Ministry of Education and Science of Ukraine
National Aviation University

AEROHYDROGASDYNAMICS AND FLIGHT DYNAMICS PART I: AEROHYDROGASDYNAMICS

SELF-STUDY METHOD GUIDE

Part I

AEROHYDROGASDYNAMICS
Plotting the Aircraft Polar
for the Students of the
Field of Study 27 “Transport”,
Specialty 272 “Aviation Transport”

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Містять декілька рекомендацій для самостійної роботи, щодо виконання розрахунково-графічної роботи з дисципліни «Аерогідрогазодинаміка та динаміка польоту», в Частині І, що стосується розділу «Аерогідрогазодинаміка» при побудові поляри повітряного судна.

Для студентів 3-го курсу галузі знань 27 «Транспорт», спеціальності 272 «Авіаційний транспорт».

A992 **Aerohydrogasdynamics and Flight Dynamics. Part I : Aerohydrogasdynamics** : Self-Study Method Guide . Part I . Aerohydrogasdynamics . Plotting the Aircraft Polar / compiler A. V. Goncharenko. – K. : NAU, 2020. – 57 p.

The **METHOD GUIDE** contains a few recommendations on the Self-Study in regards with the completion of the Calculation and Graphic Work on the academic subject of “Aerohydrogasdynamics and Flight Dynamics”, in Part I, concerning the section of “Aerohydrogasdynamics” at plotting the aircraft polar.

Designed for the 3rd year students of the Field of Study 27 “Transport”, Specialty 272 “Aviation Transport”.

CONTENTS

INTRODUCTION

This **METHOD GUIDE ON THE SELF-STUDY** (SS) is contemplated in response to the needs of our students in more detailed elaborations concerning problems stated, set or given for the students' independent work on this subject for the specified **CALCULATION AND GRAPHIC WORK** (CGW), possibly used in their **TERM PAPERING** (TP), **COURSE PROJECTING** (CP), further **GRADUATION PAPERS** or even **PH.D. STUDIES.** The whole material is split into portions. Each portion is intended to cover a fraction of probable applications aimed at **AERO-HYDRO-GAS-DYNAMICS AND FLIGHT DYNAMICS** or their adjacent problems.

The presented in this, **First Part**, **PART I** of the **METHOD GUIDE ON the SS** problems are dedicated, and a special attention is drawn here, to the scientific component of the SS work. Specifically, the objectives of the **PART I** are to help students cope with the challenging problems relating to the **AIRCRAFT** (A/C) **POLAR PLOTTING** (P/P) in regards with the A/C and its elements **AERODYNAMIC CHARACTERISTIC** (AC) determination.

The set of the considered problems is based upon the **ACADEMIC SUBJECT PROGRAM** on **"AEROHYDROGASDYNAMICS AND FLIGHT DYNAMICS"**, as well as upon the **RECOMMENDED LITERATURE SOURCES** (the list is presented, but not limited to it). The **LIST OF LITERATURE** at the end of the **METHOD GUIDE** is basic (major) and compiled partially in the alphabetic order with respect to the matter of supposed (assumed) importance.

The **REFERENCE LIST** is selected, set in the order [1-198], does not pretend for completeness, but instead it is aimed at developing the students' abilities of thinking and to analyze, contemplate in the specified directory rather than their abilities to know and memorize. However, these are very significant too. Actually, in the contemporary informative boom world, the needed or required data can easily be retrieved from the internet, found in multiple references, guidance materials [1-7], studies, dictionaries [3], comprehensive books [4] or monographs etc. The **METHOD GUIDE** is designed for the 3rd year students of the Field of Study: 27 "Transport", Specialty: 272 "Aviation Transport", Specialization: 01 "Maintenance and Repair of Aircraft and Aircraft Engines". It includes detailed solutions for an example of the

A/C P/P considered in reference [1]: **«Аэромеханика** : Методические указания и задания по выполнению курсовой работы / Составители: А. Г. Баскакова, В. Д. Трубенок. – К. : КМУГА, 1995. – 52 с.**» [Aeromechanics** : Method Guide and Tasks on Course Work Completion / Compilers: A. G. Baskakova, V. D. Trubenok. – K. : KIUCA, 1995. – 52 p.**] [in Russian].**

It is proposed to select the **VARIANTS** for the CGW, as well as for TP or CP completion in accordance with the recommendations of [1]:

1. If the last figure of the number of the **STUDENT'S CREDIT BOOK** is "5" and more, then it is the A/C with the **TURBOJET ENGINE** that must be taking for consideration.
2. If the last figure of the number of the **STUDENT'S CREDIT BOOK** is less than "5", then it is the A/C with the **TURBOPROP ENGINE** that must be taking for consideration.
3. The **VARIANT** itself is selected by the last figure of the number of the **STUDENT'S CREDIT BOOK.**
4. For the **WING** calculation it is also by the last figure of the number of the **STUDENT'S CREDIT BOOK.**
5. For the **WING MECHANIZATION** calculation it is by the third from the end figure of the number of the **STUDENT'S CREDIT BOOK.**

I. CALCULATION AND PLOTTING THE DEPENDENCE OF $C_{y_a} = f(\alpha)$ FOR THE SMALL NUMBERS OF M FOR THE FLIGHT AT THE ALTITUDE OF $H = 0$

The principal theoretical provisions can be found out in references [1-7].

