NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL MEMORANDUM

No. 1127

RUSSIAN LAMINAR FLOW AIRFOILS

3RD PART: MEASUREMENTS ON THE PROFILE

NO. 2315 BIS WITH AVA-NOSE FLAP

By F. Riegels

TRANSLATION

"Russische Laminarprofile
3.Teil: Messungen am Profil
Nr. 2315 Bis mit AVA-Nasenklappe"
Deutsche Luftfahrtforschung, Untersuchungen und Mitteilungen Nr. 3067

Washington
September 1947

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL MEMORANDUM NO. 1127

RUSSIAN LAMINAR-FLOW AIRFOILS

3RD PART: MEASUREMENTS ON THE PROFILE NO. 2315 BIS WITH AVA NOSE FLAP*

By F. Riegels

Abstract: The tests on the Russian airfoil 2315 Bis were continued. This airfoil shows, according to Moscow tests, good laminar-flow characteristics. Several tests were prepared in the large wind tunnel at Göttingen; partial results were obtained.

Outline:

I. STATEMENT OF THE PROBLEM

The laminar effect of the airfoil, according to Moscow tests, was relatively strong; a repetition of the test at high Reynolds numbers was indicated. The theoretical investigations described in the second part of the present report had ascertained that the stability limit for small c_a values is near $Re \approx 5 \times 10^6$; therefore, a laminar effect up to 18×10^6 appears quite possible if the considerable pressure gradient in the front part of the airfoil is taken into consideration. We attempted in our tests to reach the highest possible Reynolds number in order to determine at what Reynolds number the laminar effect ceases to exist. Therefore, a Reynolds number of 35×10^6 shall be attained with a wing of a 3-meter chord. The tests are made in the large wind tunnel at Göttingen at high pressure. The corresponding dynamic pressure is very high; since for technical reasons it does not allow for measurements at higher c_a-values, parallel $c_{a_{max}}$ measurements on a wing with a chord of only 0.8 meter had to be carried out.

---

*"Russische Laminarprofile 3.Teil: Messungen am Profil Nr. 2315 Bis mit AVA-Nasenklappe." Untersuchungen und Mitteilungen Nr. 3067. Zentrale für wissenschaftliches Berichtswesen der Luftfahrtforschung des Generalluftzeugmeisters (ZWB) - Berlin-Adlershof. Jan. 17, 1944.

Meanwhile good results had been obtained at the Institute Windkänale der AVA (Aerodynamische Versuchsanstalt.): According to a suggestion of W. Krüger, a nose flap had been used which increased the maximum lift of an airfoil considerably (3). It may be expected that, particularly for these airfoils with small nose radii, this novel flap will produce considerable increases of $c_{a_{max}}$. However, the above-mentioned investigations at the Institut Windkänale represented only preliminary tests at a small Reynolds number; therefore, it was deemed expedient to provide the wing of 0.8-meter chord with this flap thus obtaining measurements at $Re \approx 2.2 \times 10^6$. The aerodynamic reason for the effectiveness of this new arrangement is probably the following: an airfoil with a nose flap adjusts itself more readily to the flow pattern at high lift; therefore, the high flow velocities at the nose which usually cause a premature separation on airfoils with pointed noses are avoided.

The following specifications apply to the results of the three-component measurements for the last part of the program, that is, the wing with a chord of 0.8 meter without and with nose flap; a report on the drag at high Reynolds numbers will be made only after testing the large wing.

II. SPECIFICATIONS OF MODEL AND TESTS

The coordinates of the Russian laminar profile 2315 Bis (formerly 4-010) are known from the first and second parts of this report. Plaster-type construction was used for the model; its chord is 0.8 meter; the span (without tips), 4 meters. The construction of the tips is based on a recent investigation by B. Regenscheit (4); in the front part of the profile they have the shape of circular arcs with the diameter = local thickness; from the location of the maximum profile thickness on, they are ellipses with the axis ratio maximum profile thickness/local profile thickness. Accordingly, the aspect ratio of the wing $\Lambda = b^2/F = 5.1$. Figures 1 and 2 show the arrangement of the flaps. The flaps were screwed on; the upper side of the flaps was shaped like the underside of the profile; it was (with respect to the nonextended condition) turned by the angle of 105°, 115°, and 125°, respectively, around the pivot indicated in figure 1. It is perhaps expedient to use in this report not these angles but rather the angles formed by the straight underside of the flap and the profile chord; this precaution will guarantee better possibilities of comparison with other tests. The corresponding angles will then be:

$$\gamma = 118^\circ, 128^\circ, 138^\circ$$

In order to include the influence of the chord of the flap and of the rounding off of the nose on the maximum lift, the flaps were shortened twice more and, in the case of the optimum angle, measured each time with sharp edges and with an added circular rod. The tests were carried out in the large wind tunnel of the AVA with the 5.4- by 4-meter nozzle operating at normal pressure and at high pressure. For the profile without flaps, three-component measurements at normal pressure and at high pressure at Reynolds numbers of 0.8×10^6 to 10×10^6 were taken; corresponding tests were carried out for the profile with flaps but without high pressure at $Re \approx 2.2 \times 10^6$; no range other than the range of the higher c_a-values was measured.

Several measurements with transition wires of 0.1-, 0.3-, and 0.5-millimeter diameter were taken in order to determine whether for small c_a-values the laminar effect would be impaired by disturbances on the pressure side; such disturbances would be certain to originate for the nonextended position of the flaps.

Wind-tunnel corrections were taken into consideration in the customary manner.

III. RESULTS

The measurements confirm the expectations with regard to the maximum lifts attainable. Figures 3 to 5[1] show the results of the three-component measurements on the profile alone. First of all the variation of $c_{a_{max}}$ with Reynolds number is presented from the measurements at high pressure (fig. 6). The measurements on profiles with flaps are represented graphically in figures 7 to 12. The maximum values of $c_{a_{max}}$ at $Re \approx 2.2 \times 10^6$ and at the optimum flap angle $\gamma = 118^\circ$ are:

(a) For the profile without flap $c_{a_{max}} = 0.68$

(b) For the profile with AVA-nose flap $c_{a_{max}} = 1.29$

(c) For the profile with split flap $c_{a_{max}} = 1.43$

(d) With split flap and AVA-nose flap

(1) $\frac{l_k}{l} = 0.1$; $\frac{\rho_k}{l} = 0.02$; $c_{a_{max}} = 2.09$

[1]Translator's note: The values identifying test points in the figures were frequently illegible in the original German copy and values shown are subject to error in deciphering.

(2) $\frac{l_k}{l} = 0.09$; $\frac{\rho_k}{l} = 0.01$; $c_{a_{max}} = 2.04$

(3) $\frac{l_k}{l} = 0.08$; nose angular; $c_{a_{max}} = 1.95$

(4) $\frac{l_k}{l} = 0.05$; $\frac{\rho_k}{l} = 0.01$; $c_{a_{max}} = 1.86$

(5) $\frac{l_k}{l} = 0.04$; nose angular; $c_{a_{max}} = 1.73$

Therewith it is indicated that the new AVA nose flap will produce considerable lift increases for profiles with small nose radii even for small flap chords. The corresponding changes of moment are small and can be tolerated. The more the flight speed approaches sound velocity the more the development will lead toward profiles with small radii of the nose; accordingly, a way is shown for obtaining practically useful maximum lifts for such profiles.

However, there is another question to be reconsidered: will the laminar effect at small c_a-values actually cease to exist because of the surface disturbance which cannot be avoided in the case of a flap at the nose? Several preliminary tests with various transition wires at 5 percent, 10 percent, and 20 percent of the chord of the profile on the pressure side were carried out in order to clarify this question. These tests showed considerable changes of c_a and c_m; there resulted also an increase in drag which is apparently caused by the forward shifting of the transition point. The frictional boundary layer, which is made turbulent at the front part, grows considerably thicker than the boundary layer which is laminar for the greatest part of the profile chord on the suction side; this behavior is equivalent to a change of the direction at which the flow leaves the airfoil and, therewith, of the circulation around the profile (and thus causes the aforementioned changes of c_a and c_m). One may also express it this way: the transition wires on the pressure side correspond to an additional camber of the mean line, transition wires on the suction side correspond to a reduction of that camber. The corresponding changes in angle at zero lift were measured; they are for 0.1-millimeter wire at 5 percent of the chord, for instance, about 0.8°. There are no longer such strong effects for this thinnest wire at further aft positions; the reason is that the boundary layer grows gradually thicker and the small disturbance caused by the wire is dampened (at any rate while the pressure is falling off and the stability limit has not yet been exceeded). Figures 13 to 18 interpret the conditions further. More tests with transition wires were agreed upon with Professor Schlichting in order to obtain as correct an idea as possible of the magnitude of the permissible disturbances of the surface and all their effects. The scheme for the tests on the Russian laminar profile provided from the first for tests on pressure distribution, loss of momentum, and point of transition on a wing of 1m chord in the wind tunnel of the Aerodynamic Institute of the TH Braunschweig; these tests are now, thanks to Professor Schlichting's cooperation, supplemented by tests with transition wires.

Considering the results obtained so far, one may say that transition wires of thicknesses of the same order of magnitude as the thickness of the local boundary layer are dangerous. The question whether the disturbances originating with the installation of the nose flap will cause the same effects as transition wires will have to be investigated further. At any rate, it seems advisable not to extend the nose flap at an angle to the profile contour since a slight disturbance will always occur in the nonextended position at the junction of flap and profile; it will be preferable to keep the disturbance for small c_a values, if possible close to the stagnation point; due to the very considerable pressure decrease a transition of the boundary layer may be avoided. These deliberations seem to advocate Professor Betz' suggestion to avoid, as far as possible, extending the nose flap at an angle, and instead to fit it into the profile curve by providing a means for retracting the flap into the profile as shown in figure 19.

Translation by Mary L. Mahler
National Advisory Committee
for Aeronautics

IV. BIBLIOGRAPHY

1. Riegels, F.: Russische Laminarprofile. UM 3040.

2. Riegels, F. and Liese, J.: Russische Laminarprofile. 2.Teil: Theoretische Ergänzungen. UM 3056.

3. Krüger, W.: Ueber eine neue Möglichkeit der Steigerung des Höchstauftriebes von Hochgeschwindigkeitsprofilen. UM 3049.

4. Regenscheit, B.: Untersuchungen über den Einfluss der Randkappenform auf die Tragflügelmessergebnisse (erscheint demnächst in den Techn. Ber.)

