.4388	1.0000
4.500	0.8676	0.00504	3.00000	-0.0571	0.4261	1.0000
5.000	0.9267	0.00528	3.00000	-0.0579	0.4100	1.0000
5.500	0.9855	0.00559	3.00000	-0.0587	0.3859	1.0000
6.000	1.0435	0.00601	3.00000	-0.0594	0.3603	1.0000
6.500	1.0999	0.00671	3.00000	-0.0601	0.3090	1.0000
7.000	1.1545	0.00765	3.00000	-0.0607	0.2527	1.0000
7.500	1.2036	0.00929	3.00000	-0.0609	0.1657	1.0000
8.000	1.2531	0.01062	3.00000	-0.0610	0.1066	1.0000
8.500	1.3003	0.01200	3.00000	-0.0608	0.0670	1.0000
9.000	1.3482	0.01312	3.00000	-0.0607	0.0474	1.0000
9.500	1.3946	0.01425	3.00000	-0.0603	0.0344	1.0000
10.000	1.4395	0.01537	3.00000	-0.0598	0.0245	1.0000
10.500	1.4693	0.01722	3.00000	-0.0584	0.0218	1.0000
11.000	1.4955	0.01956	3.00000	-0.0573	0.0179	1.0000
12.000	1.5342	0.02627	3.00000	-0.0558	0.0133	1.0000
12.500	1.5496	0.03035	3.00000	-0.0553	0.0112	1.0000

XFOIL Version 5.0

Calculated polar for: FFA-W1-128

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-3.00000	-0.02300	0.00942	3.00000	-0.04940	0.01000	0.03250
-2.50000	0.03600	0.00934	3.00000	-0.04980	0.01000	0.03810
-2.00000	0.09490	0.00932	3.00000	-0.05020	0.01000	0.04450
-1.50000	0.15390	0.00930	3.00000	-0.05070	0.01000	0.05750
-1.00000	0.21260	0.00927	3.00000	-0.05120	0.01000	0.08050
-0.50000	0.27150	0.00932	3.00000	-0.05170	0.01000	0.10000
0.00000	0.33040	0.00944	3.00000	-0.05220	0.01000	0.10000
0.50000	0.38910	0.00958	3.00000	-0.05270	0.01000	0.10000
1.00000	0.44770	0.00973	3.00000	-0.05330	0.01000	0.10000
1.50000	0.50610	0.00991	3.00000	-0.05390	0.01000	0.10000
2.00000	0.56440	0.01010	3.00000	-0.05440	0.01000	0.10000
2.500	0.6234	0.01032	3.00000	-0.0552	0.0100	0.1000
3.000	0.6813	0.01055	3.00000	-0.0558	0.0100	0.1000
3.500	0.7389	0.01080	3.00000	-0.0563	0.0100	0.1000
4.000	0.7961	0.01108	3.00000	-0.0569	0.0100	0.1000
4.500	0.8531	0.01139	3.00000	-0.0574	0.0100	0.1000
5.000	0.9096	0.01172	3.00000	-0.0579	0.0100	0.1000
5.500	0.9656	0.01209	3.00000	-0.0583	0.0100	0.1000
6.000	1.0211	0.01249	3.00000	-0.0587	0.0100	0.1000
6.500	1.0759	0.01293	3.00000	-0.0590	0.0100	0.1000
7.000	1.1299	0.01341	3.00000	-0.0593	0.0100	0.1000
7.500	1.1830	0.01394	3.00000	-0.0595	0.0100	0.1000
8.000	1.2350	0.01453	3.00000	-0.0595	0.0100	0.1000
8.500	1.2856	0.01519	3.00000	-0.0594	0.0100	0.1000
9.000	1.3343	0.01594	3.00000	-0.0592	0.0100	0.1000
9.500	1.3804	0.01681	3.00000	-0.0587	0.0100	0.1000
10.000	1.4136	0.01810	3.00000	-0.0571	0.0100	0.1000
10.500	1.4445	0.01998	3.00000	-0.0561	0.0100	0.1000
11.000	1.4720	0.02232	3.00000	-0.0553	0.0100	0.1000
11.500	1.4965	0.02512	3.00000	-0.0546	0.0100	0.1000
12.000	1.5183	0.02834	3.00000	-0.0541	0.0100	0.1000
12.500	1.5381	0.03191	3.00000	-0.0536	0.0100	0.1000
13.000	1.5541	0.03605	3.00000	-0.0532	0.0092	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W1-152

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-10.00000	-0.43310	0.10078	1.00000	-0.00800	0.65280	0.00820
-9.50000	-0.44160	0.08656	1.00000	-0.01700	0.64680	0.00850
-9.00000	-0.46210	0.06550	1.00000	-0.03430	0.64250	0.00880
-8.50000	-0.50340	0.04233	1.00000	-0.04990	0.63730	0.00920
-8.00000	-0.50940	0.02928	1.00000	-0.05240	0.62970	0.00990
-7.50000	-0.47950	0.02288	1.00000	-0.05210	0.62250	0.01070
-5.00000	-0.22600	0.01138	1.00000	-0.05020	0.58510	0.01920
-4.50000	-0.16760	0.01086	1.00000	-0.05050	0.57600	0.02200
-4.00000	-0.10860	0.01055	1.00000	-0.05100	0.57080	0.02380
-3.50000	-0.04980	0.00991	1.00000	-0.05130	0.56490	0.03120
-3.00000	0.00970	0.00965	1.00000	-0.05180	0.55640	0.03840
-2.50000	0.06940	0.00937	1.00000	-0.05240	0.55200	0.05480
-1.50000	0.18860	0.00889	1.00000	-0.05380	0.53820	0.12900
-1.00000	0.24730	0.00830	1.00000	-0.05470	0.53410	0.27380
-0.50000	0.30450	0.00711	1.00000	-0.05580	0.52870	0.61570
0.00000	0.36350	0.00688	1.00000	-0.05640	0.52100	0.73100
0.50000	0.42220	0.00682	1.00000	-0.05680	0.51680	0.79060
1.000	0.4811	0.00683	1.00000	-0.0572	0.5119	0.8241
1.500	0.5387	0.00689	1.00000	-0.0574	0.5054	0.8569
2.000	0.5956	0.00698	1.00000	-0.0574	0.4991	0.8821
2.500	0.6536	0.00705	1.00000	-0.0577	0.4948	0.8983
3.000	0.7109	0.00716	1.00000	-0.0579	0.4888	0.9143
3.500	0.7669	0.00730	1.00000	-0.0579	0.4812	0.9310
4.000	0.8221	0.00738	1.00000	-0.0577	0.4767	0.9489
4.500	0.8763	0.00749	1.00000	-0.0573	0.4696	0.9713
5.500	1.0075	0.00787	1.00000	-0.0620	0.4542	1.0000
6.000	1.0670	0.00814	1.00000	-0.0632	0.4434	1.0000
6.500	1.1260	0.00840	1.00000	-0.0642	0.4334	1.0000
7.000	1.1839	0.00875	1.00000	-0.0651	0.4181	1.0000
7.500	1.2408	0.00917	1.00000	-0.0659	0.4024	1.0000
8.000	1.2963	0.00970	1.00000	-0.0664	0.3828	1.0000
8.500	1.3493	0.01040	1.00000	-0.0668	0.3572	1.0000
9.000	1.3999	0.01125	1.00000	-0.0668	0.3304	1.0000
9.500	1.4443	0.01250	1.00000	-0.0662	0.2932	1.0000
10.000	1.4648	0.01513	1.00000	-0.0634	0.2304	1.0000
10.500	1.4635	0.01917	1.00000	-0.0600	0.1825	1.0000
11.500	1.4459	0.03095	1.00000	-0.0564	0.1135	1.0000
12.000	1.4419	0.03701	1.00000	-0.0554	0.0920	1.0000
12.500	1.4370	0.04341	1.00000	-0.0547	0.0739	1.0000
13.000	1.4408	0.04909	1.00000	-0.0543	0.0603	1.0000
13.500	1.4415	0.05524	1.00000	-0.0541	0.0542	1.0000
14.000	1.4490	0.06083	1.00000	-0.0541	0.0439	1.0000
14.500	1.4515	0.06720	1.00000	-0.0544	0.0404	1.0000
15.000	1.4554	0.07358	1.00000	-0.0548	0.0350	1.0000

XFOIL Version 5.0

Calculated polar for: FFA-W1-152

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
1.000	0.4690	0.01275	1.00000	-0.0569	0.0100	0.1000
1.500	0.5263	0.01299	1.00000	-0.0574	0.0100	0.1000
2.000	0.5831	0.01326	1.00000	-0.0580	0.0100	0.1000
2.500	0.6394	0.01357	1.00000	-0.0584	0.0100	0.1000
3.000	0.6951	0.01391	1.00000	-0.0589	0.0100	0.1000
3.500	0.7501	0.01429	1.00000	-0.0593	0.0100	0.1000
4.000	0.8043	0.01472	1.00000	-0.0596	0.0100	0.1000
4.500	0.8575	0.01520	1.00000	-0.0598	0.0100	0.1000
5.000	0.9095	0.01573	1.00000	-0.0599	0.0100	0.1000
5.500	0.9599	0.01634	1.00000	-0.0598	0.0100	0.1000
6.000	1.0084	0.01704	1.00000	-0.0596	0.0100	0.1000
6.500	1.0540	0.01786	1.00000	-0.0591	0.0100	0.1000
7.000	1.0866	0.01908	1.00000	-0.0573	0.0100	0.1000
7.500	1.1209	0.02073	1.00000	-0.0565	0.0100	0.1000
8.000	1.1519	0.02276	1.00000	-0.0558	0.0100	0.1000
8.500	1.1802	0.02516	1.00000	-0.0552	0.0100	0.1000
9.000	1.2060	0.02790	1.00000	-0.0546	0.0100	0.1000
9.500	1.2301	0.03094	1.00000	-0.0541	0.0100	0.1000
10.000	1.2517	0.03431	1.00000	-0.0537	0.0100	0.1000
10.500	1.2718	0.03793	1.00000	-0.0533	0.0100	0.1000
11.000	1.2906	0.04176	1.00000	-0.0529	0.0100	0.1000
11.500	1.3081	0.04582	1.00000	-0.0527	0.0100	0.1000
12.000	1.3248	0.05010	1.00000	-0.0525	0.0100	0.1000
12.500	1.3402	0.05460	1.00000	-0.0525	0.0100	0.1000
13.000	1.3549	0.05931	1.00000	-0.0525	0.0100	0.1000
13.500	1.3686	0.06425	1.00000	-0.0527	0.0100	0.1000
14.000	1.3814	0.06935	1.00000	-0.0530	0.0100	0.1000
14.500	1.3936	0.07465	1.00000	-0.0534	0.0100	0.1000
15.000	1.4048	0.08034	1.00000	-0.0541	0.0100	0.1000
15.500	1.4154	0.08620	1.00000	-0.0549	0.0100	0.1000
16.000	1.4257	0.09213	1.00000	-0.0557	0.0100	0.1000
16.500	1.4344	0.09862	1.00000	-0.0569	0.0100	0.1000
17.000	1.4427	0.10533	1.00000	-0.0582	0.0100	0.1000
17.500	1.4504	0.11222	1.00000	-0.0597	0.0100	0.1000
18.000	1.4557	0.11975	1.00000	-0.0615	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W1-152

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 2.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-9.50000	-0.43310	0.09005	2.00000	-0.01360	0.61150	0.00760
-9.00000	-0.44610	0.07370	2.00000	-0.02560	0.60800	0.00770
-8.50000	-0.47530	0.04863	2.00000	-0.04660	0.60450	0.00780
-8.00000	-0.51100	0.02923	2.00000	-0.05180	0.59890	0.00800
-7.50000	-0.51600	0.01378	2.00000	-0.04910	0.59320	0.00960
-7.00000	-0.46310	0.01201	2.00000	-0.04860	0.58770	0.01120
-6.50000	-0.40530	0.01146	2.00000	-0.04880	0.57720	0.01220
-6.00000	-0.34680	0.01104	2.00000	-0.04910	0.57220	0.01300
-5.50000	-0.28760	0.01080	2.00000	-0.04950	0.56610	0.01340
-4.50000	-0.17180	0.00898	2.00000	-0.04970	0.55280	0.01850
-4.00000	-0.11200	0.00864	2.00000	-0.05010	0.54710	0.02250
-3.00000	0.00830	0.00820	2.00000	-0.05130	0.53480	0.03140
-2.50000	0.06840	0.00801	2.00000	-0.05200	0.52910	0.03910
-2.00000	0.12870	0.00784	2.00000	-0.05280	0.52130	0.05590
-1.50000	0.18890	0.00764	2.00000	-0.05350	0.51710	0.08820
-1.00000	0.24900	0.00738	2.00000	-0.05440	0.51290	0.14400
-0.50000	0.30880	0.00695	2.00000	-0.05540	0.50660	0.26330
0.00000	0.36770	0.00611	2.00000	-0.05660	0.50020	0.53540
0.50000	0.42780	0.00579	2.00000	-0.05760	0.49670	0.65880
1.000	0.4881	0.00564	2.00000	-0.0585	0.4916	0.7465
1.500	0.5482	0.00566	2.00000	-0.0593	0.4829	0.7878
2.000	0.6078	0.00565	2.00000	-0.0600	0.4797	0.8275
2.500	0.6675	0.00571	2.00000	-0.0607	0.4743	0.8498
3.000	0.7272	0.00581	2.00000	-0.0615	0.4665	0.8660
3.500	0.7868	0.00590	2.00000	-0.0622	0.4611	0.8813
4.000	0.8460	0.00601	2.00000	-0.0629	0.4548	0.8957
4.500	0.9048	0.00616	2.00000	-0.0636	0.4449	0.9078
5.000	0.9630	0.00628	2.00000	-0.0642	0.4392	0.9237
5.500	1.0202	0.00647	2.00000	-0.0646	0.4253	0.9402
6.000	1.0756	0.00665	2.00000	-0.0646	0.4160	0.9598
7.000	1.1937	0.00731	2.00000	-0.0666	0.3794	1.0000
7.500	1.2508	0.00781	2.00000	-0.0675	0.3548	1.0000
8.000	1.3054	0.00851	2.00000	-0.0680	0.3265	1.0000
8.500	1.3580	0.00932	2.00000	-0.0684	0.2930	1.0000
9.000	1.4042	0.01064	2.00000	-0.0681	0.2469	1.0000
9.500	1.4366	0.01289	2.00000	-0.0664	0.1778	1.0000
10.000	1.4588	0.01519	2.00000	-0.0637	0.1373	1.0000
10.500	1.4576	0.01942	2.00000	-0.0608	0.0942	1.0000
11.000	1.4652	0.02366	2.00000	-0.0593	0.0756	1.0000
11.500	1.4730	0.02821	2.00000	-0.0583	0.0631	1.0000
12.000	1.4784	0.03321	2.00000	-0.0573	0.0547	1.0000
12.500	1.4907	0.03766	2.00000	-0.0567	0.0441	1.0000
13.000	1.4934	0.04326	2.00000	-0.0561	0.0403	1.0000
13.500	1.5011	0.04851	2.00000	-0.0558	0.0345	1.0000
14.000	1.5102	0.05369	2.00000	-0.0555	0.0287	1.0000
14.500	1.5136	0.05967	2.00000	-0.0555	0.0268	1.0000
15.000	1.5179	0.06579	2.00000	-0.0557	0.0234	1.0000

XFOIL Version 5.0

Calculated polar for: FFA-W1-152

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-9.50000	-0.42850	0.09193	3.00000	-0.01200	0.59350	0.00750
-9.25000	-0.43410	0.08434	3.00000	-0.01700	0.59220	0.00750
-9.00000	-0.44010	0.07628	3.00000	-0.02300	0.59070	0.00750
-8.75000	-0.44570	0.06450	3.00000	-0.03420	0.58930	0.00750
-8.50000	-0.45680	0.05220	3.00000	-0.04480	0.58770	0.00760
-8.25000	-0.47180	0.04407	3.00000	-0.04840	0.58540	0.00760
-8.00000	-0.48240	0.03597	3.00000	-0.05080	0.58250	0.00760
-7.75000	-0.48870	0.02826	3.00000	-0.05160	0.57890	0.00760
-7.50000	-0.48980	0.02091	3.00000	-0.05100	0.57570	0.00770
-7.25000	-0.49000	0.01307	3.00000	-0.04890	0.57460	0.00820
-7.00000	-0.47020	0.01024	3.00000	-0.04760	0.57190	0.01060
-6.50000	-0.41140	0.00978	3.00000	-0.04790	0.56500	0.01230
-6.00000	-0.35200	0.00949	3.00000	-0.04820	0.55670	0.01320
-5.50000	-0.29220	0.00933	3.00000	-0.04870	0.55200	0.01350
-4.50000	-0.17310	0.00863	3.00000	-0.04960	0.53810	0.01490
-4.00000	-0.11350	0.00794	3.00000	-0.04980	0.53390	0.02030
-3.500	-0.0525	0.00778	3.00000	-0.0509	0.5237	0.0231
-2.500	0.0683	0.00737	3.00000	-0.0520	0.5144	0.0356
-2.000	0.1285	0.00719	3.00000	-0.0527	0.5085	0.0453
-1.500	0.1890	0.00707	3.00000	-0.0535	0.5023	0.0667
-1.000	0.2492	0.00690	3.00000	-0.0543	0.4983	0.0943
-0.500	0.3095	0.00671	3.00000	-0.0552	0.4922	0.1577
0.000	0.3686	0.00598	3.00000	-0.0563	0.4879	0.3927
0.500	0.4284	0.00560	3.00000	-0.0574	0.4825	0.5310
1.000	0.4886	0.00536	3.00000	-0.0583	0.4771	0.6374
1.500	0.5488	0.00519	3.00000	-0.0593	0.4714	0.7360
2.000	0.6089	0.00516	3.00000	-0.0601	0.4658	0.7873
2.500	0.6689	0.00520	3.00000	-0.0609	0.4600	0.8205
3.000	0.7288	0.00527	3.00000	-0.0618	0.4537	0.8407
3.500	0.7887	0.00536	3.00000	-0.0626	0.4465	0.8589
4.000	0.8485	0.00547	3.00000	-0.0635	0.4404	0.8703
4.500	0.9077	0.00562	3.00000	-0.0643	0.4296	0.8864
5.000	0.9667	0.00578	3.00000	-0.0650	0.4188	0.8991
5.500	1.0251	0.00597	3.00000	-0.0657	0.4069	0.9133
6.000	1.0828	0.00625	3.00000	-0.0664	0.3904	0.9263
6.500	1.1388	0.00658	3.00000	-0.0667	0.3703	0.9420
7.000	1.1919	0.00699	3.00000	-0.0665	0.3460	0.9655
7.500	1.2485	0.00765	3.00000	-0.0674	0.3101	1.0000
8.000	1.3024	0.00840	3.00000	-0.0679	0.2787	1.0000
8.500	1.3521	0.00948	3.00000	-0.0680	0.2299	1.0000
9.000	1.3966	0.01087	3.00000	-0.0675	0.1787	1.0000
9.500	1.4303	0.01289	3.00000	-0.0659	0.1225	1.0000
10.000	1.4454	0.01544	3.00000	-0.0625	0.0876	1.0000
10.500	1.4616	0.01834	3.00000	-0.0605	0.0712	1.0000
11.000	1.4821	0.02126	3.00000	-0.0592	0.0563	1.0000
11.500	1.4909	0.02547	3.00000	-0.0579	0.0498	1.0000

XFOIL Version 5.0

Calculated polar for: FFA-W1-152

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-3.500	-0.0577	0.01006	3.00000	-0.0513	0.0100	0.0239
-3.000	0.0017	0.00999	3.00000	-0.0519	0.0100	0.0260
-2.500	0.0614	0.00994	3.00000	-0.0525	0.0100	0.0360
-2.000	0.1207	0.00988	3.00000	-0.0531	0.0100	0.0463
-1.500	0.1800	0.00984	3.00000	-0.0538	0.0100	0.0662
-1.000	0.2393	0.00984	3.00000	-0.0545	0.0100	0.0989
-0.500	0.2987	0.00996	3.00000	-0.0552	0.0100	0.1000
0.000	0.3579	0.01010	3.00000	-0.0559	0.0100	0.1000
0.500	0.4168	0.01026	3.00000	-0.0566	0.0100	0.1000
1.000	0.4755	0.01044	3.00000	-0.0573	0.0100	0.1000
1.500	0.5340	0.01064	3.00000	-0.0579	0.0100	0.1000
2.000	0.5921	0.01086	3.00000	-0.0586	0.0100	0.1000
2.500	0.6499	0.01110	3.00000	-0.0593	0.0100	0.1000
3.000	0.7073	0.01136	3.00000	-0.0599	0.0100	0.1000
3.500	0.7642	0.01166	3.00000	-0.0604	0.0100	0.1000
4.000	0.8205	0.01198	3.00000	-0.0610	0.0100	0.1000
4.500	0.8763	0.01233	3.00000	-0.0614	0.0100	0.1000
5.000	0.9313	0.01272	3.00000	-0.0618	0.0100	0.1000
5.500	0.9854	0.01315	3.00000	-0.0621	0.0100	0.1000
6.000	1.0385	0.01362	3.00000	-0.0623	0.0100	0.1000
6.500	1.0903	0.01415	3.00000	-0.0624	0.0100	0.1000
7.000	1.1405	0.01475	3.00000	-0.0622	0.0100	0.1000
7.500	1.1885	0.01542	3.00000	-0.0619	0.0100	0.1000
8.000	1.2300	0.01625	3.00000	-0.0607	0.0100	0.1000
8.500	1.2604	0.01763	3.00000	-0.0588	0.0100	0.1000
9.000	1.2917	0.01940	3.00000	-0.0577	0.0100	0.1000
9.500	1.3194	0.02158	3.00000	-0.0567	0.0100	0.1000
10.000	1.3448	0.02415	3.00000	-0.0559	0.0100	0.1000
10.500	1.3675	0.02710	3.00000	-0.0551	0.0100	0.1000
11.000	1.3889	0.03034	3.00000	-0.0544	0.0100	0.1000
11.500	1.4080	0.03383	3.00000	-0.0537	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W1-152

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 8.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
2.000	0.6126	0.00483	8.00000	-0.0603	0.4385	0.5149
2.500	0.6727	0.00448	8.00000	-0.0614	0.4297	0.6971
3.000	0.7331	0.00448	8.00000	-0.0625	0.4192	0.7533
3.500	0.7935	0.00455	8.00000	-0.0635	0.4090	0.7846
4.000	0.8537	0.00467	8.00000	-0.0645	0.3969	0.8101
4.500	0.9136	0.00485	8.00000	-0.0655	0.3828	0.8258
5.000	0.9727	0.00514	8.00000	-0.0664	0.3589	0.8396
5.500	1.0315	0.00545	8.00000	-0.0673	0.3378	0.8536
6.000	1.0904	0.00570	8.00000	-0.0682	0.3225	0.8656
6.500	1.1470	0.00625	8.00000	-0.0689	0.2861	0.8829
7.000	1.1995	0.00727	8.00000	-0.0692	0.2198	0.8916
7.500	1.2530	0.00800	8.00000	-0.0695	0.1802	0.9103
8.000	1.3046	0.00882	8.00000	-0.0696	0.1463	0.9265
8.500	1.3512	0.00990	8.00000	-0.0690	0.0997	0.9525
9.000	1.3988	0.01065	8.00000	-0.0683	0.0776	1.0000
9.500	1.4436	0.01181	8.00000	-0.0677	0.0537	1.0000
10.000	1.4860	0.01296	8.00000	-0.0668	0.0452	1.0000
10.500	1.5217	0.01404	8.00000	-0.0651	0.0372	1.0000
11.000	1.5452	0.01607	8.00000	-0.0629	0.0322	1.0000
11.500	1.5612	0.01904	8.00000	-0.0610	0.0281	1.0000
12.000	1.5819	0.02198	8.00000	-0.0598	0.0239	1.0000
12.500	1.6014	0.02515	8.00000	-0.0588	0.0203	1.0000
13.000	1.6197	0.02864	8.00000	-0.0579	0.0173	1.0000
13.500	1.6224	0.03368	8.00000	-0.0567	0.0160	1.0000

XFOIL Version 5.0

Calculated polar for: FFA-W1-152

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 8.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
2.000	0.5986	0.00918	8.00000	-0.0592	0.0100	0.1000
2.500	0.6574	0.00938	8.00000	-0.0600	0.0100	0.1000
3.000	0.7159	0.00959	8.00000	-0.0607	0.0100	0.1000
3.500	0.7741	0.00982	8.00000	-0.0614	0.0100	0.1000
4.000	0.8318	0.01008	8.00000	-0.0621	0.0100	0.1000
4.500	0.8892	0.01036	8.00000	-0.0628	0.0100	0.1000
5.000	0.9461	0.01066	8.00000	-0.0634	0.0100	0.1000
5.500	1.0024	0.01098	8.00000	-0.0639	0.0100	0.1000
6.000	1.0581	0.01134	8.00000	-0.0644	0.0100	0.1000
6.500	1.1130	0.01173	8.00000	-0.0648	0.0100	0.1000
7.000	1.1670	0.01215	8.00000	-0.0651	0.0100	0.1000
7.500	1.2199	0.01262	8.00000	-0.0652	0.0100	0.1000
8.000	1.2715	0.01314	8.00000	-0.0653	0.0100	0.1000
8.500	1.3215	0.01372	8.00000	-0.0651	0.0100	0.1000
9.000	1.3690	0.01438	8.00000	-0.0646	0.0100	0.1000
9.500	1.4027	0.01529	8.00000	-0.0623	0.0100	0.1000
10.000	1.4343	0.01674	8.00000	-0.0607	0.0100	0.1000
10.500	1.4628	0.01861	8.00000	-0.0593	0.0100	0.1000
11.000	1.4890	0.02086	8.00000	-0.0581	0.0100	0.1000
11.500	1.5116	0.02358	8.00000	-0.0570	0.0100	0.1000
12.000	1.5330	0.02659	8.00000	-0.0561	0.0100	0.1000
12.250	1.5428	0.02824	8.00000	-0.0557	0.0100	0.1000
12.500	1.5508	0.03008	8.00000	-0.0553	0.0100	0.1000
12.750	1.5601	0.03175	8.00000	-0.0548	0.0100	0.1000
13.000	1.5675	0.03375	8.00000	-0.0544	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W1-182

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-7.50000	-0.51790	0.02123	1.00000	-0.04680	0.60870	0.01050
-7.00000	-0.50020	0.01481	1.00000	-0.04580	0.59900	0.01310
-6.50000	-0.44530	0.01368	1.00000	-0.04640	0.58950	0.01450
-6.00000	-0.38800	0.01293	1.00000	-0.04720	0.57960	0.01570
-5.50000	-0.32950	0.01237	1.00000	-0.04800	0.57200	0.01670
-5.00000	-0.27020	0.01197	1.00000	-0.04890	0.56210	0.01740
-4.50000	-0.21070	0.01103	1.00000	-0.04970	0.55610	0.02540
-4.00000	-0.15030	0.01066	1.00000	-0.05070	0.54760	0.03350
-3.50000	-0.08980	0.01030	1.00000	-0.05170	0.54040	0.05580
-3.00000	-0.03090	0.00849	1.00000	-0.05400	0.53460	0.32400
-2.50000	0.02950	0.00774	1.00000	-0.05550	0.52560	0.46140
-2.00000	0.09070	0.00750	1.00000	-0.05660	0.52070	0.52660
-1.50000	0.15180	0.00747	1.00000	-0.05750	0.51540	0.57140
-1.00000	0.21300	0.00754	1.00000	-0.05850	0.50730	0.58180
-0.50000	0.27430	0.00760	1.00000	-0.05940	0.50240	0.59270
0.00000	0.33540	0.00766	1.00000	-0.06040	0.49790	0.60300
0.50000	0.39640	0.00776	1.00000	-0.06140	0.49060	0.61120
1.000	0.4571	0.00788	1.00000	-0.0623	0.4849	0.6194
1.500	0.5180	0.00797	1.00000	-0.0633	0.4810	0.6278
2.000	0.5785	0.00809	1.00000	-0.0642	0.4755	0.6341
2.500	0.6387	0.00825	1.00000	-0.0651	0.4674	0.6425
3.000	0.6990	0.00838	1.00000	-0.0660	0.4638	0.6491
3.500	0.7591	0.00851	1.00000	-0.0669	0.4590	0.6542
4.000	0.8188	0.00868	1.00000	-0.0678	0.4512	0.6628
4.500	0.8783	0.00886	1.00000	-0.0686	0.4456	0.6687
5.000	0.9378	0.00902	1.00000	-0.0694	0.4405	0.6735
5.500	0.9968	0.00923	1.00000	-0.0702	0.4320	0.6819
6.000	1.0556	0.00943	1.00000	-0.0710	0.4249	0.6877
6.500	1.1140	0.00967	1.00000	-0.0716	0.4166	0.6929
7.000	1.1719	0.00992	1.00000	-0.0723	0.4055	0.7009
7.500	1.2288	0.01027	1.00000	-0.0728	0.3897	0.7069
8.000	1.2845	0.01070	1.00000	-0.0732	0.3718	0.7123
8.500	1.3381	0.01128	1.00000	-0.0733	0.3491	0.7208
9.000	1.3863	0.01224	1.00000	-0.0728	0.3125	0.7272
9.500	1.4289	0.01352	1.00000	-0.0717	0.2735	0.7338
10.000	1.4605	0.01538	1.00000	-0.0695	0.2284	0.7428
11.000	1.4597	0.02290	1.00000	-0.0615	0.1449	0.7611
11.500	1.4532	0.02861	1.00000	-0.0601	0.1216	0.7702
12.000	1.4452	0.03511	1.00000	-0.0594	0.0995	0.7833
12.500	1.4384	0.04190	1.00000	-0.0592	0.0817	0.7970
13.000	1.4337	0.04868	1.00000	-0.0591	0.0686	0.8144
13.500	1.4337	0.05510	1.00000	-0.0593	0.0566	0.8519
14.000	1.4312	0.06138	1.00000	-0.0590	0.0500	1.0000
14.500	1.4351	0.06779	1.00000	-0.0595	0.0420	1.0000
15.000	1.4345	0.07487	1.00000	-0.0603	0.0385	1.0000
15.500	1.4382	0.08167	1.00000	-0.0612	0.0335	1.0000
16.000	1.4406	0.08885	1.00000	-0.0624	0.0288	1.0000
16.500	1.4459	0.09563	1.00000	-0.0636	0.0277	1.0000
17.000	1.4479	0.10326	1.00000	-0.0651	0.0262	1.0000
17.500	1.4485	0.11137	1.00000	-0.0670	0.0238	1.0000

XFOIL Version 5.0

Calculated polar for: FFA-W1-182

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
1.000	0.4341	0.01320	1.00000	-0.0590	0.0100	0.1000
1.500	0.4921	0.01342	1.00000	-0.0598	0.0100	0.1000
2.000	0.5497	0.01367	1.00000	-0.0604	0.0100	0.1000
2.500	0.6069	0.01395	1.00000	-0.0611	0.0100	0.1000
3.000	0.6634	0.01427	1.00000	-0.0616	0.0100	0.1000
3.500	0.7193	0.01462	1.00000	-0.0621	0.0100	0.1000
4.000	0.7744	0.01502	1.00000	-0.0625	0.0100	0.1000
4.500	0.8286	0.01546	1.00000	-0.0628	0.0100	0.1000
5.000	0.8817	0.01596	1.00000	-0.0629	0.0100	0.1000
5.500	0.9334	0.01652	1.00000	-0.0630	0.0100	0.1000
6.000	0.9834	0.01716	1.00000	-0.0628	0.0100	0.1000
6.500	1.0313	0.01789	1.00000	-0.0624	0.0100	0.1000
7.000	1.0750	0.01873	1.00000	-0.0615	0.0100	0.1000
7.500	1.1054	0.01994	1.00000	-0.0591	0.0100	0.1000
8.000	1.1378	0.02165	1.00000	-0.0581	0.0100	0.1000
8.500	1.1665	0.02386	1.00000	-0.0573	0.0100	0.1000
9.000	1.1923	0.02655	1.00000	-0.0567	0.0100	0.1000
9.500	1.2155	0.02970	1.00000	-0.0564	0.0100	0.1000
10.000	1.2361	0.03328	1.00000	-0.0563	0.0100	0.1000
10.500	1.2546	0.03725	1.00000	-0.0562	0.0100	0.1000
11.000	1.2715	0.04148	1.00000	-0.0562	0.0100	0.1000
11.500	1.2867	0.04602	1.00000	-0.0563	0.0100	0.1000
12.000	1.3011	0.05076	1.00000	-0.0565	0.0100	0.1000
12.500	1.3142	0.05569	1.00000	-0.0567	0.0100	0.1000
13.000	1.3267	0.06084	1.00000	-0.0571	0.0100	0.1000
13.500	1.3381	0.06622	1.00000	-0.0576	0.0100	0.1000
14.000	1.3489	0.07175	1.00000	-0.0581	0.0100	0.1000
14.500	1.3597	0.07741	1.00000	-0.0588	0.0100	0.1000
15.000	1.3695	0.08345	1.00000	-0.0598	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W1-182