For the **WINGS** with the relative extension $\lambda > 4$ the **COEFFICIENT** of the **LIFT FORCE** C_{y_a} depends linearly upon the **ANGLE OF ATTACK** α up to the values of $C_{y_a} = 0.8C_{y_a \max}$, [1, p. 5, (1)]:

$$C_{y_a}(\alpha) = C_{y_a}^{\alpha}(\alpha - \alpha_0), \qquad (1)$$

where: $C_{y_a}^{\alpha}$ – the derivative of the **LIFT FORCE COEFFICIENT** C_{y_a} with respect to the **ANGLE OF ATTACK** α:

$$C_{y_a}^{\alpha} = \frac{dC_{y_a}(\alpha)}{d\alpha} \qquad (2)$$

α_0 – the value of the **ANGLE OF ATTACK** at $C_{y_a} = 0$:

$$C_{y_a}(\alpha = \alpha_0) = 0. \qquad (3)$$

§ 1. Determination of the parameters of the wing

The **ANGLE OF ATTACK** of α_0 in turn depends upon the wing profile curvature f and the wing twist φ_{tw}, [1, p. 5]:

$$\alpha_0 = \alpha_{0f} + \alpha_{0\varphi_{tw}}, \qquad (4)$$

where α_{0f} – the **ANGLE OF ATTACK** depending upon the wing profile curvature f, [1, p. 5]:

$$\alpha_{0f} = -65\bar{f}\left[1.165 - 1.534\left(\bar{x}_f - \bar{c}\right)\right], \qquad (5)$$

where $\overline{f}$ – the relative curvature of the wing profile (given in the **TASK**):

$$\overline{f} = \frac{f}{b}, \tag{6}$$

where b – the chord (mean/average geometric chord) of the wing profile, [1, p. 10]:

$$b = \frac{S}{l}, \tag{7}$$

where S – the wing area (given in the **TASK**); l – the wing span (given in the **TASK**);

$\overline{x}_f$ – the relative abscise of the maximum of the wing profile curvature on the chord of the wing profile (given in the **TASK**):

$$\overline{x}_f = \frac{x_f}{b}, \tag{8}$$

where x_f – the abscise of the maximum of the wing profile curvature on the chord of the wing profile;

$\overline{c}$ – the relative thickness of the wing profile (given in the **TASK**):

$$\overline{c} = \frac{c}{b}, \tag{9}$$

where c – the thickness of the wing profile;

$\alpha_{0\varphi_{tw}}$ – the **ANGLE OF ATTACK** depending upon the wing twist φ_{tw}, [1, p. 5]:

$$\alpha_{0\varphi_{tw}} = -1.1\left(1-\overline{D}_\phi\right)\varphi_{tw}\left[\frac{\eta-\overline{D}_\phi(\eta-1)}{\eta+1}-\frac{\lambda \mathrm{tg}\left(\chi_{fe}\right)(1-\zeta)\left(1-\overline{D}_\phi\right)}{6}\right], \tag{10}$$

where $\overline{D}_\phi$ – the relative diameter of the fuselage at the place of the wing mounting:

$$\overline{D}_\phi = \frac{d_\phi}{l_\phi}, \tag{11}$$

where d_ϕ – the diameter of the fuselage (given in the **TASK**); l_ϕ – the length of the fuselage (given in the **TASK**);

η – the wing tapering (narrowing):

$$\eta = \frac{b_0}{b_k}\,, \tag{12}$$

where b_0 – the wing root chord (given in the **TASK**); b_k – the wing tip (ending) chord (given in the **TASK**);

λ – the relative extension of the wing:

$$\lambda = \frac{l^2}{S}\,; \tag{13}$$

ζ – the parameter depending upon the sweep-back (arrow) shape of the leading (front) χ_{fe} and trailing (rear) χ_{re} edge of the wing, [1, p. 6]:

$$\zeta = \frac{\mathrm{tg}(\chi_{re})}{\mathrm{tg}(\chi_{fe})}\,. \tag{14}$$

Thus, calculation:

1. The **ANGLE OF ATTACK** depending upon the wing profile curvature

At:

$$\bar{f} = 0.0202\,, \qquad \bar{x}_f = 0.3\,, \qquad \bar{c} = 0.12\,, \tag{15}$$

$$\alpha_{0f} = -65\bar{f}\left[1.165 - 1.534\left(\bar{x}_f - \bar{c}\right)\right]\,, \tag{16}$$

(see (5)),

$$\alpha_{0f} = -65 \cdot 0.0202\left[1.165 - 1.534(0.3 - 0.12)\right] = -1.167 \ \ [°, \text{degrees}]. \tag{17}$$