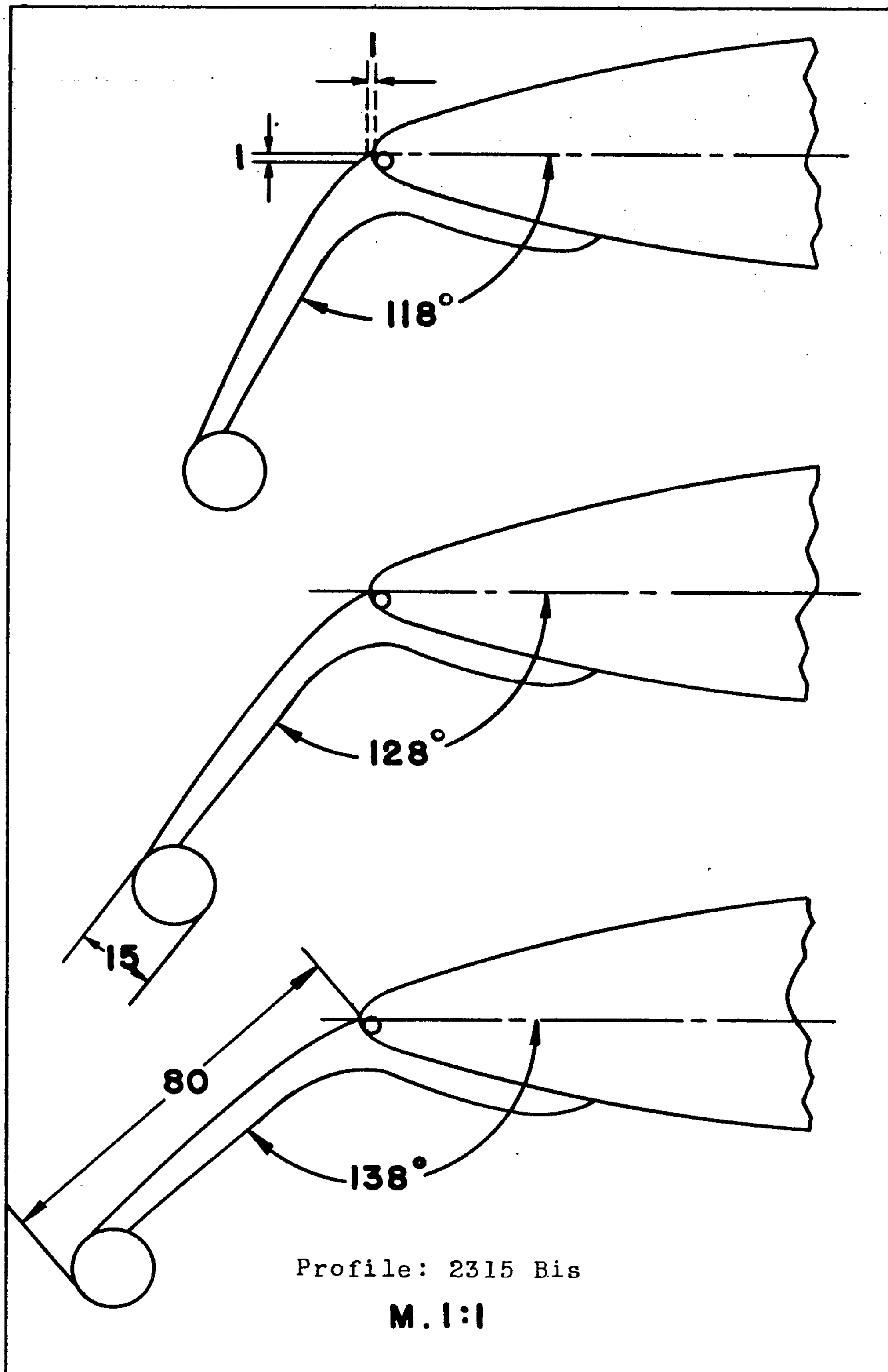

Figure 1. The angles of the nose flaps tested.

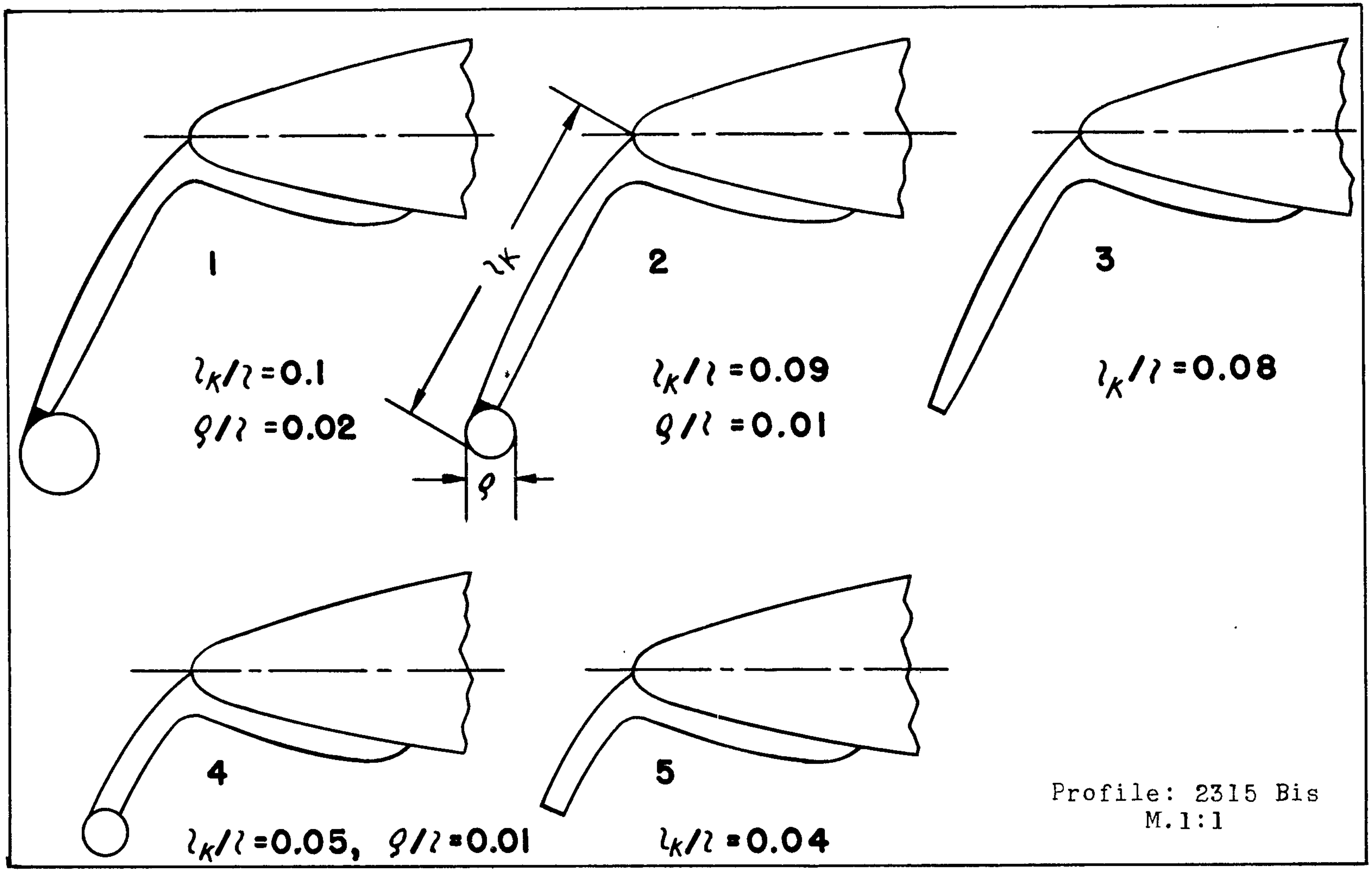

Figure 2. The chords of the nose flaps tested.

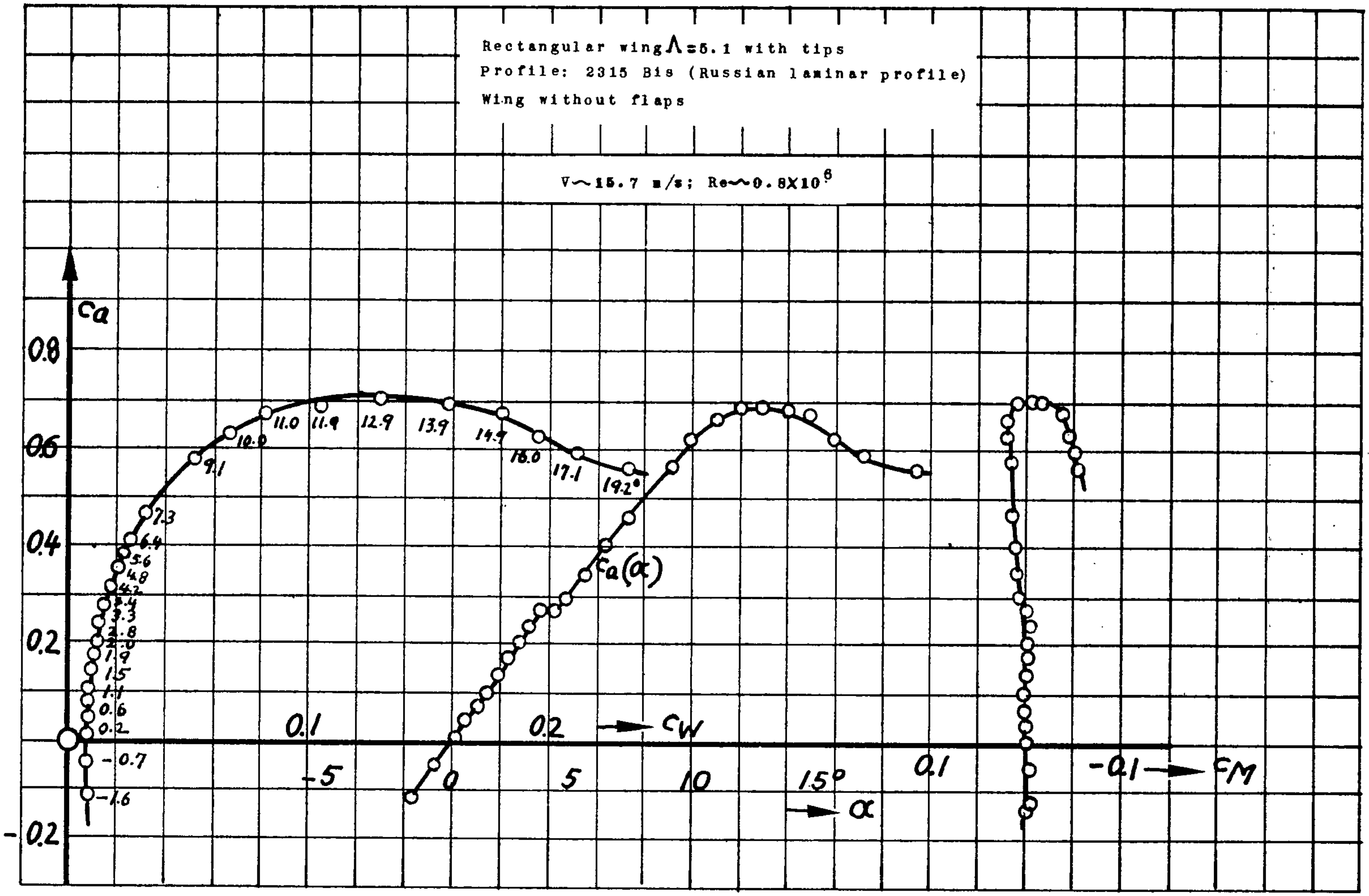

Figure 3. Wing alone. $Re = 0.8 \times 10^6$.