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 2.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-10.00000	-0.77120	0.02393	2.00000	-0.04240	0.63020	0.00800
-9.50000	-0.74990	0.02059	2.00000	-0.04330	0.61760	0.00830
-9.00000	-0.70650	0.01866	2.00000	-0.04440	0.60630	0.00870
-6.50000	-0.45070	0.01143	2.00000	-0.04670	0.55910	0.01180
-6.00000	-0.39190	0.01063	2.00000	-0.04740	0.55110	0.01360
-5.50000	-0.33190	0.01017	2.00000	-0.04830	0.54250	0.01520
-5.00000	-0.27150	0.00983	2.00000	-0.04920	0.53720	0.01650
-4.00000	-0.14980	0.00920	2.00000	-0.05110	0.52210	0.02510
-3.50000	-0.08870	0.00896	2.00000	-0.05210	0.51690	0.03270
-3.00000	-0.02770	0.00857	2.00000	-0.05320	0.50740	0.07560
-2.50000	0.03270	0.00707	2.00000	-0.05540	0.50340	0.32520
-2.00000	0.09390	0.00649	2.00000	-0.05680	0.49900	0.44070
-1.50000	0.15530	0.00626	2.00000	-0.05800	0.49110	0.50050
-1.00000	0.21680	0.00616	2.00000	-0.05910	0.48560	0.54600
-0.50000	0.27840	0.00616	2.00000	-0.06010	0.48210	0.56400
0.00000	0.34000	0.00620	2.00000	-0.06120	0.47630	0.57500
0.50000	0.40150	0.00627	2.00000	-0.06220	0.46870	0.58490
1.000	0.4629	0.00632	2.00000	-0.0633	0.4651	0.5939
1.500	0.5242	0.00639	2.00000	-0.0643	0.4612	0.6016
2.000	0.5854	0.00648	2.00000	-0.0654	0.4545	0.6108
2.500	0.6463	0.00660	2.00000	-0.0664	0.4477	0.6160
3.000	0.7074	0.00667	2.00000	-0.0674	0.4445	0.6258
3.500	0.7681	0.00679	2.00000	-0.0684	0.4390	0.6323
4.000	0.8286	0.00695	2.00000	-0.0694	0.4298	0.6375
4.500	0.8891	0.00706	2.00000	-0.0704	0.4259	0.6464
5.000	0.9493	0.00722	2.00000	-0.0713	0.4196	0.6515
5.500	1.0090	0.00741	2.00000	-0.0722	0.4090	0.6570
6.000	1.0685	0.00761	2.00000	-0.0731	0.4000	0.6654
6.500	1.1275	0.00786	2.00000	-0.0738	0.3884	0.6707
7.000	1.1853	0.00821	2.00000	-0.0745	0.3700	0.6748
7.500	1.2418	0.00866	2.00000	-0.0750	0.3484	0.6836
8.000	1.2937	0.00952	2.00000	-0.0750	0.3075	0.6892
8.500	1.3443	0.01043	2.00000	-0.0748	0.2711	0.6933
9.000	1.3900	0.01164	2.00000	-0.0741	0.2277	0.7017
9.500	1.4299	0.01317	2.00000	-0.0727	0.1805	0.7074
10.000	1.4572	0.01528	2.00000	-0.0699	0.1319	0.7122
10.500	1.4670	0.01775	2.00000	-0.0655	0.1062	0.7211
11.000	1.4724	0.02151	2.00000	-0.0629	0.0856	0.7275
11.500	1.4765	0.02613	2.00000	-0.0615	0.0696	0.7330
12.000	1.4820	0.03114	2.00000	-0.0609	0.0565	0.7426
12.500	1.4851	0.03669	2.00000	-0.0605	0.0482	0.7496
13.000	1.4896	0.04227	2.00000	-0.0603	0.0414	0.7599
13.500	1.4891	0.04858	2.00000	-0.0602	0.0362	0.7679
14.000	1.4973	0.05407	2.00000	-0.0603	0.0303	0.7810
14.500	1.5005	0.06028	2.00000	-0.0606	0.0281	0.7978
15.000	1.5030	0.06676	2.00000	-0.0612	0.0264	0.8234

XFOIL Version 5.0

Calculated polar for: FFA-W1-182

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-7.00000	-0.49630	0.01439	3.00000	-0.04710	0.55290	0.01000
-6.50000	-0.44650	0.01211	3.00000	-0.04740	0.54230	0.01020
-6.00000	-0.39060	0.01058	3.00000	-0.04800	0.53710	0.01050
-5.50000	-0.33170	0.00960	3.00000	-0.04880	0.53060	0.01120
-5.00000	-0.27120	0.00905	3.00000	-0.04960	0.52140	0.01310
-3.50000	-0.08800	0.00828	3.00000	-0.05240	0.50420	0.02820
-3.00000	-0.02670	0.00805	3.00000	-0.05340	0.49560	0.04420
-2.50000	0.03430	0.00748	3.00000	-0.05480	0.49200	0.13330
-2.00000	0.09510	0.00625	3.00000	-0.05670	0.48860	0.35490
-1.50000	0.15640	0.00584	3.00000	-0.05800	0.48220	0.45210
-1.00000	0.21790	0.00563	3.00000	-0.05920	0.47460	0.51070
-0.50000	0.27950	0.00556	3.00000	-0.06030	0.46990	0.54830
0.00000	0.34110	0.00558	3.00000	-0.06130	0.46650	0.55750
0.500	0.4021	0.00563	3.00000	-0.0623	0.4567	0.5681
1.000	0.4636	0.00568	3.00000	-0.0634	0.4537	0.5742
1.500	0.5249	0.00574	3.00000	-0.0644	0.4490	0.5869
2.000	0.5862	0.00582	3.00000	-0.0655	0.4407	0.5948
2.500	0.6473	0.00591	3.00000	-0.0665	0.4333	0.6015
3.000	0.7084	0.00598	3.00000	-0.0676	0.4308	0.6122
3.500	0.7692	0.00609	3.00000	-0.0686	0.4260	0.6198
4.000	0.8299	0.00623	3.00000	-0.0696	0.4167	0.6235
4.500	0.8903	0.00636	3.00000	-0.0706	0.4104	0.6344
5.000	0.9504	0.00653	3.00000	-0.0716	0.4021	0.6412
5.500	1.0101	0.00674	3.00000	-0.0725	0.3936	0.6458
6.000	1.0693	0.00698	3.00000	-0.0733	0.3824	0.6508
6.500	1.1272	0.00735	3.00000	-0.0740	0.3603	0.6609
7.000	1.1852	0.00768	3.00000	-0.0747	0.3434	0.6665
7.500	1.2365	0.00868	3.00000	-0.0747	0.2896	0.6711
8.000	1.2886	0.00951	3.00000	-0.0747	0.2535	0.6766
8.500	1.3376	0.01052	3.00000	-0.0743	0.2103	0.6849
9.000	1.3826	0.01175	3.00000	-0.0735	0.1677	0.6907
9.500	1.4234	0.01313	3.00000	-0.0722	0.1305	0.6954
10.000	1.4572	0.01474	3.00000	-0.0701	0.1007	0.6990
10.500	1.4717	0.01678	3.00000	-0.0657	0.0804	0.7085
11.000	1.4879	0.01945	3.00000	-0.0632	0.0633	0.7160
11.500	1.4937	0.02354	3.00000	-0.0613	0.0495	0.7204

XFOIL Version 5.0

Calculated polar for: FFA-W1-182

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-5.00000	-0.27740	0.01107	3.00000	-0.04970	0.01000	0.01280
-4.50000	-0.21720	0.01080	3.00000	-0.05050	0.01000	0.01730
-4.00000	-0.15700	0.01068	3.00000	-0.05140	0.01000	0.01840
-3.50000	-0.09640	0.01054	3.00000	-0.05220	0.01000	0.02740
-3.00000	-0.03600	0.01041	3.00000	-0.05320	0.01000	0.04120
-2.50000	0.02420	0.01009	3.00000	-0.05430	0.01000	0.10000
-2.00000	0.08470	0.01014	3.00000	-0.05520	0.01000	0.10000
-1.50000	0.14500	0.01021	3.00000	-0.05600	0.01000	0.10000
-1.00000	0.20520	0.01030	3.00000	-0.05690	0.01000	0.10000
-0.50000	0.26520	0.01040	3.00000	-0.05780	0.01000	0.10000
0.00000	0.32500	0.01052	3.00000	-0.05870	0.01000	0.10000
0.500	0.3845	0.01068	3.00000	-0.0596	0.0100	0.1000
1.000	0.4439	0.01083	3.00000	-0.0604	0.0100	0.1000
1.500	0.5030	0.01101	3.00000	-0.0613	0.0100	0.1000
2.000	0.5619	0.01122	3.00000	-0.0621	0.0100	0.1000
2.500	0.6203	0.01144	3.00000	-0.0628	0.0100	0.1000
3.000	0.6784	0.01169	3.00000	-0.0635	0.0100	0.1000
3.500	0.7361	0.01196	3.00000	-0.0642	0.0100	0.1000
4.000	0.7932	0.01226	3.00000	-0.0648	0.0100	0.1000
4.500	0.8497	0.01259	3.00000	-0.0653	0.0100	0.1000
5.000	0.9055	0.01295	3.00000	-0.0658	0.0100	0.1000
5.500	0.9606	0.01335	3.00000	-0.0662	0.0100	0.1000
6.000	1.0147	0.01380	3.00000	-0.0664	0.0100	0.1000
6.500	1.0677	0.01428	3.00000	-0.0665	0.0100	0.1000
7.000	1.1192	0.01483	3.00000	-0.0665	0.0100	0.1000
7.500	1.1691	0.01544	3.00000	-0.0662	0.0100	0.1000
8.000	1.2165	0.01613	3.00000	-0.0656	0.0100	0.1000
8.500	1.2542	0.01695	3.00000	-0.0636	0.0100	0.1000
9.000	1.2846	0.01827	3.00000	-0.0614	0.0100	0.1000
9.500	1.3135	0.02007	3.00000	-0.0599	0.0100	0.1000
10.000	1.3390	0.02242	3.00000	-0.0587	0.0100	0.1000
10.500	1.3612	0.02529	3.00000	-0.0579	0.0100	0.1000
11.000	1.3817	0.02865	3.00000	-0.0574	0.0100	0.1000
11.500	1.3999	0.03234	3.00000	-0.0569	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W1-211

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.42860	0.00849	1.00000	-0.06030	0.57650	0.40050
-5.50000	-0.36590	0.00841	1.00000	-0.06140	0.56430	0.41820
-5.00000	-0.30340	0.00853	1.00000	-0.06230	0.55160	0.43550
-4.50000	-0.24070	0.00858	1.00000	-0.06330	0.53860	0.43980
-4.00000	-0.17800	0.00853	1.00000	-0.06450	0.52960	0.44490
-3.50000	-0.11520	0.00849	1.00000	-0.06560	0.51650	0.45040
-3.00000	-0.05270	0.00851	1.00000	-0.06670	0.50850	0.45420
-2.50000	0.00990	0.00855	1.00000	-0.06780	0.49730	0.45720
-2.00000	0.07220	0.00860	1.00000	-0.06880	0.49040	0.46010
-1.50000	0.13470	0.00858	1.00000	-0.07000	0.47880	0.46550
-1.00000	0.19710	0.00860	1.00000	-0.07110	0.47300	0.47020
-0.50000	0.25930	0.00867	1.00000	-0.07220	0.46490	0.47330
0.00000	0.32120	0.00876	1.00000	-0.07320	0.45720	0.47640
0.50000	0.38310	0.00885	1.00000	-0.07410	0.45080	0.47890
1.00000	0.44500	0.00891	1.00000	-0.07520	0.44210	0.48380
1.000	0.4450	0.00891	1.00000	-0.0752	0.4421	0.4838
1.500	0.5066	0.00899	1.00000	-0.0762	0.4368	0.4888
2.000	0.5681	0.00910	1.00000	-0.0771	0.4309	0.4920
2.500	0.6292	0.00927	1.00000	-0.0780	0.4226	0.4949
3.000	0.6902	0.00942	1.00000	-0.0789	0.4177	0.4978
3.500	0.7512	0.00953	1.00000	-0.0798	0.4125	0.5025
4.000	0.8117	0.00970	1.00000	-0.0806	0.4043	0.5075
4.500	0.8719	0.00991	1.00000	-0.0813	0.3988	0.5110
5.000	0.9320	0.01008	1.00000	-0.0820	0.3940	0.5141
5.500	0.9915	0.01033	1.00000	-0.0826	0.3862	0.5172
6.000	1.0507	0.01056	1.00000	-0.0832	0.3793	0.5228
6.500	1.1099	0.01077	1.00000	-0.0838	0.3727	0.5276
7.000	1.1679	0.01110	1.00000	-0.0842	0.3615	0.5312
7.500	1.2259	0.01139	1.00000	-0.0845	0.3523	0.5345
8.000	1.2826	0.01178	1.00000	-0.0847	0.3395	0.5377
8.500	1.3380	0.01223	1.00000	-0.0848	0.3230	0.5452
9.000	1.3920	0.01277	1.00000	-0.0847	0.3090	0.5493
9.500	1.4409	0.01364	1.00000	-0.0839	0.2824	0.5528
10.000	1.4838	0.01485	1.00000	-0.0825	0.2514	0.5560
10.500	1.5146	0.01661	1.00000	-0.0798	0.2125	0.5629
11.000	1.5135	0.01965	1.00000	-0.0739	0.1703	0.5670
11.500	1.5108	0.02394	1.00000	-0.0706	0.1439	0.5704
12.000	1.5073	0.02929	1.00000	-0.0691	0.1165	0.5737
12.500	1.4998	0.03582	1.00000	-0.0686	0.0976	0.5777
13.000	1.4932	0.04282	1.00000	-0.0688	0.0852	0.5832
13.500	1.4868	0.05025	1.00000	-0.0694	0.0753	0.5869
14.000	1.4834	0.05756	1.00000	-0.0702	0.0633	0.5902
14.500	1.4749	0.06578	1.00000	-0.0714	0.0581	0.5930
15.000	1.4743	0.07319	1.00000	-0.0725	0.0504	0.5977
15.500	1.4716	0.08111	1.00000	-0.0739	0.0463	0.6031
16.000	1.4678	0.08939	1.00000	-0.0756	0.0437	0.6075
16.500	1.4652	0.09766	1.00000	-0.0773	0.0399	0.6108
17.000	1.4648	0.10588	1.00000	-0.0791	0.0353	0.6141
17.500	1.4631	0.11458	1.00000	-0.0813	0.0328	0.6209
18.000	1.4626	0.12328	1.00000	-0.0836	0.0317	0.6257

XFOIL Version 5.0

Calculated polar for: FFA-W1-211

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
1.000	0.4169	0.01403	1.00000	-0.0706	0.0100	0.1000
1.500	0.4764	0.01422	1.00000	-0.0714	0.0100	0.1000
2.000	0.5353	0.01443	1.00000	-0.0722	0.0100	0.1000
2.500	0.5938	0.01468	1.00000	-0.0729	0.0100	0.1000
3.000	0.6518	0.01497	1.00000	-0.0735	0.0100	0.1000
3.500	0.7090	0.01530	1.00000	-0.0741	0.0100	0.1000
4.000	0.7655	0.01566	1.00000	-0.0745	0.0100	0.1000
4.500	0.8212	0.01607	1.00000	-0.0748	0.0100	0.1000
5.000	0.8758	0.01654	1.00000	-0.0750	0.0100	0.1000
5.500	0.9293	0.01706	1.00000	-0.0750	0.0100	0.1000
6.000	0.9813	0.01764	1.00000	-0.0749	0.0100	0.1000
6.500	1.0315	0.01831	1.00000	-0.0746	0.0100	0.1000
7.000	1.0792	0.01907	1.00000	-0.0740	0.0100	0.1000
7.500	1.1211	0.01995	1.00000	-0.0725	0.0100	0.1000
8.000	1.1512	0.02116	1.00000	-0.0698	0.0100	0.1000
8.500	1.1834	0.02286	1.00000	-0.0685	0.0100	0.1000
9.000	1.2113	0.02510	1.00000	-0.0675	0.0100	0.1000
9.500	1.2359	0.02790	1.00000	-0.0668	0.0100	0.1000
10.000	1.2578	0.03127	1.00000	-0.0666	0.0100	0.1000
10.500	1.2769	0.03517	1.00000	-0.0666	0.0100	0.1000
11.000	1.2938	0.03956	1.00000	-0.0669	0.0100	0.1000
11.500	1.3084	0.04438	1.00000	-0.0674	0.0100	0.1000
12.000	1.3213	0.04961	1.00000	-0.0680	0.0100	0.1000
12.500	1.3319	0.05520	1.00000	-0.0687	0.0100	0.1000
13.000	1.3411	0.06116	1.00000	-0.0695	0.0100	0.1000
13.500	1.3495	0.06734	1.00000	-0.0705	0.0100	0.1000
14.000	1.3567	0.07376	1.00000	-0.0716	0.0100	0.1000
14.500	1.3635	0.08043	1.00000	-0.0728	0.0100	0.1000
15.000	1.3699	0.08730	1.00000	-0.0742	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W1-211

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 2.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-8.00000	-0.68050	0.01370	2.00000	-0.04690	0.59320	0.04150
-7.50000	-0.61790	0.01246	2.00000	-0.05010	0.57850	0.06060
-7.00000	-0.55420	0.01085	2.00000	-0.05390	0.56720	0.12030
-6.50000	-0.48910	0.00816	2.00000	-0.05900	0.55220	0.28550
-6.00000	-0.42570	0.00734	2.00000	-0.06110	0.53780	0.35240
-5.50000	-0.36270	0.00704	2.00000	-0.06250	0.52950	0.38160
-5.00000	-0.29970	0.00687	2.00000	-0.06370	0.51660	0.40410
-4.50000	-0.23710	0.00683	2.00000	-0.06480	0.50680	0.42540
-4.00000	-0.17400	0.00679	2.00000	-0.06590	0.49610	0.43260
-3.50000	-0.11110	0.00680	2.00000	-0.06700	0.48820	0.43690
-3.00000	-0.04820	0.00683	2.00000	-0.06810	0.47770	0.43960
-2.50000	0.01450	0.00687	2.00000	-0.06920	0.47040	0.44170
-2.00000	0.07750	0.00686	2.00000	-0.07040	0.45950	0.44860
-1.50000	0.14020	0.00688	2.00000	-0.07150	0.45490	0.45350
-1.00000	0.20290	0.00693	2.00000	-0.07260	0.44750	0.45640
-0.50000	0.26530	0.00700	2.00000	-0.07360	0.43870	0.45890
0.00000	0.32770	0.00707	2.00000	-0.07470	0.43410	0.46080
0.50000	0.39030	0.00710	2.00000	-0.07580	0.42670	0.46740
1.000	0.4526	0.00718	2.00000	-0.0768	0.4197	0.4718
1.500	0.5148	0.00726	2.00000	-0.0778	0.4152	0.4748
2.000	0.5767	0.00736	2.00000	-0.0788	0.4091	0.4774
2.500	0.6385	0.00750	2.00000	-0.0798	0.4007	0.4794
3.000	0.7004	0.00757	2.00000	-0.0808	0.3975	0.4844
3.500	0.7620	0.00769	2.00000	-0.0817	0.3917	0.4900
4.000	0.8233	0.00785	2.00000	-0.0826	0.3823	0.4934
4.500	0.8845	0.00798	2.00000	-0.0834	0.3785	0.4960
5.000	0.9454	0.00815	2.00000	-0.0842	0.3739	0.4982
5.500	1.0059	0.00834	2.00000	-0.0850	0.3627	0.5029
6.000	1.0662	0.00853	2.00000	-0.0857	0.3572	0.5090
6.500	1.1259	0.00879	2.00000	-0.0864	0.3440	0.5127
7.000	1.1845	0.00911	2.00000	-0.0868	0.3327	0.5155
7.500	1.2424	0.00949	2.00000	-0.0872	0.3179	0.5178
8.000	1.2997	0.00988	2.00000	-0.0875	0.3022	0.5240
8.500	1.3533	0.01058	2.00000	-0.0874	0.2769	0.5291
9.000	1.4042	0.01145	2.00000	-0.0869	0.2450	0.5331
9.500	1.4495	0.01268	2.00000	-0.0858	0.2070	0.5358
10.000	1.4840	0.01454	2.00000	-0.0835	0.1593	0.5397
10.500	1.5112	0.01637	2.00000	-0.0801	0.1264	0.5458
11.000	1.5240	0.01861	2.00000	-0.0755	0.1012	0.5498
11.500	1.5332	0.02187	2.00000	-0.0726	0.0837	0.5529
12.000	1.5371	0.02629	2.00000	-0.0708	0.0727	0.5553
12.500	1.5399	0.03150	2.00000	-0.0701	0.0606	0.5592
13.000	1.5449	0.03699	2.00000	-0.0699	0.0524	0.5656
13.500	1.5484	0.04292	2.00000	-0.0701	0.0466	0.5693
14.000	1.5468	0.04979	2.00000	-0.0706	0.0435	0.5721
14.500	1.5485	0.05641	2.00000	-0.0712	0.0390	0.5745
15.000	1.5528	0.06294	2.00000	-0.0719	0.0343	0.5788

XFOIL Version 5.0

Calculated polar for: FFA-W1-211

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-10.00000	-0.92370	0.01870	3.00000	-0.03350	0.63420	0.01960
-9.50000	-0.86580	0.01657	3.00000	-0.03830	0.61680	0.01990
-9.00000	-0.80310	0.01447	3.00000	-0.04340	0.59970	0.02320
-8.50000	-0.74030	0.01331	3.00000	-0.04660	0.58450	0.02780
-8.00000	-0.67830	0.01240	3.00000	-0.04910	0.57270	0.03270
-7.50000	-0.61590	0.01165	3.00000	-0.05130	0.55580	0.04400
-7.00000	-0.55310	0.01075	3.00000	-0.05380	0.54520	0.06930
-6.50000	-0.48890	0.00904	3.00000	-0.05740	0.53120	0.15980
-6.00000	-0.42490	0.00733	3.00000	-0.06070	0.52270	0.27950
-5.50000	-0.36160	0.00667	3.00000	-0.06260	0.50870	0.34400
-5.00000	-0.29850	0.00636	3.00000	-0.06400	0.50230	0.37330
-4.50000	-0.23550	0.00614	3.00000	-0.06520	0.48720	0.41140
-4.00000	-0.17270	0.00611	3.00000	-0.06630	0.48160	0.42220
-3.50000	-0.10970	0.00610	3.00000	-0.06750	0.47300	0.42990
-3.00000	-0.04670	0.00611	3.00000	-0.06860	0.46220	0.43430
-2.50000	0.01620	0.00612	3.00000	-0.06970	0.45790	0.43780
-2.00000	0.07890	0.00618	3.00000	-0.07070	0.44840	0.43960
-1.50000	0.14170	0.00621	3.00000	-0.07180	0.43900	0.44290
-1.00000	0.20450	0.00622	3.00000	-0.07300	0.43550	0.44900
-0.50000	0.26720	0.00627	3.00000	-0.07400	0.43000	0.45470
0.00000	0.32980	0.00632	3.00000	-0.07510	0.42160	0.45800
0.50000	0.39230	0.00638	3.00000	-0.07620	0.41460	0.46040
1.000	0.4541	0.00645	3.00000	-0.0766	0.4067	0.4634
1.500	0.5163	0.00652	3.00000	-0.0776	0.4040	0.4672
2.000	0.5783	0.00661	3.00000	-0.0786	0.3993	0.4705
2.500	0.6402	0.00671	3.00000	-0.0796	0.3927	0.4724
3.000	0.7019	0.00684	3.00000	-0.0806	0.3835	0.4739
3.500	0.7638	0.00691	3.00000	-0.0815	0.3812	0.4791
4.000	0.8252	0.00704	3.00000	-0.0825	0.3762	0.4855
4.500	0.8865	0.00718	3.00000	-0.0833	0.3675	0.4895
5.000	0.9475	0.00733	3.00000	-0.0842	0.3601	0.4924
5.500	1.0082	0.00750	3.00000	-0.0850	0.3551	0.4946
6.000	1.0683	0.00773	3.00000	-0.0857	0.3412	0.4962
6.500	1.1281	0.00797	3.00000	-0.0863	0.3333	0.5033
7.000	1.1861	0.00837	3.00000	-0.0868	0.3121	0.5088
7.500	1.2445	0.00871	3.00000	-0.0873	0.2946	0.5127
8.000	1.3007	0.00922	3.00000	-0.0875	0.2762	0.5156
8.500	1.3507	0.01026	3.00000	-0.0869	0.2332	0.5177
9.000	1.3990	0.01134	3.00000	-0.0862	0.1978	0.5216
9.500	1.4438	0.01257	3.00000	-0.0850	0.1583	0.5280
10.000	1.4835	0.01402	3.00000	-0.0832	0.1246	0.5337
10.500	1.5105	0.01588	3.00000	-0.0798	0.0917	0.5368
11.000	1.5287	0.01765	3.00000	-0.0754	0.0747	0.5391
11.500	1.5430	0.02029	3.00000	-0.0722	0.0633	0.5408

XFOIL Version 5.0

Calculated polar for: FFA-W1-211

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
1.000	0.4296	0.01149	3.00000	-0.0722	0.0100	0.1000
1.500	0.4898	0.01164	3.00000	-0.0731	0.0100	0.1000
2.000	0.5498	0.01182	3.00000	-0.0740	0.0100	0.1000
2.500	0.6093	0.01202	3.00000	-0.0747	0.0100	0.1000
3.000	0.6685	0.01225	3.00000	-0.0755	0.0100	0.1000
3.500	0.7272	0.01250	3.00000	-0.0761	0.0100	0.1000
4.000	0.7854	0.01278	3.00000	-0.0767	0.0100	0.1000
4.500	0.8431	0.01309	3.00000	-0.0772	0.0100	0.1000
5.000	0.9001	0.01343	3.00000	-0.0776	0.0100	0.1000
5.500	0.9564	0.01381	3.00000	-0.0779	0.0100	0.1000
6.000	1.0118	0.01423	3.00000	-0.0781	0.0100	0.1000
6.500	1.0662	0.01469	3.00000	-0.0782	0.0100	0.1000
7.000	1.1193	0.01520	3.00000	-0.0781	0.0100	0.1000
7.500	1.1711	0.01576	3.00000	-0.0778	0.0100	0.1000
8.000	1.2209	0.01640	3.00000	-0.0773	0.0100	0.1000
8.500	1.2682	0.01712	3.00000	-0.0765	0.0100	0.1000
9.000	1.3026	0.01796	3.00000	-0.0736	0.0100	0.1000
9.500	1.3345	0.01923	3.00000	-0.0712	0.0100	0.1000
10.000	1.3632	0.02098	3.00000	-0.0694	0.0100	0.1000
10.500	1.3875	0.02332	3.00000	-0.0679	0.0100	0.1000
11.000	1.4092	0.02625	3.00000	-0.0669	0.0100	0.1000
11.500	1.4281	0.02977	3.00000	-0.0664	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W1-242

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.50000	-0.57060	0.01066	1.00000	-0.04940	0.60020	0.41500
-6.00000	-0.50390	0.01048	1.00000	-0.05180	0.58320	0.41900
-5.50000	-0.43730	0.01029	1.00000	-0.05410	0.57130	0.42180
-5.00000	-0.37050	0.01008	1.00000	-0.05660	0.55710	0.42550
-4.50000	-0.30320	0.00985	1.00000	-0.05910	0.54320	0.43200
-4.00000	-0.23720	0.00974	1.00000	-0.06110	0.53100	0.43560
-3.50000	-0.17150	0.00967	1.00000	-0.06310	0.51980	0.43850
-3.00000	-0.10610	0.00964	1.00000	-0.06490	0.50560	0.44120
-2.50000	-0.04010	0.00953	1.00000	-0.06700	0.49700	0.44650
-2.00000	0.02550	0.00947	1.00000	-0.06890	0.48480	0.45170
-1.50000	0.09040	0.00948	1.00000	-0.07050	0.47470	0.45510
-1.00000	0.15510	0.00951	1.00000	-0.07220	0.46370	0.45810
-0.50000	0.21950	0.00956	1.00000	-0.07370	0.45420	0.46070
0.00000	0.28440	0.00955	1.00000	-0.07540	0.44290	0.46690
0.50000	0.34870	0.00961	1.00000	-0.07690	0.43240	0.47150
1.000	0.4112	0.00974	1.00000	-0.0780	0.4222	0.4731
1.500	0.4752	0.00980	1.00000	-0.0795	0.4120	0.4782
2.000	0.5389	0.00989	1.00000	-0.0808	0.4010	0.4835
2.500	0.6018	0.01004	1.00000	-0.0820	0.3914	0.4869
3.000	0.6645	0.01022	1.00000	-0.0831	0.3798	0.4899
3.500	0.7267	0.01042	1.00000	-0.0842	0.3707	0.4926
4.000	0.7891	0.01059	1.00000	-0.0853	0.3585	0.4987
4.500	0.8509	0.01080	1.00000	-0.0863	0.3501	0.5034
5.000	0.9120	0.01106	1.00000	-0.0871	0.3393	0.5068
5.500	0.9724	0.01136	1.00000	-0.0878	0.3295	0.5098
6.000	1.0324	0.01169	1.00000	-0.0884	0.3179	0.5124
6.500	1.0921	0.01199	1.00000	-0.0890	0.3086	0.5187
7.000	1.1507	0.01238	1.00000	-0.0894	0.2958	0.5235
7.500	1.2082	0.01281	1.00000	-0.0896	0.2872	0.5267
8.000	1.2644	0.01331	1.00000	-0.0896	0.2733	0.5297
8.500	1.3192	0.01387	1.00000	-0.0894	0.2612	0.5323
9.000	1.3719	0.01453	1.00000	-0.0890	0.2457	0.5393
9.500	1.4231	0.01523	1.00000	-0.0883	0.2326	0.5435
10.000	1.4716	0.01603	1.00000	-0.0872	0.2195	0.5467
10.500	1.5140	0.01709	1.00000	-0.0853	0.2059	0.5496
11.000	1.5436	0.01818	1.00000	-0.0813	0.1934	0.5532
11.500	1.5695	0.01974	1.00000	-0.0776	0.1807	0.5590
12.000	1.5882	0.02201	1.00000	-0.0740	0.1711	0.5628
12.500	1.6012	0.02511	1.00000	-0.0712	0.1573	0.5657
13.000	1.6064	0.02948	1.00000	-0.0694	0.1481	0.5683
13.500	1.6088	0.03481	1.00000	-0.0687	0.1363	0.5705
14.000	1.6021	0.04177	1.00000	-0.0690	0.1205	0.5758
14.500	1.5933	0.04938	1.00000	-0.0697	0.1124	0.5797
15.000	1.5859	0.05725	1.00000	-0.0708	0.1009	0.5823
15.500	1.5736	0.06605	1.00000	-0.0724	0.0921	0.5846
16.000	1.5642	0.07474	1.00000	-0.0740	0.0833	0.5867
16.500	1.5509	0.08444	1.00000	-0.0763	0.0770	0.5886
17.000	1.5431	0.09358	1.00000	-0.0785	0.0683	0.5911
17.500	1.5328	0.10351	1.00000	-0.0813	0.0652	0.5957
18.000	1.5225	0.11378	1.00000	-0.0843	0.0605	0.5990