2. The **ANGLE OF ATTACK** depending upon the wing twist

At:

$$d_\phi = 6.08\,, \quad l_\phi = 56.1\,, \quad b_0 = 12.02\,, \quad b_k = 3.05\,, \quad l = 48.6\,,$$

$$S = 361.0\,, \qquad \chi_{re} = 21.82\,, \qquad \chi_{fe} = 32\,, \qquad \varphi_{tw} = -2.502\,, \tag{18}$$

$$\bar{D}_\phi = \frac{d_\phi}{l_\phi} = \frac{6.08}{56.1} = 0.108\,, \qquad \eta = \frac{b_0}{b_k} = \frac{12.02}{3.05} = 3.941\,,$$

$$\lambda = \frac{l^2}{S} = \frac{48.6^2}{361.0} = 6.543\,, \qquad \zeta = \frac{\mathrm{tg}(\chi_{re})}{\mathrm{tg}(\chi_{fe})} = \frac{\mathrm{tg}\left(21.82 \cdot \frac{\pi}{180}\right)}{\mathrm{tg}\left(32 \cdot \frac{\pi}{180}\right)} = 0.641\,.$$

(19)

$$\alpha_{0\varphi_{tw}} = -1.1\left(1-\overline{D}_\phi\right)\varphi_{tw}\left[\frac{\eta-\overline{D}_\phi(\eta-1)}{\eta+1} - \frac{\lambda \mathrm{tg}(\chi_{fe})(1-\zeta)\left(1-\overline{D}_\phi\right)}{6}\right], \quad (20)$$

(see (10)),

$$\alpha_{0\varphi_{tw}} = -1.1(1-0.108)(-2.502)\times$$

$$\times\left[\frac{3.941-0.108(3.941-1)}{3.941+1} - -\frac{6.543\cdot \mathrm{tg}\left(32\cdot\frac{\pi}{180}\right)(1-0.641)(1-0.108)}{6}\right],$$

$$\alpha_{0\varphi_{tw}} = 1.233\,. \quad (21)$$

<u>3.</u> <u>The zero **LIFT FORCE ANGLE OF ATTACK** of α_0:</u>

$$\alpha_0 = \alpha_{0f} + \alpha_{0\varphi_{tw}}\,, \quad (22)$$

(see (4, 17, 21)),

$$\alpha_0 = -1.167 + 1.233 = 0.066\,. \quad (23)$$

§ 2. Determination of the derivative of the lift force coefficient

The derivative of the **LIFT FORCE COEFFICIENT** C_{y_a} with respect to the **ANGLE OF ATTACK** α for the sweep-back (arrow) shape wings of the heavy A/C is determined by the formula of [1, p. 6, (2)]:

$$C_{y_a}^{\alpha} = \frac{4\pi\lambda_{ef}}{\sqrt{D^2+\beta^2\lambda_{ef}^2} + \sqrt{D^2\zeta^2+\beta^2\lambda_{ef}^2} + D\frac{1-\zeta}{\eta-1} + 4} \text{ [1/radians]}, \quad (24)$$

where λ_{ef} – the relative effective extension of the wing, is calculated by the formula of [1, p. 6]:

$$\lambda_{ef} = \lambda \frac{k_{\chi}^{1}}{1+\frac{S_i}{S}}; \tag{25}$$

where k_{χ}^{1} – the coefficient that takes into account the sweep-back (arrow) shape of the wing. For the first approximation, for the wings with the sweep-back (arrow) shape of the $\chi_{1/4} = 20\ldots40$ °, it is possible to accept:

$$k_{\chi}^{1} = 0.95\ldots0.80; \tag{26}$$

S_i – the area of the wing occupied by the fuselage, engines compartments etc. (given in the **TASK**);

D – the parameter depending upon the tapering (narrowing) and sweep-back (arrow) shape of the wing, is calculated by the formula of

$$D = \frac{4(\eta-1)}{(\eta+1)(\zeta+1)}; \tag{27}$$

β – the coefficient depending upon the number of M_∞ of the undisturbed flow:

$$\beta = \sqrt{1-M_\infty^2}; \tag{28}$$

$$M_\infty = \frac{V_{land}}{a_{H=0}}, \tag{29}$$

where V_{land} – the aircraft speed at landing (in meters per second), [1, p. 7, (5)]:

$$V_{land} = 3\sqrt{\frac{m_{land}}{S}}, \tag{30}$$

where m_{land} – the aircraft mass at landing:

$$m_{land} = m_{toff} - 0.8m_f, \tag{31}$$

where m_{toff} – the aircraft mass at takeoff (given in the **TASK**);

m_f – the aircraft fuel mass:

$$m_f = (0.2\ldots0.4)m_{toff} \,; \tag{32}$$

$a_{H=0}$ – the speed of sound at the altitude of $H = 0$.

The derivative of the **LIFT FORCE COEFFICIENT** C_{y_a} with respect to the **ANGLE OF ATTACK** α for the rectangular shape wings of the light A/C is better to determine by the formula of [1, p. 6, (3)]:

$$C_{y_a}^{\alpha} = \frac{2\pi\lambda_{ef}}{\overline{p}\lambda_{ef} + 2} \,, \tag{33}$$

where $\overline{p}$ – the ratio of the wing half-perimeter to its span [1, p. 6]:

$$\overline{p} = \frac{p}{2l} \,, \tag{34}$$

where p – the wing perimeter.