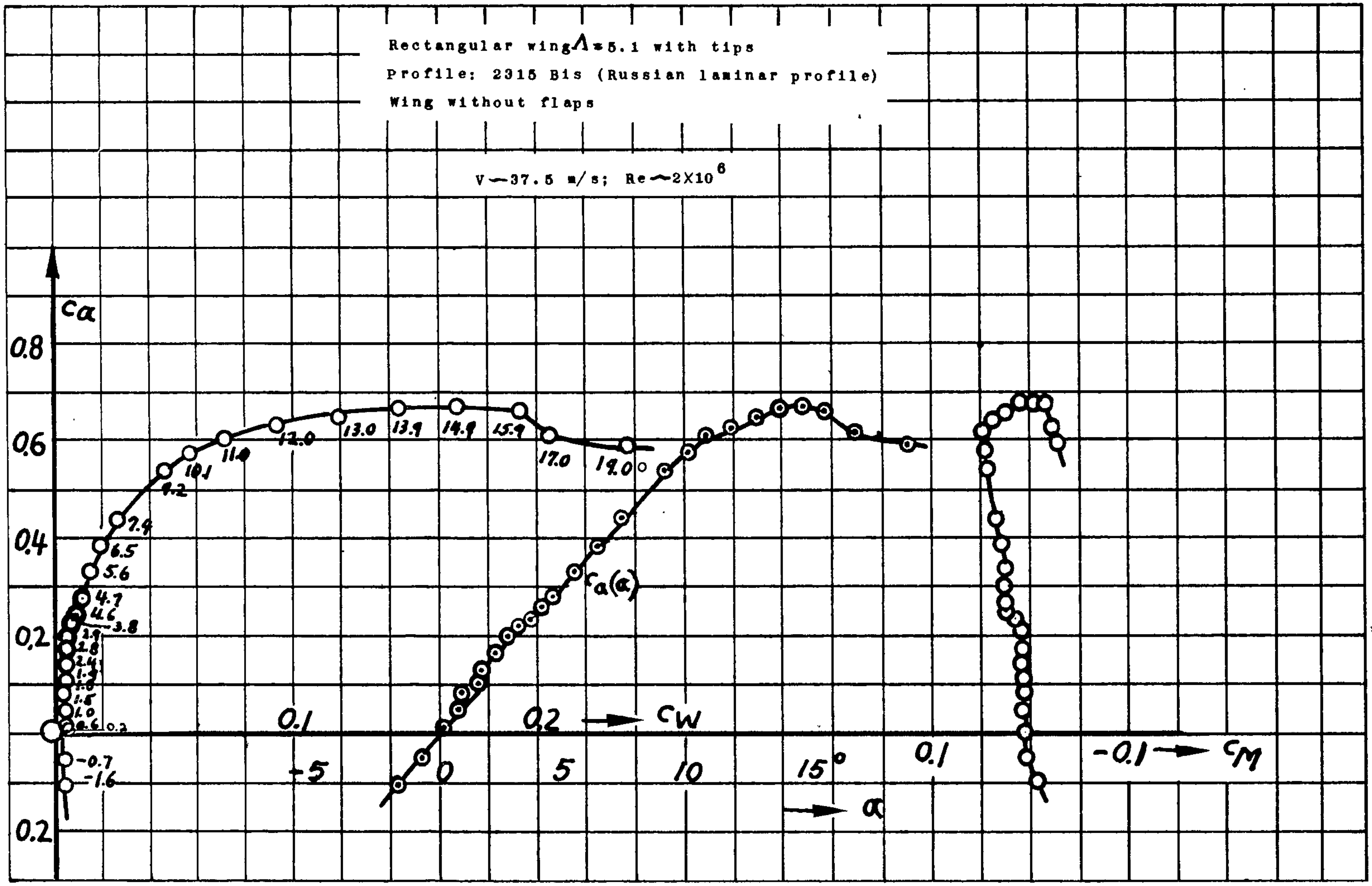

Figure 4. Wing alone. Re = 2×10^6.

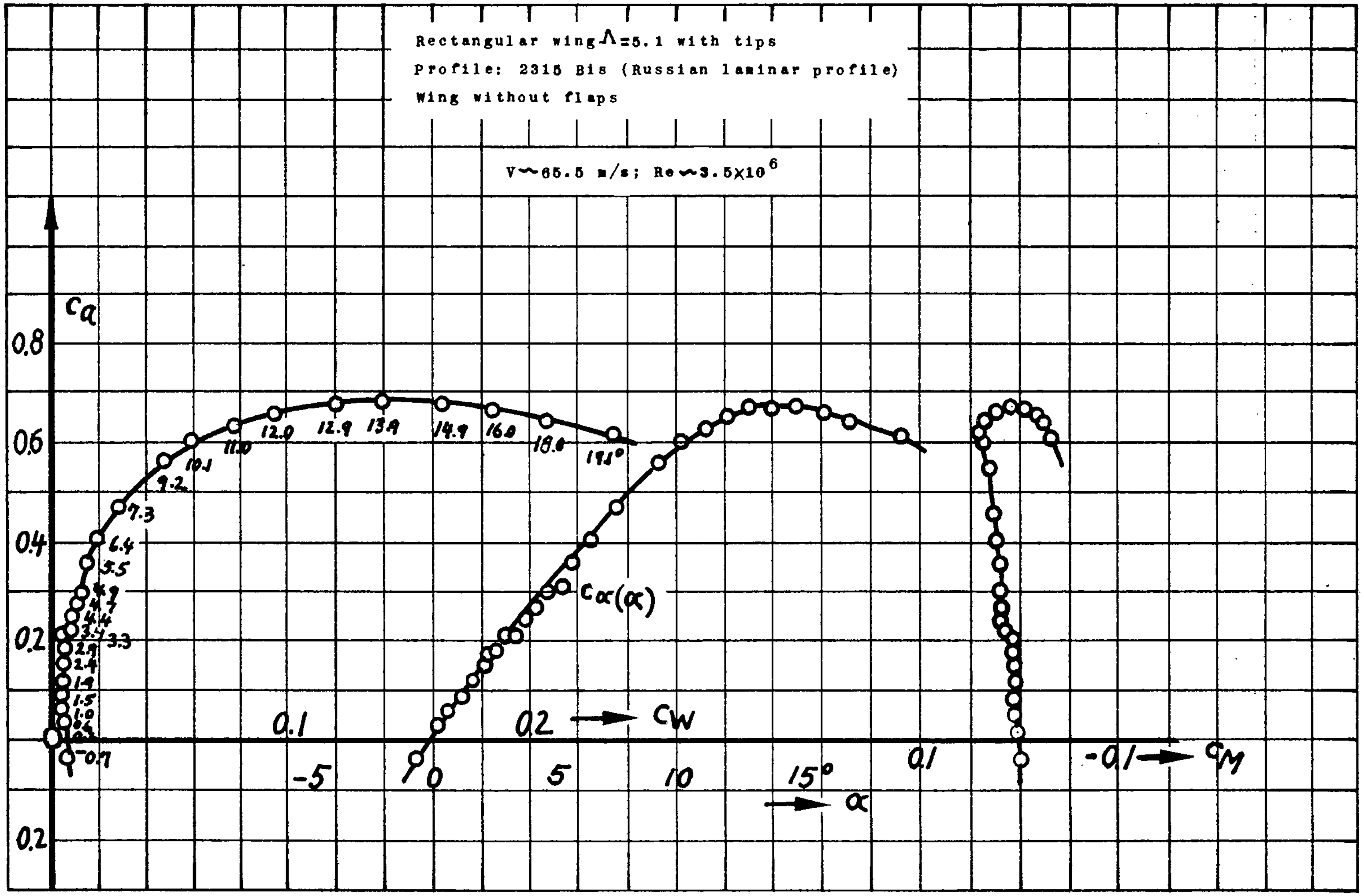

Figure 5. Wing alone. Re = 3.5X10^6.

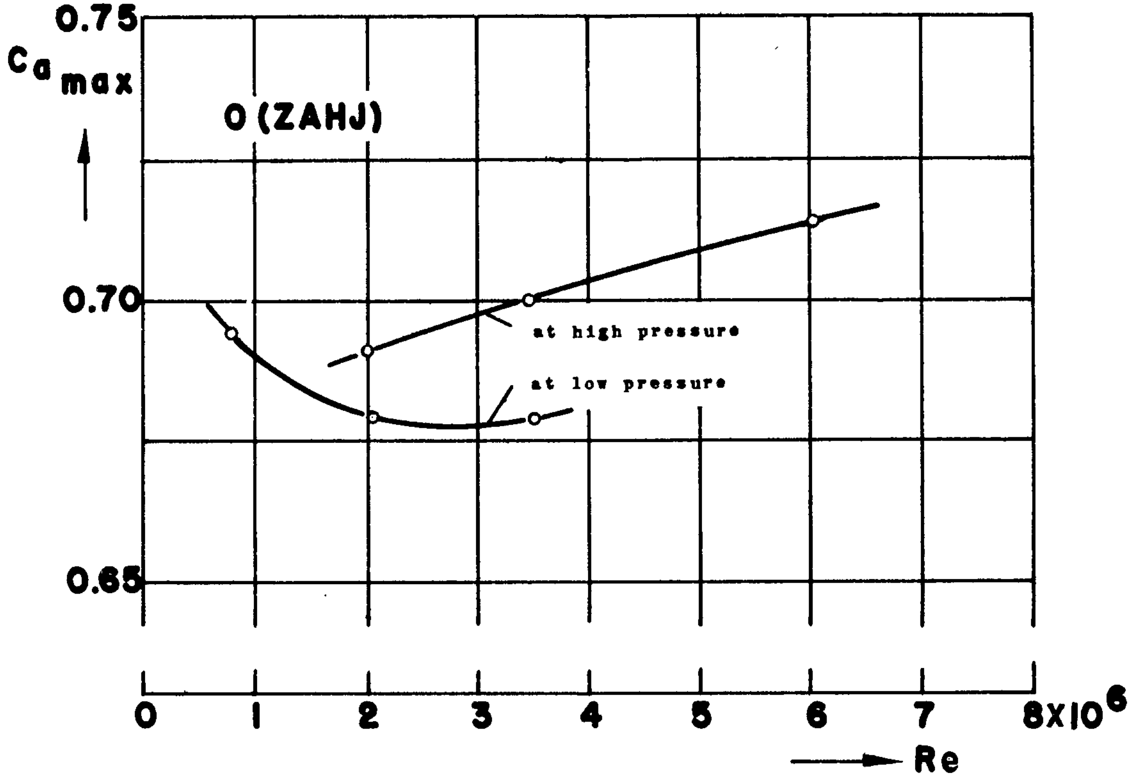

Figure 6. $c_{a\ max}$ as a function of the Re-number.