XFOIL Version 5.0

Calculated polar for: FFA-W1-242

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
1.000	0.3502	0.01615	1.00000	-0.0635	0.0100	0.1000
1.500	0.4146	0.01617	1.00000	-0.0657	0.0100	0.1000
2.000	0.4782	0.01624	1.00000	-0.0677	0.0100	0.1000
2.500	0.5409	0.01637	1.00000	-0.0694	0.0100	0.1000
3.000	0.6027	0.01655	1.00000	-0.0710	0.0100	0.1000
3.500	0.6636	0.01678	1.00000	-0.0723	0.0100	0.1000
4.000	0.7235	0.01706	1.00000	-0.0734	0.0100	0.1000
4.500	0.7823	0.01739	1.00000	-0.0744	0.0100	0.1000
5.000	0.8398	0.01778	1.00000	-0.0751	0.0100	0.1000
5.500	0.8960	0.01822	1.00000	-0.0756	0.0100	0.1000
6.000	0.9506	0.01874	1.00000	-0.0758	0.0100	0.1000
6.500	1.0033	0.01932	1.00000	-0.0758	0.0100	0.1000
7.000	1.0537	0.02000	1.00000	-0.0754	0.0100	0.1000
7.500	1.1010	0.02078	1.00000	-0.0746	0.0100	0.1000
8.000	1.1331	0.02169	1.00000	-0.0714	0.0100	0.1000
8.500	1.1638	0.02301	1.00000	-0.0687	0.0100	0.1000
9.000	1.1941	0.02477	1.00000	-0.0669	0.0100	0.1000
9.500	1.2199	0.02711	1.00000	-0.0655	0.0100	0.1000
10.000	1.2425	0.03010	1.00000	-0.0648	0.0100	0.1000
10.500	1.2628	0.03375	1.00000	-0.0648	0.0100	0.1000
11.000	1.2807	0.03800	1.00000	-0.0652	0.0100	0.1000
11.500	1.2967	0.04276	1.00000	-0.0659	0.0100	0.1000
12.000	1.3105	0.04801	1.00000	-0.0669	0.0100	0.1000
12.500	1.3224	0.05363	1.00000	-0.0680	0.0100	0.1000
13.000	1.3326	0.05970	1.00000	-0.0694	0.0100	0.1000
13.500	1.3415	0.06605	1.00000	-0.0708	0.0100	0.1000
14.000	1.3488	0.07275	1.00000	-0.0724	0.0100	0.1000
14.500	1.3547	0.07988	1.00000	-0.0742	0.0100	0.1000
15.000	1.3597	0.08731	1.00000	-0.0761	0.0100	0.1000

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Calculated polar for: FFA-W1-242

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 2.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-9.00000	-0.91030	0.01332	2.00000	-0.03160	0.63920	0.30560
-8.50000	-0.82350	0.00984	2.00000	-0.04460	0.62050	0.36040
-8.00000	-0.75400	0.00926	2.00000	-0.04810	0.60180	0.37090
-7.50000	-0.68600	0.00888	2.00000	-0.05100	0.58680	0.38210
-7.00000	-0.61900	0.00861	2.00000	-0.05340	0.57260	0.39090
-6.50000	-0.55220	0.00840	2.00000	-0.05570	0.55920	0.39870
-6.00000	-0.48680	0.00832	2.00000	-0.05750	0.54350	0.40400
-5.50000	-0.42010	0.00813	2.00000	-0.05980	0.53300	0.40910
-5.00000	-0.35380	0.00801	2.00000	-0.06180	0.52010	0.41430
-4.50000	-0.28800	0.00792	2.00000	-0.06370	0.50620	0.41800
-4.00000	-0.22250	0.00787	2.00000	-0.06550	0.49880	0.42090
-3.50000	-0.15730	0.00785	2.00000	-0.06720	0.48490	0.42290
-3.00000	-0.09160	0.00778	2.00000	-0.06900	0.47650	0.42770
-2.50000	-0.02620	0.00774	2.00000	-0.07080	0.46510	0.43320
-2.00000	0.03880	0.00774	2.00000	-0.07250	0.45460	0.43700
-1.50000	0.10360	0.00776	2.00000	-0.07400	0.44460	0.44020
-1.00000	0.16810	0.00781	2.00000	-0.07550	0.43310	0.44220
-0.50000	0.23300	0.00781	2.00000	-0.07710	0.42430	0.44800
0.00000	0.29750	0.00786	2.00000	-0.07870	0.41180	0.45330
0.50000	0.36170	0.00792	2.00000	-0.08010	0.40400	0.45730
1.000	0.4256	0.00803	2.00000	-0.0814	0.3905	0.4598
1.500	0.4894	0.00813	2.00000	-0.0827	0.3837	0.4627
2.000	0.5534	0.00821	2.00000	-0.0841	0.3718	0.4692
2.500	0.6167	0.00835	2.00000	-0.0854	0.3634	0.4737
3.000	0.6799	0.00849	2.00000	-0.0865	0.3506	0.4772
3.500	0.7425	0.00869	2.00000	-0.0876	0.3430	0.4795
4.000	0.8052	0.00885	2.00000	-0.0887	0.3305	0.4842
4.500	0.8675	0.00905	2.00000	-0.0897	0.3225	0.4898
5.000	0.9295	0.00927	2.00000	-0.0906	0.3105	0.4936
5.500	0.9907	0.00954	2.00000	-0.0914	0.3018	0.4964
6.000	1.0516	0.00981	2.00000	-0.0921	0.2905	0.4989
6.500	1.1120	0.01011	2.00000	-0.0928	0.2801	0.5048
7.000	1.1721	0.01042	2.00000	-0.0934	0.2699	0.5098
7.500	1.2308	0.01082	2.00000	-0.0938	0.2562	0.5133
8.000	1.2875	0.01135	2.00000	-0.0939	0.2410	0.5159
8.500	1.3431	0.01193	2.00000	-0.0937	0.2262	0.5180
9.000	1.3989	0.01244	2.00000	-0.0937	0.2126	0.5254
9.500	1.4526	0.01305	2.00000	-0.0933	0.1996	0.5298
10.000	1.5019	0.01391	2.00000	-0.0923	0.1832	0.5329
10.500	1.5497	0.01477	2.00000	-0.0910	0.1711	0.5354
11.000	1.5943	0.01572	2.00000	-0.0894	0.1562	0.5393
11.500	1.6274	0.01686	2.00000	-0.0859	0.1422	0.5457
12.000	1.6424	0.01865	2.00000	-0.0803	0.1325	0.5493
12.500	1.6579	0.02101	2.00000	-0.0763	0.1191	0.5520
13.000	1.6649	0.02449	2.00000	-0.0730	0.1064	0.5541
13.500	1.6623	0.02962	2.00000	-0.0711	0.0981	0.5572
14.000	1.6546	0.03615	2.00000	-0.0706	0.0841	0.5630
14.500	1.6484	0.04324	2.00000	-0.0710	0.0762	0.5664
15.000	1.6450	0.05035	2.00000	-0.0717	0.0682	0.5691

XFOIL Version 5.0

Calculated polar for: FFA-W1-242

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-8.50000	-0.81740	0.00907	3.00000	-0.04660	0.59670	0.32970
-8.00000	-0.74670	0.00832	3.00000	-0.05070	0.57900	0.35390
-7.50000	-0.67850	0.00786	3.00000	-0.05360	0.56250	0.36830
-7.00000	-0.61170	0.00761	3.00000	-0.05590	0.55310	0.38180
-6.50000	-0.54600	0.00748	3.00000	-0.05780	0.53510	0.38720
-6.00000	-0.47970	0.00730	3.00000	-0.05990	0.52720	0.39780
-5.50000	-0.41390	0.00718	3.00000	-0.06180	0.51170	0.40500
-5.00000	-0.34850	0.00712	3.00000	-0.06350	0.50320	0.40770
-4.50000	-0.28320	0.00708	3.00000	-0.06520	0.48890	0.40950
-4.00000	-0.21820	0.00707	3.00000	-0.06680	0.48220	0.41080
-3.50000	-0.15270	0.00701	3.00000	-0.06860	0.46800	0.41730
-3.00000	-0.08740	0.00697	3.00000	-0.07030	0.45970	0.42270
-2.50000	-0.02240	0.00697	3.00000	-0.07190	0.45080	0.42720
-2.00000	0.04240	0.00697	3.00000	-0.07350	0.43730	0.43020
-1.50000	0.10710	0.00700	3.00000	-0.07500	0.43050	0.43250
-1.00000	0.17160	0.00704	3.00000	-0.07640	0.41770	0.43410
-0.50000	0.23620	0.00707	3.00000	-0.07800	0.40930	0.44030
0.00000	0.30070	0.00712	3.00000	-0.07940	0.39860	0.44630
0.50000	0.36500	0.00718	3.00000	-0.08090	0.38880	0.45090
1.000	0.4273	0.00728	3.00000	-0.0819	0.3800	0.4536
1.500	0.4911	0.00738	3.00000	-0.0832	0.3649	0.4588
2.000	0.5547	0.00748	3.00000	-0.0845	0.3592	0.4616
2.500	0.6179	0.00762	3.00000	-0.0857	0.3442	0.4636
3.000	0.6811	0.00776	3.00000	-0.0869	0.3383	0.4676
3.500	0.7442	0.00790	3.00000	-0.0880	0.3279	0.4739
4.000	0.8068	0.00808	3.00000	-0.0891	0.3172	0.4788
4.500	0.8691	0.00827	3.00000	-0.0901	0.3082	0.4826
5.000	0.9312	0.00847	3.00000	-0.0910	0.2971	0.4847
5.500	0.9925	0.00874	3.00000	-0.0918	0.2881	0.4868
6.000	1.0541	0.00896	3.00000	-0.0926	0.2753	0.4933
6.500	1.1144	0.00928	3.00000	-0.0933	0.2665	0.4986
7.000	1.1749	0.00956	3.00000	-0.0939	0.2529	0.5023
7.500	1.2329	0.01004	3.00000	-0.0942	0.2345	0.5050
8.000	1.2897	0.01059	3.00000	-0.0943	0.2203	0.5072
8.500	1.3475	0.01101	3.00000	-0.0945	0.2070	0.5120
9.000	1.4032	0.01156	3.00000	-0.0944	0.1952	0.5178
9.500	1.4579	0.01212	3.00000	-0.0941	0.1833	0.5223
10.000	1.5078	0.01297	3.00000	-0.0932	0.1628	0.5253
10.500	1.5569	0.01379	3.00000	-0.0921	0.1479	0.5277
11.000	1.6001	0.01487	3.00000	-0.0902	0.1353	0.5307
11.500	1.6392	0.01593	3.00000	-0.0877	0.1197	0.5370
12.000	1.6559	0.01744	3.00000	-0.0818	0.1063	0.5418
12.500	1.6663	0.01982	3.00000	-0.0767	0.0962	0.5451
13.000	1.6783	0.02266	3.00000	-0.0732	0.0830	0.5477
13.500	1.6756	0.02739	3.00000	-0.0706	0.0768	0.5495

XFOIL Version 5.0

Calculated polar for: FFA-W1-242

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
1.000	0.3807	0.01306	3.00000	-0.0710	0.0100	0.1000
1.500	0.4444	0.01313	3.00000	-0.0727	0.0100	0.1000
2.000	0.5076	0.01323	3.00000	-0.0742	0.0100	0.1000
2.500	0.5702	0.01336	3.00000	-0.0757	0.0100	0.1000
3.000	0.6323	0.01353	3.00000	-0.0770	0.0100	0.1000
3.500	0.6938	0.01372	3.00000	-0.0782	0.0100	0.1000
4.000	0.7546	0.01395	3.00000	-0.0792	0.0100	0.1000
4.500	0.8148	0.01421	3.00000	-0.0801	0.0100	0.1000
5.000	0.8742	0.01452	3.00000	-0.0809	0.0100	0.1000
5.500	0.9327	0.01485	3.00000	-0.0815	0.0100	0.1000
6.000	0.9902	0.01523	3.00000	-0.0820	0.0100	0.1000
6.500	1.0467	0.01565	3.00000	-0.0823	0.0100	0.1000
7.000	1.1018	0.01612	3.00000	-0.0823	0.0100	0.1000
7.500	1.1555	0.01664	3.00000	-0.0822	0.0100	0.1000
8.000	1.2073	0.01723	3.00000	-0.0818	0.0100	0.1000
8.500	1.2568	0.01789	3.00000	-0.0810	0.0100	0.1000
9.000	1.3027	0.01864	3.00000	-0.0798	0.0100	0.1000
9.500	1.3268	0.01958	3.00000	-0.0749	0.0100	0.1000
10.000	1.3576	0.02094	3.00000	-0.0721	0.0100	0.1000
10.500	1.3838	0.02276	3.00000	-0.0696	0.0100	0.1000
11.000	1.4051	0.02523	3.00000	-0.0675	0.0100	0.1000
11.500	1.4242	0.02839	3.00000	-0.0663	0.0100	0.1000
12.000	1.4404	0.03228	3.00000	-0.0659	0.0100	0.1000
12.500	1.4544	0.03680	3.00000	-0.0660	0.0100	0.1000
13.000	1.4673	0.04181	3.00000	-0.0665	0.0100	0.1000
13.500	1.4785	0.04724	3.00000	-0.0673	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W1-271

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-7.00000	-0.72140	0.01512	1.00000	-0.03120	0.62840	0.39270
-6.50000	-0.65000	0.01432	1.00000	-0.03610	0.60940	0.39670
-6.00000	-0.57800	0.01371	1.00000	-0.04070	0.59570	0.39970
-5.50000	-0.50610	0.01322	1.00000	-0.04490	0.57890	0.40240
-5.00000	-0.43450	0.01283	1.00000	-0.04890	0.56090	0.40460
-4.50000	-0.35940	0.01220	1.00000	-0.05410	0.54910	0.41080
-4.00000	-0.28620	0.01178	1.00000	-0.05840	0.53300	0.41600
-3.50000	-0.21560	0.01156	1.00000	-0.06170	0.51940	0.41930
-3.00000	-0.14640	0.01143	1.00000	-0.06460	0.50610	0.42190
-2.50000	-0.07770	0.01135	1.00000	-0.06720	0.49270	0.42430
-2.00000	-0.01000	0.01134	1.00000	-0.06960	0.48010	0.42660
-1.50000	0.06000	0.01113	1.00000	-0.07270	0.46520	0.43250
-1.00000	0.12840	0.01105	1.00000	-0.07530	0.45510	0.43750
-0.50000	0.19570	0.01107	1.00000	-0.07750	0.44020	0.44080
0.00000	0.26250	0.01109	1.00000	-0.07960	0.43250	0.44350
1.000	0.3948	0.01117	1.00000	-0.0834	0.4075	0.4513
1.500	0.4606	0.01127	1.00000	-0.0852	0.3985	0.4539
2.000	0.5259	0.01139	1.00000	-0.0869	0.3888	0.4564
2.500	0.5912	0.01152	1.00000	-0.0885	0.3772	0.4601
3.000	0.6569	0.01159	1.00000	-0.0903	0.3692	0.4667
3.500	0.7213	0.01178	1.00000	-0.0918	0.3570	0.4706
4.000	0.7853	0.01197	1.00000	-0.0931	0.3498	0.4736
4.500	0.8486	0.01222	1.00000	-0.0943	0.3383	0.4763
5.000	0.9112	0.01248	1.00000	-0.0953	0.3302	0.4787
5.500	0.9742	0.01270	1.00000	-0.0965	0.3213	0.4851
6.000	1.0360	0.01300	1.00000	-0.0975	0.3105	0.4904
6.500	1.0970	0.01334	1.00000	-0.0982	0.3035	0.4938
7.000	1.1564	0.01376	1.00000	-0.0986	0.2904	0.4968
7.500	1.2153	0.01418	1.00000	-0.0990	0.2835	0.4995
8.000	1.2728	0.01466	1.00000	-0.0991	0.2699	0.5034
8.500	1.3293	0.01515	1.00000	-0.0992	0.2628	0.5107
9.000	1.3830	0.01578	1.00000	-0.0987	0.2488	0.5145
9.500	1.4335	0.01650	1.00000	-0.0978	0.2410	0.5176
10.000	1.4796	0.01737	1.00000	-0.0961	0.2268	0.5205
10.500	1.5205	0.01832	1.00000	-0.0938	0.2200	0.5229
11.000	1.5398	0.01975	1.00000	-0.0882	0.2064	0.5304
11.500	1.5628	0.02166	1.00000	-0.0844	0.1999	0.5346
12.000	1.5818	0.02418	1.00000	-0.0811	0.1869	0.5377
12.500	1.5926	0.02773	1.00000	-0.0784	0.1805	0.5404
13.000	1.6013	0.03204	1.00000	-0.0766	0.1674	0.5428
13.500	1.6028	0.03749	1.00000	-0.0755	0.1591	0.5448
14.000	1.6059	0.04339	1.00000	-0.0754	0.1471	0.5512
14.500	1.6026	0.05040	1.00000	-0.0757	0.1386	0.5553
15.000	1.5988	0.05772	1.00000	-0.0763	0.1282	0.5584
15.500	1.5903	0.06611	1.00000	-0.0775	0.1190	0.5608
16.000	1.5819	0.07473	1.00000	-0.0790	0.1106	0.5630
16.500	1.5706	0.08414	1.00000	-0.0809	0.1020	0.5649
17.000	1.5593	0.09396	1.00000	-0.0832	0.0949	0.5666
17.500	1.5452	0.10462	1.00000	-0.0860	0.0872	0.5712
18.000	1.5348	0.11502	1.00000	-0.0889	0.0808	0.5755

XFOIL Version 5.0

Calculated polar for: FFA-W1-271

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
1.000	0.2907	0.01960	1.00000	-0.0556	0.0100	0.1000
1.500	0.3617	0.01928	1.00000	-0.0600	0.0100	0.1000
2.000	0.4317	0.01907	1.00000	-0.0639	0.0100	0.1000
2.500	0.5010	0.01895	1.00000	-0.0676	0.0100	0.1000
3.000	0.5689	0.01891	1.00000	-0.0709	0.0100	0.1000
3.500	0.6353	0.01897	1.00000	-0.0738	0.0100	0.1000
4.000	0.6994	0.01914	1.00000	-0.0760	0.0100	0.1000
4.500	0.7611	0.01942	1.00000	-0.0778	0.0100	0.1000
5.000	0.8209	0.01978	1.00000	-0.0791	0.0100	0.1000
5.500	0.8785	0.02023	1.00000	-0.0800	0.0100	0.1000
6.000	0.9337	0.02076	1.00000	-0.0805	0.0100	0.1000
6.500	0.9859	0.02139	1.00000	-0.0805	0.0100	0.1000
7.000	1.0331	0.02215	1.00000	-0.0797	0.0100	0.1000
7.500	1.0554	0.02318	1.00000	-0.0749	0.0100	0.1000
8.000	1.0878	0.02470	1.00000	-0.0728	0.0100	0.1000
8.500	1.1167	0.02669	1.00000	-0.0710	0.0100	0.1000
9.000	1.1427	0.02923	1.00000	-0.0699	0.0100	0.1000
9.500	1.1662	0.03231	1.00000	-0.0693	0.0100	0.1000
10.000	1.1880	0.03593	1.00000	-0.0692	0.0100	0.1000
10.500	1.2076	0.04009	1.00000	-0.0695	0.0100	0.1000
11.000	1.2256	0.04468	1.00000	-0.0701	0.0100	0.1000
11.500	1.2416	0.04981	1.00000	-0.0711	0.0100	0.1000
12.000	1.2562	0.05527	1.00000	-0.0721	0.0100	0.1000
12.500	1.2687	0.06119	1.00000	-0.0734	0.0100	0.1000
13.000	1.2794	0.06759	1.00000	-0.0750	0.0100	0.1000
13.500	1.2888	0.07434	1.00000	-0.0766	0.0100	0.1000
14.000	1.2965	0.08143	1.00000	-0.0784	0.0100	0.1000
14.500	1.3033	0.08886	1.00000	-0.0804	0.0100	0.1000
15.000	1.3093	0.09658	1.00000	-0.0825	0.0100	0.1000
15.500	1.3139	0.10472	1.00000	-0.0849	0.0100	0.1000
16.000	1.3169	0.11334	1.00000	-0.0874	0.0100	0.1000
16.500	1.3194	0.12231	1.00000	-0.0901	0.0100	0.1000
17.000	1.3223	0.13139	1.00000	-0.0930	0.0100	0.1000
17.500	1.3238	0.14096	1.00000	-0.0960	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W1-271

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 2.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-9.50000	-1.04890	0.01594	2.00000	-0.01930	0.67180	0.35700
-9.00000	-0.98040	0.01480	2.00000	-0.02420	0.65260	0.36150
-8.50000	-0.90910	0.01378	2.00000	-0.02940	0.63090	0.36540
-8.00000	-0.83380	0.01263	2.00000	-0.03560	0.61750	0.37530
-7.50000	-0.76010	0.01189	2.00000	-0.04060	0.60100	0.37890
-7.00000	-0.68780	0.01139	2.00000	-0.04490	0.58100	0.38130
-6.50000	-0.61630	0.01099	2.00000	-0.04870	0.56740	0.38340
-6.00000	-0.54190	0.01042	2.00000	-0.05340	0.55170	0.38800
-5.50000	-0.46840	0.00995	2.00000	-0.05770	0.53450	0.39380
-5.00000	-0.39800	0.00970	2.00000	-0.06090	0.52450	0.39790
-4.50000	-0.32890	0.00955	2.00000	-0.06370	0.50800	0.40080
-4.00000	-0.26070	0.00947	2.00000	-0.06620	0.49560	0.40320
-3.50000	-0.19280	0.00941	2.00000	-0.06850	0.48350	0.40480
-3.00000	-0.12420	0.00931	2.00000	-0.07110	0.46780	0.40860
-2.50000	-0.05560	0.00920	2.00000	-0.07370	0.45840	0.41430
-2.00000	0.01220	0.00915	2.00000	-0.07600	0.44500	0.41860
-1.50000	0.07930	0.00915	2.00000	-0.07810	0.43420	0.42210
-1.00000	0.14620	0.00917	2.00000	-0.08010	0.42090	0.42430
-0.50000	0.21270	0.00922	2.00000	-0.08200	0.41250	0.42600
0.00000	0.27920	0.00927	2.00000	-0.08390	0.39910	0.42830
0.50000	0.34630	0.00927	2.00000	-0.08590	0.39030	0.43460
1.000	0.4125	0.00934	2.00000	-0.0878	0.3792	0.4393
1.500	0.4784	0.00941	2.00000	-0.0895	0.3688	0.4433
2.000	0.5440	0.00952	2.00000	-0.0911	0.3620	0.4457
2.500	0.6091	0.00967	2.00000	-0.0926	0.3474	0.4476
3.000	0.6742	0.00980	2.00000	-0.0941	0.3424	0.4497
3.500	0.7395	0.00991	2.00000	-0.0957	0.3329	0.4564
4.000	0.8042	0.01008	2.00000	-0.0971	0.3224	0.4614
4.500	0.8682	0.01027	2.00000	-0.0984	0.3157	0.4654
5.000	0.9318	0.01050	2.00000	-0.0995	0.3020	0.4679
5.500	0.9950	0.01073	2.00000	-0.1006	0.2970	0.4699
6.000	1.0578	0.01100	2.00000	-0.1016	0.2849	0.4742
6.500	1.1201	0.01128	2.00000	-0.1026	0.2763	0.4804
7.000	1.1816	0.01161	2.00000	-0.1033	0.2676	0.4850
7.500	1.2422	0.01197	2.00000	-0.1039	0.2562	0.4888
8.000	1.3009	0.01245	2.00000	-0.1042	0.2438	0.4911
8.500	1.3589	0.01293	2.00000	-0.1043	0.2331	0.4932
9.000	1.4156	0.01346	2.00000	-0.1043	0.2196	0.5005
9.500	1.4695	0.01412	2.00000	-0.1038	0.2096	0.5058
10.000	1.5222	0.01474	2.00000	-0.1031	0.1954	0.5094
10.500	1.5690	0.01563	2.00000	-0.1015	0.1878	0.5125
11.000	1.6124	0.01654	2.00000	-0.0993	0.1742	0.5145
11.500	1.6315	0.01796	2.00000	-0.0934	0.1630	0.5199
12.000	1.6504	0.01988	2.00000	-0.0885	0.1528	0.5254
12.500	1.6630	0.02258	2.00000	-0.0842	0.1411	0.5293
13.000	1.6687	0.02640	2.00000	-0.0809	0.1330	0.5322
13.500	1.6745	0.03084	2.00000	-0.0788	0.1205	0.5347
14.000	1.6701	0.03690	2.00000	-0.0776	0.1143	0.5363
14.500	1.6662	0.04348	2.00000	-0.0772	0.1019	0.5397
15.000	1.6584	0.05103	2.00000	-0.0775	0.0959	0.5448

XFOIL Version 5.0

Calculated polar for: FFA-W1-271

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-10.00000	-1.10610	0.01429	3.00000	-0.02240	0.66520	0.33580
-9.50000	-1.03130	0.01266	3.00000	-0.02950	0.64250	0.34650
-9.00000	-0.95650	0.01164	3.00000	-0.03530	0.62460	0.35550
-8.50000	-0.88120	0.01075	3.00000	-0.04090	0.60700	0.36540
-8.00000	-0.80790	0.01015	3.00000	-0.04540	0.58770	0.36860
-7.00000	-0.66580	0.00943	3.00000	-0.05260	0.55710	0.37320
-6.50000	-0.59410	0.00905	3.00000	-0.05620	0.54310	0.37940
-6.00000	-0.52450	0.00884	3.00000	-0.05920	0.52950	0.38480
-5.50000	-0.45570	0.00868	3.00000	-0.06180	0.51760	0.38920
-5.00000	-0.38730	0.00855	3.00000	-0.06440	0.50130	0.39290
-4.50000	-0.31950	0.00848	3.00000	-0.06670	0.48990	0.39500
-4.00000	-0.25210	0.00844	3.00000	-0.06890	0.47530	0.39640
-3.50000	-0.18520	0.00844	3.00000	-0.07090	0.46400	0.39740
-3.00000	-0.11690	0.00834	3.00000	-0.07330	0.44830	0.40280
-2.50000	-0.04940	0.00828	3.00000	-0.07550	0.44170	0.40810
-2.00000	0.01770	0.00827	3.00000	-0.07760	0.42460	0.41380
-1.50000	0.08460	0.00826	3.00000	-0.07960	0.41680	0.41710
-1.00000	0.15120	0.00830	3.00000	-0.08150	0.40690	0.41950
-0.50000	0.21760	0.00834	3.00000	-0.08330	0.39310	0.42120
0.00000	0.28350	0.00842	3.00000	-0.08510	0.38590	0.42240
0.50000	0.34980	0.00847	3.00000	-0.08690	0.37430	0.42540
1.000	0.4152	0.00853	3.00000	-0.0885	0.3630	0.4311
1.500	0.4810	0.00861	3.00000	-0.0902	0.3569	0.4360
2.000	0.5467	0.00869	3.00000	-0.0918	0.3457	0.4398
2.500	0.6120	0.00881	3.00000	-0.0933	0.3367	0.4430
3.000	0.6769	0.00896	3.00000	-0.0948	0.3290	0.4448
3.500	0.7418	0.00910	3.00000	-0.0961	0.3172	0.4461
4.000	0.8064	0.00926	3.00000	-0.0976	0.3106	0.4517
4.500	0.8708	0.00943	3.00000	-0.0989	0.3013	0.4573
5.000	0.9348	0.00962	3.00000	-0.1001	0.2900	0.4619
5.500	0.9982	0.00985	3.00000	-0.1012	0.2840	0.4651
6.000	1.0612	0.01009	3.00000	-0.1022	0.2698	0.4679
6.500	1.1236	0.01037	3.00000	-0.1031	0.2642	0.4694
7.000	1.1853	0.01070	3.00000	-0.1039	0.2524	0.4751
7.500	1.2465	0.01104	3.00000	-0.1045	0.2413	0.4811
8.000	1.3061	0.01149	3.00000	-0.1049	0.2285	0.4858
8.500	1.3645	0.01197	3.00000	-0.1052	0.2164	0.4891
9.000	1.4221	0.01247	3.00000	-0.1052	0.2048	0.4914
9.500	1.4763	0.01315	3.00000	-0.1048	0.1917	0.4941
10.000	1.5309	0.01372	3.00000	-0.1043	0.1773	0.5009
10.500	1.5791	0.01459	3.00000	-0.1029	0.1681	0.5066
11.000	1.6257	0.01540	3.00000	-0.1012	0.1527	0.5106
11.500	1.6607	0.01662	3.00000	-0.0979	0.1435	0.5134
12.000	1.6756	0.01811	3.00000	-0.0913	0.1311	0.5152
12.500	1.6831	0.02069	3.00000	-0.0855	0.1236	0.5166
13.000	1.6956	0.02362	3.00000	-0.0819	0.1109	0.5240
13.500	1.6943	0.02826	3.00000	-0.0788	0.1054	0.5287

XFOIL Version 5.0

Calculated polar for: FFA-W1-271

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
1.000	0.3427	0.01544	3.00000	-0.0698	0.0100	0.1000
1.500	0.4129	0.01527	3.00000	-0.0733	0.0100	0.1000
2.000	0.4819	0.01518	3.00000	-0.0765	0.0100	0.1000
2.500	0.5488	0.01521	3.00000	-0.0790	0.0100	0.1000
3.000	0.6144	0.01531	3.00000	-0.0811	0.0100	0.1000
3.500	0.6790	0.01545	3.00000	-0.0830	0.0100	0.1000
4.000	0.7427	0.01565	3.00000	-0.0847	0.0100	0.1000
4.500	0.8053	0.01588	3.00000	-0.0861	0.0100	0.1000
5.000	0.8668	0.01616	3.00000	-0.0874	0.0100	0.1000
5.500	0.9271	0.01649	3.00000	-0.0883	0.0100	0.1000
6.000	0.9861	0.01687	3.00000	-0.0891	0.0100	0.1000
6.500	1.0435	0.01731	3.00000	-0.0895	0.0100	0.1000
7.000	1.0991	0.01780	3.00000	-0.0897	0.0100	0.1000
7.500	1.1524	0.01837	3.00000	-0.0894	0.0100	0.1000
8.000	1.2025	0.01902	3.00000	-0.0887	0.0100	0.1000
8.500	1.2467	0.01978	3.00000	-0.0870	0.0100	0.1000
9.000	1.2631	0.02093	3.00000	-0.0809	0.0100	0.1000
9.500	1.2914	0.02254	3.00000	-0.0778	0.0100	0.1000
10.000	1.3144	0.02469	3.00000	-0.0751	0.0100	0.1000
10.500	1.3356	0.02742	3.00000	-0.0733	0.0100	0.1000
11.000	1.3551	0.03075	3.00000	-0.0721	0.0100	0.1000
11.500	1.3719	0.03466	3.00000	-0.0715	0.0100	0.1000
12.000	1.3874	0.03916	3.00000	-0.0715	0.0100	0.1000
12.500	1.4028	0.04395	3.00000	-0.0717	0.0100	0.1000
13.000	1.4147	0.04946	3.00000	-0.0724	0.0100	0.1000
13.500	1.4251	0.05528	3.00000	-0.0732	0.0100	0.1000

ISES polar driver Version 1.3

Calculated polar for: AIRFOIL FFA-W2-152

Vortex + doublet far field
Entropy conserved

Mach = 0.150 Re = 3.000 e 6 Acrit = 9.000

alpha	CL	CD	CDi	CM	S xtr	P xtr
4.000	0.7762	0.00514	0.00000	-0.0587	0.4816	0.8451
4.500	0.8346	0.00530	0.00000	-0.0596	0.4663	0.8602
5.000	0.8927	0.00542	0.00000	-0.0604	0.4548	0.8756
5.500	0.9501	0.00562	0.00000	-0.0612	0.4378	0.8901
6.000	1.0066	0.00592	0.00000	-0.0619	0.4162	0.9033
6.250	1.0343	0.00611	0.00000	-0.0622	0.4017	0.9098
6.500	1.0615	0.00635	0.00000	-0.0625	0.3830	0.9161
6.750	1.0875	0.00672	0.00000	-0.0626	0.3558	0.9233
7.000	1.1109	0.00727	0.00000	-0.0625	0.3183	0.9320
7.250	1.1320	0.00797	0.00000	-0.0621	0.2710	0.9417
7.500	1.1518	0.00866	0.00000	-0.0615	0.2295	0.9527
7.750	1.1686	0.00935	0.00000	-0.0604	0.1936	0.9702
8.000	1.1910	0.01001	0.00000	-0.0606	0.1607	1.0000
8.250	1.2093	0.01072	0.00000	-0.0600	0.1345	1.0000
8.500	1.2262	0.01143	0.00000	-0.0594	0.1133	1.0000
8.750	1.2402	0.01221	0.00000	-0.0583	0.0959	1.0000
9.000	1.2398	0.01326	0.00000	-0.0555	0.0854	1.0000
9.250	1.2422	0.01453	0.00000	-0.0544	0.0759	1.0000
9.500	1.2419	0.01601	0.00000	-0.0534	0.0685	1.0000
9.750	1.2384	0.01762	0.00000	-0.0512	0.0618	1.0000
10.000	1.2358	0.01949	0.00000	-0.0501	0.0552	1.0000
10.250	1.2311	0.02121	0.00000	-0.0474	0.0503	1.0000
10.500	1.2327	0.02313	0.00000	-0.0483	0.0452	1.0000
10.667	1.2280	0.02439	0.00000	-0.0468	0.0424	1.0000