<u>**Thus, calculation:**</u>

<u>**4.** The relative effective extension of the wing</u>

At:

$$k_{\chi}^{1} = 0.8112 \,, \qquad S_i = 73.1 \,, \tag{35}$$

$$\lambda_{ef} = \lambda \frac{k_{\chi}^{1}}{1 + \dfrac{S_i}{S}} \,, \tag{36}$$

(see (18, 19, 25)),

$$\lambda_{ef} = 6.543 \frac{0.8112}{1 + \dfrac{73.1}{361.0}} = 4.414 \,. \tag{37}$$

<u>**5.** The parameter depending upon the tapering (narrowing) and sweep-back (arrow) shape of the wing:</u>

$$D = \frac{4(\eta - 1)}{(\eta + 1)(\zeta + 1)} \,, \tag{38}$$

(see (19, 27)),

$$D = \frac{4(3.941-1)}{(3.941+1)(0.641+1)} = 1.451. \tag{39}$$

6. The aircraft fuel mass

At:

$$m_{toff} = 206{,}000\ , \tag{40}$$

$$m_f = 0.45 m_{toff} = 0.45 \cdot 206{,}000 = 92{,}700\ , \tag{41}$$

(see (32)).

7. The aircraft mass at landing:

$$m_{land} = m_{toff} - 0.8 m_f = 206{,}000 - 0.8 \cdot 92{,}700 = 131{,}840\ , \tag{42}$$

(see (31)).

8. The aircraft mass at landing:

$$V_{land} = 3\sqrt{\frac{m_{land}}{S}} = 3\sqrt{\frac{131{,}840}{361.0}} = 57.331, \tag{43}$$

(see (30)).

9. The number of M_∞ of the undisturbed flow

At:

$$a_{H=0} = 340.1, \tag{44}$$

$$M_\infty = \frac{V_{land}}{a_{H=0}} = \frac{57.331}{340.1} = 0.169, \tag{45}$$

(see (29)).

10. The coefficient depending upon the number of M_∞:

$$\beta = \sqrt{1 - M_\infty^2} = \sqrt{1 - 0.169^2} = 0.986, \tag{46}$$

(see (28)).

11. The derivative of the **LIFT FORCE COEFFICIENT** C_{y_a} with respect to the **ANGLE OF ATTACK** α for the sweep-back (arrow) shape wings of the heavy A/C:

$$C_{y_a}^{\alpha} = \frac{4\pi\lambda_{ef}}{\sqrt{D^2+\beta^2\lambda_{ef}^2}+\sqrt{D^2\zeta^2+\beta^2\lambda_{ef}^2}+D\dfrac{1-\zeta}{\eta-1}+4}, \tag{47}$$

(see (24)),

$$C_{y_a}^{\alpha} = \frac{4\cdot 3.14\cdot 4.414}{\begin{gathered}\sqrt{1.451^2+0.986^2\cdot 4.414^2}+\\ +\sqrt{1.451^2\cdot 0.641^2+0.986^2\cdot 4.414^2}+\\ +1.451\cdot\dfrac{1-0.641}{3.941-1}+4\end{gathered}} = 4.198\,. \tag{48}$$

For plotting the dependence of $C_{y_a} = f(\alpha)$ it is necessary to determine the **LIFT FORCE COEFFICIENT MAXIMUM** value $C_{y_a\max}$, the **CRITICAL ANGLE OF ATTACK** value α_{cr}, and the **ALLOWABLE ANGLE OF ATTACK** value α_{allow} corresponding to the beginning of the flow break out off the top surface of the wing. Thus, α_{cr} corresponds with the $C_{y_a\max}$; and α_{allow}: to the $C_{y_a allow}$, [1, p. 6].

§ 3. Determination of the lift force coefficient maximum

The **LIFT FORCE COEFFICIENT MAXIMUM** value is determined by the formula of [1, p. 7]:

$$C_{y_a\max} = C_{y_0} k_\eta k_\chi k_{\mathrm{Re}}\,, \tag{49}$$

where C_{y_0} – the experimentally found **LIFT FORCE COEFFICIENT MAXIMUM** value of the wing profile at the specified number of Re_0 (given in the **TASK**); k_η – the correction (adjustment, tuning) coefficient depending upon the wing tapering (narrowing), [1, p. 7]:

$$k_\eta = 0.91\eta^{0.017}\left(0.95 + 0.057\eta - 0.012\eta^2\right); \tag{50}$$

k_χ – the correction (adjustment, tuning) coefficient depending upon the wing sweep-back (arrow) shape and tapering (narrowing), [1, p. 7]:

$$k_\chi = \frac{1}{\eta+1}\left[0.09 + 0.51\eta + (0.91 + 0.49\eta)\cos^2\left(\chi_{1/4}\right)\right]; \tag{51}$$

where $\cos\left(\chi_{1/4}\right)$ – cosine of the wing sweep-back (arrow) shape angle by the line of the $1/4$-th of the chord (given in the **TASK**);

k_{Re} – the correction (adjustment, tuning) coefficient depending upon the Reynolds' number, [1, p. 7]:

$$\mathrm{Re} = \frac{V_{land} b_{mg}}{\nu}, \tag{52}$$

where b_{mg} – mean (average) geometric chord of the wing (7); ν – kinematics viscosity; and Re_0 – the experimental Reynolds' number (given in the **TASK**), [1, p. 7]:

$$k_{\mathrm{Re}} = 1.49\left(\frac{\log \mathrm{Re}}{\log \mathrm{Re}_0}\right)^{1.62} \exp\left(-0.35\frac{\log \mathrm{Re}}{\log \mathrm{Re}_0}\right). \tag{53}$$