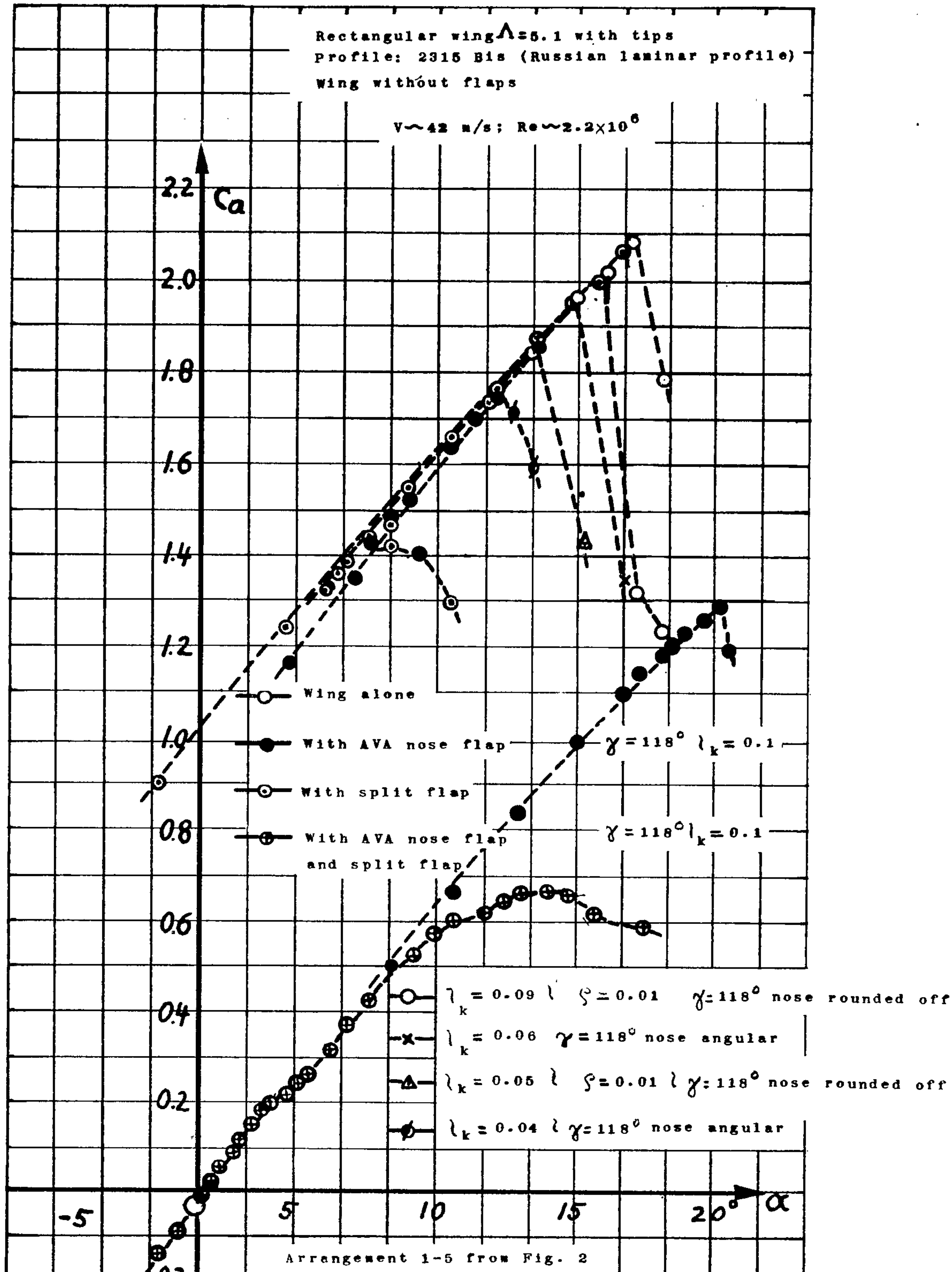

Figure 7. Wing with split flap and AVA nose flap. $c_a(\alpha)$. Survey of the optimum arrangement at various flap chords.

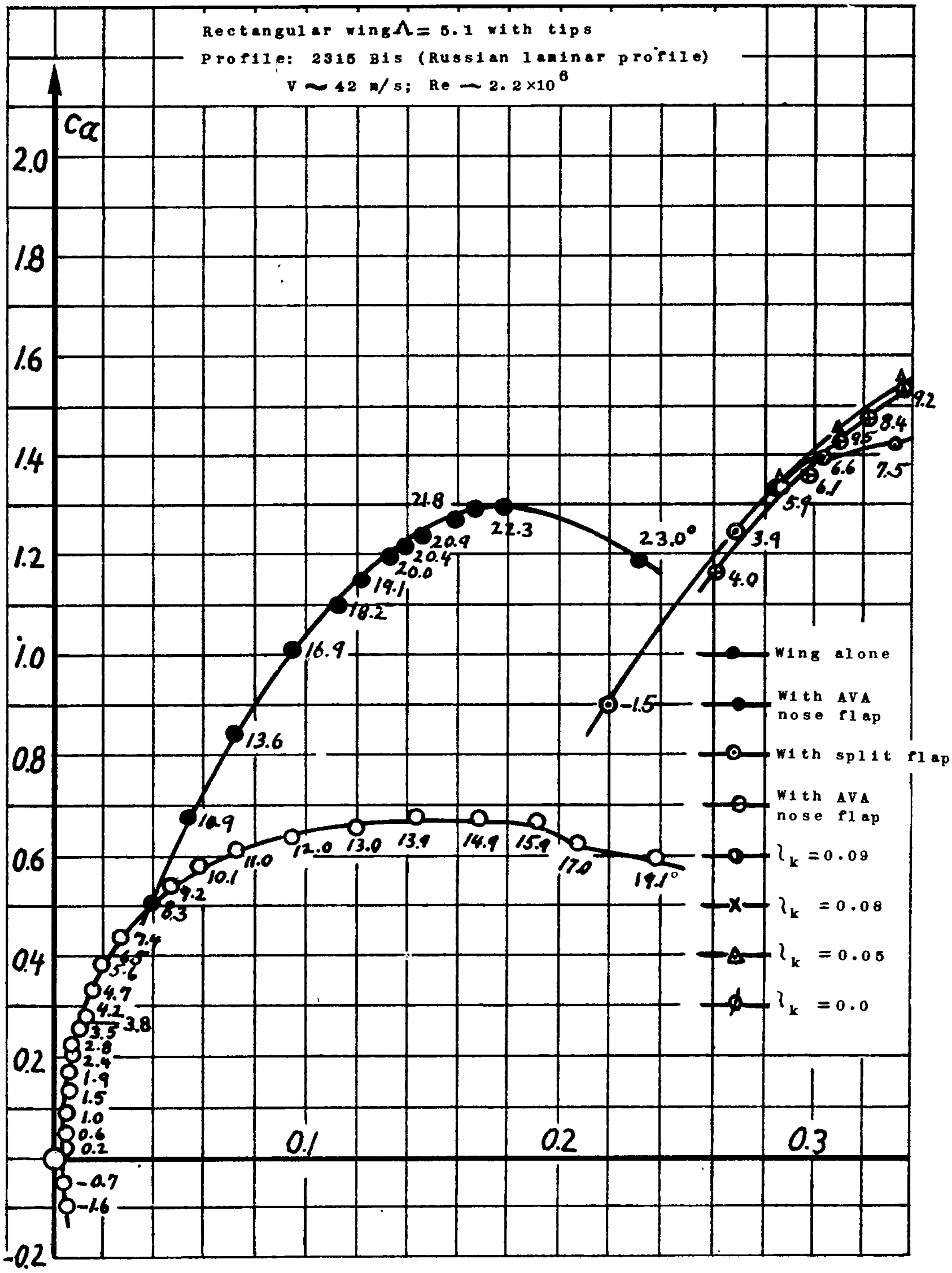

Figure 8. Wing with split flap and AVA nose flap. $c_a (c_w)$, c_a (c_m). Survey of the optimum arrangement $\gamma = 118^\circ$ at various flap chords.