ISES polar driver Version 1.3

Calculated polar for: AIRFOIL FFA-W2-152

Vortex + doublet far field
Entropy conserved

Mach = 0.150 Re = 3.000 e 6 Acrit = 9.000

alpha	CL	CD	CDi	CM	S xtr	P xtr
0.000	0.2925	0.00988	0.00000	-0.0506	0.0100	0.1000
0.500	0.3498	0.01001	0.00000	-0.0514	0.0100	0.1000
1.000	0.4073	0.01017	0.00000	-0.0521	0.0100	0.1000
1.500	0.4649	0.01033	0.00000	-0.0529	0.0100	0.1000
2.000	0.5212	0.01052	0.00000	-0.0536	0.0100	0.1000
2.500	0.5776	0.01073	0.00000	-0.0543	0.0100	0.1000
3.000	0.6344	0.01096	0.00000	-0.0549	0.0100	0.1000
3.500	0.6900	0.01121	0.00000	-0.0555	0.0100	0.1000
4.000	0.7449	0.01150	0.00000	-0.0560	0.0100	0.1000
4.500	0.7990	0.01181	0.00000	-0.0565	0.0100	0.1000
5.000	0.8522	0.01216	0.00000	-0.0568	0.0100	0.1000
5.500	0.9042	0.01255	0.00000	-0.0571	0.0100	0.1000
6.000	0.9545	0.01299	0.00000	-0.0572	0.0100	0.1000
6.250	0.9784	0.01323	0.00000	-0.0572	0.0100	0.1000
6.500	1.0033	0.01349	0.00000	-0.0571	0.0100	0.1000
6.750	1.0266	0.01377	0.00000	-0.0569	0.0100	0.1000
7.000	1.0490	0.01407	0.00000	-0.0567	0.0100	0.1000
7.250	1.0697	0.01443	0.00000	-0.0564	0.0100	0.1000
7.500	1.0817	0.01500	0.00000	-0.0548	0.0100	0.1000
7.750	1.0933	0.01572	0.00000	-0.0542	0.0100	0.1000
8.000	1.1031	0.01653	0.00000	-0.0531	0.0100	0.1000
8.250	1.1123	0.01746	0.00000	-0.0520	0.0100	0.1000
8.500	1.1200	0.01849	0.00000	-0.0510	0.0100	0.1000
8.750	1.1267	0.01959	0.00000	-0.0495	0.0100	0.1000
9.000	1.1328	0.02082	0.00000	-0.0485	0.0100	0.1000
9.250	1.1416	0.02211	0.00000	-0.0493	0.0100	0.1000
9.500	1.1416	0.02349	0.00000	-0.0466	0.0100	0.1000
9.750	1.1460	0.02486	0.00000	-0.0448	0.0100	0.1000
10.000	1.1549	0.02627	0.00000	-0.0470	0.0100	0.1000
10.250	1.1559	0.02775	0.00000	-0.0440	0.0100	0.1000
10.417	1.1607	0.02869	0.00000	-0.0435	0.0100	0.1000
10.583	1.1582	0.02976	0.00000	-0.0415	0.0100	0.1000
10.750	1.1623	0.03071	0.00000	-0.0410	0.0100	0.1000
10.917	1.1693	0.03183	0.00000	-0.0431	0.0100	0.1000
11.083	1.1703	0.03287	0.00000	-0.0416	0.0100	0.1000
11.194	1.1673	0.03352	0.00000	-0.0381	0.0100	0.1000
11.333	1.1791	0.03455	0.00000	-0.0435	0.0100	0.1000
11.444	1.1726	0.03522	0.00000	-0.0396	0.0100	0.1000
11.583	1.1762	0.03605	0.00000	-0.0395	0.0100	0.1000
11.750	1.1861	0.03731	0.00000	-0.0435	0.0100	0.1000
11.861	1.1736	0.03806	0.00000	-0.0350	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W2-152

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-5.50000	-0.34150	0.01138	1.00000	-0.04450	0.62310	0.02420
-5.00000	-0.28400	0.01092	1.00000	-0.04480	0.61530	0.02750
-4.50000	-0.22570	0.01064	1.00000	-0.04530	0.61020	0.02940
-3.50000	-0.10920	0.00971	1.00000	-0.04600	0.59910	0.04600
-3.00000	-0.05040	0.00939	1.00000	-0.04650	0.59310	0.06500
-2.50000	0.00840	0.00905	1.00000	-0.04720	0.58900	0.10810
-2.00000	0.06590	0.00837	1.00000	-0.04800	0.58390	0.23140
-1.50000	0.12350	0.00774	1.00000	-0.04880	0.57650	0.38070
-1.00000	0.18150	0.00709	1.00000	-0.04960	0.57270	0.55010
0.00000	0.29950	0.00660	1.00000	-0.05110	0.56170	0.73480
0.50000	0.35820	0.00657	1.00000	-0.05160	0.55650	0.78560
1.00000	0.41730	0.00658	1.00000	-0.05210	0.55260	0.81310
1.50000	0.47650	0.00663	1.00000	-0.05280	0.54740	0.83360
2.000	0.5352	0.00675	1.00000	-0.0534	0.5401	0.8518
2.500	0.5939	0.00680	1.00000	-0.0539	0.5360	0.8686
3.000	0.6523	0.00687	1.00000	-0.0544	0.5311	0.8841
3.500	0.7102	0.00700	1.00000	-0.0549	0.5233	0.8988
4.000	0.7671	0.00710	1.00000	-0.0551	0.5186	0.9144
4.500	0.8236	0.00719	1.00000	-0.0552	0.5129	0.9286
5.000	0.8792	0.00735	1.00000	-0.0552	0.5032	0.9439
5.500	0.9342	0.00740	1.00000	-0.0551	0.4921	0.9616
6.000	0.9913	0.00755	1.00000	-0.0555	0.4793	0.9847
6.500	1.0608	0.00783	1.00000	-0.0588	0.4639	1.0000
7.000	1.1198	0.00820	1.00000	-0.0601	0.4446	1.0000
7.500	1.1752	0.00893	1.00000	-0.0609	0.4034	1.0000
8.000	1.2189	0.01069	1.00000	-0.0605	0.3125	1.0000
8.500	1.2407	0.01390	1.00000	-0.0582	0.1910	1.0000
9.000	1.2517	0.01707	1.00000	-0.0553	0.1336	1.0000
9.500	1.2603	0.02100	1.00000	-0.0535	0.0982	1.0000
10.000	1.2676	0.02540	1.00000	-0.0522	0.0741	1.0000
10.500	1.2761	0.02993	1.00000	-0.0511	0.0547	1.0000
11.000	1.2854	0.03450	1.00000	-0.0502	0.0398	1.0000
11.500	1.2980	0.03893	1.00000	-0.0494	0.0285	1.0000
12.000	1.3100	0.04354	1.00000	-0.0489	0.0220	1.0000
12.500	1.3238	0.04814	1.00000	-0.0485	0.0175	1.0000
13.000	1.3363	0.05294	1.00000	-0.0482	0.0158	1.0000
13.500	1.3476	0.05812	1.00000	-0.0481	0.0133	1.0000
14.000	1.3435	0.06542	1.00000	-0.0479	0.0100	1.0000
14.500	1.3574	0.07060	1.00000	-0.0484	0.0097	1.0000
15.000	1.3684	0.07639	1.00000	-0.0490	0.0092	1.0000
15.500	1.3770	0.08268	1.00000	-0.0499	0.0086	1.0000
16.000	1.3826	0.08955	1.00000	-0.0509	0.0079	1.0000
16.500	1.3821	0.09776	1.00000	-0.0524	0.0070	1.0000
17.000	1.3654	0.10927	1.00000	-0.0548	0.0059	1.0000

XFOIL Version 5.0

Calculated polar for: FFA-W2-152

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-5.50000	-0.34150	0.01138	1.00000	-0.04450	0.62310	0.02420
-5.00000	-0.28400	0.01092	1.00000	-0.04480	0.61530	0.02750
-4.50000	-0.22570	0.01064	1.00000	-0.04530	0.61020	0.02940
-3.00000	-0.05860	0.01192	1.00000	-0.04690	0.01000	0.06350
-2.50000	-0.00090	0.01171	1.00000	-0.04740	0.01000	0.10000
-2.00000	0.05770	0.01176	1.00000	-0.04800	0.01000	0.10000
-1.50000	0.11630	0.01183	1.00000	-0.04860	0.01000	0.10000
-1.00000	0.17460	0.01193	1.00000	-0.04930	0.01000	0.10000
0.00000	0.29090	0.01218	1.00000	-0.05060	0.01000	0.10000
0.50000	0.34870	0.01235	1.00000	-0.05120	0.01000	0.10000
1.00000	0.40620	0.01254	1.00000	-0.05190	0.01000	0.10000
1.50000	0.46340	0.01276	1.00000	-0.05250	0.01000	0.10000
2.000	0.5202	0.01301	1.00000	-0.0531	0.0100	0.1000
2.500	0.5765	0.01329	1.00000	-0.0537	0.0100	0.1000
3.000	0.6322	0.01361	1.00000	-0.0542	0.0100	0.1000
3.500	0.6873	0.01397	1.00000	-0.0546	0.0100	0.1000
4.000	0.7416	0.01437	1.00000	-0.0550	0.0100	0.1000
4.500	0.7949	0.01483	1.00000	-0.0552	0.0100	0.1000
5.000	0.8471	0.01534	1.00000	-0.0554	0.0100	0.1000
5.500	0.8977	0.01593	1.00000	-0.0554	0.0100	0.1000
6.000	0.9463	0.01660	1.00000	-0.0552	0.0100	0.1000
6.500	0.9916	0.01742	1.00000	-0.0547	0.0100	0.1000
7.000	1.0236	0.01869	1.00000	-0.0529	0.0100	0.1000
7.500	1.0584	0.02034	1.00000	-0.0523	0.0100	0.1000
8.000	1.0905	0.02230	1.00000	-0.0516	0.0100	0.1000
8.500	1.1202	0.02456	1.00000	-0.0509	0.0100	0.1000
9.000	1.1481	0.02707	1.00000	-0.0503	0.0100	0.1000
9.500	1.1743	0.02983	1.00000	-0.0497	0.0100	0.1000
10.000	1.1989	0.03279	1.00000	-0.0491	0.0100	0.1000
10.500	1.2221	0.03598	1.00000	-0.0485	0.0100	0.1000
11.000	1.2439	0.03939	1.00000	-0.0480	0.0100	0.1000
11.500	1.2644	0.04302	1.00000	-0.0475	0.0100	0.1000
12.000	1.2841	0.04686	1.00000	-0.0472	0.0100	0.1000
12.500	1.3025	0.05093	1.00000	-0.0469	0.0100	0.1000
13.000	1.3197	0.05524	1.00000	-0.0467	0.0100	0.1000
13.500	1.3347	0.05995	1.00000	-0.0466	0.0100	0.1000
14.500	1.3486	0.07190	1.00000	-0.0469	0.0097	0.1000
15.000	1.3599	0.07761	1.00000	-0.0476	0.0092	0.1000
15.500	1.3685	0.08385	1.00000	-0.0484	0.0087	0.1000
16.000	1.3745	0.09066	1.00000	-0.0495	0.0080	0.1000
16.500	1.3751	0.09869	1.00000	-0.0510	0.0071	0.1000
17.000	1.3610	0.10968	1.00000	-0.0533	0.0060	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W2-152

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 2.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.40310	0.01070	2.00000	-0.04370	0.60550	0.01690
-5.00000	-0.28940	0.00905	2.00000	-0.04370	0.59320	0.02400
-4.50000	-0.23040	0.00872	2.00000	-0.04420	0.58910	0.02830
-4.00000	-0.17100	0.00852	2.00000	-0.04490	0.58320	0.02990
-3.50000	-0.11150	0.00825	2.00000	-0.04540	0.57630	0.03890
-3.00000	-0.05200	0.00800	2.00000	-0.04610	0.57300	0.04630
-2.50000	0.00770	0.00777	2.00000	-0.04680	0.56860	0.07230
-2.00000	0.06700	0.00744	2.00000	-0.04760	0.56100	0.12050
-1.50000	0.12590	0.00694	2.00000	-0.04850	0.55690	0.22860
-1.00000	0.18510	0.00648	2.00000	-0.04950	0.55280	0.34160
0.00000	0.30380	0.00565	2.00000	-0.05150	0.54040	0.59990
0.50000	0.36390	0.00542	2.00000	-0.05250	0.53710	0.68670
1.00000	0.42400	0.00531	2.00000	-0.05340	0.53300	0.75220
1.50000	0.48430	0.00533	2.00000	-0.05430	0.52610	0.78050
2.000	0.5445	0.00537	2.00000	-0.0552	0.5206	0.8045
2.500	0.6046	0.00541	2.00000	-0.0561	0.5164	0.8237
3.000	0.6647	0.00548	2.00000	-0.0571	0.5101	0.8384
3.500	0.7245	0.00558	2.00000	-0.0579	0.5024	0.8526
4.000	0.7842	0.00565	2.00000	-0.0588	0.4978	0.8681
4.500	0.8432	0.00580	2.00000	-0.0596	0.4825	0.8834
5.000	0.9019	0.00595	2.00000	-0.0603	0.4706	0.8982
5.500	0.9603	0.00614	2.00000	-0.0610	0.4592	0.9090
6.000	1.0179	0.00638	2.00000	-0.0616	0.4419	0.9229
6.500	1.0730	0.00689	2.00000	-0.0619	0.4045	0.9370
7.000	1.1206	0.00810	2.00000	-0.0613	0.3235	0.9548
7.500	1.1611	0.00957	2.00000	-0.0598	0.2393	0.9882
8.000	1.2036	0.01182	2.00000	-0.0601	0.1447	1.0000
8.500	1.2436	0.01344	2.00000	-0.0593	0.0971	1.0000
9.000	1.2670	0.01568	2.00000	-0.0572	0.0716	1.0000
9.500	1.2868	0.01857	2.00000	-0.0558	0.0513	1.0000
10.000	1.3028	0.02207	2.00000	-0.0546	0.0366	1.0000
10.500	1.3179	0.02585	2.00000	-0.0535	0.0266	1.0000
11.000	1.3367	0.02941	2.00000	-0.0527	0.0181	1.0000
11.500	1.3503	0.03356	2.00000	-0.0518	0.0159	1.0000
12.000	1.3673	0.03753	2.00000	-0.0512	0.0127	1.0000
12.500	1.3823	0.04177	2.00000	-0.0505	0.0100	1.0000
13.000	1.3984	0.04604	2.00000	-0.0501	0.0096	1.0000
13.500	1.4118	0.05068	2.00000	-0.0498	0.0090	1.0000
14.000	1.4230	0.05573	2.00000	-0.0496	0.0081	1.0000
15.000	1.4209	0.06945	2.00000	-0.0495	0.0050	1.0000
15.500	1.4333	0.07494	2.00000	-0.0501	0.0049	1.0000
16.000	1.4447	0.08069	2.00000	-0.0508	0.0048	1.0000
16.500	1.4530	0.08714	2.00000	-0.0518	0.0047	1.0000
17.000	1.4596	0.09405	2.00000	-0.0531	0.0046	1.0000

XFOIL Version 5.0

Calculated polar for: FFA-W2-152

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.40860	0.00931	3.00000	-0.04270	0.59210	0.01700
-5.50000	-0.34890	0.00926	3.00000	-0.04340	0.58740	0.01730
-5.00000	-0.29120	0.00859	3.00000	-0.04360	0.58020	0.01870
-4.50000	-0.23240	0.00801	3.00000	-0.04380	0.57580	0.02600
-4.00000	-0.17280	0.00777	3.00000	-0.04440	0.57170	0.02950
-3.50000	-0.11300	0.00757	3.00000	-0.04510	0.56640	0.03340
-3.00000	-0.05290	0.00741	3.00000	-0.04590	0.55890	0.04250
-2.50000	0.00700	0.00721	3.00000	-0.04660	0.55590	0.05670
-2.00000	0.06680	0.00695	3.00000	-0.04750	0.55210	0.09130
-1.50000	0.12630	0.00659	3.00000	-0.04840	0.54510	0.15950
-1.00000	0.18590	0.00620	3.00000	-0.04930	0.53960	0.25890
0.00000	0.30530	0.00536	3.00000	-0.05150	0.53170	0.50480
0.50000	0.36540	0.00510	3.00000	-0.05250	0.52480	0.60310
1.00000	0.42570	0.00490	3.00000	-0.05360	0.52000	0.69130
1.500	0.4862	0.00483	3.00000	-0.0550	0.5170	0.7400
2.000	0.5468	0.00484	3.00000	-0.0556	0.5099	0.7721
2.500	0.6074	0.00489	3.00000	-0.0566	0.5025	0.7923
3.000	0.6678	0.00494	3.00000	-0.0576	0.4984	0.8117
3.500	0.7281	0.00501	3.00000	-0.0586	0.4912	0.8292
4.000	0.7881	0.00512	3.00000	-0.0596	0.4810	0.8451
4.500	0.8477	0.00528	3.00000	-0.0605	0.4648	0.8590
5.000	0.9070	0.00547	3.00000	-0.0614	0.4479	0.8709
5.500	0.9656	0.00570	3.00000	-0.0622	0.4301	0.8871
6.000	1.0229	0.00611	3.00000	-0.0629	0.4025	0.9008
7.000	1.1210	0.00878	3.00000	-0.0628	0.2220	0.9267
7.500	1.1659	0.01014	3.00000	-0.0621	0.1566	0.9448
8.000	1.2045	0.01152	3.00000	-0.0603	0.1005	0.9770
8.500	1.2500	0.01286	3.00000	-0.0602	0.0693	1.0000
9.000	1.2872	0.01429	3.00000	-0.0591	0.0469	1.0000
9.500	1.3082	0.01676	3.00000	-0.0572	0.0328	1.0000
10.000	1.3267	0.01987	3.00000	-0.0559	0.0242	1.0000
10.500	1.3487	0.02293	3.00000	-0.0550	0.0174	1.0000
11.000	1.3660	0.02650	3.00000	-0.0540	0.0142	1.0000
11.500	1.3872	0.02986	3.00000	-0.0532	0.0106	1.0000
12.000	1.4040	0.03370	3.00000	-0.0525	0.0097	1.0000
12.500	1.4197	0.03776	3.00000	-0.0517	0.0089	1.0000
13.000	1.4344	0.04201	3.00000	-0.0511	0.0079	1.0000
13.500	1.4372	0.04772	3.00000	-0.0502	0.0051	1.0000
14.000	1.4520	0.05223	3.00000	-0.0500	0.0050	1.0000
15.000	1.4792	0.06202	3.00000	-0.0500	0.0048	1.0000
15.500	1.4906	0.06726	3.00000	-0.0502	0.0046	1.0000
16.000	1.5006	0.07294	3.00000	-0.0506	0.0044	1.0000
16.500	1.5086	0.07916	3.00000	-0.0513	0.0043	1.0000
17.000	1.5151	0.08575	3.00000	-0.0522	0.0040	1.0000

XFOIL Version 5.0

Calculated polar for: FFA-W2-152

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
2.000	0.5297	0.01063	3.00000	-0.0540	0.0100	0.1000
2.500	0.5877	0.01085	3.00000	-0.0547	0.0100	0.1000
3.000	0.6452	0.01109	3.00000	-0.0554	0.0100	0.1000
3.500	0.7024	0.01136	3.00000	-0.0561	0.0100	0.1000
4.000	0.7591	0.01166	3.00000	-0.0567	0.0100	0.1000
4.500	0.8152	0.01198	3.00000	-0.0573	0.0100	0.1000
5.000	0.8706	0.01234	3.00000	-0.0578	0.0100	0.1000
5.500	0.9253	0.01274	3.00000	-0.0582	0.0100	0.1000
6.000	0.9790	0.01319	3.00000	-0.0585	0.0100	0.1000
6.500	1.0316	0.01368	3.00000	-0.0587	0.0100	0.1000
7.000	1.0826	0.01424	3.00000	-0.0587	0.0100	0.1000
7.500	1.1317	0.01488	3.00000	-0.0585	0.0100	0.1000
8.000	1.1770	0.01566	3.00000	-0.0579	0.0100	0.1000
8.500	1.2069	0.01698	3.00000	-0.0558	0.0100	0.1000
9.000	1.2401	0.01866	3.00000	-0.0549	0.0100	0.1000
9.500	1.2708	0.02069	3.00000	-0.0541	0.0100	0.1000
10.000	1.2991	0.02303	3.00000	-0.0533	0.0100	0.1000
10.500	1.3253	0.02564	3.00000	-0.0525	0.0100	0.1000
11.000	1.3497	0.02854	3.00000	-0.0517	0.0100	0.1000
11.500	1.3724	0.03166	3.00000	-0.0510	0.0100	0.1000
12.000	1.3920	0.03517	3.00000	-0.0503	0.0098	0.1000
12.500	1.4075	0.03915	3.00000	-0.0495	0.0090	0.1000
13.000	1.4223	0.04336	3.00000	-0.0489	0.0081	0.1000
15.000	1.4673	0.06317	3.00000	-0.0479	0.0048	0.1000
15.500	1.4788	0.06844	3.00000	-0.0481	0.0047	0.1000
16.000	1.4897	0.07395	3.00000	-0.0485	0.0045	0.1000
16.500	1.4980	0.08011	3.00000	-0.0493	0.0043	0.1000
17.000	1.5045	0.08665	3.00000	-0.0502	0.0041	0.1000
1.500	0.4715	0.01044	3.00000	-0.0532	0.0100	0.1000
1.000	0.4130	0.01026	3.00000	-0.0524	0.0100	0.1000
0.500	0.3543	0.01010	3.00000	-0.0516	0.0100	0.1000
0.000	0.2954	0.00996	3.00000	-0.0509	0.0100	0.1000
-1.000	0.1771	0.00974	3.00000	-0.0493	0.0100	0.1000
-1.500	0.1178	0.00965	3.00000	-0.0485	0.0100	0.1000
-2.000	0.0586	0.00964	3.00000	-0.0478	0.0100	0.0858
-2.500	-0.0002	0.00979	3.00000	-0.0470	0.0100	0.0547
-3.000	-0.0593	0.00987	3.00000	-0.0464	0.0100	0.0418
-3.500	-0.1130	0.00757	3.00000	-0.0451	0.5664	0.0334
-4.000	-0.1728	0.00777	3.00000	-0.0444	0.5717	0.0295
-4.500	-0.2324	0.00801	3.00000	-0.0438	0.5758	0.0260
-5.000	-0.2912	0.00859	3.00000	-0.0436	0.5802	0.0187
-5.500	-0.3489	0.00926	3.00000	-0.0434	0.5874	0.0173
-6.000	-0.4086	0.00931	3.00000	-0.0427	0.5921	0.0170

XFOIL Version 5.0

Calculated polar for: FFA-W2-210

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.56900	0.00889	1.00000	-0.02470	0.58470	0.38780
-5.50000	-0.50680	0.00858	1.00000	-0.02610	0.57560	0.41480
-5.00000	-0.44450	0.00838	1.00000	-0.02740	0.56830	0.44320
-4.50000	-0.38180	0.00829	1.00000	-0.02860	0.56010	0.45450
-4.00000	-0.31910	0.00821	1.00000	-0.02990	0.55160	0.46140
-3.50000	-0.25640	0.00813	1.00000	-0.03120	0.54570	0.47010
-3.00000	-0.19370	0.00812	1.00000	-0.03250	0.53570	0.47530
-2.50000	-0.13100	0.00805	1.00000	-0.03370	0.53040	0.48390
-2.00000	-0.06820	0.00804	1.00000	-0.03500	0.52400	0.49010
-1.50000	-0.00570	0.00807	1.00000	-0.03630	0.51540	0.49470
-1.00000	0.05700	0.00803	1.00000	-0.03760	0.51000	0.50330
0.00000	0.18190	0.00813	1.00000	-0.04000	0.49560	0.51380
0.50000	0.24440	0.00812	1.00000	-0.04130	0.49010	0.52170
1.00000	0.30680	0.00819	1.00000	-0.04250	0.48300	0.52780
1.50000	0.36890	0.00829	1.00000	-0.04370	0.47570	0.53220
2.000	0.4310	0.00833	1.00000	-0.0449	0.4707	0.5403
2.500	0.4930	0.00845	1.00000	-0.0461	0.4622	0.5459
3.000	0.5547	0.00857	1.00000	-0.0472	0.4564	0.5504
3.500	0.6164	0.00866	1.00000	-0.0483	0.4505	0.5579
4.000	0.6777	0.00884	1.00000	-0.0493	0.4407	0.5639
4.500	0.7390	0.00897	1.00000	-0.0503	0.4356	0.5688
5.000	0.8001	0.00912	1.00000	-0.0514	0.4280	0.5755
5.500	0.8608	0.00932	1.00000	-0.0523	0.4185	0.5818
6.000	0.9211	0.00956	1.00000	-0.0532	0.4039	0.5865
6.500	0.9810	0.00979	1.00000	-0.0540	0.3942	0.5930
7.000	1.0402	0.01011	1.00000	-0.0548	0.3791	0.5998
7.500	1.0982	0.01053	1.00000	-0.0554	0.3596	0.6047
8.000	1.1550	0.01103	1.00000	-0.0558	0.3399	0.6109
8.500	1.2092	0.01173	1.00000	-0.0560	0.3108	0.6180
9.000	1.2532	0.01324	1.00000	-0.0552	0.2510	0.6229
9.500	1.2873	0.01530	1.00000	-0.0535	0.1925	0.6279
10.000	1.3040	0.01779	1.00000	-0.0500	0.1483	0.6361
10.500	1.3074	0.02169	1.00000	-0.0478	0.1168	0.6411
11.000	1.3093	0.02668	1.00000	-0.0472	0.0964	0.6454
11.500	1.3084	0.03238	1.00000	-0.0470	0.0780	0.6534
12.000	1.3150	0.03755	1.00000	-0.0470	0.0653	0.6587
12.500	1.3185	0.04319	1.00000	-0.0470	0.0590	0.6636
13.000	1.3269	0.04857	1.00000	-0.0473	0.0497	0.6714
13.500	1.3325	0.05433	1.00000	-0.0477	0.0472	0.6770
14.000	1.3375	0.06036	1.00000	-0.0483	0.0434	0.6817
14.500	1.3451	0.06631	1.00000	-0.0491	0.0384	0.6902
15.000	1.3482	0.07298	1.00000	-0.0500	0.0341	0.6961
15.500	1.3567	0.07911	1.00000	-0.0511	0.0331	0.7011
16.000	1.3633	0.08569	1.00000	-0.0525	0.0318	0.7098
16.500	1.3676	0.09268	1.00000	-0.0539	0.0301	0.7168
17.000	1.3706	0.10024	1.00000	-0.0558	0.0280	0.7242

XFOIL Version 5.0

Calculated polar for: FFA-W2-210

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-4.00000	-0.32280	0.01375	1.00000	-0.02660	0.01000	0.10000
-3.50000	-0.26190	0.01362	1.00000	-0.02790	0.01000	0.10000
-3.00000	-0.20100	0.01352	1.00000	-0.02910	0.01000	0.10000
-2.50000	-0.14000	0.01345	1.00000	-0.03040	0.01000	0.10000
-2.00000	-0.07900	0.01341	1.00000	-0.03170	0.01000	0.10000
-1.50000	-0.01810	0.01339	1.00000	-0.03290	0.01000	0.10000
-1.00000	0.04280	0.01341	1.00000	-0.03420	0.01000	0.10000
0.00000	0.16410	0.01351	1.00000	-0.03650	0.01000	0.10000
0.50000	0.22450	0.01361	1.00000	-0.03770	0.01000	0.10000
1.00000	0.28460	0.01373	1.00000	-0.03880	0.01000	0.10000
1.50000	0.34440	0.01389	1.00000	-0.03990	0.01000	0.10000
2.000	0.4039	0.01408	1.00000	-0.0409	0.0100	0.1000
2.500	0.4630	0.01430	1.00000	-0.0418	0.0100	0.1000
3.000	0.5215	0.01455	1.00000	-0.0427	0.0100	0.1000
3.500	0.5795	0.01485	1.00000	-0.0436	0.0100	0.1000
4.000	0.6369	0.01518	1.00000	-0.0443	0.0100	0.1000
4.500	0.6934	0.01556	1.00000	-0.0449	0.0100	0.1000
5.000	0.7490	0.01600	1.00000	-0.0455	0.0100	0.1000
5.500	0.8035	0.01649	1.00000	-0.0459	0.0100	0.1000
6.000	0.8566	0.01705	1.00000	-0.0461	0.0100	0.1000
6.500	0.9078	0.01770	1.00000	-0.0462	0.0100	0.1000
7.000	0.9567	0.01845	1.00000	-0.0460	0.0100	0.1000
7.500	1.0021	0.01934	1.00000	-0.0456	0.0100	0.1000
8.000	1.0307	0.02068	1.00000	-0.0433	0.0100	0.1000
8.500	1.0623	0.02274	1.00000	-0.0430	0.0100	0.1000
9.000	1.0903	0.02534	1.00000	-0.0431	0.0100	0.1000
9.500	1.1153	0.02837	1.00000	-0.0432	0.0100	0.1000
10.000	1.1379	0.03179	1.00000	-0.0434	0.0100	0.1000
10.500	1.1579	0.03554	1.00000	-0.0435	0.0100	0.1000
11.000	1.1767	0.03953	1.00000	-0.0438	0.0100	0.1000
11.500	1.1936	0.04381	1.00000	-0.0440	0.0100	0.1000
12.000	1.2095	0.04835	1.00000	-0.0445	0.0100	0.1000
12.500	1.2242	0.05310	1.00000	-0.0449	0.0100	0.1000
13.000	1.2378	0.05808	1.00000	-0.0455	0.0100	0.1000
13.500	1.2511	0.06325	1.00000	-0.0462	0.0100	0.1000
14.000	1.2635	0.06859	1.00000	-0.0470	0.0100	0.1000
14.500	1.2751	0.07424	1.00000	-0.0480	0.0100	0.1000
15.000	1.2862	0.08015	1.00000	-0.0492	0.0100	0.1000
15.500	1.2969	0.08621	1.00000	-0.0504	0.0100	0.1000
16.000	1.3074	0.09237	1.00000	-0.0518	0.0100	0.1000
16.500	1.3166	0.09902	1.00000	-0.0534	0.0100	0.1000
17.000	1.3250	0.10595	1.00000	-0.0553	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W2-210

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 2.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.56740	0.00789	2.00000	-0.02500	0.55540	0.32140
-5.50000	-0.50520	0.00740	2.00000	-0.02650	0.54900	0.36100
-5.00000	-0.44280	0.00702	2.00000	-0.02800	0.54140	0.39900
-4.50000	-0.38010	0.00678	2.00000	-0.02940	0.53340	0.42890
-4.00000	-0.31720	0.00670	2.00000	-0.03070	0.52820	0.43830
-3.50000	-0.25410	0.00664	2.00000	-0.03200	0.51840	0.44840
-3.00000	-0.19110	0.00659	2.00000	-0.03330	0.51310	0.45450
-2.50000	-0.12820	0.00656	2.00000	-0.03450	0.50760	0.46050
-2.00000	-0.06510	0.00655	2.00000	-0.03590	0.49730	0.46910
-1.50000	-0.00220	0.00653	2.00000	-0.03710	0.49330	0.47480
-1.00000	0.06070	0.00654	2.00000	-0.03840	0.48760	0.48040
0.00000	0.18640	0.00657	2.00000	-0.04100	0.47410	0.49360
0.50000	0.24920	0.00660	2.00000	-0.04230	0.46830	0.49990
1.00000	0.31190	0.00666	2.00000	-0.04350	0.45900	0.50730
1.50000	0.37450	0.00670	2.00000	-0.04480	0.45510	0.51240
2.000	0.4370	0.00677	2.00000	-0.0460	0.4481	0.5175
2.500	0.4995	0.00685	2.00000	-0.0472	0.4400	0.5252
3.000	0.5618	0.00692	2.00000	-0.0484	0.4353	0.5306
3.500	0.6239	0.00703	2.00000	-0.0496	0.4272	0.5349
4.000	0.6860	0.00714	2.00000	-0.0507	0.4196	0.5438
4.500	0.7477	0.00728	2.00000	-0.0518	0.4121	0.5487
5.000	0.8092	0.00745	2.00000	-0.0528	0.4009	0.5523
5.500	0.8703	0.00764	2.00000	-0.0539	0.3866	0.5616
6.000	0.9310	0.00788	2.00000	-0.0548	0.3759	0.5668
6.500	0.9911	0.00818	2.00000	-0.0557	0.3590	0.5705
7.000	1.0497	0.00862	2.00000	-0.0564	0.3341	0.5793
7.500	1.1074	0.00913	2.00000	-0.0570	0.3068	0.5848
8.000	1.1593	0.01020	2.00000	-0.0571	0.2557	0.5886
8.500	1.2073	0.01153	2.00000	-0.0567	0.2031	0.5957
9.000	1.2507	0.01309	2.00000	-0.0559	0.1516	0.6024
9.500	1.2898	0.01475	2.00000	-0.0546	0.1125	0.6065
10.000	1.3209	0.01661	2.00000	-0.0525	0.0809	0.6125
10.500	1.3372	0.01898	2.00000	-0.0497	0.0659	0.6192
11.000	1.3513	0.02258	2.00000	-0.0489	0.0576	0.6243
11.500	1.3674	0.02642	2.00000	-0.0487	0.0491	0.6289
12.000	1.3765	0.03113	2.00000	-0.0485	0.0453	0.6361
12.500	1.3889	0.03570	2.00000	-0.0484	0.0398	0.6410
13.000	1.4021	0.04029	2.00000	-0.0484	0.0346	0.6453
13.500	1.4086	0.04573	2.00000	-0.0486	0.0333	0.6530
14.000	1.4147	0.05133	2.00000	-0.0488	0.0315	0.6584
14.500	1.4205	0.05714	2.00000	-0.0492	0.0294	0.6625
15.000	1.4292	0.06282	2.00000	-0.0498	0.0269	0.6689
15.500	1.4364	0.06879	2.00000	-0.0505	0.0245	0.6763
16.000	1.4382	0.07568	2.00000	-0.0515	0.0219	0.6807
16.500	1.4408	0.08270	2.00000	-0.0527	0.0208	0.6869