Thus, calculation:

12. The correction coefficient depending upon the wing tapering:

$$k_\eta = 0.91\eta^{0.017}\left(0.95 + 0.057\eta - 0.012\eta^2\right), \tag{54}$$

(see (19, 50)),

$$k_\eta = 0.91 \cdot 3.941^{0.017}\left(0.95 + 0.057 \cdot 3.941 - 0.012 \cdot 3.941^2\right) = 0.921. \tag{55}$$

<u>13.</u> <u>The correction coefficient depending upon the wing sweep-back (arrow) shape and tapering</u>

At:

$$\chi_{1/4} = 38.5, \tag{56}$$

$$k_{\chi} = \frac{1}{\eta+1}\left[0.09 + 0.51\eta + (0.91 + 0.49\eta)\cos^2\left(\chi_{1/4}\right)\right], \tag{57}$$

(see (19, 51)),

$$k_{\chi} = \frac{1}{3.941+1}\begin{bmatrix} 0.09 + 0.51\cdot 3.941 + \\ +(0.91 + 0.49\cdot 3.941)\cos^2\left(38.5\cdot\dfrac{\pi}{180}\right)\end{bmatrix} = 0.777. \tag{58}$$

<u>14.</u> <u>The mean geometric chord of the wing:</u>

$$b_{mg} = \frac{S}{l} = \frac{361.0}{48.6} = 7.428, \tag{59}$$

(see (7)).

<u>15.</u> <u>The Reynolds' number</u>

At:

$$\nu = 1.44607 \cdot 10^{-5}, \tag{60}$$

$$\mathrm{Re} = \frac{V_{land} b_{mg}}{\nu} = \frac{57.331\cdot 7.428}{1.44607\cdot 10^{-5}} = 2.945\cdot 10^{7}, \tag{61}$$

(see (43, 52, 59)).

<u>16.</u> <u>The correction coefficient depending upon the Reynolds' number</u>

At:

$$\mathrm{Re}_0 = 4.4765, \tag{62}$$

$$k_{\mathrm{Re}} = 1.49\left(\frac{\log \mathrm{Re}}{\log \mathrm{Re}_0}\right)^{1.62} \exp\left(-0.35\frac{\log \mathrm{Re}}{\log \mathrm{Re}_0}\right), \tag{63}$$

(see (53, 61)),

$$k_{\mathrm{Re}} = 1.49\left(\frac{\log 2.945\cdot 10^{7}}{\log 4.4765}\right)^{1.62} \exp\left(-0.35\frac{\log 2.945\cdot 10^{7}}{\log 4.4765}\right) = 1.399. \tag{64}$$

17. The **LIFT FORCE COEFFICIENT MAXIMUM** value

At:

$$C_{y_0} = 1.1, \tag{65}$$

$$C_{y_a \max} = C_{y_0} k_\eta k_\chi k_{\mathrm{Re}} = 1.1 \cdot 0.921 \cdot 0.777 \cdot 1.399 = 1.101, \tag{66}$$

(see (55, 58, 64)).

§ 4. Determination of the critical angle of attack

The **CRITICAL ANGLE OF ATTACK** value α_{cr} is determined by the formula of [1, p. 7]:

$$\alpha_{cr} = \frac{57.3 C_{y_a \max}}{C_{y_a}^{\alpha}} + \alpha_0^0 + \Delta\alpha^0, \tag{67}$$

where: α_0^0 – the correction coefficient; $\Delta\alpha^0$ – the correction coefficient with the value range of 1 … 3 °.

Thus, calculation:

18. The correction coefficient depending upon the wing tapering

At:

$$\alpha_0^0 = 1, \qquad \Delta\alpha^0 = 2, \tag{68}$$

$$\alpha_{cr} = \frac{57.3 C_{y_a \max}}{C_{y_a}^{\alpha}} + \alpha_0^0 + \Delta\alpha^0 = \frac{57.3 \cdot 1.101}{4.198} + 1 + 2 = 18.027, \tag{69}$$

(see (48, 66, 67)).

§ 5. Determination of the allowable lift force coefficient value

The **ALLOWABLE LIFT FORCE COEFFICIENT** value $C_{y_a allow}$ is determined on condition that, [1, p. 7]:

$$C_{y_a allow} = 0.8 C_{y_a \max} \,. \tag{70}$$

<u>Thus, calculation:</u>

<u>**19.** The **ALLOWABLE LIFT FORCE COEFFICIENT** value $C_{y_a allow}$:</u>

$$C_{y_a allow} = 0.8 C_{y_a \max} = 0.8 \cdot 1.101 = 0.881 \,, \tag{71}$$

(see (66)).

§ 6. Determination of the allowable angle of attack

The **ALLOWABLE ANGLE OF ATTACK** value α_{allow} is determined as it follows from the formulae of (1, 70):

$$C_{y_a allow} = C_{y_a}\left(\alpha = \alpha_{allow}\right) = C_{y_a}^{\alpha}\left(\alpha_{allow} - \alpha_0\right), \tag{72}$$

$$\alpha_{allow} = \frac{C_{y_a allow}}{C_{y_a}^{\alpha}} + \alpha_0 \,. \tag{73}$$

<u>Thus, calculation:</u>

<u>**20.** The **ALLOWABLE ANGLE OF ATTACK** value α_{allow}:</u>

$$\alpha_{allow} = \left(\frac{C_{y_a allow}}{C_{y_a}^{\alpha}} + \alpha_0\right) = \left(\frac{0.881}{4.198} + 0.066 \cdot \frac{\pi}{180}\right)\frac{180}{\pi} = 12.086 \,, \tag{74}$$

(see (23, 48, 71, 73)).