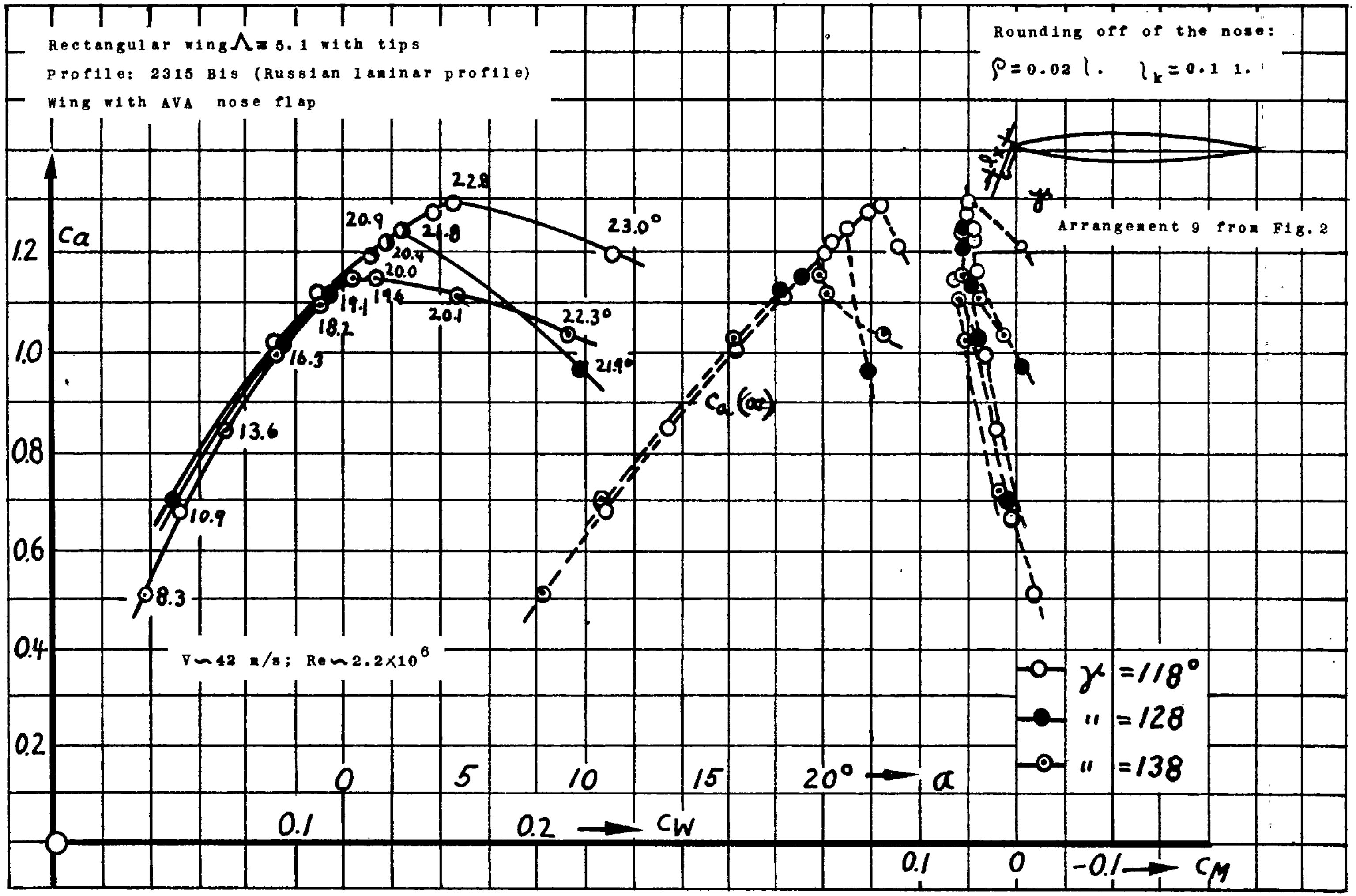

Figure 9. Wing with AVA nose flap. Influence of the angle of the flap.

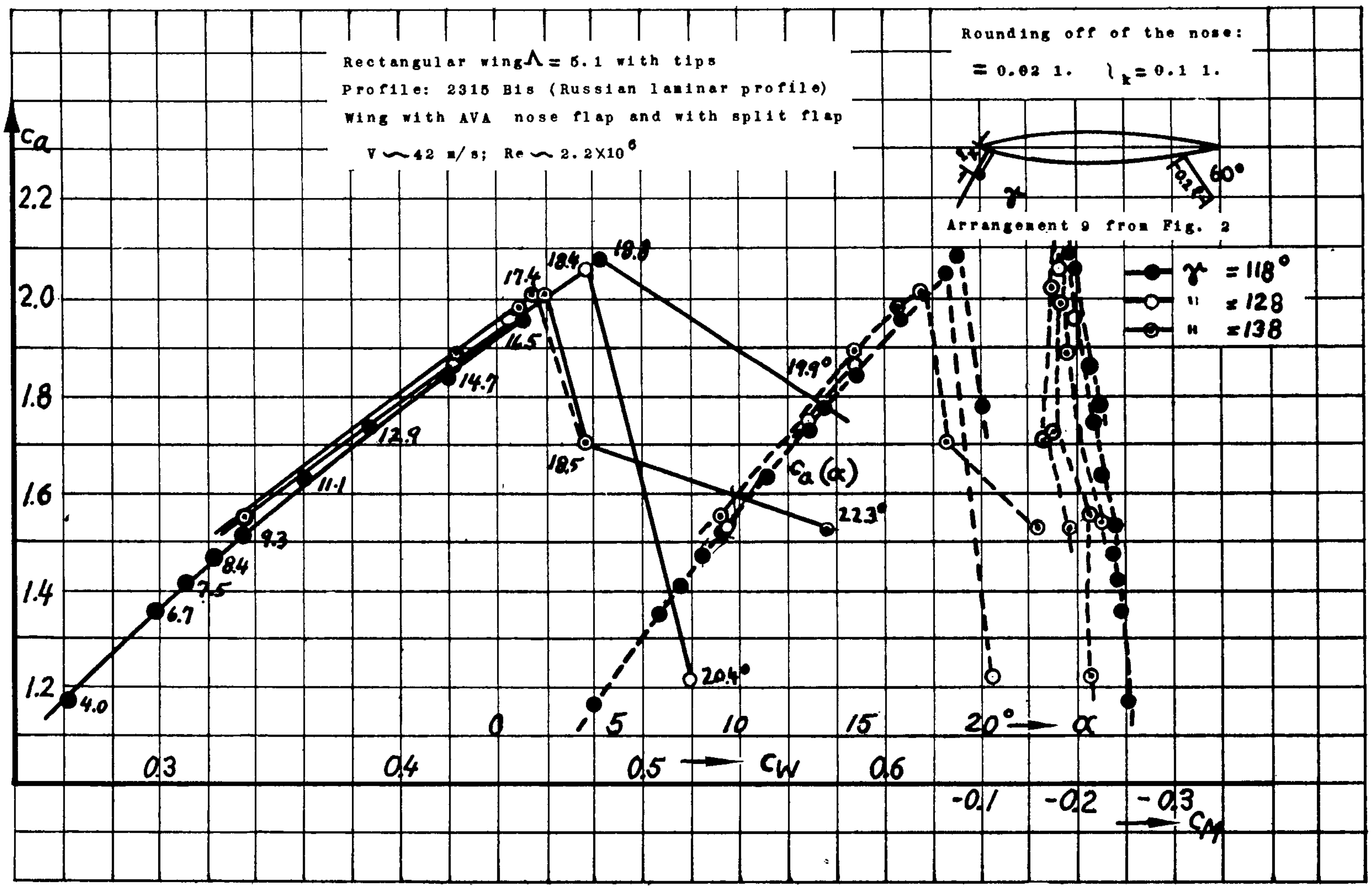

Figure 10. Wing with split flap and AVA nose flap. Influence of the angle of the flap.

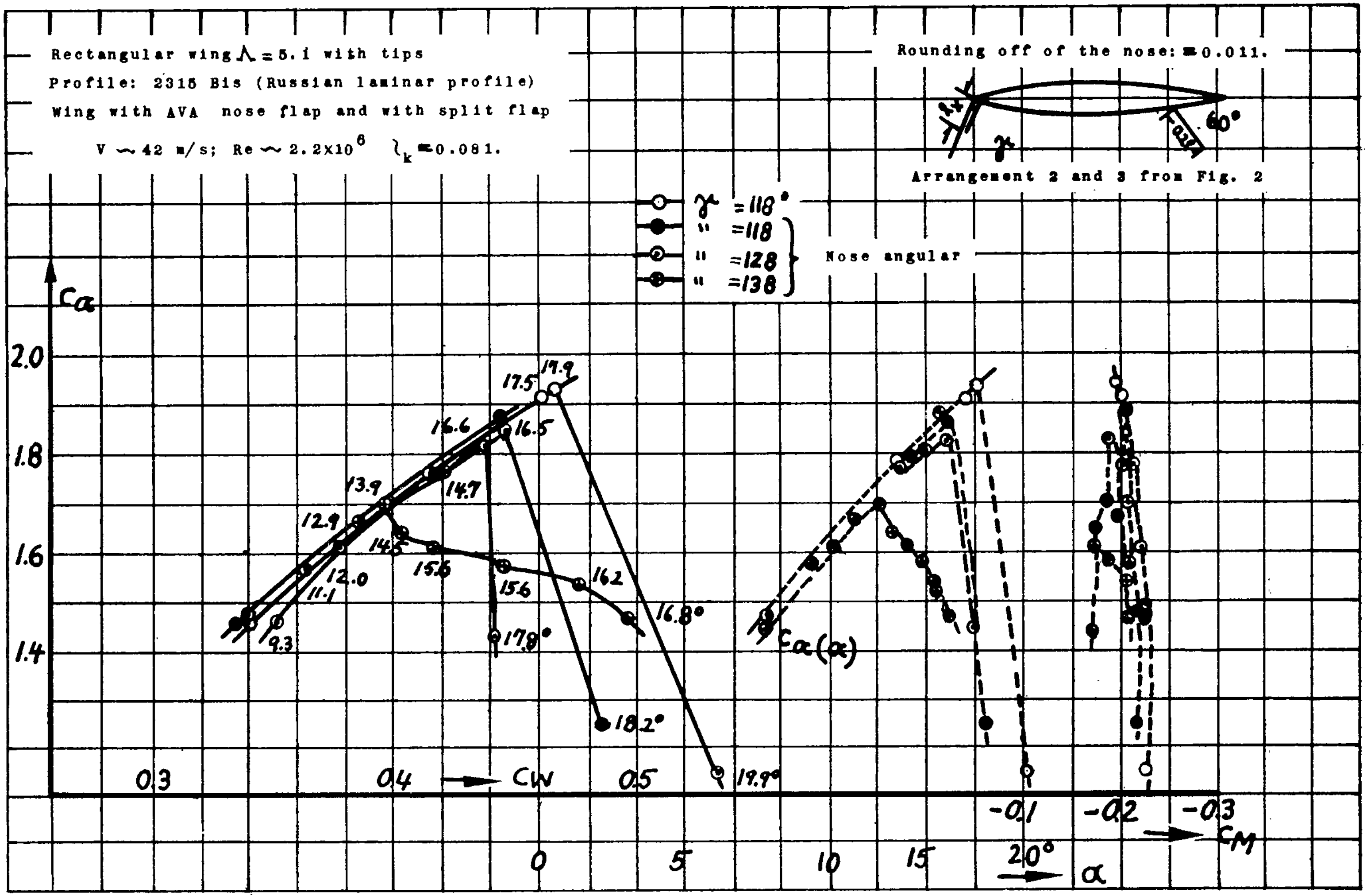

Figure 11. Wing with split flap and AVA nose flap shortened once. Influence of the angle of the flap and of the rounding off of the nose.