XFOIL Version 5.0

Calculated polar for: FFA-W2-210

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.56650	0.00789	3.00000	-0.02460	0.54130	0.25420
-5.50000	-0.50460	0.00709	3.00000	-0.02650	0.53350	0.32020
-5.00000	-0.44190	0.00669	3.00000	-0.02800	0.52840	0.35650
-4.50000	-0.37920	0.00628	3.00000	-0.02960	0.51970	0.40000
-4.00000	-0.31630	0.00609	3.00000	-0.03100	0.51320	0.42730
-3.50000	-0.25320	0.00603	3.00000	-0.03230	0.50760	0.43580
-3.00000	-0.19000	0.00599	3.00000	-0.03360	0.49810	0.44020
-2.50000	-0.12690	0.00597	3.00000	-0.03490	0.49350	0.44990
-2.00000	-0.06380	0.00595	3.00000	-0.03620	0.48750	0.45600
-1.50000	-0.00080	0.00595	3.00000	-0.03750	0.47840	0.46130
-1.00000	0.06230	0.00594	3.00000	-0.03880	0.47420	0.47060
0.00000	0.18820	0.00597	3.00000	-0.04140	0.45940	0.48160
0.50000	0.25120	0.00600	3.00000	-0.04260	0.45530	0.48920
1.00000	0.31400	0.00604	3.00000	-0.04390	0.44930	0.49420
1.50000	0.37670	0.00609	3.00000	-0.04520	0.44010	0.49930
2.000	0.4394	0.00613	3.00000	-0.0464	0.4365	0.5080
2.500	0.5020	0.00621	3.00000	-0.0477	0.4290	0.5124
3.000	0.5644	0.00630	3.00000	-0.0489	0.4205	0.5166
3.500	0.6268	0.00638	3.00000	-0.0501	0.4154	0.5257
4.000	0.6890	0.00649	3.00000	-0.0512	0.4051	0.5305
4.500	0.7509	0.00663	3.00000	-0.0523	0.3979	0.5352
5.000	0.8126	0.00680	3.00000	-0.0534	0.3836	0.5435
5.500	0.8736	0.00703	3.00000	-0.0545	0.3662	0.5486
6.000	0.9341	0.00731	3.00000	-0.0554	0.3499	0.5526
6.500	0.9935	0.00773	3.00000	-0.0562	0.3242	0.5613
7.000	1.0520	0.00821	3.00000	-0.0570	0.2999	0.5667
7.500	1.1084	0.00890	3.00000	-0.0575	0.2641	0.5707
8.000	1.1578	0.01026	3.00000	-0.0573	0.1934	0.5786
8.500	1.2061	0.01155	3.00000	-0.0570	0.1486	0.5844
9.000	1.2525	0.01283	3.00000	-0.0564	0.1106	0.5883
9.500	1.2944	0.01428	3.00000	-0.0553	0.0787	0.5954
10.000	1.3364	0.01552	3.00000	-0.0543	0.0628	0.6019
10.500	1.3668	0.01684	3.00000	-0.0517	0.0504	0.6060
11.000	1.3828	0.01974	3.00000	-0.0499	0.0462	0.6104
11.500	1.4023	0.02303	3.00000	-0.0495	0.0403	0.6189
12.000	1.4227	0.02648	3.00000	-0.0492	0.0346	0.6235
12.500	1.4320	0.03115	3.00000	-0.0490	0.0330	0.6266
13.000	1.4401	0.03614	3.00000	-0.0489	0.0309	0.6340
13.500	1.4489	0.04121	3.00000	-0.0489	0.0285	0.6397
14.000	1.4602	0.04613	3.00000	-0.0490	0.0259	0.6441
14.500	1.4705	0.05126	3.00000	-0.0491	0.0236	0.6499
15.000	1.4770	0.05698	3.00000	-0.0494	0.0213	0.6567
16.000	1.4913	0.06870	3.00000	-0.0508	0.0203	0.6665
16.500	1.4955	0.07521	3.00000	-0.0518	0.0196	0.6743
17.000	1.4989	0.08205	3.00000	-0.0530	0.0188	0.6792

XFOIL Version 5.0

Calculated polar for: FFA-W2-210

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.56350	0.00981	3.00000	-0.02280	0.54110	0.10000
-5.50000	-0.50160	0.00959	3.00000	-0.02410	0.53340	0.10000
-5.00000	-0.43950	0.00938	3.00000	-0.02540	0.52830	0.10000
-4.50000	-0.37730	0.00920	3.00000	-0.02670	0.51950	0.10000
-4.00000	-0.32040	0.01106	3.00000	-0.02830	0.01000	0.10000
-3.50000	-0.25860	0.01100	3.00000	-0.02960	0.01000	0.10000
-3.00000	-0.19680	0.01095	3.00000	-0.03090	0.01000	0.10000
-2.50000	-0.13500	0.01092	3.00000	-0.03210	0.01000	0.10000
-2.00000	-0.07330	0.01091	3.00000	-0.03340	0.01000	0.10000
-1.50000	-0.01160	0.01092	3.00000	-0.03460	0.01000	0.10000
-1.00000	0.05000	0.01095	3.00000	-0.03590	0.01000	0.10000
0.00000	0.17290	0.01106	3.00000	-0.03830	0.01000	0.10000
0.50000	0.23410	0.01115	3.00000	-0.03950	0.01000	0.10000
1.00000	0.29520	0.01126	3.00000	-0.04070	0.01000	0.10000
1.50000	0.35600	0.01139	3.00000	-0.04180	0.01000	0.10000
2.000	0.4167	0.01154	3.00000	-0.0429	0.0100	0.1000
2.500	0.4770	0.01171	3.00000	-0.0439	0.0100	0.1000
3.000	0.5370	0.01191	3.00000	-0.0449	0.0100	0.1000
3.500	0.5966	0.01214	3.00000	-0.0459	0.0100	0.1000
4.000	0.6558	0.01239	3.00000	-0.0468	0.0100	0.1000
4.500	0.7145	0.01267	3.00000	-0.0476	0.0100	0.1000
5.000	0.7726	0.01299	3.00000	-0.0483	0.0100	0.1000
5.500	0.8301	0.01334	3.00000	-0.0490	0.0100	0.1000
6.000	0.8868	0.01372	3.00000	-0.0496	0.0100	0.1000
6.500	0.9425	0.01416	3.00000	-0.0500	0.0100	0.1000
7.000	0.9972	0.01464	3.00000	-0.0503	0.0100	0.1000
7.500	1.0504	0.01519	3.00000	-0.0504	0.0100	0.1000
8.000	1.1020	0.01581	3.00000	-0.0504	0.0100	0.1000
8.500	1.1512	0.01652	3.00000	-0.0501	0.0100	0.1000
9.000	1.1968	0.01735	3.00000	-0.0494	0.0100	0.1000
9.500	1.2229	0.01868	3.00000	-0.0466	0.0100	0.1000
10.000	1.2518	0.02081	3.00000	-0.0459	0.0100	0.1000
10.500	1.2775	0.02352	3.00000	-0.0457	0.0100	0.1000
11.000	1.3002	0.02672	3.00000	-0.0456	0.0100	0.1000
11.500	1.3205	0.03031	3.00000	-0.0456	0.0100	0.1000
12.000	1.3382	0.03423	3.00000	-0.0456	0.0100	0.1000
12.500	1.3539	0.03852	3.00000	-0.0458	0.0100	0.1000
13.000	1.3685	0.04296	3.00000	-0.0459	0.0100	0.1000
13.500	1.3821	0.04769	3.00000	-0.0461	0.0100	0.1000
14.000	1.3941	0.05267	3.00000	-0.0465	0.0100	0.1000
14.500	1.4060	0.05780	3.00000	-0.0469	0.0100	0.1000
15.000	1.4176	0.06311	3.00000	-0.0475	0.0100	0.1000
15.500	1.4276	0.06873	3.00000	-0.0483	0.0100	0.1000
16.000	1.4367	0.07464	3.00000	-0.0493	0.0100	0.1000
16.500	1.4456	0.08075	3.00000	-0.0504	0.0100	0.1000
17.000	1.4549	0.08681	3.00000	-0.0515	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: 50% NACA 63-618 AND 50% FFA-W3-2

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.31600	0.00898	1.00000	-0.08630	0.64380	0.28500
-5.50000	-0.25110	0.00810	1.00000	-0.08900	0.63350	0.36880
-5.00000	-0.18780	0.00777	1.00000	-0.09060	0.62090	0.41160
-4.50000	-0.12520	0.00766	1.00000	-0.09190	0.60720	0.43110
-4.00000	-0.06280	0.00761	1.00000	-0.09300	0.59740	0.44630
-3.50000	-0.00080	0.00764	1.00000	-0.09410	0.58440	0.46670
-3.00000	0.06130	0.00766	1.00000	-0.09510	0.57570	0.47960
-2.50000	0.12320	0.00772	1.00000	-0.09610	0.56290	0.48520
-2.00000	0.18500	0.00777	1.00000	-0.09710	0.55430	0.49050
-1.00000	0.30850	0.00788	1.00000	-0.09920	0.53430	0.50330
-0.50000	0.36980	0.00798	1.00000	-0.10010	0.52360	0.50760
0.00000	0.43130	0.00803	1.00000	-0.10110	0.51560	0.51480
0.50000	0.49230	0.00815	1.00000	-0.10210	0.50480	0.52040
1.00000	0.55340	0.00825	1.00000	-0.10290	0.49860	0.52450
1.500	0.6140	0.00839	1.00000	-0.1038	0.4882	0.5283
2.000	0.6748	0.00848	1.00000	-0.1047	0.4811	0.5356
2.500	0.7351	0.00863	1.00000	-0.1055	0.4730	0.5405
3.000	0.7949	0.00882	1.00000	-0.1061	0.4637	0.5448
3.500	0.8547	0.00898	1.00000	-0.1068	0.4571	0.5484
4.000	0.9141	0.00918	1.00000	-0.1075	0.4450	0.5560
4.500	0.9734	0.00936	1.00000	-0.1081	0.4386	0.5612
5.000	1.0319	0.00962	1.00000	-0.1085	0.4262	0.5654
5.500	1.0902	0.00985	1.00000	-0.1090	0.4186	0.5697
6.000	1.1479	0.01015	1.00000	-0.1093	0.4043	0.5775
6.500	1.2048	0.01049	1.00000	-0.1096	0.3895	0.5823
7.000	1.2600	0.01093	1.00000	-0.1095	0.3705	0.5870
7.500	1.3144	0.01140	1.00000	-0.1094	0.3542	0.5942
8.000	1.3650	0.01211	1.00000	-0.1087	0.3244	0.6005
8.500	1.4114	0.01305	1.00000	-0.1075	0.2941	0.6054
9.000	1.4494	0.01440	1.00000	-0.1051	0.2498	0.6122
9.500	1.4633	0.01651	1.00000	-0.0994	0.2002	0.6194
10.000	1.4730	0.01912	1.00000	-0.0943	0.1625	0.6246
10.500	1.4735	0.02289	1.00000	-0.0899	0.1273	0.6300
11.000	1.4767	0.02716	1.00000	-0.0870	0.1028	0.6390
11.500	1.4804	0.03198	1.00000	-0.0851	0.0832	0.6447
12.000	1.4821	0.03741	1.00000	-0.0838	0.0706	0.6509
12.500	1.4872	0.04293	1.00000	-0.0831	0.0602	0.6594
13.000	1.4934	0.04852	1.00000	-0.0826	0.0510	0.6657
13.500	1.4962	0.05487	1.00000	-0.0825	0.0459	0.6749
14.000	1.5008	0.06120	1.00000	-0.0827	0.0416	0.6828
14.500	1.5068	0.06753	1.00000	-0.0829	0.0361	0.6904
15.000	1.5112	0.07427	1.00000	-0.0835	0.0343	0.7008

XFOIL Version 5.0

Calculated polar for: 50% NACA 63-618 AND 50% FFA-W3-2

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
0.000	0.3879	0.01340	1.00000	-0.0928	0.0100	0.1000
0.500	0.4470	0.01357	1.00000	-0.0936	0.0100	0.1000
1.000	0.5057	0.01376	1.00000	-0.0943	0.0100	0.1000
1.500	0.5638	0.01398	1.00000	-0.0950	0.0100	0.1000
2.000	0.6214	0.01423	1.00000	-0.0955	0.0100	0.1000
2.500	0.6784	0.01452	1.00000	-0.0960	0.0100	0.1000
3.000	0.7347	0.01485	1.00000	-0.0963	0.0100	0.1000
3.500	0.7902	0.01521	1.00000	-0.0966	0.0100	0.1000
4.000	0.8448	0.01561	1.00000	-0.0967	0.0100	0.1000
4.500	0.8983	0.01607	1.00000	-0.0966	0.0100	0.1000
5.000	0.9504	0.01658	1.00000	-0.0964	0.0100	0.1000
5.500	1.0010	0.01714	1.00000	-0.0959	0.0100	0.1000
6.000	1.0493	0.01778	1.00000	-0.0952	0.0100	0.1000
6.500	1.0895	0.01850	1.00000	-0.0930	0.0100	0.1000
7.000	1.1265	0.01942	1.00000	-0.0907	0.0100	0.1000
7.500	1.1632	0.02062	1.00000	-0.0889	0.0100	0.1000
8.000	1.1961	0.02214	1.00000	-0.0871	0.0100	0.1000
8.500	1.2260	0.02402	1.00000	-0.0854	0.0100	0.1000
9.000	1.2534	0.02631	1.00000	-0.0839	0.0100	0.1000
9.500	1.2783	0.02900	1.00000	-0.0827	0.0100	0.1000
10.000	1.3014	0.03210	1.00000	-0.0817	0.0100	0.1000
10.500	1.3221	0.03560	1.00000	-0.0810	0.0100	0.1000
11.000	1.3412	0.03949	1.00000	-0.0804	0.0100	0.1000
11.500	1.3587	0.04374	1.00000	-0.0801	0.0100	0.1000
12.000	1.3747	0.04828	1.00000	-0.0799	0.0100	0.1000
12.500	1.3893	0.05319	1.00000	-0.0799	0.0100	0.1000
13.000	1.4024	0.05846	1.00000	-0.0801	0.0100	0.1000
13.500	1.4140	0.06400	1.00000	-0.0804	0.0100	0.1000
14.000	1.4249	0.06981	1.00000	-0.0809	0.0100	0.1000
14.500	1.4351	0.07588	1.00000	-0.0815	0.0100	0.1000
15.000	1.4444	0.08218	1.00000	-0.0823	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: 50% NACA 63-618 AND 50% FFA-W3-2

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 2.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.31420	0.00899	2.00000	-0.08520	0.61020	0.13350
-5.50000	-0.24710	0.00775	2.00000	-0.08900	0.59580	0.24460
-5.00000	-0.18070	0.00679	2.00000	-0.09200	0.58570	0.34690
-4.50000	-0.11690	0.00649	2.00000	-0.09370	0.57460	0.39040
-4.00000	-0.05380	0.00636	2.00000	-0.09510	0.56350	0.41380
-3.50000	0.00890	0.00630	2.00000	-0.09630	0.55320	0.42680
-3.00000	0.07160	0.00623	2.00000	-0.09760	0.54100	0.45480
-2.50000	0.13400	0.00624	2.00000	-0.09870	0.53450	0.46630
-2.00000	0.19610	0.00630	2.00000	-0.09970	0.52060	0.47170
-1.50000	0.25840	0.00632	2.00000	-0.10090	0.51400	0.47960
-1.00000	0.32050	0.00637	2.00000	-0.10190	0.50640	0.48770
-0.50000	0.38230	0.00645	2.00000	-0.10300	0.49380	0.49260
0.00000	0.44400	0.00653	2.00000	-0.10390	0.48830	0.49580
0.50000	0.50580	0.00659	2.00000	-0.10500	0.48030	0.50310
1.00000	0.56740	0.00669	2.00000	-0.10600	0.46930	0.51050
1.50000	0.62880	0.00678	2.00000	-0.10690	0.46410	0.51470
2.000	0.6898	0.00691	2.00000	-0.1078	0.4568	0.5181
2.500	0.7506	0.00706	2.00000	-0.1086	0.4450	0.5204
3.000	0.8118	0.00715	2.00000	-0.1095	0.4398	0.5287
3.500	0.8723	0.00731	2.00000	-0.1103	0.4303	0.5353
4.000	0.9325	0.00748	2.00000	-0.1111	0.4198	0.5392
4.500	0.9923	0.00767	2.00000	-0.1117	0.4131	0.5421
5.000	1.0514	0.00791	2.00000	-0.1123	0.3967	0.5446
5.500	1.1100	0.00819	2.00000	-0.1128	0.3821	0.5530
6.000	1.1678	0.00851	2.00000	-0.1132	0.3665	0.5597
6.500	1.2240	0.00894	2.00000	-0.1133	0.3439	0.5638
7.000	1.2791	0.00944	2.00000	-0.1132	0.3213	0.5669
7.500	1.3313	0.01011	2.00000	-0.1128	0.2935	0.5717
8.000	1.3753	0.01135	2.00000	-0.1112	0.2404	0.5795
8.500	1.4163	0.01267	2.00000	-0.1093	0.1950	0.5858
9.000	1.4485	0.01436	2.00000	-0.1062	0.1424	0.5897
9.500	1.4691	0.01594	2.00000	-0.1010	0.1132	0.5926
10.000	1.4859	0.01802	2.00000	-0.0964	0.0910	0.6012
10.500	1.5001	0.02062	2.00000	-0.0926	0.0721	0.6079
11.000	1.5100	0.02403	2.00000	-0.0895	0.0610	0.6125
11.500	1.5178	0.02803	2.00000	-0.0870	0.0504	0.6165
12.000	1.5299	0.03212	2.00000	-0.0854	0.0440	0.6255
12.500	1.5454	0.03613	2.00000	-0.0843	0.0376	0.6328
13.000	1.5524	0.04125	2.00000	-0.0833	0.0355	0.6377
13.500	1.5591	0.04667	2.00000	-0.0826	0.0334	0.6423
14.000	1.5668	0.05225	2.00000	-0.0823	0.0309	0.6541
14.500	1.5747	0.05791	2.00000	-0.0821	0.0281	0.6603
15.000	1.5808	0.06403	2.00000	-0.0821	0.0252	0.6649
15.500	1.5838	0.07084	2.00000	-0.0825	0.0231	0.6764
16.000	1.5911	0.07721	2.00000	-0.0830	0.0228	0.6842
16.500	1.5953	0.08411	2.00000	-0.0838	0.0223	0.6905
17.000	1.5974	0.09131	2.00000	-0.0846	0.0214	0.7033

XFOIL Version 5.0

Calculated polar for: 50% NACA 63-618 AND 50% FFA-W3-2

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.31190	0.00868	3.00000	-0.08570	0.58450	0.09400
-5.50000	-0.24590	0.00770	3.00000	-0.08880	0.57640	0.17570
-5.00000	-0.18010	0.00688	3.00000	-0.09160	0.56370	0.26350
-4.50000	-0.11520	0.00631	3.00000	-0.09390	0.55550	0.32960
-4.00000	-0.05110	0.00596	3.00000	-0.09570	0.54280	0.38490
-3.50000	0.01190	0.00585	3.00000	-0.09700	0.53430	0.40630
-3.00000	0.07480	0.00577	3.00000	-0.09830	0.52360	0.42810
-2.50000	0.13730	0.00574	3.00000	-0.09950	0.51480	0.44750
-2.00000	0.19970	0.00574	3.00000	-0.10060	0.50410	0.46140
-1.50000	0.26190	0.00579	3.00000	-0.10170	0.49670	0.46710
-1.00000	0.32400	0.00583	3.00000	-0.10280	0.48510	0.47320
-0.50000	0.38620	0.00588	3.00000	-0.10380	0.48040	0.48090
0.000	0.4480	0.00595	3.00000	-0.1048	0.4696	0.4860
0.500	0.5097	0.00603	3.00000	-0.1058	0.4629	0.4893
1.000	0.5714	0.00611	3.00000	-0.1068	0.4555	0.4982
1.500	0.6329	0.00620	3.00000	-0.1078	0.4451	0.5038
2.000	0.6942	0.00631	3.00000	-0.1087	0.4402	0.5070
2.500	0.7552	0.00643	3.00000	-0.1096	0.4303	0.5113
3.000	0.8163	0.00654	3.00000	-0.1104	0.4227	0.5183
3.500	0.8769	0.00669	3.00000	-0.1112	0.4148	0.5230
4.000	0.9372	0.00686	3.00000	-0.1120	0.4032	0.5267
4.500	0.9971	0.00705	3.00000	-0.1126	0.3940	0.5292
5.000	1.0566	0.00727	3.00000	-0.1133	0.3806	0.5365
5.500	1.1148	0.00759	3.00000	-0.1137	0.3610	0.5419
6.000	1.1723	0.00796	3.00000	-0.1140	0.3411	0.5460
6.500	1.2276	0.00848	3.00000	-0.1140	0.3129	0.5485
7.000	1.2799	0.00923	3.00000	-0.1136	0.2748	0.5554
7.500	1.3298	0.01011	3.00000	-0.1129	0.2348	0.5614
8.000	1.3767	0.01114	3.00000	-0.1117	0.1906	0.5658
8.500	1.4162	0.01260	3.00000	-0.1095	0.1450	0.5685
9.000	1.4532	0.01403	3.00000	-0.1070	0.1070	0.5761
9.500	1.4833	0.01524	3.00000	-0.1031	0.0824	0.5821
10.000	1.5067	0.01684	3.00000	-0.0989	0.0651	0.5861
10.500	1.5233	0.01911	3.00000	-0.0948	0.0511	0.5911
11.000	1.5390	0.02186	3.00000	-0.0917	0.0448	0.5992
11.500	1.5589	0.02469	3.00000	-0.0896	0.0379	0.6044
12.000	1.5745	0.02818	3.00000	-0.0877	0.0344	0.6080
12.500	1.5864	0.03232	3.00000	-0.0862	0.0324	0.6175
13.000	1.5991	0.03670	3.00000	-0.0851	0.0296	0.6237
13.500	1.6132	0.04115	3.00000	-0.0843	0.0267	0.6278
14.000	1.6224	0.04635	3.00000	-0.0838	0.0245	0.6366
14.500	1.6311	0.05179	3.00000	-0.0835	0.0239	0.6445
15.000	1.6343	0.05809	3.00000	-0.0834	0.0225	0.6488

XFOIL Version 5.0

Calculated polar for: 50% NACA 63-618 AND 50% FFA-W3-2

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-4.50000	-0.14040	0.01058	3.00000	-0.08760	0.01000	0.10000
-4.00000	-0.07870	0.01055	3.00000	-0.08880	0.01000	0.10000
-3.50000	-0.01710	0.01054	3.00000	-0.08990	0.01000	0.10000
-3.00000	0.04440	0.01055	3.00000	-0.09100	0.01000	0.10000
-2.50000	0.10570	0.01058	3.00000	-0.09210	0.01000	0.10000
-2.00000	0.16690	0.01063	3.00000	-0.09320	0.01000	0.10000
-1.50000	0.22780	0.01070	3.00000	-0.09420	0.01000	0.10000
-1.00000	0.28850	0.01078	3.00000	-0.09520	0.01000	0.10000
-0.50000	0.34900	0.01088	3.00000	-0.09610	0.01000	0.10000
0.000	0.4093	0.01100	3.00000	-0.0970	0.0100	0.1000
0.500	0.4693	0.01114	3.00000	-0.0979	0.0100	0.1000
1.000	0.5289	0.01131	3.00000	-0.0987	0.0100	0.1000
1.500	0.5882	0.01149	3.00000	-0.0994	0.0100	0.1000
2.000	0.6471	0.01170	3.00000	-0.1001	0.0100	0.1000
2.500	0.7055	0.01193	3.00000	-0.1007	0.0100	0.1000
3.000	0.7635	0.01218	3.00000	-0.1013	0.0100	0.1000
3.500	0.8209	0.01246	3.00000	-0.1017	0.0100	0.1000
4.000	0.8777	0.01277	3.00000	-0.1021	0.0100	0.1000
4.500	0.9338	0.01311	3.00000	-0.1023	0.0100	0.1000
5.000	0.9890	0.01349	3.00000	-0.1024	0.0100	0.1000
5.500	1.0433	0.01390	3.00000	-0.1024	0.0100	0.1000
6.000	1.0965	0.01436	3.00000	-0.1022	0.0100	0.1000
6.500	1.1483	0.01486	3.00000	-0.1018	0.0100	0.1000
7.000	1.1984	0.01542	3.00000	-0.1011	0.0100	0.1000
7.500	1.2460	0.01604	3.00000	-0.1001	0.0100	0.1000
8.000	1.2809	0.01674	3.00000	-0.0968	0.0100	0.1000
8.500	1.3178	0.01770	3.00000	-0.0945	0.0100	0.1000
9.000	1.3509	0.01894	3.00000	-0.0920	0.0100	0.1000
9.500	1.3801	0.02055	3.00000	-0.0896	0.0100	0.1000
10.000	1.4063	0.02256	3.00000	-0.0874	0.0100	0.1000
10.500	1.4300	0.02503	3.00000	-0.0856	0.0100	0.1000
11.000	1.4527	0.02784	3.00000	-0.0840	0.0100	0.1000
11.500	1.4715	0.03120	3.00000	-0.0827	0.0100	0.1000
12.000	1.4900	0.03483	3.00000	-0.0817	0.0100	0.1000
12.500	1.5064	0.03891	3.00000	-0.0809	0.0100	0.1000
13.000	1.5200	0.04346	3.00000	-0.0803	0.0100	0.1000
13.500	1.5328	0.04829	3.00000	-0.0799	0.0100	0.1000
14.000	1.5449	0.05341	3.00000	-0.0797	0.0100	0.1000
14.500	1.5560	0.05882	3.00000	-0.0797	0.0100	0.1000
15.000	1.5657	0.06448	3.00000	-0.0799	0.0100	0.1000
15.500	1.5749	0.07028	3.00000	-0.0801	0.0100	0.1000
16.000	1.5833	0.07644	3.00000	-0.0806	0.0100	0.1000
16.500	1.5898	0.08311	3.00000	-0.0814	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W3-211

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.39160	0.00851	1.00000	-0.07080	0.57380	0.40900
-5.50000	-0.32740	0.00837	1.00000	-0.07240	0.55920	0.43050
-5.00000	-0.26440	0.00845	1.00000	-0.07340	0.54920	0.44730
-4.50000	-0.20040	0.00836	1.00000	-0.07490	0.53690	0.45490
-4.00000	-0.13690	0.00838	1.00000	-0.07620	0.52450	0.46020
-3.50000	-0.07380	0.00842	1.00000	-0.07730	0.51530	0.46430
-3.00000	-0.01050	0.00845	1.00000	-0.07860	0.50420	0.46780
-2.50000	0.05300	0.00842	1.00000	-0.07990	0.49640	0.47500
-2.00000	0.11590	0.00849	1.00000	-0.08110	0.48560	0.47900
-1.50000	0.17880	0.00856	1.00000	-0.08220	0.47830	0.48220
-1.00000	0.24140	0.00865	1.00000	-0.08330	0.47080	0.48510
-0.50000	0.30440	0.00867	1.00000	-0.08460	0.46190	0.49070
0.00000	0.36700	0.00875	1.00000	-0.08570	0.45560	0.49520
0.50000	0.42930	0.00886	1.00000	-0.08680	0.44720	0.49830
1.000	0.4915	0.00898	1.00000	-0.0879	0.4411	0.5012
1.500	0.5534	0.00911	1.00000	-0.0889	0.4352	0.5039
2.000	0.6155	0.00921	1.00000	-0.0900	0.4269	0.5086
2.500	0.6774	0.00933	1.00000	-0.0910	0.4218	0.5130
3.000	0.7390	0.00947	1.00000	-0.0919	0.4165	0.5162
3.500	0.8000	0.00967	1.00000	-0.0928	0.4081	0.5192
4.000	0.8609	0.00987	1.00000	-0.0936	0.4029	0.5218
4.500	0.9218	0.01003	1.00000	-0.0944	0.3979	0.5244
5.000	0.9825	0.01021	1.00000	-0.0953	0.3900	0.5302
5.500	1.0425	0.01045	1.00000	-0.0960	0.3835	0.5340
6.000	1.1025	0.01067	1.00000	-0.0966	0.3779	0.5369
6.500	1.1614	0.01099	1.00000	-0.0972	0.3668	0.5397
7.000	1.2207	0.01123	1.00000	-0.0977	0.3579	0.5422
7.500	1.2790	0.01155	1.00000	-0.0982	0.3455	0.5481
8.000	1.3364	0.01195	1.00000	-0.0985	0.3320	0.5521
8.500	1.3925	0.01242	1.00000	-0.0986	0.3202	0.5553
9.000	1.4456	0.01312	1.00000	-0.0983	0.2974	0.5583
9.500	1.4951	0.01404	1.00000	-0.0976	0.2696	0.5610
10.000	1.5371	0.01544	1.00000	-0.0962	0.2322	0.5672
10.500	1.5620	0.01782	1.00000	-0.0930	0.1846	0.5712
11.000	1.5653	0.02063	1.00000	-0.0873	0.1517	0.5742
11.500	1.5630	0.02501	1.00000	-0.0842	0.1232	0.5770
12.000	1.5565	0.03084	1.00000	-0.0829	0.1022	0.5795
12.500	1.5512	0.03738	1.00000	-0.0827	0.0873	0.5840
13.000	1.5419	0.04488	1.00000	-0.0830	0.0781	0.5889
13.500	1.5381	0.05215	1.00000	-0.0837	0.0667	0.5920
14.000	1.5275	0.06054	1.00000	-0.0847	0.0595	0.5947
14.500	1.5252	0.06819	1.00000	-0.0857	0.0513	0.5973
15.000	1.5201	0.07631	1.00000	-0.0870	0.0488	0.6001

XFOIL Version 5.0

Calculated polar for: FFA-W3-211

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
0.000	0.3329	0.01395	1.00000	-0.0787	0.0100	0.1000
0.500	0.3945	0.01406	1.00000	-0.0800	0.0100	0.1000
1.000	0.4557	0.01420	1.00000	-0.0812	0.0100	0.1000
1.500	0.5165	0.01438	1.00000	-0.0824	0.0100	0.1000
2.000	0.5768	0.01460	1.00000	-0.0834	0.0100	0.1000
2.500	0.6366	0.01485	1.00000	-0.0844	0.0100	0.1000
3.000	0.6958	0.01513	1.00000	-0.0853	0.0100	0.1000
3.500	0.7543	0.01546	1.00000	-0.0860	0.0100	0.1000
4.000	0.8121	0.01582	1.00000	-0.0867	0.0100	0.1000
4.500	0.8691	0.01624	1.00000	-0.0873	0.0100	0.1000
5.000	0.9251	0.01670	1.00000	-0.0877	0.0100	0.1000
5.500	0.9800	0.01722	1.00000	-0.0880	0.0100	0.1000
6.000	1.0336	0.01781	1.00000	-0.0881	0.0100	0.1000
6.500	1.0855	0.01847	1.00000	-0.0880	0.0100	0.1000
7.000	1.1355	0.01922	1.00000	-0.0877	0.0100	0.1000
7.500	1.1827	0.02008	1.00000	-0.0870	0.0100	0.1000
8.000	1.2241	0.02109	1.00000	-0.0857	0.0100	0.1000
8.500	1.2510	0.02252	1.00000	-0.0827	0.0100	0.1000
9.000	1.2792	0.02461	1.00000	-0.0813	0.0100	0.1000
9.500	1.3033	0.02739	1.00000	-0.0806	0.0100	0.1000
10.000	1.3242	0.03084	1.00000	-0.0803	0.0100	0.1000
10.500	1.3419	0.03490	1.00000	-0.0803	0.0100	0.1000
11.000	1.3572	0.03952	1.00000	-0.0806	0.0100	0.1000
11.500	1.3699	0.04462	1.00000	-0.0811	0.0100	0.1000
12.000	1.3810	0.05017	1.00000	-0.0818	0.0100	0.1000
12.500	1.3901	0.05604	1.00000	-0.0825	0.0100	0.1000
13.000	1.3980	0.06228	1.00000	-0.0834	0.0100	0.1000
13.500	1.4048	0.06877	1.00000	-0.0845	0.0100	0.1000
14.000	1.4104	0.07552	1.00000	-0.0856	0.0100	0.1000
14.500	1.4157	0.08252	1.00000	-0.0869	0.0100	0.1000
15.000	1.4206	0.08975	1.00000	-0.0884	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W3-211