§ 7. Plotting the diagram for the dependence of the lift force coefficient upon the angle of attack

Plotting the diagram for the dependence of the **LIFT FORCE COEFFICIENT** of C_{y_a} upon the **ANGLE OF ATTACK** value α: $C_{y_a}(\alpha)$, is implemented (realized) in the following way, [1, p. 8, ## 1-6, Fig. 1]:

1. In the coordinate (reference) system of the abscise axis of α and the ordinate axis of C_{y_a} respectively, it is put the points (dots) with the coordinates of the **ANGLES OF ATTACK** values of α_0, α_{allow}, and α_{cr} against of the coordinates of the **LIFT FORCE COEFFICIENT** of C_{y_a} values of $C_{y_a}(\alpha = \alpha_0) = 0$, $C_{y_a allow}$, and $C_{y_a \max}$ in correspondence.
2. The points with the coordinates of $(\alpha_0, 0)$ and $(\alpha_{allow}, C_{y_a allow})$ are connected by drawing a straight line; whereas the points with the coordinates of $(\alpha_{allow}, C_{y_a allow})$ and $(\alpha_{cr}, C_{y_a \max})$ are connected by drawing a smooth curve line.

Thus, plotting:

1. The diagram of the dependence of the **LIFT FORCE COEFFICIENT** C_{y_a} upon the **ANGLE OF ATTACK** value α

The diagram of the dependence of the **LIFT FORCE COEFFICIENT** C_{y_a} upon the **ANGLE OF ATTACK** value α, plotted by the calculated data of the **CALCULATION SECTION ITEMS OF 1-20**, in corresponding scales is presented in Fig. 1.

In Fig. 1 the points of allowable and critical/maximal values with the coordinates of $(\alpha_{allow}, C_{y_a allow})$ and $(\alpha_{cr}, C_{y_a \max})$ are indicated correspondingly as $\alpha\mathrm{dop}$, Cydop and $\alpha\mathrm{cr}$, Cyamax.

The calculated above values of $\alpha\mathrm{dop} = 12.086$, formula (74), $\mathrm{Cydop} = 0.881$, (71); and $\alpha\mathrm{cr} = 18.027$, (69), $\mathrm{Cyamax} = 1.101$, (66), are also portrayed there (see Fig. 1).

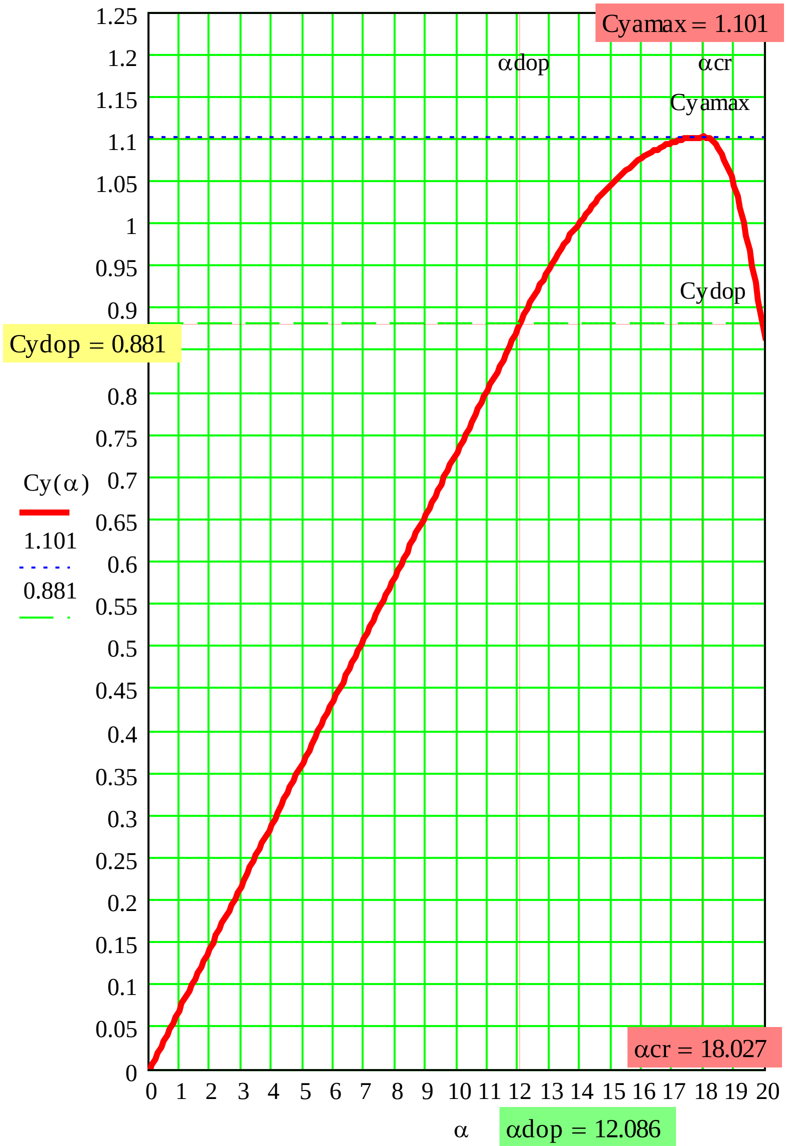

Fig. 1 – The diagram of the dependence of the **LIFT FORCE COEFFICIENT** C_{y_a} upon the **ANGLE OF ATTACK** value α