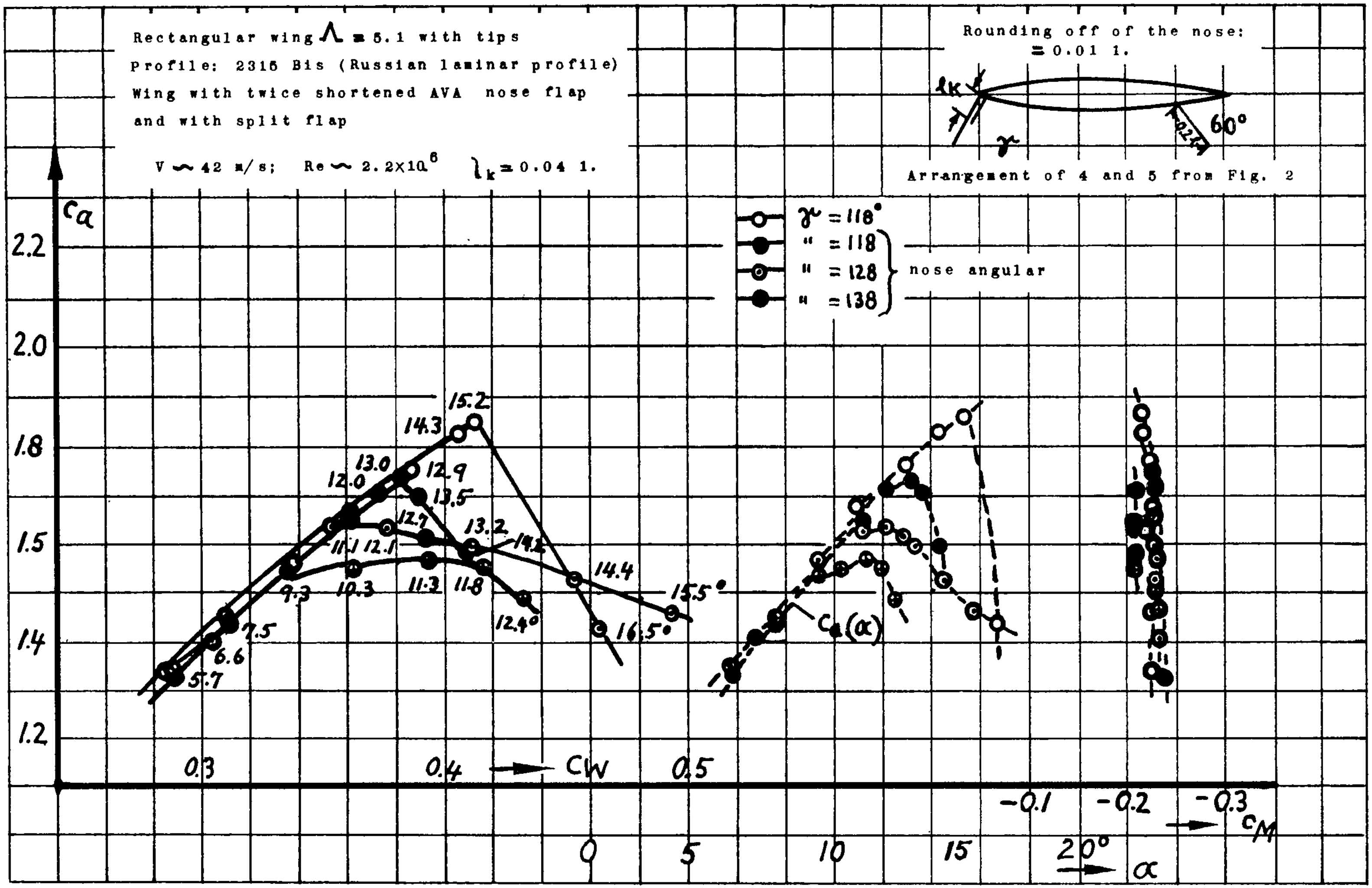

Figure 12.Wing with split flap and AVA nose flap shortened twice. Influence of the angle of the flap and of the rounding off of the nose.

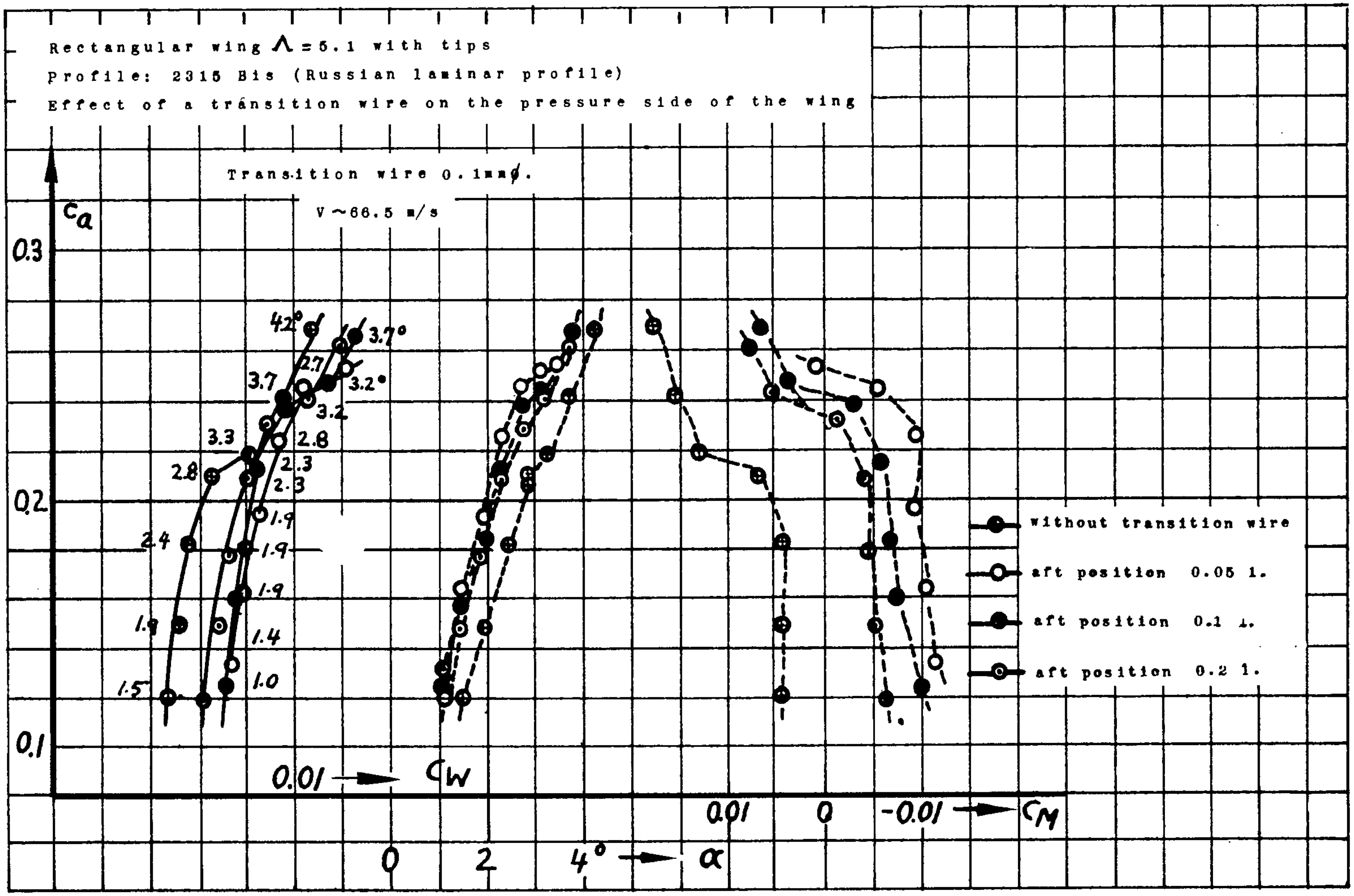

Figure 13. Results of the measurements with transition wire on the wing alone.

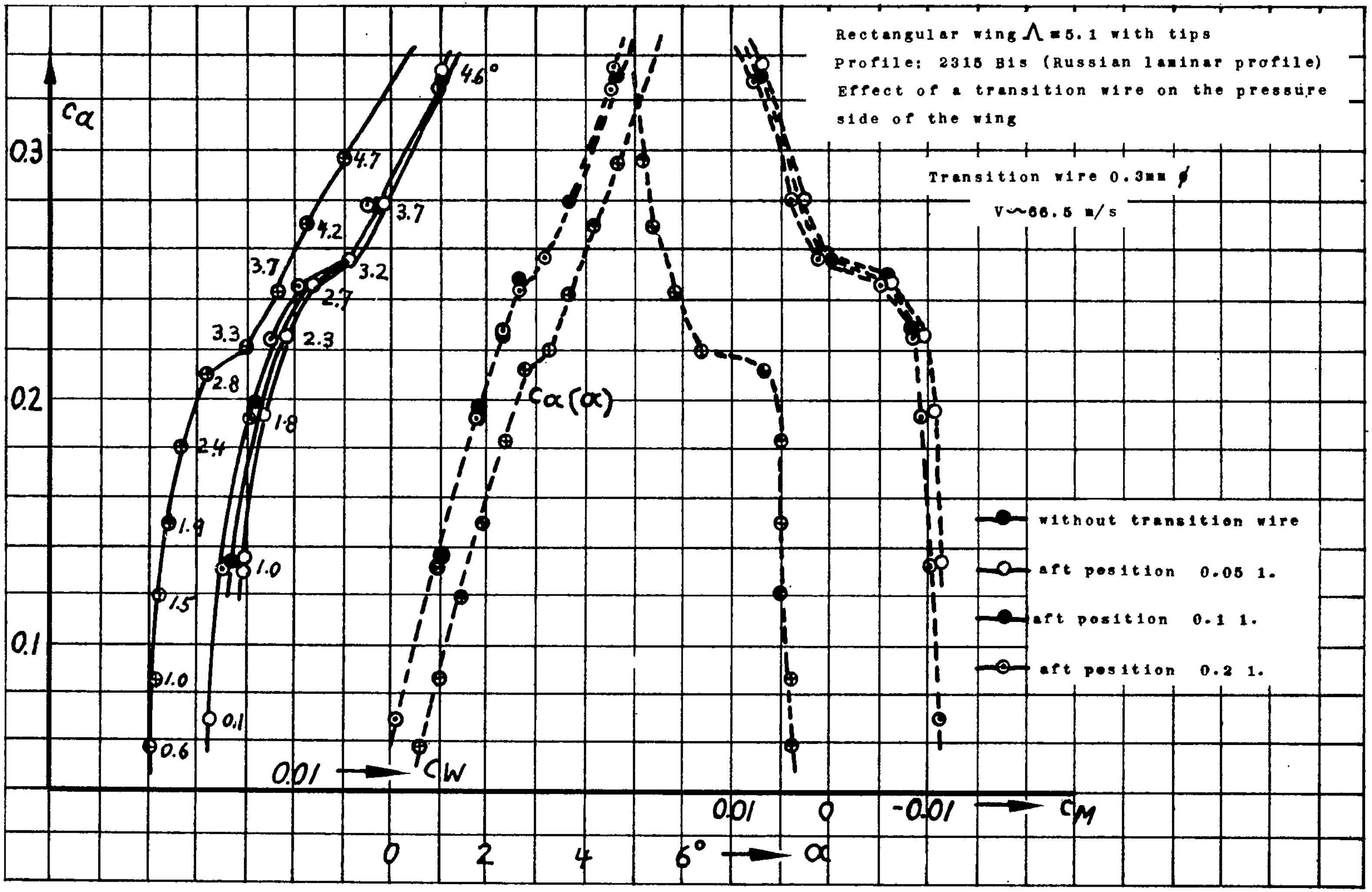

Figure 14. Results of the measurements with transition wire on the wing alone.