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 2.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.38480	0.00734	2.00000	-0.07240	0.53550	0.35720
-5.50000	-0.32000	0.00706	2.00000	-0.07420	0.52770	0.38810
-5.00000	-0.25550	0.00684	2.00000	-0.07590	0.51180	0.41630
-4.50000	-0.19180	0.00672	2.00000	-0.07730	0.50530	0.43990
-4.00000	-0.12830	0.00672	2.00000	-0.07850	0.49350	0.44390
-3.50000	-0.06490	0.00672	2.00000	-0.07970	0.48580	0.44840
-3.00000	-0.00130	0.00671	2.00000	-0.08110	0.47830	0.45620
-2.50000	0.06200	0.00674	2.00000	-0.08230	0.46590	0.46200
-2.00000	0.12520	0.00678	2.00000	-0.08340	0.46110	0.46530
-1.50000	0.18810	0.00685	2.00000	-0.08460	0.45330	0.46800
-1.00000	0.25130	0.00688	2.00000	-0.08580	0.44360	0.47460
-0.50000	0.31440	0.00691	2.00000	-0.08700	0.43980	0.48030
0.000	0.3772	0.00699	2.00000	-0.0881	0.4329	0.4841
0.500	0.4398	0.00708	2.00000	-0.0892	0.4237	0.4871
1.000	0.5022	0.00718	2.00000	-0.0903	0.4192	0.4892
1.500	0.5645	0.00729	2.00000	-0.0913	0.4145	0.4909
2.000	0.6273	0.00734	2.00000	-0.0925	0.4081	0.4978
2.500	0.6895	0.00747	2.00000	-0.0935	0.3994	0.5031
3.000	0.7515	0.00758	2.00000	-0.0945	0.3957	0.5062
3.500	0.8132	0.00772	2.00000	-0.0955	0.3909	0.5086
4.000	0.8748	0.00787	2.00000	-0.0964	0.3841	0.5106
4.500	0.9359	0.00808	2.00000	-0.0972	0.3753	0.5126
5.000	0.9975	0.00819	2.00000	-0.0981	0.3715	0.5173
5.500	1.0586	0.00837	2.00000	-0.0990	0.3646	0.5227
6.000	1.1190	0.00861	2.00000	-0.0997	0.3523	0.5264
6.500	1.1787	0.00889	2.00000	-0.1003	0.3424	0.5291
7.000	1.2378	0.00922	2.00000	-0.1008	0.3274	0.5316
7.500	1.2964	0.00958	2.00000	-0.1013	0.3152	0.5336
8.000	1.3525	0.01016	2.00000	-0.1014	0.2903	0.5352
8.500	1.4075	0.01080	2.00000	-0.1015	0.2638	0.5414
9.000	1.4572	0.01188	2.00000	-0.1009	0.2318	0.5463
9.500	1.4968	0.01369	2.00000	-0.0992	0.1714	0.5501
10.000	1.5349	0.01537	2.00000	-0.0973	0.1323	0.5526
10.500	1.5659	0.01727	2.00000	-0.0945	0.1042	0.5547
11.000	1.5777	0.01938	2.00000	-0.0893	0.0850	0.5565
11.500	1.5867	0.02256	2.00000	-0.0861	0.0709	0.5598
12.000	1.5899	0.02707	2.00000	-0.0843	0.0629	0.5652
12.500	1.5951	0.03210	2.00000	-0.0836	0.0548	0.5694
13.000	1.6007	0.03745	2.00000	-0.0832	0.0471	0.5731
13.500	1.5994	0.04399	2.00000	-0.0833	0.0449	0.5754
14.000	1.5957	0.05113	2.00000	-0.0837	0.0419	0.5773
14.500	1.5967	0.05784	2.00000	-0.0842	0.0382	0.5791
15.000	1.5988	0.06469	2.00000	-0.0849	0.0347	0.5833
15.500	1.5996	0.07190	2.00000	-0.0858	0.0314	0.5885
16.000	1.5963	0.07970	2.00000	-0.0868	0.0297	0.5925
16.500	1.5945	0.08757	2.00000	-0.0882	0.0292	0.5956

XFOIL Version 5.0

Calculated polar for: FFA-W3-211

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.38520	0.00738	3.00000	-0.07190	0.51810	0.29310
-5.50000	-0.31790	0.00665	3.00000	-0.07480	0.50410	0.36200
-5.00000	-0.25330	0.00643	3.00000	-0.07640	0.49680	0.38540
-4.50000	-0.18860	0.00613	3.00000	-0.07820	0.48490	0.42650
-4.00000	-0.12510	0.00612	3.00000	-0.07940	0.47840	0.43200
-3.50000	-0.06150	0.00612	3.00000	-0.08070	0.46680	0.43980
-3.00000	0.00190	0.00612	3.00000	-0.08190	0.46030	0.44500
-2.50000	0.06520	0.00615	3.00000	-0.08310	0.45330	0.44770
-2.00000	0.12860	0.00616	3.00000	-0.08430	0.44370	0.45530
-1.50000	0.19180	0.00619	3.00000	-0.08550	0.43860	0.46130
-1.00000	0.25490	0.00624	3.00000	-0.08670	0.43080	0.46490
-0.50000	0.31780	0.00630	3.00000	-0.08780	0.42380	0.46700
0.00000	0.38080	0.00635	3.00000	-0.08900	0.41910	0.47420
0.50000	0.44370	0.00641	3.00000	-0.09010	0.41210	0.47900
1.000	0.5064	0.00649	3.00000	-0.0912	0.4050	0.4826
1.500	0.5689	0.00658	3.00000	-0.0922	0.4012	0.4847
2.000	0.6314	0.00668	3.00000	-0.0933	0.3958	0.4875
2.500	0.6938	0.00678	3.00000	-0.0944	0.3863	0.4933
3.000	0.7561	0.00687	3.00000	-0.0954	0.3831	0.4981
3.500	0.8181	0.00700	3.00000	-0.0963	0.3787	0.5009
4.000	0.8799	0.00714	3.00000	-0.0973	0.3698	0.5029
4.500	0.9415	0.00729	3.00000	-0.0981	0.3647	0.5045
5.000	1.0030	0.00744	3.00000	-0.0990	0.3590	0.5099
5.500	1.0641	0.00764	3.00000	-0.0999	0.3472	0.5145
6.000	1.1247	0.00788	3.00000	-0.1006	0.3387	0.5178
6.500	1.1844	0.00819	3.00000	-0.1012	0.3233	0.5205
7.000	1.2433	0.00857	3.00000	-0.1017	0.3054	0.5223
7.500	1.3017	0.00899	3.00000	-0.1022	0.2880	0.5238
8.000	1.3568	0.00973	3.00000	-0.1022	0.2530	0.5292
8.500	1.4092	0.01066	3.00000	-0.1020	0.2206	0.5342
9.000	1.4554	0.01209	3.00000	-0.1010	0.1715	0.5372
9.500	1.5001	0.01347	3.00000	-0.0998	0.1343	0.5395
10.000	1.5425	0.01486	3.00000	-0.0983	0.1020	0.5413
10.500	1.5805	0.01638	3.00000	-0.0963	0.0819	0.5429
11.500	1.6214	0.02025	3.00000	-0.0886	0.0531	0.5533
12.000	1.6291	0.02390	3.00000	-0.0861	0.0486	0.5562
12.500	1.6390	0.02812	3.00000	-0.0850	0.0428	0.5585
13.000	1.6512	0.03259	3.00000	-0.0845	0.0368	0.5603
13.500	1.6508	0.03874	3.00000	-0.0844	0.0354	0.5619
14.000	1.6490	0.04550	3.00000	-0.0847	0.0332	0.5679
14.500	1.6492	0.05224	3.00000	-0.0851	0.0295	0.5720
15.000	1.6514	0.05898	3.00000	-0.0858	0.0268	0.5751

XFOIL Version 5.0

Calculated polar for: FFA-W3-211

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.40200	0.01010	3.00000	-0.06430	0.51930	0.10000
-5.50000	-0.33750	0.00985	3.00000	-0.06620	0.50770	0.10000
-5.00000	-0.27990	0.01158	3.00000	-0.06780	0.01000	0.10000
-4.50000	-0.21610	0.01145	3.00000	-0.06950	0.01000	0.10000
-4.00000	-0.15230	0.01136	3.00000	-0.07110	0.01000	0.10000
-3.50000	-0.08880	0.01129	3.00000	-0.07270	0.01000	0.10000
-3.00000	-0.02540	0.01124	3.00000	-0.07430	0.01000	0.10000
-2.50000	0.03790	0.01121	3.00000	-0.07570	0.01000	0.10000
-2.00000	0.10090	0.01121	3.00000	-0.07720	0.01000	0.10000
-1.50000	0.16370	0.01123	3.00000	-0.07860	0.01000	0.10000
-1.00000	0.22640	0.01127	3.00000	-0.08000	0.01000	0.10000
-0.50000	0.28880	0.01134	3.00000	-0.08130	0.01000	0.10000
0.000	0.3510	0.01142	3.00000	-0.0825	0.0100	0.1000
0.500	0.4129	0.01153	3.00000	-0.0837	0.0100	0.1000
1.000	0.4745	0.01166	3.00000	-0.0849	0.0100	0.1000
1.500	0.5359	0.01181	3.00000	-0.0860	0.0100	0.1000
2.000	0.5969	0.01199	3.00000	-0.0871	0.0100	0.1000
2.500	0.6576	0.01219	3.00000	-0.0880	0.0100	0.1000
3.000	0.7178	0.01241	3.00000	-0.0890	0.0100	0.1000
3.500	0.7776	0.01267	3.00000	-0.0898	0.0100	0.1000
4.000	0.8370	0.01295	3.00000	-0.0906	0.0100	0.1000
4.500	0.8957	0.01326	3.00000	-0.0913	0.0100	0.1000
5.000	0.9539	0.01361	3.00000	-0.0919	0.0100	0.1000
5.500	1.0113	0.01399	3.00000	-0.0924	0.0100	0.1000
6.000	1.0680	0.01440	3.00000	-0.0927	0.0100	0.1000
6.500	1.1237	0.01486	3.00000	-0.0930	0.0100	0.1000
7.000	1.1783	0.01537	3.00000	-0.0931	0.0100	0.1000
7.500	1.2316	0.01593	3.00000	-0.0930	0.0100	0.1000
8.000	1.2833	0.01656	3.00000	-0.0927	0.0100	0.1000
8.500	1.3330	0.01726	3.00000	-0.0922	0.0100	0.1000
9.000	1.3800	0.01807	3.00000	-0.0913	0.0100	0.1000
9.500	1.4190	0.01900	3.00000	-0.0892	0.0100	0.1000
10.000	1.4427	0.02044	3.00000	-0.0855	0.0100	0.1000
10.500	1.4664	0.02259	3.00000	-0.0835	0.0100	0.1000
11.000	1.4864	0.02550	3.00000	-0.0821	0.0100	0.1000
11.500	1.5030	0.02916	3.00000	-0.0814	0.0100	0.1000
12.000	1.5174	0.03342	3.00000	-0.0811	0.0100	0.1000
12.500	1.5288	0.03823	3.00000	-0.0810	0.0100	0.1000
13.000	1.5388	0.04354	3.00000	-0.0812	0.0100	0.1000
13.500	1.5465	0.04928	3.00000	-0.0816	0.0100	0.1000
14.000	1.5521	0.05546	3.00000	-0.0821	0.0100	0.1000
14.500	1.5575	0.06184	3.00000	-0.0828	0.0100	0.1000
15.000	1.5617	0.06854	3.00000	-0.0837	0.0100	0.1000
15.500	1.5641	0.07554	3.00000	-0.0847	0.0100	0.1000
16.000	1.5660	0.08279	3.00000	-0.0858	0.0100	0.1000
16.500	1.5693	0.08998	3.00000	-0.0870	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W3-241

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-3.50000	-0.09630	0.00940	1.00000	-0.08170	0.51900	0.43940
-3.00000	-0.03040	0.00941	1.00000	-0.08360	0.50270	0.44390
-2.50000	0.03490	0.00945	1.00000	-0.08520	0.49440	0.44780
-2.00000	0.10110	0.00941	1.00000	-0.08710	0.48010	0.45460
-1.50000	0.16690	0.00940	1.00000	-0.08890	0.47270	0.46130
-1.00000	0.23170	0.00949	1.00000	-0.09040	0.45950	0.46590
-0.50000	0.29620	0.00958	1.00000	-0.09180	0.45090	0.46930
0.000	0.3605	0.00969	1.00000	-0.0932	0.4426	0.4721
0.500	0.4257	0.00973	1.00000	-0.0949	0.4298	0.4805
1.000	0.4900	0.00982	1.00000	-0.0963	0.4235	0.4854
1.500	0.5537	0.00997	1.00000	-0.0975	0.4118	0.4889
2.000	0.6168	0.01015	1.00000	-0.0986	0.4040	0.4922
2.500	0.6799	0.01033	1.00000	-0.0997	0.3973	0.4948
3.000	0.7435	0.01046	1.00000	-0.1010	0.3843	0.5013
3.500	0.8064	0.01061	1.00000	-0.1022	0.3787	0.5066
4.000	0.8685	0.01084	1.00000	-0.1031	0.3716	0.5105
4.500	0.9299	0.01114	1.00000	-0.1039	0.3587	0.5135
5.000	0.9913	0.01137	1.00000	-0.1046	0.3534	0.5162
5.500	1.0518	0.01168	1.00000	-0.1052	0.3429	0.5185
6.000	1.1128	0.01192	1.00000	-0.1061	0.3333	0.5258
6.500	1.1723	0.01225	1.00000	-0.1065	0.3252	0.5306
7.000	1.2305	0.01268	1.00000	-0.1068	0.3109	0.5339
7.500	1.2883	0.01308	1.00000	-0.1070	0.3045	0.5368
8.000	1.3443	0.01360	1.00000	-0.1069	0.2890	0.5395
8.500	1.3983	0.01420	1.00000	-0.1064	0.2795	0.5419
9.000	1.4525	0.01473	1.00000	-0.1062	0.2643	0.5495
9.500	1.5016	0.01553	1.00000	-0.1052	0.2532	0.5537
10.000	1.5485	0.01636	1.00000	-0.1038	0.2387	0.5570
10.500	1.5882	0.01739	1.00000	-0.1013	0.2276	0.5599
11.000	1.6093	0.01873	1.00000	-0.0960	0.2147	0.5625
11.500	1.6301	0.02072	1.00000	-0.0920	0.2031	0.5649
12.000	1.6416	0.02368	1.00000	-0.0885	0.1879	0.5714
13.000	1.6528	0.03226	1.00000	-0.0840	0.1635	0.5787
13.500	1.6544	0.03782	1.00000	-0.0830	0.1484	0.5814
14.000	1.6484	0.04458	1.00000	-0.0826	0.1407	0.5837
15.000	1.6288	0.06054	1.00000	-0.0836	0.1181	0.5878
15.500	1.6229	0.06863	1.00000	-0.0848	0.1056	0.5927
16.000	1.6133	0.07746	1.00000	-0.0862	0.1014	0.5971
16.500	1.6028	0.08676	1.00000	-0.0880	0.0944	0.6001

XFOIL Version 5.0

Calculated polar for: FFA-W3-241

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
0.000	0.2940	0.01606	1.00000	-0.0771	0.0100	0.1000
0.500	0.3600	0.01605	1.00000	-0.0796	0.0100	0.1000
1.000	0.4252	0.01610	1.00000	-0.0818	0.0100	0.1000
1.500	0.4894	0.01620	1.00000	-0.0838	0.0100	0.1000
2.000	0.5528	0.01635	1.00000	-0.0856	0.0100	0.1000
2.500	0.6153	0.01655	1.00000	-0.0871	0.0100	0.1000
3.000	0.6768	0.01680	1.00000	-0.0885	0.0100	0.1000
3.500	0.7372	0.01709	1.00000	-0.0897	0.0100	0.1000
4.000	0.7965	0.01744	1.00000	-0.0906	0.0100	0.1000
4.500	0.8545	0.01785	1.00000	-0.0913	0.0100	0.1000
5.000	0.9110	0.01831	1.00000	-0.0918	0.0100	0.1000
5.500	0.9658	0.01884	1.00000	-0.0920	0.0100	0.1000
6.000	1.0186	0.01944	1.00000	-0.0920	0.0100	0.1000
6.500	1.0688	0.02014	1.00000	-0.0915	0.0100	0.1000
7.000	1.1153	0.02095	1.00000	-0.0906	0.0100	0.1000
7.500	1.1406	0.02197	1.00000	-0.0862	0.0100	0.1000
8.000	1.1727	0.02344	1.00000	-0.0839	0.0100	0.1000
8.500	1.2013	0.02539	1.00000	-0.0821	0.0100	0.1000
9.000	1.2263	0.02790	1.00000	-0.0807	0.0100	0.1000
9.500	1.2486	0.03098	1.00000	-0.0798	0.0100	0.1000
10.000	1.2684	0.03463	1.00000	-0.0794	0.0100	0.1000
10.500	1.2869	0.03879	1.00000	-0.0794	0.0100	0.1000
11.000	1.3028	0.04352	1.00000	-0.0798	0.0100	0.1000
11.500	1.3166	0.04864	1.00000	-0.0803	0.0100	0.1000
12.000	1.3285	0.05431	1.00000	-0.0812	0.0100	0.1000
12.500	1.3392	0.06034	1.00000	-0.0822	0.0100	0.1000
13.000	1.3482	0.06670	1.00000	-0.0833	0.0100	0.1000
13.500	1.3555	0.07342	1.00000	-0.0846	0.0100	0.1000
14.000	1.3621	0.08048	1.00000	-0.0861	0.0100	0.1000
14.500	1.3680	0.08781	1.00000	-0.0877	0.0100	0.1000
15.000	1.3736	0.09531	1.00000	-0.0894	0.0100	0.1000
15.500	1.3774	0.10329	1.00000	-0.0914	0.0100	0.1000
16.000	1.3799	0.11169	1.00000	-0.0936	0.0100	0.1000
16.500	1.3830	0.12015	1.00000	-0.0958	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W3-241

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 2.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-4.00000	-0.14810	0.00782	2.00000	-0.08380	0.49680	0.41520
-3.50000	-0.08200	0.00778	2.00000	-0.08560	0.47980	0.42100
-3.00000	-0.01660	0.00778	2.00000	-0.08720	0.47310	0.42510
-2.50000	0.04940	0.00776	2.00000	-0.08900	0.46100	0.43200
-2.00000	0.11500	0.00776	2.00000	-0.09070	0.45090	0.43950
-1.50000	0.18020	0.00778	2.00000	-0.09220	0.44160	0.44500
-1.00000	0.24510	0.00784	2.00000	-0.09370	0.43020	0.44800
-0.50000	0.31010	0.00789	2.00000	-0.09520	0.42300	0.45300
0.000	0.3752	0.00793	2.00000	-0.0968	0.4118	0.4615
0.500	0.4398	0.00800	2.00000	-0.0982	0.4046	0.4661
1.000	0.5039	0.00811	2.00000	-0.0995	0.3967	0.4695
1.500	0.5679	0.00824	2.00000	-0.1007	0.3851	0.4721
2.000	0.6323	0.00831	2.00000	-0.1021	0.3799	0.4789
2.500	0.6960	0.00845	2.00000	-0.1033	0.3714	0.4850
3.000	0.7596	0.00859	2.00000	-0.1044	0.3598	0.4897
3.500	0.8226	0.00877	2.00000	-0.1055	0.3542	0.4925
4.000	0.8853	0.00897	2.00000	-0.1065	0.3459	0.4947
4.500	0.9481	0.00916	2.00000	-0.1075	0.3350	0.4992
5.000	1.0104	0.00936	2.00000	-0.1084	0.3286	0.5065
5.500	1.0721	0.00962	2.00000	-0.1092	0.3149	0.5108
6.000	1.1333	0.00989	2.00000	-0.1099	0.3082	0.5138
6.500	1.1935	0.01023	2.00000	-0.1105	0.2964	0.5162
7.000	1.2529	0.01061	2.00000	-0.1109	0.2844	0.5184
7.500	1.3122	0.01098	2.00000	-0.1113	0.2743	0.5244
8.000	1.3700	0.01143	2.00000	-0.1115	0.2603	0.5301
8.500	1.4260	0.01198	2.00000	-0.1114	0.2415	0.5341
9.000	1.4801	0.01261	2.00000	-0.1110	0.2328	0.5372
9.500	1.5334	0.01324	2.00000	-0.1104	0.2167	0.5395
10.000	1.5827	0.01406	2.00000	-0.1093	0.2054	0.5414
10.500	1.6304	0.01489	2.00000	-0.1080	0.1910	0.5468
11.000	1.6673	0.01612	2.00000	-0.1051	0.1702	0.5525
11.500	1.6839	0.01763	2.00000	-0.0991	0.1589	0.5563
12.000	1.7017	0.01968	2.00000	-0.0946	0.1461	0.5595
12.500	1.7112	0.02268	2.00000	-0.0905	0.1336	0.5617
13.000	1.7068	0.02741	2.00000	-0.0872	0.1211	0.5634
13.500	1.7135	0.03198	2.00000	-0.0855	0.1084	0.5650
14.000	1.7108	0.03805	2.00000	-0.0847	0.1023	0.5701
14.500	1.7062	0.04478	2.00000	-0.0844	0.0946	0.5748
15.000	1.7066	0.05144	2.00000	-0.0846	0.0850	0.5784
15.500	1.7004	0.05912	2.00000	-0.0852	0.0820	0.5816
16.000	1.6898	0.06771	2.00000	-0.0862	0.0773	0.5836
16.500	1.6880	0.07538	2.00000	-0.0872	0.0708	0.5854

XFOIL Version 5.0

Calculated polar for: FFA-W3-241

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.40830	0.00751	3.00000	-0.07760	0.52390	0.37360
-5.50000	-0.34170	0.00740	3.00000	-0.07960	0.51100	0.37880
-5.00000	-0.27500	0.00730	3.00000	-0.08160	0.49980	0.38890
-4.50000	-0.20870	0.00723	3.00000	-0.08340	0.48480	0.39670
-4.00000	-0.14270	0.00718	3.00000	-0.08520	0.47550	0.40170
-3.50000	-0.07670	0.00714	3.00000	-0.08700	0.46500	0.40860
-3.00000	-0.01090	0.00712	3.00000	-0.08870	0.45370	0.41700
-2.50000	0.05470	0.00711	3.00000	-0.09030	0.44440	0.42290
-2.00000	0.11990	0.00714	3.00000	-0.09190	0.43120	0.42560
-1.50000	0.18540	0.00714	3.00000	-0.09350	0.42640	0.43300
-1.00000	0.25050	0.00718	3.00000	-0.09500	0.41440	0.44150
-0.50000	0.31550	0.00722	3.00000	-0.09650	0.40580	0.44560
0.000	0.3802	0.00729	3.00000	-0.0979	0.4003	0.4483
0.500	0.4449	0.00735	3.00000	-0.0993	0.3877	0.4547
1.000	0.5095	0.00742	3.00000	-0.1007	0.3812	0.4617
1.500	0.5739	0.00752	3.00000	-0.1020	0.3755	0.4673
2.000	0.6379	0.00763	3.00000	-0.1033	0.3630	0.4703
2.500	0.7016	0.00776	3.00000	-0.1044	0.3570	0.4722
3.000	0.7653	0.00791	3.00000	-0.1057	0.3488	0.4784
3.500	0.8291	0.00803	3.00000	-0.1068	0.3376	0.4858
4.000	0.8923	0.00819	3.00000	-0.1079	0.3324	0.4899
4.500	0.9548	0.00841	3.00000	-0.1088	0.3222	0.4929
5.000	1.0173	0.00861	3.00000	-0.1097	0.3112	0.4948
5.500	1.0791	0.00887	3.00000	-0.1106	0.3039	0.4999
6.000	1.1410	0.00911	3.00000	-0.1114	0.2880	0.5060
6.500	1.2013	0.00948	3.00000	-0.1120	0.2795	0.5107
7.000	1.2621	0.00976	3.00000	-0.1126	0.2658	0.5140
7.500	1.3204	0.01024	3.00000	-0.1128	0.2546	0.5168
8.000	1.3792	0.01063	3.00000	-0.1131	0.2403	0.5184
8.500	1.4354	0.01121	3.00000	-0.1130	0.2239	0.5235
9.000	1.4910	0.01177	3.00000	-0.1129	0.2126	0.5292
9.500	1.5458	0.01234	3.00000	-0.1125	0.1957	0.5344
10.000	1.5956	0.01319	3.00000	-0.1115	0.1844	0.5374
10.500	1.6404	0.01426	3.00000	-0.1098	0.1641	0.5396
11.000	1.6840	0.01521	3.00000	-0.1078	0.1476	0.5412
11.500	1.7110	0.01647	3.00000	-0.1032	0.1347	0.5447
12.000	1.7272	0.01823	3.00000	-0.0976	0.1241	0.5502
12.500	1.7363	0.02093	3.00000	-0.0928	0.1067	0.5555
13.000	1.7376	0.02489	3.00000	-0.0891	0.1011	0.5585
13.500	1.7432	0.02919	3.00000	-0.0869	0.0920	0.5609
14.000	1.7484	0.03416	3.00000	-0.0856	0.0843	0.5628
14.500	1.7434	0.04063	3.00000	-0.0849	0.0805	0.5642
15.000	1.7419	0.04721	3.00000	-0.0848	0.0745	0.5656
15.500	1.7418	0.05397	3.00000	-0.0850	0.0655	0.5708
16.000	1.7343	0.06206	3.00000	-0.0859	0.0636	0.5762
16.500	1.7242	0.07058	3.00000	-0.0869	0.0607	0.5794

XFOIL Version 5.0

Calculated polar for: FFA-W3-241

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-5.00000	-0.35790	0.01577	3.00000	-0.05370	0.01000	0.10000
-4.50000	-0.28750	0.01510	3.00000	-0.05810	0.01000	0.10000
-4.00000	-0.21660	0.01454	3.00000	-0.06230	0.01000	0.10000
-3.50000	-0.14570	0.01406	3.00000	-0.06630	0.01000	0.10000
-3.00000	-0.07500	0.01366	3.00000	-0.07010	0.01000	0.10000
-2.50000	-0.00490	0.01334	3.00000	-0.07370	0.01000	0.10000
-2.00000	0.06350	0.01316	3.00000	-0.07660	0.01000	0.10000
-1.50000	0.13070	0.01307	3.00000	-0.07910	0.01000	0.10000
-1.00000	0.19740	0.01302	3.00000	-0.08150	0.01000	0.10000
-0.50000	0.26350	0.01300	3.00000	-0.08360	0.01000	0.10000
0.000	0.3291	0.01302	3.00000	-0.0857	0.0100	0.1000
0.500	0.3942	0.01307	3.00000	-0.0876	0.0100	0.1000
1.000	0.4588	0.01316	3.00000	-0.0893	0.0100	0.1000
1.500	0.5228	0.01327	3.00000	-0.0910	0.0100	0.1000
2.000	0.5863	0.01341	3.00000	-0.0925	0.0100	0.1000
2.500	0.6492	0.01359	3.00000	-0.0939	0.0100	0.1000
3.000	0.7115	0.01379	3.00000	-0.0951	0.0100	0.1000
3.500	0.7732	0.01403	3.00000	-0.0963	0.0100	0.1000
4.000	0.8341	0.01430	3.00000	-0.0973	0.0100	0.1000
4.500	0.8941	0.01461	3.00000	-0.0981	0.0100	0.1000
5.000	0.9533	0.01495	3.00000	-0.0988	0.0100	0.1000
5.500	1.0115	0.01534	3.00000	-0.0993	0.0100	0.1000
6.000	1.0685	0.01577	3.00000	-0.0996	0.0100	0.1000
6.500	1.1241	0.01625	3.00000	-0.0997	0.0100	0.1000
7.000	1.1782	0.01679	3.00000	-0.0995	0.0100	0.1000
7.500	1.2303	0.01739	3.00000	-0.0991	0.0100	0.1000
8.000	1.2798	0.01807	3.00000	-0.0983	0.0100	0.1000
8.500	1.3250	0.01886	3.00000	-0.0968	0.0100	0.1000
9.000	1.3457	0.01990	3.00000	-0.0914	0.0100	0.1000
9.500	1.3754	0.02140	3.00000	-0.0885	0.0100	0.1000
10.000	1.4000	0.02342	3.00000	-0.0859	0.0100	0.1000
10.500	1.4217	0.02604	3.00000	-0.0840	0.0100	0.1000
11.000	1.4397	0.02934	3.00000	-0.0826	0.0100	0.1000
11.500	1.4564	0.03321	3.00000	-0.0819	0.0100	0.1000
12.000	1.4713	0.03767	3.00000	-0.0816	0.0100	0.1000
12.500	1.4832	0.04271	3.00000	-0.0818	0.0100	0.1000
13.000	1.4943	0.04810	3.00000	-0.0821	0.0100	0.1000
13.500	1.5040	0.05394	3.00000	-0.0827	0.0100	0.1000
14.000	1.5113	0.06035	3.00000	-0.0837	0.0100	0.1000
14.500	1.5170	0.06704	3.00000	-0.0847	0.0100	0.1000
15.000	1.5211	0.07411	3.00000	-0.0859	0.0100	0.1000
15.500	1.5245	0.08149	3.00000	-0.0872	0.0100	0.1000
16.000	1.5278	0.08904	3.00000	-0.0888	0.0100	0.1000
16.500	1.5304	0.09687	3.00000	-0.0905	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W3-270

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
0.000	0.3439	0.01049	1.00000	-0.0930	0.4265	0.4596
0.500	0.4098	0.01060	1.00000	-0.0947	0.4155	0.4630
1.000	0.4749	0.01075	1.00000	-0.0963	0.4079	0.4662
1.500	0.5415	0.01081	1.00000	-0.0983	0.3949	0.4735
2.000	0.6070	0.01091	1.00000	-0.0999	0.3895	0.4792
2.500	0.6714	0.01109	1.00000	-0.1013	0.3797	0.4829
3.000	0.7348	0.01133	1.00000	-0.1025	0.3691	0.4864
3.500	0.7983	0.01154	1.00000	-0.1037	0.3636	0.4891
4.000	0.8621	0.01172	1.00000	-0.1051	0.3531	0.4951
4.500	0.9249	0.01196	1.00000	-0.1062	0.3441	0.5011
5.000	0.9867	0.01224	1.00000	-0.1071	0.3385	0.5052
5.500	1.0475	0.01258	1.00000	-0.1078	0.3277	0.5083
6.000	1.1073	0.01295	1.00000	-0.1083	0.3196	0.5111
6.500	1.1665	0.01333	1.00000	-0.1087	0.3141	0.5134
7.000	1.2256	0.01369	1.00000	-0.1092	0.3030	0.5216
7.500	1.2823	0.01415	1.00000	-0.1093	0.2954	0.5259
8.000	1.3375	0.01466	1.00000	-0.1091	0.2886	0.5292
8.500	1.3905	0.01525	1.00000	-0.1085	0.2787	0.5321
9.000	1.4402	0.01593	1.00000	-0.1075	0.2704	0.5347
9.500	1.4856	0.01668	1.00000	-0.1057	0.2644	0.5374
10.000	1.5143	0.01759	1.00000	-0.1014	0.2553	0.5448
10.500	1.5440	0.01899	1.00000	-0.0980	0.2467	0.5488
11.000	1.5699	0.02080	1.00000	-0.0948	0.2410	0.5518
11.500	1.5916	0.02317	1.00000	-0.0921	0.2315	0.5545
12.000	1.6071	0.02631	1.00000	-0.0896	0.2228	0.5569
12.500	1.6199	0.03007	1.00000	-0.0877	0.2165	0.5592
13.000	1.6328	0.03431	1.00000	-0.0867	0.2058	0.5639
13.500	1.6416	0.03921	1.00000	-0.0860	0.1996	0.5690
14.000	1.6446	0.04496	1.00000	-0.0855	0.1930	0.5722