II. CALCULATION AND PLOTTING THE DEPENDENCE OF $C_{x_a} = f(\alpha)$ FOR THE SMALL NUMBERS OF M FOR THE FLIGHT AT THE ALTITUDE OF $H = 0$

The principal theoretical provisions can be found out in references [1-7].

For the **SMALL NUMBERS** of M the **COEFFICIENT** of the **DRAG FORCE** C_{x_a} of an A/C can be determined by the formula of [1, p. 8, (6)]:

$$C_{x_a} = C_{x_{a0}} + C_{x_{ai}} + \Delta C_{x_p}, \tag{75}$$

where: $C_{x_{a0}}$ – the coefficient of the A/C **DRAG FORCE** resistance when the **LIFT FORCE COEFFICIENT** of $C_{y_a} = 0$ (coefficient of the resistance because of (due to) the **FORCES OF FRICTION**); $C_{x_{ai}}$ – the coefficient of the A/C **INDUCTIVE** resistance; ΔC_{x_p} – the correction (adjustment, tuning) coefficient taking into account the alteration (change, variation) of the profile resistance due to (because of, caused by) the pressures redistribution over the A/C **WING SURFACE** at $\alpha \neq \alpha_0$.

§ 8. Determination of the A/C drag coefficient at the zero lift force coefficient value

The **COEFFICIENT OF THE A/C DRAG FORCE RESISTANCE** $C_{x_{a0}}$ at the **LIFT FORCE COEFFICIENT** of $C_{y_a} = 0$, in case of the SS work on the TP or CP completion, can be determined on the basis of the methods described in reference [1, pp. 9-13, Section 2.1, (7-10), Fig. 2-4].

For the considered CGW performance the SS is simplified, nevertheless, as well as for TP or CP completion in accordance with the recommendations of [1] it is reduced down to [1, p. 13, (9)]:

$$C_{x_{a0}} = (1.03\ldots1.05)\frac{\sum nC_{x_{ak}} S_k}{S}, \qquad (76)$$

where $(1.03\ldots1.05)$ – the multiplier of the additional resistance that takes into account some small A/C details and parts (such as, for instance, aerial/antenna, different distinguishing parts etc.) influence; n – the number of the A/C one-type (unified) elements under consideration; $C_{x_{ak}}$ – the coefficient of the specified element resistance determined by the formula of [1, p. 13, (10)]:

$$C_{x_{ak}} = 2C_F K_C K_i, \qquad (77)$$

where $2C_F$ – the summarized (summed up, cumulative, integral, total, whole, full amount, entire, full) coefficient of the equivalent flat plate friction resistance that is determined in turn as [1, pp. 9, 10, Fig. 2]:

$$2C_F = f\left(\mathrm{Re}, \bar{x}_t\right), \qquad (78)$$

where Re – the Reynolds' number, similar with (analogous to, likewise) (52), however, with making allowance (taking into account, making necessary corrections/modifications/adjustments) for the specified conditions of flowing (blowing); $\bar{x}_t$ – the relative abscise of the point where the laminar boundary layer transmits into the turbulent one, described in reference [1, p. 9];

K_C – the coefficient that takes into account the impact (influence) of the A/C specified element relative thickness upon its resistance, described in reference [1, p. 9]; K_i – the coefficient that takes into account the aerodynamic interference (interferencion/interferention, mutual impact/influence) of the A/C specified elements. The interference between the **WING** and **FUSELAGE** at the subsonic speeds of flight is estimated by the coefficient determined with the dependence of [1, p. 11]:

$$K_i = 1 - k\frac{S_i}{S}, \qquad (79)$$

where k – the coefficient that takes into account the mutual positioning (situating) of the **WING** and **FUSELAGE**, the shape of the **WING** at the plane projection; and that coefficient has the values following the reference [1, p. 11];

S_k – the computational area of the A/C element, [1, p. 9]:

$$S_k = \frac{S_{om}}{2}, \qquad (80)$$

where S_{om} – the A/C element full (total, entire, whole) side surface (area) “flown” (flushed, touched) by the flow [1, p. 9].

The computations results performed with the help of (76-80) can be presented in the view of a Table, [1, pp. 13, 14, Table 1].