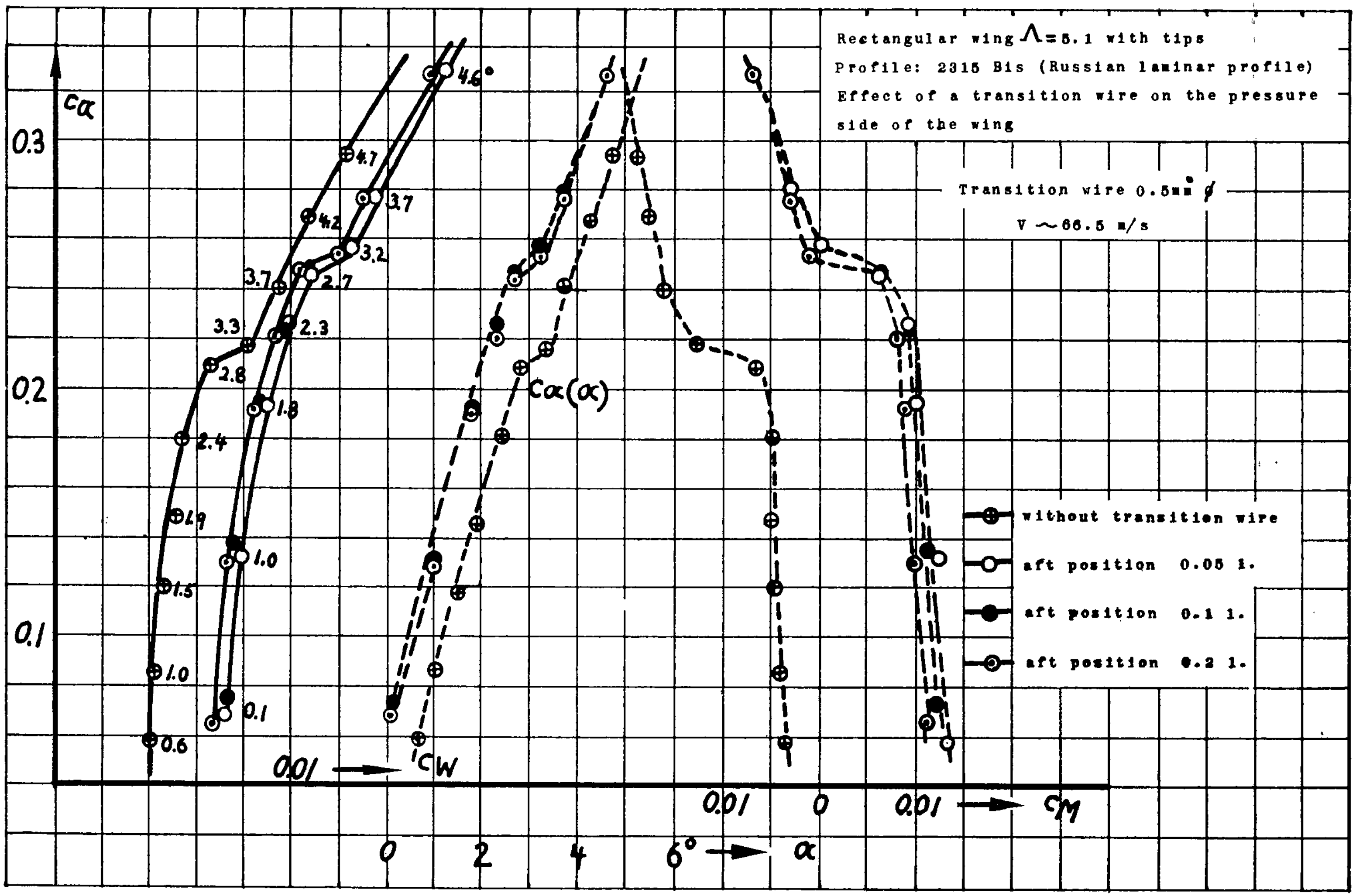

Figure 15. Results of the measurements with transition wire on the wing alone.

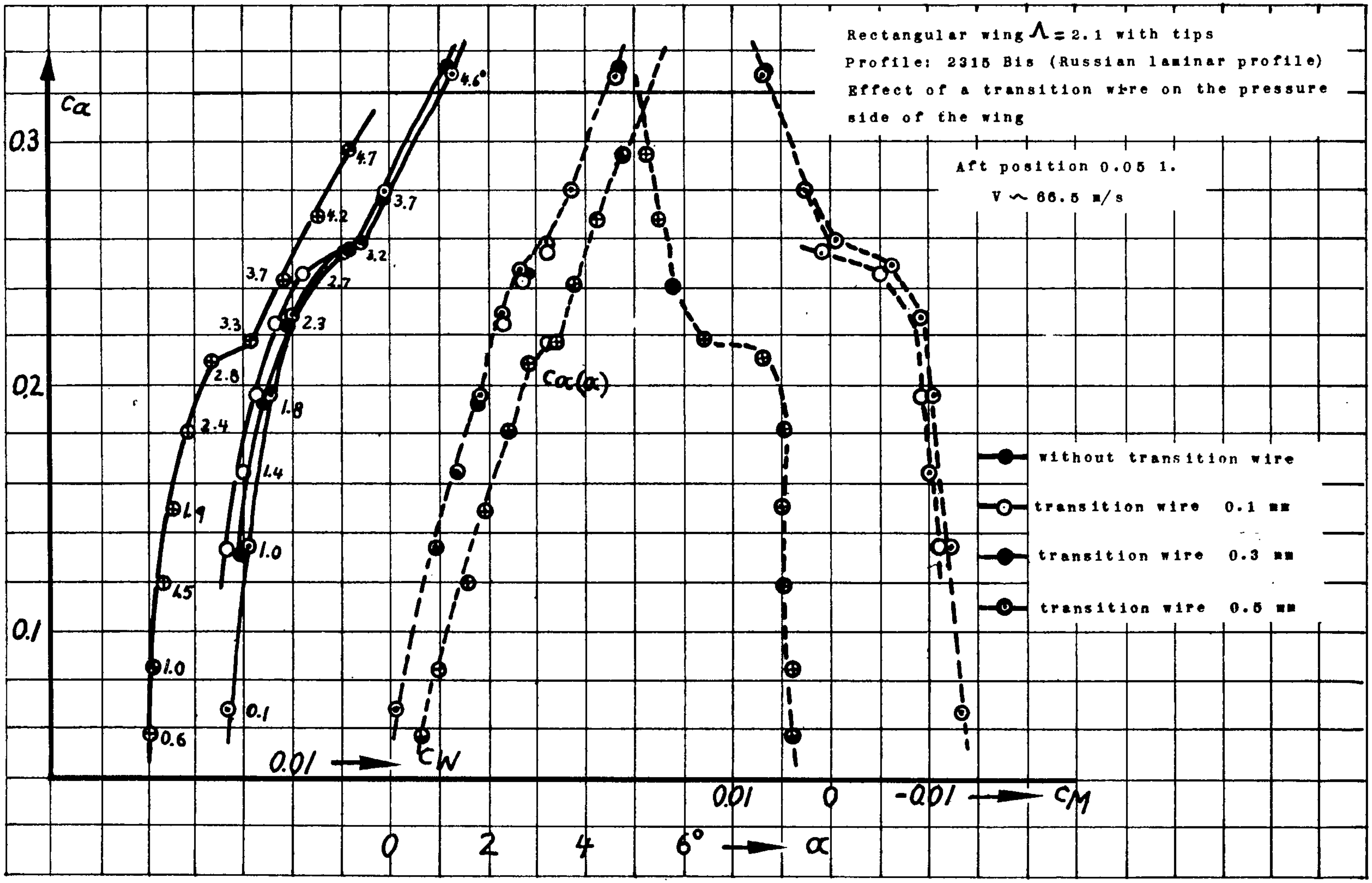

Figure 16. Results of the measurements with transition wire on the wing alone.