XFOIL Version 5.0

Calculated polar for: FFA-W3-270

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
0.000	0.2152	0.01956	1.00000	-0.0585	0.0100	0.1000
0.500	0.2857	0.01919	1.00000	-0.0629	0.0100	0.1000
1.000	0.3558	0.01894	1.00000	-0.0670	0.0100	0.1000
1.500	0.4255	0.01876	1.00000	-0.0709	0.0100	0.1000
2.000	0.4942	0.01868	1.00000	-0.0745	0.0100	0.1000
2.500	0.5618	0.01868	1.00000	-0.0777	0.0100	0.1000
3.000	0.6277	0.01877	1.00000	-0.0806	0.0100	0.1000
3.500	0.6913	0.01896	1.00000	-0.0829	0.0100	0.1000
4.000	0.7521	0.01928	1.00000	-0.0846	0.0100	0.1000
4.500	0.8104	0.01968	1.00000	-0.0857	0.0100	0.1000
5.000	0.8660	0.02018	1.00000	-0.0864	0.0100	0.1000
5.500	0.9180	0.02078	1.00000	-0.0866	0.0100	0.1000
6.000	0.9539	0.02152	1.00000	-0.0840	0.0100	0.1000
6.500	0.9844	0.02267	1.00000	-0.0812	0.0100	0.1000
7.000	1.0196	0.02421	1.00000	-0.0800	0.0100	0.1000
7.500	1.0502	0.02619	1.00000	-0.0789	0.0100	0.1000
8.000	1.0781	0.02869	1.00000	-0.0783	0.0100	0.1000
8.500	1.1029	0.03169	1.00000	-0.0779	0.0100	0.1000
9.000	1.1260	0.03518	1.00000	-0.0779	0.0100	0.1000
9.500	1.1478	0.03908	1.00000	-0.0782	0.0100	0.1000
10.000	1.1669	0.04347	1.00000	-0.0788	0.0100	0.1000
10.500	1.1842	0.04827	1.00000	-0.0795	0.0100	0.1000
11.000	1.2003	0.05341	1.00000	-0.0804	0.0100	0.1000
11.500	1.2147	0.05895	1.00000	-0.0814	0.0100	0.1000
12.000	1.2272	0.06487	1.00000	-0.0826	0.0100	0.1000
12.500	1.2378	0.07115	1.00000	-0.0839	0.0100	0.1000
13.000	1.2473	0.07779	1.00000	-0.0853	0.0100	0.1000
13.500	1.2560	0.08476	1.00000	-0.0870	0.0100	0.1000
14.000	1.2643	0.09191	1.00000	-0.0887	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W3-270

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 2.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.46510	0.01002	2.00000	-0.06620	0.53380	0.37060
-5.50000	-0.39070	0.00954	2.00000	-0.07070	0.51600	0.37820
-5.00000	-0.31890	0.00925	2.00000	-0.07420	0.50710	0.38710
-4.50000	-0.24850	0.00905	2.00000	-0.07720	0.49180	0.39230
-4.00000	-0.17980	0.00896	2.00000	-0.07970	0.48190	0.39610
-3.50000	-0.10990	0.00880	2.00000	-0.08250	0.46730	0.40480
-3.00000	-0.04130	0.00871	2.00000	-0.08500	0.46010	0.41190
-2.50000	0.02680	0.00866	2.00000	-0.08730	0.44630	0.41630
-2.00000	0.09390	0.00868	2.00000	-0.08930	0.43740	0.41960
-1.50000	0.16200	0.00863	2.00000	-0.09160	0.42600	0.42750
-1.00000	0.22930	0.00863	2.00000	-0.09360	0.41670	0.43440
-0.50000	0.29590	0.00868	2.00000	-0.09550	0.40740	0.43880
0.000	0.3621	0.00875	2.00000	-0.0973	0.3946	0.4422
0.500	0.4285	0.00879	2.00000	-0.0991	0.3898	0.4472
1.000	0.4948	0.00886	2.00000	-0.1009	0.3812	0.4545
1.500	0.5605	0.00896	2.00000	-0.1025	0.3692	0.4604
2.000	0.6257	0.00908	2.00000	-0.1040	0.3639	0.4637
2.500	0.6904	0.00924	2.00000	-0.1054	0.3546	0.4661
3.000	0.7555	0.00937	2.00000	-0.1069	0.3451	0.4717
3.500	0.8200	0.00953	2.00000	-0.1083	0.3388	0.4794
4.000	0.8839	0.00972	2.00000	-0.1095	0.3291	0.4836
4.500	0.9474	0.00993	2.00000	-0.1106	0.3206	0.4868
5.000	1.0100	0.01019	2.00000	-0.1116	0.3142	0.4891
5.500	1.0727	0.01043	2.00000	-0.1126	0.3044	0.4945
6.000	1.1348	0.01070	2.00000	-0.1135	0.2955	0.5010
6.500	1.1959	0.01100	2.00000	-0.1142	0.2900	0.5056
7.000	1.2564	0.01133	2.00000	-0.1147	0.2805	0.5089
7.500	1.3155	0.01172	2.00000	-0.1150	0.2715	0.5116
8.000	1.3735	0.01215	2.00000	-0.1152	0.2659	0.5135
8.500	1.4310	0.01259	2.00000	-0.1153	0.2527	0.5207
9.000	1.4868	0.01307	2.00000	-0.1151	0.2471	0.5261
9.500	1.5386	0.01373	2.00000	-0.1143	0.2374	0.5299
10.000	1.5891	0.01436	2.00000	-0.1132	0.2261	0.5330
10.500	1.6322	0.01521	2.00000	-0.1111	0.2194	0.5351
11.000	1.6594	0.01619	2.00000	-0.1062	0.2100	0.5368
11.500	1.6881	0.01752	2.00000	-0.1024	0.2016	0.5434
12.000	1.7076	0.01955	2.00000	-0.0983	0.1963	0.5484
12.500	1.7252	0.02211	2.00000	-0.0951	0.1866	0.5520
13.000	1.7315	0.02588	2.00000	-0.0920	0.1766	0.5549
13.500	1.7353	0.03048	2.00000	-0.0899	0.1692	0.5569
14.000	1.7470	0.03483	2.00000	-0.0887	0.1576	0.5587
14.500	1.7431	0.04100	2.00000	-0.0876	0.1532	0.5600
15.000	1.7399	0.04758	2.00000	-0.0873	0.1452	0.5651
15.500	1.7402	0.05418	2.00000	-0.0874	0.1348	0.5697
16.000	1.7329	0.06183	2.00000	-0.0877	0.1308	0.5731
16.500	1.7232	0.07000	2.00000	-0.0884	0.1233	0.5762

XFOIL Version 5.0

Calculated polar for: FFA-W3-270

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
0.000	0.3695	0.00802	3.00000	-0.0990	0.3839	0.4351
0.500	0.4360	0.00805	3.00000	-0.1008	0.3721	0.4401
1.000	0.5016	0.00816	3.00000	-0.1024	0.3657	0.4426
1.500	0.5673	0.00826	3.00000	-0.1040	0.3574	0.4475
2.000	0.6331	0.00835	3.00000	-0.1056	0.3460	0.4561
2.500	0.6984	0.00845	3.00000	-0.1071	0.3422	0.4609
3.000	0.7630	0.00862	3.00000	-0.1085	0.3320	0.4642
3.500	0.8275	0.00878	3.00000	-0.1097	0.3215	0.4663
4.000	0.8920	0.00893	3.00000	-0.1111	0.3176	0.4730
4.500	0.9557	0.00915	3.00000	-0.1122	0.3072	0.4794
5.000	1.0194	0.00934	3.00000	-0.1134	0.2973	0.4840
5.500	1.0824	0.00956	3.00000	-0.1144	0.2935	0.4872
6.000	1.1442	0.00987	3.00000	-0.1152	0.2832	0.4896
6.500	1.2065	0.01012	3.00000	-0.1160	0.2734	0.4943
7.000	1.2674	0.01045	3.00000	-0.1167	0.2679	0.5007
7.500	1.3279	0.01078	3.00000	-0.1172	0.2591	0.5058
8.000	1.3872	0.01116	3.00000	-0.1176	0.2484	0.5098
8.500	1.4450	0.01160	3.00000	-0.1176	0.2430	0.5120
9.000	1.5016	0.01208	3.00000	-0.1175	0.2290	0.5136
9.500	1.5570	0.01258	3.00000	-0.1172	0.2233	0.5201
10.000	1.6080	0.01328	3.00000	-0.1163	0.2110	0.5258
10.500	1.6580	0.01390	3.00000	-0.1151	0.2018	0.5306
11.000	1.7002	0.01478	3.00000	-0.1128	0.1955	0.5334
11.500	1.7266	0.01575	3.00000	-0.1077	0.1854	0.5353
12.000	1.7426	0.01762	3.00000	-0.1023	0.1756	0.5367
12.500	1.7584	0.01993	3.00000	-0.0981	0.1666	0.5417
13.000	1.7742	0.02273	3.00000	-0.0950	0.1567	0.5469
13.500	1.7771	0.02691	3.00000	-0.0920	0.1519	0.5517
14.000	1.7830	0.03149	3.00000	-0.0902	0.1426	0.5545

XFOIL Version 5.0

Calculated polar for: FFA-W3-270

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.50900	0.02499	3.00000	-0.02720	0.01000	0.10000
-5.50000	-0.46030	0.02349	3.00000	-0.02950	0.01000	0.10000
-5.00000	-0.40680	0.02205	3.00000	-0.03270	0.01000	0.10000
-4.50000	-0.34900	0.02093	3.00000	-0.03590	0.01000	0.10000
-4.00000	-0.28690	0.01976	3.00000	-0.04000	0.01000	0.10000
-3.50000	-0.22190	0.01890	3.00000	-0.04400	0.01000	0.10000
-3.00000	-0.15410	0.01811	3.00000	-0.04840	0.01000	0.10000
-2.50000	-0.08380	0.01737	3.00000	-0.05310	0.01000	0.10000
-2.00000	-0.01300	0.01682	3.00000	-0.05750	0.01000	0.10000
-1.50000	0.05930	0.01629	3.00000	-0.06220	0.01000	0.10000
-1.00000	0.13150	0.01589	3.00000	-0.06660	0.01000	0.10000
-0.50000	0.20390	0.01554	3.00000	-0.07090	0.01000	0.10000
0.000	0.2758	0.01528	3.00000	-0.0750	0.0100	0.1000
0.500	0.3473	0.01506	3.00000	-0.0789	0.0100	0.1000
1.000	0.4177	0.01493	3.00000	-0.0824	0.0100	0.1000
1.500	0.4870	0.01486	3.00000	-0.0856	0.0100	0.1000
2.000	0.5543	0.01490	3.00000	-0.0882	0.0100	0.1000
2.500	0.6199	0.01502	3.00000	-0.0904	0.0100	0.1000
3.000	0.6844	0.01519	3.00000	-0.0923	0.0100	0.1000
3.500	0.7477	0.01540	3.00000	-0.0939	0.0100	0.1000
4.000	0.8099	0.01566	3.00000	-0.0953	0.0100	0.1000
4.500	0.8707	0.01597	3.00000	-0.0965	0.0100	0.1000
5.000	0.9301	0.01633	3.00000	-0.0973	0.0100	0.1000
5.500	0.9878	0.01674	3.00000	-0.0979	0.0100	0.1000
6.000	1.0437	0.01721	3.00000	-0.0982	0.0100	0.1000
6.500	1.0971	0.01776	3.00000	-0.0981	0.0100	0.1000
7.000	1.1472	0.01839	3.00000	-0.0975	0.0100	0.1000
7.500	1.1799	0.01915	3.00000	-0.0939	0.0100	0.1000
8.000	1.2056	0.02036	3.00000	-0.0898	0.0100	0.1000
8.500	1.2353	0.02197	3.00000	-0.0875	0.0100	0.1000
9.000	1.2609	0.02411	3.00000	-0.0854	0.0100	0.1000
9.500	1.2841	0.02679	3.00000	-0.0840	0.0100	0.1000
10.000	1.3045	0.03003	3.00000	-0.0830	0.0100	0.1000
10.500	1.3240	0.03376	3.00000	-0.0826	0.0100	0.1000
11.000	1.3416	0.03798	3.00000	-0.0825	0.0100	0.1000
11.500	1.3568	0.04263	3.00000	-0.0827	0.0100	0.1000
12.000	1.3699	0.04786	3.00000	-0.0831	0.0100	0.1000
12.500	1.3825	0.05335	3.00000	-0.0838	0.0100	0.1000
13.000	1.3936	0.05920	3.00000	-0.0846	0.0100	0.1000
13.500	1.4027	0.06542	3.00000	-0.0855	0.0100	0.1000
14.000	1.4101	0.07203	3.00000	-0.0866	0.0100	0.1000
14.500	1.4166	0.07901	3.00000	-0.0880	0.0100	0.1000
15.000	1.4220	0.08631	3.00000	-0.0895	0.0100	0.1000
15.500	1.4280	0.09357	3.00000	-0.0910	0.0100	0.1000
16.000	1.4307	0.10157	3.00000	-0.0928	0.0100	0.1000
16.500	1.4320	0.11004	3.00000	-0.0949	0.0100	0.1000
17.000	1.4349	0.11835	3.00000	-0.0970	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W3-301

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-4.00000	-0.24430	0.01368	1.00000	-0.06390	0.47880	0.39260
-3.50000	-0.16470	0.01320	1.00000	-0.06980	0.46260	0.39900
-3.00000	-0.08760	0.01290	1.00000	-0.07470	0.45280	0.40330
-2.50000	-0.00720	0.01252	1.00000	-0.08050	0.43630	0.41240
-2.00000	0.07040	0.01228	1.00000	-0.08540	0.42680	0.42000
-1.50000	0.14360	0.01223	1.00000	-0.08900	0.41250	0.42470
-1.00000	0.21500	0.01223	1.00000	-0.09210	0.40530	0.42830
-0.50000	0.28870	0.01217	1.00000	-0.09580	0.39290	0.43750
0.000	0.3598	0.01220	1.00000	-0.0988	0.3833	0.4440
0.500	0.4292	0.01230	1.00000	-0.1013	0.3751	0.4482
1.500	0.5671	0.01253	1.00000	-0.1062	0.3576	0.4590
2.000	0.6357	0.01266	1.00000	-0.1085	0.3502	0.4660
2.500	0.7021	0.01290	1.00000	-0.1104	0.3374	0.4703
3.000	0.7680	0.01311	1.00000	-0.1120	0.3332	0.4742
3.500	0.8330	0.01337	1.00000	-0.1135	0.3272	0.4770
4.000	0.8994	0.01358	1.00000	-0.1155	0.3176	0.4854
4.500	0.9635	0.01387	1.00000	-0.1168	0.3104	0.4909
5.000	1.0263	0.01420	1.00000	-0.1178	0.3060	0.4950
5.500	1.0878	0.01457	1.00000	-0.1187	0.2997	0.4982
6.000	1.1470	0.01506	1.00000	-0.1191	0.2904	0.5009
6.500	1.2077	0.01541	1.00000	-0.1199	0.2860	0.5088
7.000	1.2657	0.01584	1.00000	-0.1201	0.2815	0.5145
7.500	1.3206	0.01640	1.00000	-0.1198	0.2761	0.5180
8.000	1.3711	0.01709	1.00000	-0.1188	0.2683	0.5211
8.500	1.4160	0.01792	1.00000	-0.1170	0.2625	0.5238
9.000	1.4465	0.01884	1.00000	-0.1127	0.2600	0.5285
9.500	1.4824	0.02007	1.00000	-0.1100	0.2557	0.5356
10.000	1.5124	0.02176	1.00000	-0.1071	0.2505	0.5394
10.500	1.5354	0.02409	1.00000	-0.1040	0.2435	0.5424
11.000	1.5543	0.02695	1.00000	-0.1012	0.2385	0.5452
11.500	1.5815	0.02955	1.00000	-0.0996	0.2354	0.5476
12.000	1.6026	0.03292	1.00000	-0.0983	0.2319	0.5520
12.500	1.6200	0.03701	1.00000	-0.0974	0.2268	0.5583
13.000	1.6284	0.04216	1.00000	-0.0966	0.2197	0.5619
13.500	1.6374	0.04761	1.00000	-0.0963	0.2141	0.5648
14.000	1.6521	0.05271	1.00000	-0.0964	0.2113	0.5675
14.500	1.6606	0.05873	1.00000	-0.0967	0.2074	0.5699
15.000	1.6641	0.06550	1.00000	-0.0971	0.2015	0.5720
15.500	1.6535	0.07418	1.00000	-0.0980	0.1922	0.5749
16.000	1.6630	0.08078	1.00000	-0.0992	0.1897	0.5813
16.500	1.6643	0.08856	1.00000	-0.1005	0.1861	0.5849

XFOIL Version 5.0

Calculated polar for: FFA-W3-301

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
4.000	0.7320	0.02195	1.00000	-0.0822	0.0100	0.1000
4.500	0.7922	0.02235	1.00000	-0.0843	0.0100	0.1000
5.000	0.8287	0.02299	1.00000	-0.0823	0.0100	0.1000
5.500	0.8680	0.02408	1.00000	-0.0813	0.0100	0.1000
6.000	0.9089	0.02550	1.00000	-0.0810	0.0100	0.1000
6.500	0.9457	0.02730	1.00000	-0.0806	0.0100	0.1000
7.000	0.9790	0.02954	1.00000	-0.0804	0.0100	0.1000
7.500	1.0080	0.03230	1.00000	-0.0803	0.0100	0.1000
8.000	1.0352	0.03556	1.00000	-0.0806	0.0100	0.1000
8.500	1.0601	0.03936	1.00000	-0.0813	0.0100	0.1000
9.000	1.0828	0.04368	1.00000	-0.0823	0.0100	0.1000
9.500	1.1031	0.04850	1.00000	-0.0835	0.0100	0.1000
10.000	1.1220	0.05374	1.00000	-0.0849	0.0100	0.1000
10.500	1.1387	0.05952	1.00000	-0.0865	0.0100	0.1000
11.000	1.1531	0.06577	1.00000	-0.0883	0.0100	0.1000
11.500	1.1651	0.07243	1.00000	-0.0902	0.0100	0.1000
12.000	1.1751	0.07950	1.00000	-0.0921	0.0100	0.1000
12.500	1.1841	0.08692	1.00000	-0.0941	0.0100	0.1000
13.000	1.1919	0.09469	1.00000	-0.0963	0.0100	0.1000
13.500	1.1978	0.10302	1.00000	-0.0987	0.0100	0.1000
14.000	1.2023	0.11170	1.00000	-0.1013	0.0100	0.1000
14.500	1.2066	0.12051	1.00000	-0.1038	0.0100	0.1000
15.000	1.2105	0.12971	1.00000	-0.1065	0.0100	0.1000
15.500	1.2126	0.13958	1.00000	-0.1095	0.0100	0.1000
16.000	1.2166	0.14922	1.00000	-0.1125	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W3-301

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 2.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.51560	0.01337	2.00000	-0.05390	0.49750	0.34350
-5.50000	-0.43530	0.01266	2.00000	-0.06030	0.48080	0.35150
-5.00000	-0.35350	0.01198	2.00000	-0.06690	0.46860	0.36220
-4.50000	-0.27310	0.01146	2.00000	-0.07280	0.45360	0.36720
-4.00000	-0.19380	0.01106	2.00000	-0.07820	0.44280	0.37240
-3.50000	-0.11530	0.01073	2.00000	-0.08330	0.42600	0.38250
-3.00000	-0.03950	0.01052	2.00000	-0.08760	0.41680	0.38980
-2.50000	0.03290	0.01045	2.00000	-0.09090	0.40580	0.39330
-2.00000	0.10610	0.01036	2.00000	-0.09440	0.39380	0.40060
-1.50000	0.17790	0.01032	2.00000	-0.09750	0.38700	0.40910
-1.00000	0.24910	0.01030	2.00000	-0.10040	0.37180	0.41510
-0.50000	0.31870	0.01035	2.00000	-0.10290	0.36580	0.41820
0.000	0.3890	0.01038	2.00000	-0.1056	0.3584	0.4252
0.500	0.4587	0.01042	2.00000	-0.1081	0.3467	0.4332
1.000	0.5275	0.01051	2.00000	-0.1104	0.3399	0.4391
1.500	0.5952	0.01064	2.00000	-0.1124	0.3335	0.4422
2.000	0.6632	0.01076	2.00000	-0.1144	0.3235	0.4468
2.500	0.7311	0.01088	2.00000	-0.1165	0.3165	0.4549
3.000	0.7982	0.01103	2.00000	-0.1183	0.3124	0.4615
3.500	0.8641	0.01125	2.00000	-0.1199	0.3038	0.4652
4.000	0.9296	0.01146	2.00000	-0.1214	0.2942	0.4676
4.500	0.9955	0.01165	2.00000	-0.1230	0.2903	0.4753
5.000	1.0603	0.01189	2.00000	-0.1243	0.2864	0.4820
5.500	1.1241	0.01217	2.00000	-0.1255	0.2804	0.4866
6.000	1.1870	0.01248	2.00000	-0.1264	0.2701	0.4899
6.500	1.2488	0.01282	2.00000	-0.1272	0.2659	0.4925
7.000	1.3109	0.01313	2.00000	-0.1280	0.2628	0.4990
7.500	1.3710	0.01352	2.00000	-0.1285	0.2583	0.5056
8.000	1.4292	0.01396	2.00000	-0.1286	0.2496	0.5103
9.000	1.5407	0.01494	2.00000	-0.1279	0.2404	0.5163
9.500	1.5930	0.01552	2.00000	-0.1271	0.2373	0.5196
10.000	1.6399	0.01628	2.00000	-0.1254	0.2309	0.5270
10.500	1.6672	0.01730	2.00000	-0.1204	0.2248	0.5321
11.000	1.6953	0.01879	2.00000	-0.1163	0.2181	0.5363
11.500	1.7269	0.02030	2.00000	-0.1132	0.2165	0.5389
12.000	1.7488	0.02249	2.00000	-0.1097	0.2128	0.5409
12.500	1.7666	0.02530	2.00000	-0.1065	0.2083	0.5426
13.000	1.7877	0.02831	2.00000	-0.1043	0.2020	0.5495
13.500	1.7989	0.03243	2.00000	-0.1022	0.1942	0.5547
14.000	1.8147	0.03640	2.00000	-0.1009	0.1922	0.5592
14.500	1.8226	0.04154	2.00000	-0.0998	0.1889	0.5619
15.000	1.8275	0.04735	2.00000	-0.0992	0.1840	0.5640
15.500	1.8351	0.05314	2.00000	-0.0990	0.1775	0.5658
16.000	1.8347	0.06000	2.00000	-0.0990	0.1707	0.5672
16.500	1.8349	0.06715	2.00000	-0.0996	0.1682	0.5722
17.000	1.8265	0.07573	2.00000	-0.1005	0.1641	0.5770

XFOIL Version 5.0

Calculated polar for: FFA-W3-301

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-4.00000	-0.17070	0.00991	3.00000	-0.08460	0.41810	0.36260
-3.50000	-0.09660	0.00976	3.00000	-0.08830	0.40930	0.37330
-3.00000	-0.02360	0.00965	3.00000	-0.09180	0.39830	0.37830
-2.50000	0.04830	0.00958	3.00000	-0.09480	0.38690	0.38130
-2.00000	0.11960	0.00955	3.00000	-0.09770	0.38030	0.39060
-1.50000	0.19090	0.00951	3.00000	-0.10070	0.36510	0.39940
-1.00000	0.26100	0.00952	3.00000	-0.10330	0.35950	0.40340
-0.50000	0.33060	0.00957	3.00000	-0.10580	0.35180	0.40730
0.000	0.4002	0.00959	3.00000	-0.1082	0.3405	0.4160
0.500	0.4695	0.00964	3.00000	-0.1105	0.3354	0.4238
1.000	0.5377	0.00975	3.00000	-0.1127	0.3272	0.4274
1.500	0.6059	0.00984	3.00000	-0.1147	0.3173	0.4304
2.000	0.6739	0.00994	3.00000	-0.1167	0.3115	0.4387
2.500	0.7415	0.01007	3.00000	-0.1187	0.3071	0.4466
3.000	0.8083	0.01024	3.00000	-0.1204	0.2974	0.4504
3.500	0.8748	0.01040	3.00000	-0.1220	0.2883	0.4528
4.000	0.9411	0.01057	3.00000	-0.1237	0.2857	0.4590
4.500	1.0068	0.01078	3.00000	-0.1252	0.2812	0.4676
5.000	1.0720	0.01099	3.00000	-0.1265	0.2743	0.4722
5.500	1.1366	0.01123	3.00000	-0.1278	0.2642	0.4753
6.000	1.2003	0.01149	3.00000	-0.1288	0.2618	0.4785
6.500	1.2637	0.01177	3.00000	-0.1299	0.2582	0.4860
7.000	1.3260	0.01209	3.00000	-0.1307	0.2526	0.4922
7.500	1.3876	0.01242	3.00000	-0.1313	0.2436	0.4964
8.000	1.4479	0.01278	3.00000	-0.1317	0.2389	0.4997
8.500	1.5070	0.01317	3.00000	-0.1319	0.2365	0.5015
9.000	1.5642	0.01364	3.00000	-0.1318	0.2325	0.5089
9.500	1.6191	0.01416	3.00000	-0.1314	0.2244	0.5150
10.000	1.6732	0.01467	3.00000	-0.1307	0.2176	0.5193
10.500	1.7231	0.01530	3.00000	-0.1293	0.2146	0.5228
11.000	1.7541	0.01627	3.00000	-0.1249	0.2108	0.5246
11.500	1.7789	0.01768	3.00000	-0.1199	0.2062	0.5285
12.000	1.8078	0.01922	3.00000	-0.1162	0.2001	0.5347
12.500	1.8348	0.02109	3.00000	-0.1129	0.1939	0.5398
13.000	1.8513	0.02378	3.00000	-0.1093	0.1904	0.5440
13.500	1.8641	0.02714	3.00000	-0.1061	0.1871	0.5464
14.000	1.8752	0.03111	3.00000	-0.1037	0.1822	0.5481
14.500	1.8904	0.03511	3.00000	-0.1021	0.1761	0.5496
15.500	1.9015	0.04594	3.00000	-0.1002	0.1669	0.5603
16.000	1.9002	0.05261	3.00000	-0.0998	0.1629	0.5652
16.500	1.9000	0.05963	3.00000	-0.1000	0.1570	0.5678

XFOIL Version 5.0

Calculated polar for: FFA-W3-301

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.42870	0.03149	3.00000	-0.03100	0.01000	0.10000
-5.50000	-0.39690	0.02965	3.00000	-0.03070	0.01000	0.10000
-5.00000	-0.38250	0.02750	3.00000	-0.02830	0.01000	0.10000
-4.50000	-0.34000	0.02600	3.00000	-0.02970	0.01000	0.10000
-4.00000	-0.29220	0.02452	3.00000	-0.03200	0.01000	0.10000
-3.50000	-0.23960	0.02310	3.00000	-0.03500	0.01000	0.10000
-3.00000	-0.18010	0.02204	3.00000	-0.03850	0.01000	0.10000
-2.50000	-0.11630	0.02093	3.00000	-0.04300	0.01000	0.10000
-2.00000	-0.04860	0.02016	3.00000	-0.04750	0.01000	0.10000
-1.50000	0.02230	0.01948	3.00000	-0.05240	0.01000	0.10000
-1.00000	0.09560	0.01884	3.00000	-0.05770	0.01000	0.10000
-0.50000	0.16950	0.01840	3.00000	-0.06260	0.01000	0.10000
0.000	0.2449	0.01799	3.00000	-0.0679	0.0100	0.1000
0.500	0.3202	0.01770	3.00000	-0.0729	0.0100	0.1000
1.000	0.3953	0.01748	3.00000	-0.0777	0.0100	0.1000
1.500	0.4703	0.01732	3.00000	-0.0825	0.0100	0.1000
2.000	0.5442	0.01725	3.00000	-0.0868	0.0100	0.1000
2.500	0.6170	0.01724	3.00000	-0.0909	0.0100	0.1000
3.000	0.6885	0.01730	3.00000	-0.0946	0.0100	0.1000
3.500	0.7572	0.01746	3.00000	-0.0976	0.0100	0.1000
4.000	0.8227	0.01774	3.00000	-0.0999	0.0100	0.1000
4.500	0.8860	0.01808	3.00000	-0.1016	0.0100	0.1000
5.000	0.9469	0.01849	3.00000	-0.1030	0.0100	0.1000
5.500	1.0046	0.01898	3.00000	-0.1037	0.0100	0.1000
6.000	1.0581	0.01955	3.00000	-0.1038	0.0100	0.1000
6.500	1.1019	0.02027	3.00000	-0.1021	0.0100	0.1000
7.000	1.1201	0.02152	3.00000	-0.0966	0.0100	0.1000
7.500	1.1539	0.02307	3.00000	-0.0946	0.0100	0.1000
8.000	1.1829	0.02506	3.00000	-0.0926	0.0100	0.1000
8.500	1.2075	0.02758	3.00000	-0.0909	0.0100	0.1000
9.000	1.2309	0.03060	3.00000	-0.0898	0.0100	0.1000
9.500	1.2522	0.03419	3.00000	-0.0892	0.0100	0.1000
10.000	1.2701	0.03842	3.00000	-0.0891	0.0100	0.1000
10.500	1.2878	0.04308	3.00000	-0.0894	0.0100	0.1000
11.000	1.3038	0.04828	3.00000	-0.0902	0.0100	0.1000
11.500	1.3173	0.05414	3.00000	-0.0913	0.0100	0.1000
12.000	1.3294	0.06031	3.00000	-0.0925	0.0100	0.1000
12.500	1.3388	0.06692	3.00000	-0.0939	0.0100	0.1000
13.000	1.3465	0.07402	3.00000	-0.0954	0.0100	0.1000
13.500	1.3524	0.08160	3.00000	-0.0971	0.0100	0.1000
14.000	1.3567	0.08961	3.00000	-0.0991	0.0100	0.1000
14.500	1.3601	0.09792	3.00000	-0.1011	0.0100	0.1000
15.000	1.3634	0.10615	3.00000	-0.1031	0.0100	0.1000
15.500	1.3630	0.11548	3.00000	-0.1055	0.0100	0.1000
16.000	1.3617	0.12528	3.00000	-0.1082	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W3-332

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-4.000	-0.3143	0.01711	1.00000	-0.0446	0.4411	0.3742
-3.500	-0.2353	0.01630	1.00000	-0.0510	0.4280	0.3827
-3.000	-0.1556	0.01574	1.00000	-0.0569	0.4139	0.3880
-2.500	-0.0714	0.01516	1.00000	-0.0639	0.4026	0.3982
-2.000	0.0111	0.01479	1.00000	-0.0700	0.3894	0.4051
-1.500	0.0933	0.01452	1.00000	-0.0758	0.3790	0.4112
-1.000	0.1770	0.01427	1.00000	-0.0819	0.3667	0.4204
-0.500	0.2565	0.01418	1.00000	-0.0868	0.3552	0.4253
0.000	0.3372	0.01409	1.00000	-0.0919	0.3464	0.4330
0.500	0.4155	0.01409	1.00000	-0.0964	0.3362	0.4398
1.000	0.4894	0.01421	1.00000	-0.0997	0.3269	0.4441
1.500	0.5630	0.01435	1.00000	-0.1030	0.3186	0.4499
2.000	0.6363	0.01450	1.00000	-0.1061	0.3094	0.4571
2.500	0.7069	0.01472	1.00000	-0.1086	0.3027	0.4614
3.000	0.7758	0.01501	1.00000	-0.1107	0.2926	0.4653
3.500	0.8462	0.01523	1.00000	-0.1132	0.2876	0.4721
4.000	0.9141	0.01554	1.00000	-0.1152	0.2807	0.4775
4.000	0.9141	0.01554	1.00000	-0.1152	0.2807	0.4775
4.500	0.9794	0.01596	1.00000	-0.1166	0.2713	0.4816
5.000	1.0445	0.01630	1.00000	-0.1179	0.2677	0.4848
5.500	1.1097	0.01668	1.00000	-0.1193	0.2627	0.4914
6.000	1.1714	0.01719	1.00000	-0.1201	0.2549	0.4969
6.000	1.1714	0.01719	1.00000	-0.1201	0.2549	0.4969
6.500	1.2310	0.01773	1.00000	-0.1205	0.2493	0.5005
7.000	1.2887	0.01831	1.00000	-0.1205	0.2453	0.5036
7.500	1.3442	0.01898	1.00000	-0.1203	0.2396	0.5091
8.000	1.3920	0.01993	1.00000	-0.1189	0.2321	0.5150
8.000	1.3920	0.01993	1.00000	-0.1189	0.2321	0.5150
8.500	1.4226	0.02102	1.00000	-0.1145	0.2297	0.5185
9.000	1.4549	0.02255	1.00000	-0.1111	0.2257	0.5216
9.500	1.4821	0.02458	1.00000	-0.1078	0.2208	0.5245
10.000	1.5027	0.02741	1.00000	-0.1050	0.2140	0.5314
10.500	1.5241	0.03053	1.00000	-0.1030	0.2112	0.5355
11.000	1.5433	0.03426	1.00000	-0.1016	0.2084	0.5387
11.500	1.5575	0.03889	1.00000	-0.1008	0.2047	0.5416
12.000	1.5655	0.04456	1.00000	-0.1006	0.1976	0.5451
12.500	1.5723	0.05092	1.00000	-0.1012	0.1932	0.5510
13.000	1.5832	0.05703	1.00000	-0.1019	0.1907	0.5548
13.500	1.5888	0.06389	1.00000	-0.1028	0.1873	0.5578
14.000	1.5880	0.07180	1.00000	-0.1040	0.1823	0.5608
14.500	1.5754	0.08153	1.00000	-0.1056	0.1757	0.5629
15.000	1.5816	0.08913	1.00000	-0.1073	0.1736	0.5688
15.500	1.5817	0.09762	1.00000	-0.1091	0.1709	0.5729
16.000	1.5767	0.10709	1.00000	-0.1112	0.1671	0.5758