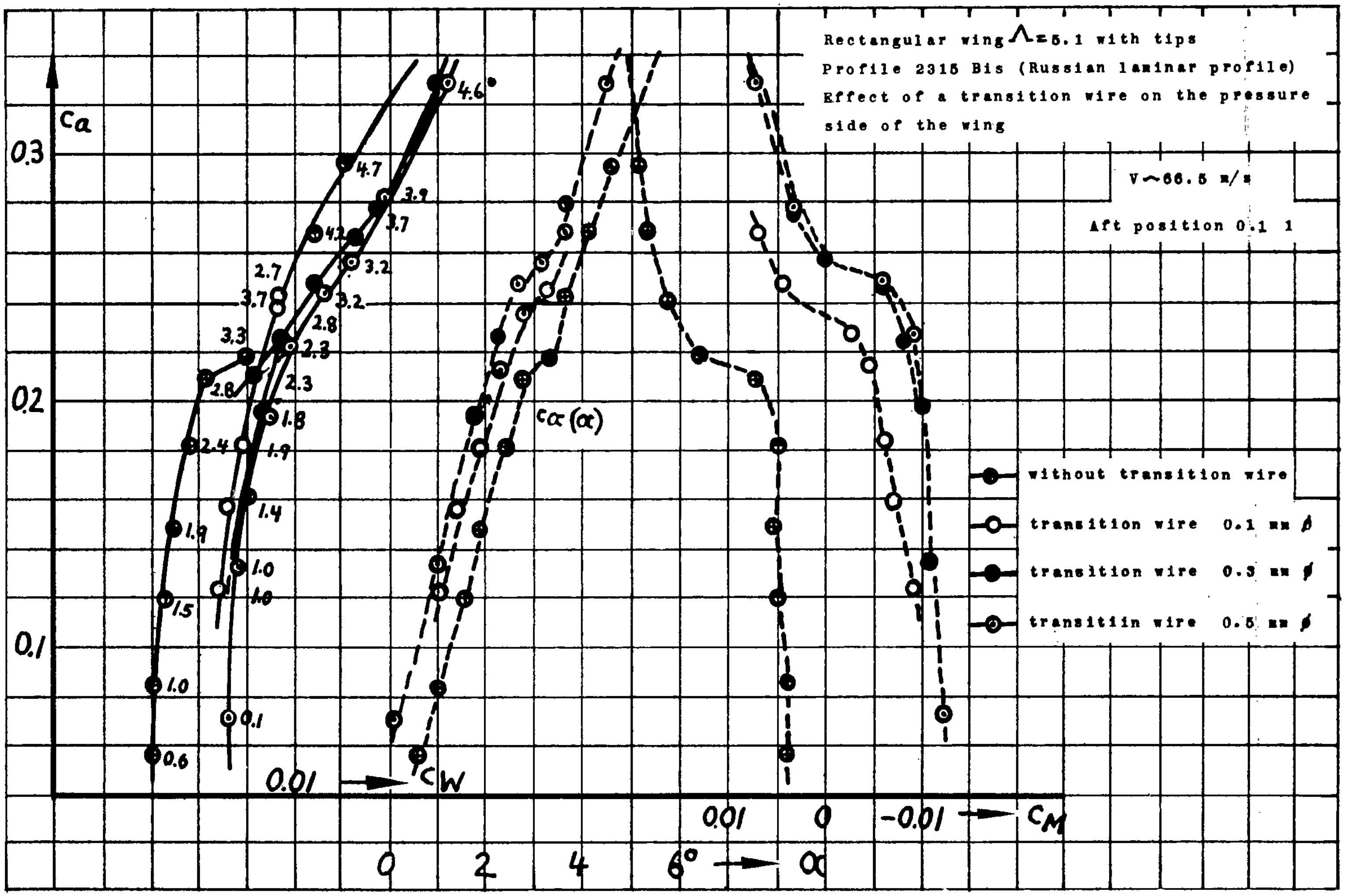

Figure 17. Results of the measurements with transition wire on the wing alone.

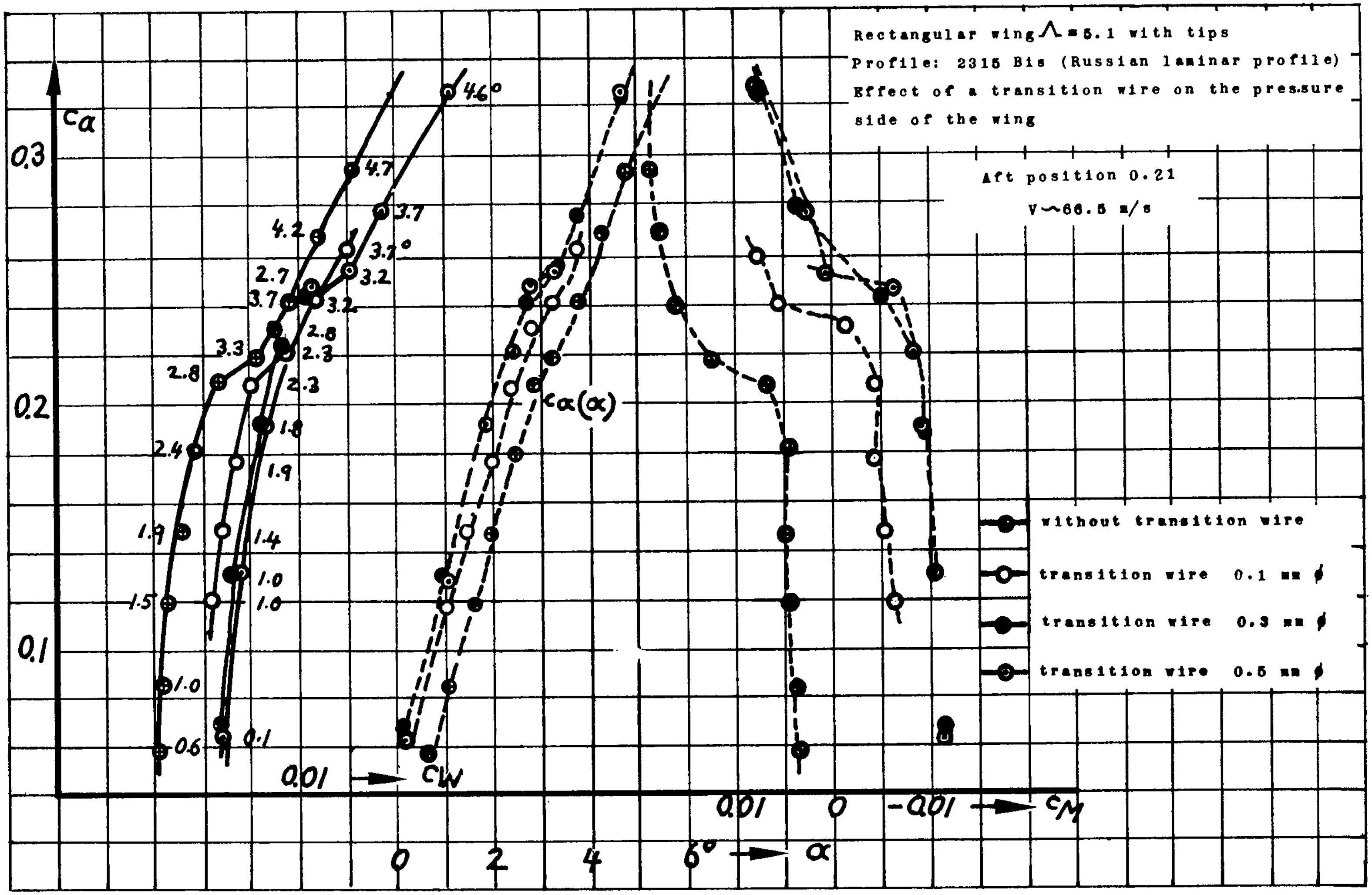

Figure 18. Results of the measurements with transition wire on the wing alone.

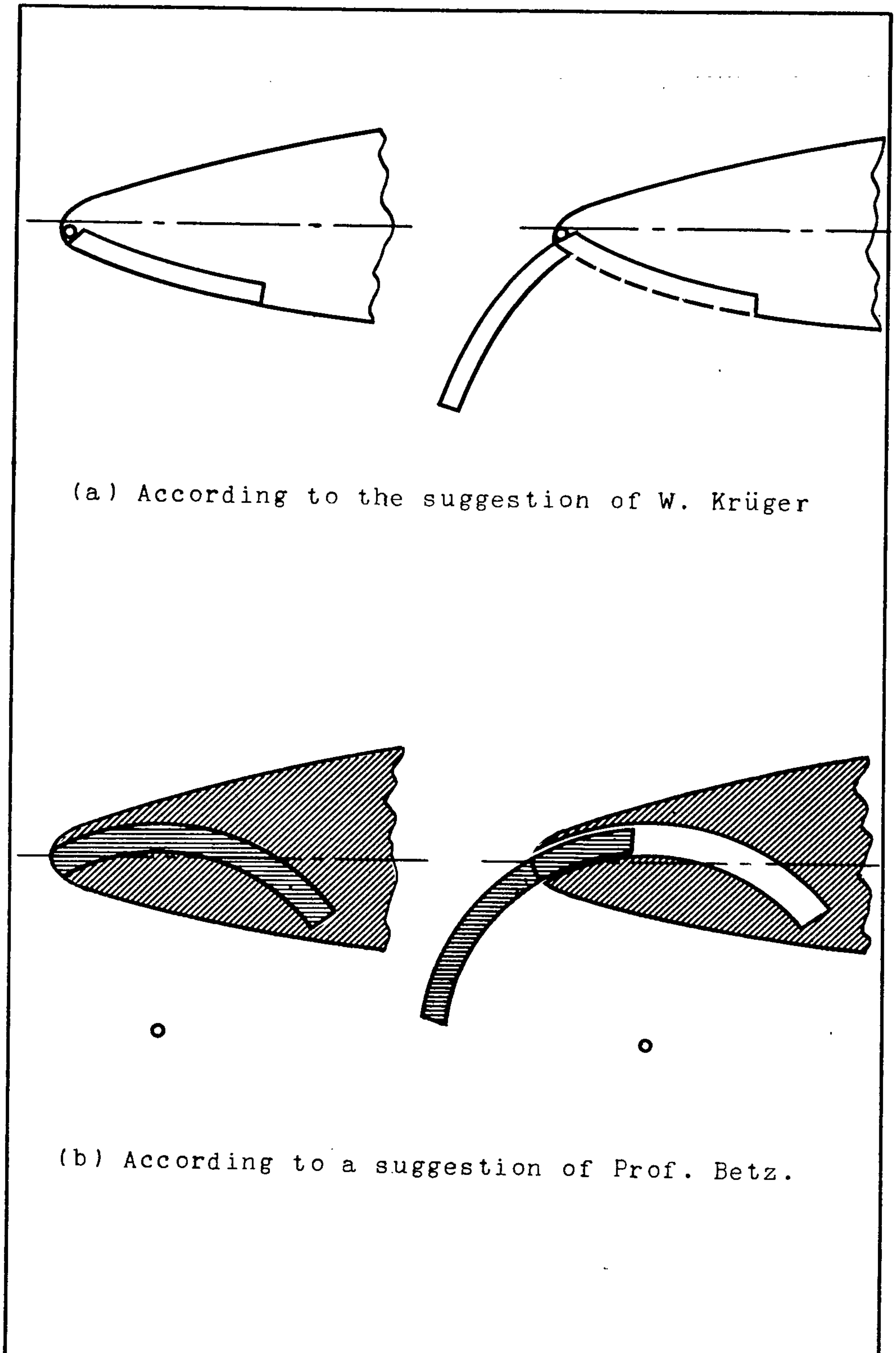

Figure 19. Two different constructions of the AVA nose flap.