XFOIL Version 5.0

Calculated polar for : FFA-W3-332

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-4.000	-0.1328	0.03911	1.00000	-0.0349	0.0100	0.1000
-3.500	-0.1030	0.03706	1.00000	-0.0345	0.0100	0.1000
-3.000	-0.0742	0.03515	1.00000	-0.0337	0.0100	0.1000
-2.500	-0.0499	0.03327	1.00000	-0.0321	0.0100	0.1000
-2.000	-0.0417	0.03150	1.00000	-0.0282	0.0100	0.1000
-1.500	-0.0073	0.03005	1.00000	-0.0281	0.0100	0.1000
-1.000	0.0342	0.02868	1.00000	-0.0292	0.0100	0.1000
-0.500	0.0835	0.02756	1.00000	-0.0314	0.0100	0.1000
0.000	0.1389	0.02660	1.00000	-0.0342	0.0100	0.1000
0.500	0.2002	0.02582	1.00000	-0.0379	0.0100	0.1000
1.000	0.2652	0.02525	1.00000	-0.0419	0.0100	0.1000
1.500	0.3332	0.02481	1.00000	-0.0463	0.0100	0.1000
2.000	0.4023	0.02456	1.00000	-0.0506	0.0100	0.1000
2.500	0.4722	0.02441	1.00000	-0.0550	0.0100	0.1000
3.000	0.5411	0.02444	1.00000	-0.0591	0.0100	0.1000
3.500	0.6085	0.02459	1.00000	-0.0629	0.0100	0.1000
4.000	0.6672	0.02491	1.00000	-0.0651	0.0100	0.1000
4.500	0.7054	0.02557	1.00000	-0.0641	0.0100	0.1000
5.000	0.7568	0.02664	1.00000	-0.0658	0.0100	0.1000
5.500	0.8060	0.02810	1.00000	-0.0676	0.0100	0.1000
6.000	0.8495	0.03006	1.00000	-0.0693	0.0100	0.1000
6.500	0.8880	0.03261	1.00000	-0.0710	0.0100	0.1000
7.000	0.9204	0.03586	1.00000	-0.0725	0.0100	0.1000
7.500	0.9496	0.03975	1.00000	-0.0743	0.0100	0.1000
8.000	0.9764	0.04425	1.00000	-0.0765	0.0100	0.1000
8.500	1.0002	0.04934	1.00000	-0.0789	0.0100	0.1000
9.000	1.0215	0.05500	1.00000	-0.0814	0.0100	0.1000
9.500	1.0404	0.06115	1.00000	-0.0841	0.0100	0.1000
10.000	1.0568	0.06776	1.00000	-0.0868	0.0100	0.1000
10.500	1.0706	0.07483	1.00000	-0.0895	0.0100	0.1000
11.000	1.0822	0.08244	1.00000	-0.0923	0.0100	0.1000
11.500	1.0924	0.09041	1.00000	-0.0952	0.0100	0.1000
12.000	1.1007	0.09876	1.00000	-0.0981	0.0100	0.1000
12.500	1.1074	0.10756	1.00000	-0.1011	0.0100	0.1000
13.000	1.1131	0.11678	1.00000	-0.1042	0.0100	0.1000
13.500	1.1179	0.12640	1.00000	-0.1075	0.0100	0.1000
14.000	1.1219	0.13629	1.00000	-0.1108	0.0100	0.1000
14.500	1.1259	0.14644	1.00000	-0.1142	0.0100	0.1000
15.000	1.1284	0.15729	1.00000	-0.1178	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: FFA-W3-332

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 2.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-4.000	-0.2651	0.01400	2.00000	-0.0603	0.4073	0.3557
-3.500	-0.1812	0.01350	2.00000	-0.0668	0.3912	0.3644
-3.000	-0.0986	0.01313	2.00000	-0.0727	0.3822	0.3696
-2.500	-0.0136	0.01274	2.00000	-0.0792	0.3695	0.3800
-2.000	0.0695	0.01248	2.00000	-0.0850	0.3571	0.3868
-1.500	0.1519	0.01226	2.00000	-0.0905	0.3476	0.3947
-1.000	0.2323	0.01212	2.00000	-0.0954	0.3370	0.4026
-0.500	0.3094	0.01208	2.00000	-0.0995	0.3283	0.4077
0.000	0.3853	0.01210	2.00000	-0.1032	0.3179	0.4157
0.500	0.4597	0.01215	2.00000	-0.1065	0.3092	0.4232
1.000	0.5321	0.01226	2.00000	-0.1093	0.3031	0.4270
1.500	0.6050	0.01237	2.00000	-0.1122	0.2910	0.4345
2.000	0.6766	0.01249	2.00000	-0.1148	0.2869	0.4417
2.500	0.7467	0.01268	2.00000	-0.1170	0.2797	0.4454
3.000	0.8166	0.01287	2.00000	-0.1192	0.2704	0.4506
3.500	0.8862	0.01307	2.00000	-0.1214	0.2664	0.4585
4.000	0.9544	0.01332	2.00000	-0.1231	0.2598	0.4626
4.000	0.9544	0.01332	2.00000	-0.1231	0.2598	0.4626
4.500	1.0213	0.01361	2.00000	-0.1246	0.2516	0.4655
5.000	1.0889	0.01386	2.00000	-0.1263	0.2487	0.4724
5.500	1.1544	0.01422	2.00000	-0.1276	0.2426	0.4781
6.000	1.2192	0.01456	2.00000	-0.1287	0.2349	0.4818
6.000	1.2192	0.01456	2.00000	-0.1287	0.2349	0.4818
6.500	1.2824	0.01496	2.00000	-0.1295	0.2308	0.4845
7.000	1.3451	0.01537	2.00000	-0.1302	0.2275	0.4902
7.500	1.4054	0.01588	2.00000	-0.1306	0.2205	0.4961
8.000	1.4638	0.01645	2.00000	-0.1305	0.2131	0.5001
8.000	1.4638	0.01645	2.00000	-0.1305	0.2131	0.5001
8.500	1.5207	0.01700	2.00000	-0.1303	0.2110	0.5033
9.000	1.5737	0.01770	2.00000	-0.1294	0.2076	0.5065
9.500	1.6228	0.01852	2.00000	-0.1279	0.2023	0.5132
10.000	1.6472	0.01982	2.00000	-0.1223	0.1963	0.5181
10.500	1.6735	0.02155	2.00000	-0.1180	0.1927	0.5211
11.000	1.6970	0.02371	2.00000	-0.1141	0.1901	0.5234
11.500	1.7144	0.02659	2.00000	-0.1106	0.1864	0.5277
12.000	1.7316	0.02998	2.00000	-0.1081	0.1811	0.5334
12.500	1.7423	0.03442	2.00000	-0.1062	0.1758	0.5379
13.000	1.7559	0.03905	2.00000	-0.1052	0.1742	0.5407
13.500	1.7636	0.04467	2.00000	-0.1047	0.1718	0.5430
14.000	1.7678	0.05114	2.00000	-0.1049	0.1683	0.5471
14.500	1.7725	0.05791	2.00000	-0.1054	0.1633	0.5533
15.000	1.7703	0.06562	2.00000	-0.1062	0.1575	0.5568
15.500	1.7714	0.07326	2.00000	-0.1072	0.1560	0.5597
16.000	1.7684	0.08157	2.00000	-0.1084	0.1537	0.5618

XFOIL Version 5.0

Calculated polar for: FFA-W3-332

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-4.000	-0.2330	0.01251	3.00000	-0.0694	0.3868	0.3464
-3.500	-0.1480	0.01209	3.00000	-0.0758	0.3718	0.3509
-3.000	-0.0636	0.01176	3.00000	-0.0820	0.3646	0.3630
-2.500	0.0195	0.01151	3.00000	-0.0877	0.3500	0.3682
-2.000	0.1010	0.01132	3.00000	-0.0929	0.3416	0.3752
-1.500	0.1799	0.01121	3.00000	-0.0974	0.3302	0.3846
-1.000	0.2556	0.01120	3.00000	-0.1010	0.3235	0.3890
-0.500	0.3307	0.01120	3.00000	-0.1045	0.3105	0.3983
0.000	0.4052	0.01123	3.00000	-0.1078	0.3052	0.4054
0.500	0.4782	0.01129	3.00000	-0.1107	0.2951	0.4088
1.000	0.5507	0.01138	3.00000	-0.1135	0.2881	0.4195
1.500	0.6224	0.01150	3.00000	-0.1160	0.2813	0.4247
2.000	0.6931	0.01162	3.00000	-0.1183	0.2715	0.4278
2.500	0.7638	0.01178	3.00000	-0.1207	0.2672	0.4379
3.000	0.8335	0.01196	3.00000	-0.1228	0.2606	0.4431
3.500	0.9026	0.01214	3.00000	-0.1246	0.2520	0.4463
4.000	0.9713	0.01235	3.00000	-0.1265	0.2492	0.4534
4.000	0.9713	0.01235	3.00000	-0.1265	0.2492	0.4534
4.500	1.0388	0.01263	3.00000	-0.1281	0.2429	0.4596
5.000	1.1062	0.01287	3.00000	-0.1296	0.2354	0.4634
5.500	1.1721	0.01317	3.00000	-0.1309	0.2311	0.4662
6.000	1.2380	0.01348	3.00000	-0.1322	0.2278	0.4728
6.000	1.2380	0.01348	3.00000	-0.1322	0.2278	0.4728
6.500	1.3026	0.01384	3.00000	-0.1332	0.2222	0.4785
7.000	1.3663	0.01420	3.00000	-0.1340	0.2132	0.4827
7.500	1.4282	0.01463	3.00000	-0.1344	0.2111	0.4850
8.000	1.4888	0.01511	3.00000	-0.1347	0.2073	0.4898
8.000	1.4888	0.01511	3.00000	-0.1347	0.2073	0.4898
8.500	1.5481	0.01561	3.00000	-0.1348	0.2014	0.4959
9.000	1.6061	0.01613	3.00000	-0.1346	0.1947	0.5006
9.500	1.6591	0.01684	3.00000	-0.1336	0.1918	0.5034
10.000	1.7080	0.01766	3.00000	-0.1320	0.1881	0.5056
10.500	1.7380	0.01878	3.00000	-0.1272	0.1830	0.5130
11.000	1.7652	0.02029	3.00000	-0.1225	0.1775	0.5177
11.500	1.7882	0.02231	3.00000	-0.1181	0.1750	0.5208
12.000	1.8060	0.02493	3.00000	-0.1141	0.1729	0.5232
12.500	1.8179	0.02843	3.00000	-0.1107	0.1698	0.5259
13.000	1.8303	0.03250	3.00000	-0.1084	0.1654	0.5330
13.500	1.8433	0.03701	3.00000	-0.1070	0.1605	0.5373
14.000	1.8481	0.04268	3.00000	-0.1062	0.1569	0.5403
14.500	1.8528	0.04887	3.00000	-0.1060	0.1551	0.5426
15.000	1.8524	0.05603	3.00000	-0.1063	0.1524	0.5462
15.500	1.8524	0.06345	3.00000	-0.1069	0.1485	0.5517
16.000	1.8550	0.07066	3.00000	-0.1077	0.1442	0.5561

XFOIL Version 5.0

Calculated polar for: FFA-W3-332

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.34170	0.03905	3.00000	-0.03360	0.01000	0.10000
-5.50000	-0.31130	0.03665	3.00000	-0.03390	0.01000	0.10000
-5.00000	-0.28010	0.03455	3.00000	-0.03390	0.01000	0.10000
-4.50000	-0.24850	0.03260	3.00000	-0.03370	0.01000	0.10000
-4.00000	-0.22220	0.03038	3.00000	-0.03300	0.01000	0.10000
-3.50000	-0.21200	0.02856	3.00000	-0.02930	0.01000	0.10000
-3.00000	-0.17630	0.02678	3.00000	-0.03000	0.01000	0.10000
-2.50000	-0.13010	0.02545	3.00000	-0.03180	0.01000	0.10000
-2.00000	-0.07700	0.02408	3.00000	-0.03490	0.01000	0.10000
-1.50000	-0.01750	0.02313	3.00000	-0.03840	0.01000	0.10000
-1.00000	0.04810	0.02220	3.00000	-0.04290	0.01000	0.10000
-0.50000	0.11730	0.02151	3.00000	-0.04770	0.01000	0.10000
0.000	0.1892	0.02097	3.00000	-0.0527	0.0100	0.1000
0.500	0.2643	0.02047	3.00000	-0.0582	0.0100	0.1000
1.000	0.3393	0.02017	3.00000	-0.0633	0.0100	0.1000
1.500	0.4157	0.01991	3.00000	-0.0687	0.0100	0.1000
2.000	0.4916	0.01977	3.00000	-0.0737	0.0100	0.1000
2.500	0.5674	0.01971	3.00000	-0.0787	0.0100	0.1000
3.000	0.6424	0.01974	3.00000	-0.0834	0.0100	0.1000
3.500	0.7160	0.01985	3.00000	-0.0877	0.0100	0.1000
4.000	0.7880	0.02004	3.00000	-0.0917	0.0100	0.1000
4.500	0.8575	0.02032	3.00000	-0.0952	0.0100	0.1000
5.000	0.9234	0.02070	3.00000	-0.0979	0.0100	0.1000
5.500	0.9839	0.02124	3.00000	-0.0997	0.0100	0.1000
6.000	1.0185	0.02207	3.00000	-0.0967	0.0100	0.1000
6.500	1.0501	0.02346	3.00000	-0.0939	0.0100	0.1000
7.000	1.0846	0.02522	3.00000	-0.0924	0.0100	0.1000
7.500	1.1139	0.02753	3.00000	-0.0910	0.0100	0.1000
8.000	1.1389	0.03050	3.00000	-0.0901	0.0100	0.1000
8.500	1.1598	0.03416	3.00000	-0.0897	0.0100	0.1000
9.000	1.1795	0.03849	3.00000	-0.0901	0.0100	0.1000
9.500	1.1973	0.04354	3.00000	-0.0911	0.0100	0.1000
10.000	1.2129	0.04930	3.00000	-0.0926	0.0100	0.1000
10.500	1.2275	0.05537	3.00000	-0.0943	0.0100	0.1000
11.000	1.2372	0.06236	3.00000	-0.0962	0.0100	0.1000
11.500	1.2457	0.06973	3.00000	-0.0983	0.0100	0.1000
12.000	1.2522	0.07758	3.00000	-0.1005	0.0100	0.1000
12.500	1.2564	0.08593	3.00000	-0.1028	0.0100	0.1000
13.000	1.2583	0.09479	3.00000	-0.1052	0.0100	0.1000
13.500	1.2577	0.10411	3.00000	-0.1078	0.0100	0.1000
14.000	1.2564	0.11390	3.00000	-0.1105	0.0100	0.1000
14.500	1.2575	0.12342	3.00000	-0.1132	0.0100	0.1000
15.000	1.2551	0.13402	3.00000	-0.1163	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: V360NR1

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-4.000	-0.2509	0.01951	1.00000	-0.0443	0.4172	0.3700
-3.500	-0.1801	0.01830	1.00000	-0.0503	0.4053	0.3803
-3.000	-0.1032	0.01746	1.00000	-0.0565	0.3975	0.3869
-2.500	-0.0262	0.01699	1.00000	-0.0620	0.3840	0.3917
-2.000	0.0590	0.01633	1.00000	-0.0695	0.3769	0.4017
-1.500	0.1414	0.01598	1.00000	-0.0757	0.3647	0.4083
-1.000	0.2220	0.01578	1.00000	-0.0812	0.3584	0.4123
-0.500	0.3083	0.01550	1.00000	-0.0882	0.3480	0.4225
0.000	0.3894	0.01542	1.00000	-0.0937	0.3394	0.4280
0.500	0.4681	0.01543	1.00000	-0.0984	0.3336	0.4320
1.000	0.5497	0.01543	1.00000	-0.1039	0.3238	0.4403
1.500	0.6276	0.01552	1.00000	-0.1084	0.3176	0.4463
2.000	0.7010	0.01572	1.00000	-0.1119	0.3117	0.4503
2.500	0.7709	0.01602	1.00000	-0.1145	0.3035	0.4538
3.000	0.8432	0.01627	1.00000	-0.1178	0.2973	0.4620
3.500	0.9114	0.01657	1.00000	-0.1202	0.2934	0.4669
4.000	0.9762	0.01699	1.00000	-0.1218	0.2872	0.4705
4.000	0.9762	0.01699	1.00000	-0.1218	0.2872	0.4705
4.500	1.0368	0.01755	1.00000	-0.1227	0.2791	0.4739
5.000	1.0995	0.01801	1.00000	-0.1241	0.2757	0.4802
5.500	1.1578	0.01857	1.00000	-0.1247	0.2724	0.4858
6.000	1.1979	0.01941	1.00000	-0.1222	0.2676	0.4893
6.000	1.1979	0.01941	1.00000	-0.1222	0.2676	0.4893
6.500	1.2304	0.02076	1.00000	-0.1188	0.2617	0.4925
7.000	1.2612	0.02254	1.00000	-0.1159	0.2565	0.4951
7.500	1.2995	0.02421	1.00000	-0.1148	0.2543	0.5014
8.000	1.3307	0.02646	1.00000	-0.1132	0.2515	0.5063
8.000	1.3307	0.02646	1.00000	-0.1132	0.2515	0.5063
8.500	1.3554	0.02939	1.00000	-0.1117	0.2480	0.5099
9.000	1.3736	0.03319	1.00000	-0.1105	0.2418	0.5127
9.500	1.3820	0.03836	1.00000	-0.1097	0.2364	0.5151
10.000	1.4062	0.04252	1.00000	-0.1102	0.2343	0.5190
10.500	1.4259	0.04745	1.00000	-0.1109	0.2318	0.5254
11.000	1.4399	0.05299	1.00000	-0.1115	0.2285	0.5289
11.500	1.4470	0.05956	1.00000	-0.1123	0.2235	0.5317
12.000	1.4380	0.06815	1.00000	-0.1132	0.2168	0.5340
12.500	1.4550	0.07390	1.00000	-0.1143	0.2146	0.5364
13.000	1.4663	0.08055	1.00000	-0.1157	0.2121	0.5406
13.500	1.4723	0.08807	1.00000	-0.1173	0.2086	0.5464
14.000	1.4710	0.09668	1.00000	-0.1190	0.2032	0.5496
14.500	1.4575	0.10709	1.00000	-0.1210	0.1971	0.5520
15.000	1.4681	0.11439	1.00000	-0.1226	0.1950	0.5546

XFOIL Version 5.0

Calculated polar for: V360NR1

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 1.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
2.000	0.3200	0.02884	1.00000	-0.0310	0.0100	0.1000
2.500	0.3737	0.02928	1.00000	-0.0345	0.0100	0.1000
3.000	0.4264	0.03012	1.00000	-0.0380	0.0100	0.1000
3.500	0.4785	0.03135	1.00000	-0.0420	0.0100	0.1000
4.000	0.5280	0.03309	1.00000	-0.0460	0.0100	0.1000
4.500	0.5754	0.03533	1.00000	-0.0502	0.0100	0.1000
5.000	0.6206	0.03811	1.00000	-0.0546	0.0100	0.1000
5.500	0.6630	0.04147	1.00000	-0.0592	0.0100	0.1000
6.000	0.7023	0.04539	1.00000	-0.0638	0.0100	0.1000
6.000	0.7023	0.04539	1.00000	-0.0638	0.0100	0.1000
6.500	0.7394	0.04982	1.00000	-0.0684	0.0100	0.1000
7.000	0.7729	0.05480	1.00000	-0.0729	0.0100	0.1000
7.500	0.8027	0.06029	1.00000	-0.0772	0.0100	0.1000
8.000	0.8294	0.06622	1.00000	-0.0812	0.0100	0.1000
8.000	0.8294	0.06622	1.00000	-0.0812	0.0100	0.1000
8.500	0.8525	0.07259	1.00000	-0.0849	0.0100	0.1000
9.000	0.8721	0.07939	1.00000	-0.0884	0.0100	0.1000
9.500	0.8893	0.08662	1.00000	-0.0918	0.0100	0.1000
10.000	0.9047	0.09428	1.00000	-0.0952	0.0100	0.1000
10.500	0.9184	0.10234	1.00000	-0.0986	0.0100	0.1000
11.000	0.9305	0.11071	1.00000	-0.1020	0.0100	0.1000
11.500	0.9423	0.11927	1.00000	-0.1053	0.0100	0.1000
12.000	0.9523	0.12840	1.00000	-0.1087	0.0100	0.1000
12.500	0.9610	0.13803	1.00000	-0.1123	0.0100	0.1000
13.000	0.9703	0.14761	1.00000	-0.1158	0.0100	0.1000
13.500	0.9779	0.15781	1.00000	-0.1193	0.0100	0.1000

XFOIL Version 5.0

Calculated polar for: V360NR1

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 2.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.50120	0.01957	2.00000	-0.03740	0.43120	0.32810
-5.50000	-0.43820	0.01806	2.00000	-0.04240	0.41640	0.33720
-5.00000	-0.36730	0.01688	2.00000	-0.04810	0.41100	0.34110
-4.50000	-0.28970	0.01600	2.00000	-0.05440	0.40090	0.34960
-4.00000	-0.20930	0.01528	2.00000	-0.06080	0.38630	0.35920
-3.50000	-0.12430	0.01456	2.00000	-0.06820	0.37960	0.36610
-3.00000	-0.04400	0.01426	2.00000	-0.07360	0.36920	0.36890
-2.50000	0.04310	0.01378	2.00000	-0.08080	0.35650	0.37930
-2.00000	0.12880	0.01341	2.00000	-0.08740	0.35180	0.38900
-1.50000	0.21180	0.01322	2.00000	-0.09320	0.34360	0.39360
-1.00000	0.29380	0.01307	2.00000	-0.09860	0.32950	0.39830
-0.50000	0.37740	0.01291	2.00000	-0.10440	0.32500	0.40820
0.000	0.4584	0.01285	2.00000	-0.1095	0.3195	0.4156
0.500	0.5372	0.01285	2.00000	-0.1141	0.3109	0.4208
1.000	0.6111	0.01296	2.00000	-0.1173	0.3011	0.4233
1.500	0.6875	0.01302	2.00000	-0.1212	0.2976	0.4319
2.000	0.7604	0.01319	2.00000	-0.1243	0.2926	0.4398
2.500	0.8318	0.01339	2.00000	-0.1270	0.2864	0.4448
3.000	0.9014	0.01362	2.00000	-0.1293	0.2786	0.4487
3.500	0.9688	0.01393	2.00000	-0.1312	0.2718	0.4507
4.000	1.0391	0.01414	2.00000	-0.1337	0.2692	0.4583
4.500	1.1063	0.01446	2.00000	-0.1356	0.2657	0.4654
5.000	1.1712	0.01483	2.00000	-0.1370	0.2609	0.4713
5.500	1.2344	0.01523	2.00000	-0.1381	0.2548	0.4746
6.000	1.2948	0.01570	2.00000	-0.1386	0.2483	0.4769
6.500	1.3527	0.01623	2.00000	-0.1388	0.2445	0.4787
7.000	1.4113	0.01674	2.00000	-0.1392	0.2425	0.4854
7.500	1.4625	0.01745	2.00000	-0.1383	0.2396	0.4931
8.000	1.4852	0.01876	2.00000	-0.1327	0.2363	0.4974
8.500	1.5184	0.02020	2.00000	-0.1295	0.2317	0.5008
9.000	1.5484	0.02199	2.00000	-0.1265	0.2266	0.5033
9.500	1.5701	0.02449	2.00000	-0.1232	0.2214	0.5053
10.000	1.5930	0.02723	2.00000	-0.1208	0.2195	0.5071
10.500	1.6180	0.03022	2.00000	-0.1194	0.2176	0.5131
11.000	1.6351	0.03421	2.00000	-0.1181	0.2153	0.5190
11.500	1.6475	0.03903	2.00000	-0.1173	0.2122	0.5237
12.000	1.6602	0.04414	2.00000	-0.1169	0.2079	0.5273
12.500	1.6706	0.04975	2.00000	-0.1169	0.2028	0.5301
13.000	1.6731	0.05638	2.00000	-0.1170	0.1974	0.5323
13.500	1.6793	0.06287	2.00000	-0.1174	0.1957	0.5344
14.000	1.6850	0.06963	2.00000	-0.1180	0.1937	0.5357
14.500	1.6849	0.07743	2.00000	-0.1191	0.1911	0.5408
15.000	1.6839	0.08533	2.00000	-0.1201	0.1874	0.5463
15.500	1.6795	0.09397	2.00000	-0.1215	0.1824	0.5509
16.000	1.6672	0.10414	2.00000	-0.1232	0.1767	0.5544

XFOIL Version 5.0

Calculated polar for: V360NR1

1 Reynolds number fixed

xtrf = 1.000 (suction) 1.000 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-4.000	-0.1836	0.01394	3.00000	-0.0684	0.3693	0.3466
-3.500	-0.0994	0.01348	3.00000	-0.0749	0.3608	0.3502
-3.000	-0.0131	0.01306	3.00000	-0.0818	0.3524	0.3600
-2.500	0.0739	0.01268	3.00000	-0.0886	0.3412	0.3680
-2.000	0.1585	0.01244	3.00000	-0.0948	0.3343	0.3715
-1.500	0.2430	0.01222	3.00000	-0.1007	0.3233	0.3828
-1.000	0.3265	0.01206	3.00000	-0.1064	0.3177	0.3886
-0.500	0.4080	0.01198	3.00000	-0.1116	0.3115	0.3930
0.000	0.4860	0.01197	3.00000	-0.1158	0.2998	0.4022
0.500	0.5624	0.01201	3.00000	-0.1196	0.2963	0.4091
1.000	0.6361	0.01214	3.00000	-0.1228	0.2911	0.4123
1.500	0.7102	0.01223	3.00000	-0.1261	0.2832	0.4202
2.000	0.7831	0.01236	3.00000	-0.1290	0.2773	0.4281
2.500	0.8540	0.01255	3.00000	-0.1316	0.2734	0.4318
3.000	0.9242	0.01276	3.00000	-0.1340	0.2680	0.4360
3.500	0.9943	0.01295	3.00000	-0.1363	0.2611	0.4451
4.000	1.0628	0.01321	3.00000	-0.1384	0.2560	0.4498
4.000	1.0628	0.01321	3.00000	-0.1384	0.2560	0.4498
4.500	1.1299	0.01350	3.00000	-0.1401	0.2533	0.4528
5.000	1.1958	0.01384	3.00000	-0.1416	0.2491	0.4563
5.500	1.2617	0.01416	3.00000	-0.1431	0.2433	0.4645
6.000	1.3260	0.01451	3.00000	-0.1443	0.2373	0.4692
6.000	1.3260	0.01451	3.00000	-0.1443	0.2373	0.4692
6.500	1.3878	0.01495	3.00000	-0.1451	0.2350	0.4724
7.000	1.4469	0.01544	3.00000	-0.1454	0.2323	0.4744
7.500	1.5040	0.01600	3.00000	-0.1454	0.2284	0.4806
8.000	1.5586	0.01660	3.00000	-0.1449	0.2233	0.4863
8.000	1.5586	0.01660	3.00000	-0.1449	0.2233	0.4863
8.500	1.5932	0.01752	3.00000	-0.1410	0.2184	0.4902
9.000	1.6243	0.01884	3.00000	-0.1370	0.2163	0.4934
9.500	1.6551	0.02038	3.00000	-0.1335	0.2147	0.4952
10.000	1.6777	0.02261	3.00000	-0.1298	0.2115	0.4990
10.500	1.6994	0.02525	3.00000	-0.1269	0.2078	0.5045
11.000	1.7195	0.02835	3.00000	-0.1245	0.2034	0.5090
11.500	1.7365	0.03208	3.00000	-0.1227	0.1988	0.5127
12.000	1.7509	0.03647	3.00000	-0.1216	0.1970	0.5149
12.500	1.7643	0.04133	3.00000	-0.1209	0.1954	0.5165
13.000	1.7733	0.04694	3.00000	-0.1207	0.1929	0.5211
13.500	1.7745	0.05376	3.00000	-0.1208	0.1881	0.5261
14.000	1.7829	0.05997	3.00000	-0.1211	0.1834	0.5310
14.500	1.7823	0.06754	3.00000	-0.1218	0.1790	0.5336
15.000	1.7868	0.07456	3.00000	-0.1226	0.1778	0.5358
15.500	1.7842	0.08264	3.00000	-0.1235	0.1758	0.5373
16.000	1.7772	0.09162	3.00000	-0.1248	0.1727	0.5413

XFOIL Version 5.0

Calculated polar for: FFA-W3-360

1 Reynolds number fixed

xtrf = 0.010 (suction) 0.100 (pressure)
Mach = 0.150 Re = 3.000 e 6 Ncrit = 9.000

alpha	CL	CD	Re(CL)	CM	S xtr	P xtr
-6.00000	-0.19160	0.04295	3.00000	-0.04630	0.01000	0.10000
-5.50000	-0.15910	0.04078	3.00000	-0.04660	0.01000	0.10000
-5.00000	-0.13090	0.03824	3.00000	-0.04680	0.01000	0.10000
-4.50000	-0.09790	0.03645	3.00000	-0.04670	0.01000	0.10000
-4.00000	-0.06710	0.03464	3.00000	-0.04620	0.01000	0.10000
-3.50000	-0.04300	0.03240	3.00000	-0.04540	0.01000	0.10000
-3.00000	-0.01660	0.03079	3.00000	-0.04390	0.01000	0.10000
-2.50000	-0.01800	0.02895	3.00000	-0.03860	0.01000	0.10000
-2.00000	0.01330	0.02743	3.00000	-0.03830	0.01000	0.10000
-1.50000	0.05720	0.02632	3.00000	-0.03950	0.01000	0.10000
-1.00000	0.10700	0.02502	3.00000	-0.04210	0.01000	0.10000
-0.50000	0.16500	0.02426	3.00000	-0.04520	0.01000	0.10000
0.000	0.2277	0.02365	3.00000	-0.0489	0.0100	0.1000
0.500	0.2952	0.02303	3.00000	-0.0534	0.0100	0.1000
1.000	0.3643	0.02265	3.00000	-0.0579	0.0100	0.1000
1.500	0.4334	0.02247	3.00000	-0.0621	0.0100	0.1000
2.000	0.5027	0.02237	3.00000	-0.0663	0.0100	0.1000
2.500	0.5712	0.02234	3.00000	-0.0704	0.0100	0.1000
3.000	0.6056	0.02265	3.00000	-0.0685	0.0100	0.1000
3.500	0.6562	0.02328	3.00000	-0.0698	0.0100	0.1000
4.000	0.7134	0.02420	3.00000	-0.0725	0.0100	0.1000
4.500	0.7631	0.02552	3.00000	-0.0744	0.0100	0.1000
5.000	0.8086	0.02727	3.00000	-0.0764	0.0100	0.1000
5.500	0.8501	0.02954	3.00000	-0.0783	0.0100	0.1000
6.000	0.8889	0.03232	3.00000	-0.0806	0.0100	0.1000
6.500	0.9215	0.03583	3.00000	-0.0827	0.0100	0.1000
7.000	0.9494	0.04007	3.00000	-0.0848	0.0100	0.1000
7.500	0.9755	0.04474	3.00000	-0.0870	0.0100	0.1000
8.000	0.9991	0.04991	3.00000	-0.0893	0.0100	0.1000
8.500	1.0190	0.05557	3.00000	-0.0916	0.0100	0.1000
9.000	1.0366	0.06172	3.00000	-0.0939	0.0100	0.1000
9.500	1.0535	0.06799	3.00000	-0.0962	0.0100	0.1000
10.000	1.0669	0.07497	3.00000	-0.0986	0.0100	0.1000
10.500	1.0766	0.08273	3.00000	-0.1012	0.0100	0.1000
11.000	1.0845	0.09070	3.00000	-0.1038	0.0100	0.1000
11.500	1.0942	0.09849	3.00000	-0.1062	0.0100	0.1000
12.000	1.0981	0.10774	3.00000	-0.1091	0.0100	0.1000
12.500	1.1068	0.11609	3.00000	-0.1118	0.0100	0.1000
13.000	1.1083	0.12628	3.00000	-0.1150	0.0100	0.1000
13.500	1.1141	0.13549	3.00000	-0.1178	0.0100	0.1000