Thus, calculation:

21. The **COEFFICIENT OF THE A/C DRAG FORCE RESISTANCE** $C_{x_{a0}}$ at the **LIFT FORCE COEFFICIENT** of $C_{y_a} = 0$

At:

$$C_{F_W} = 2.75 \cdot 10^{-3}, \quad K_{C_W} = 1.29, \quad K_{i_W} = 0.962, \quad S = 361.0, \qquad (81)$$

for the **WING**;

$$C_{F_{Hst}} = 2.992 \cdot 10^{-3}, \quad K_{C_{Hst}} = 1.29, \quad K_{i_{Hst}} = 0.925, \quad S_{Hst} = 93.9, \qquad (82)$$

for the **HORIZONTAL EMPENNAGE**;

$$C_{F_{Vst}} = 2.634 \cdot 10^{-3}, \quad K_{C_{Vst}} = 1.29, \quad K_{i_{Vst}} = 1, \quad S_{Vst} = 56.06, \qquad (83)$$

for the **VERTICAL EMPENNAGE**;

$$C_{F_F} = 1.803 \cdot 10^{-3}, \quad K_{C_F} = 1.09, \quad K_{i_F} = 1, \quad S_F = 798.46 \qquad (84)$$

for the **FUSELAGE**;

$$C_{F_{en}} = 2.579 \cdot 10^{-3}, \quad K_{C_{en}} = 1.7, \quad K_{i_{en}} = 1, \quad S_{en} = 40.916, \quad n_{en} = 4, \qquad (85)$$

for the **ENGINES**;

$$C_{F_{leg}} = 2.597 \cdot 10^{-3}, \quad K_{C_{leg}} = 1, \quad K_{i_{leg}} = 1, \quad S_{leg} = 2, \quad n_{leg} = 3, \qquad (86)$$

for the **LANDING GEAR**;

$$C_{x_{a_0}} = 1.05\left(\begin{aligned} &2C_{F_W}K_{C_W}K_{i_W} + 2C_{F_{Hst}}K_{C_{Hst}}K_{i_{Hst}}\frac{S_{Hst}}{S} + \\ &+2C_{F_{Vst}}K_{C_{Vst}}K_{i_{Vst}}\frac{S_{Vst}}{S} + 2C_{F_F}K_{C_F}K_{i_F}\frac{S_F}{S} + \\ &+2C_{F_{en}}K_{C_{en}}K_{i_{en}}\frac{S_{en}}{2S}n_{en} + 2C_{F_{leg}}K_{C_{leg}}K_{i_{leg}}\frac{S_{leg}}{2S}n_{leg} \end{aligned}\right), \quad (87)$$

(see (76)),

$$C_{x_{a_0}} = 1.05\left(\begin{aligned} &2\cdot 2.75\cdot 10^{-3}\cdot 1.29\cdot 0.962 + \\ &+2\cdot 2.992\cdot 10^{-3}\cdot 1.29\cdot 0.925\cdot\frac{93.9}{361.0} + \\ &+2\cdot 2.634\cdot 10^{-3}\cdot 1.29\cdot 1\cdot\frac{56.06}{361.0} + \\ &+2\cdot 1.803\cdot 10^{-3}\cdot 1.09\cdot 1\cdot\frac{798.46}{361.0} + \\ &+2\cdot 2.579\cdot 10^{-3}\cdot 1.7\cdot 1\cdot\frac{40.916}{2\cdot 361.0}\cdot 4 + \\ &+2\cdot 2.597\cdot 10^{-3}\cdot 1\cdot 1\cdot\frac{2}{2\cdot 361.0}\cdot 3 \end{aligned}\right) = 0.021. \quad (88)$$

§ 9. Determination of the coefficient of the A/C inductive resistance

The **COEFFICIENT OF THE A/C INDUCTIVE RESISTANCE** $C_{x_{a_i}}$ depends upon the **LIFT FORCE COEFFICIENT** C_{y_a}, **EFFECTIVE RELATIVE EXTENSION OF THE WING** λ_{ef}, and the coefficient of δ, depending in turn upon the **WING** shape in the plane projection.

The **COEFFICIENT OF THE A/C INDUCTIVE RESISTANCE** $C_{x_{a_i}}$ is determined by the formula of [1, p. 13, (11)]:

$$C_{x_{a_i}} = \frac{C_{y_a}^2(1+\delta)}{\pi\lambda_{ef}}, \quad (89)$$

where coefficient of δ can be determined by the formula from the reference of [1, p. 13]:

$$\delta = \frac{0.02\lambda_{ef}}{\cos\left(\chi_{1/4}\right)}\left(3.1 - \frac{14}{\eta} + \frac{20}{\eta^2} - \frac{8}{\eta^3}\right), \tag{90}$$

or by the diagram of the dependence of $\delta = f(\lambda, \eta)$ presented at the reference [1, p. 15, Fig. 5].

Thus, calculation:

22. The **COEFFICIENT** of δ depending upon the **WING** shape in the plane projection

At:

$$\eta = 3.941, \qquad \lambda_{ef} = 4.414, \qquad \chi_{1/4} = 38.5, \tag{91}$$

$$\delta = \frac{0.02\lambda_{ef}}{\cos\left(\chi_{1/4}\right)}\left(3.1 - \frac{14}{\eta} + \frac{20}{\eta^2} - \frac{8}{\eta^3}\right), \tag{92}$$

(see (19, 37, 56, 90)),

$$\delta = \frac{0.02 \cdot 4.414}{\cos\left(38.5 \cdot \frac{\pi}{180}\right)}\left(3.1 - \frac{14}{3.941} + \frac{20}{3.941^2} - \frac{8}{3.941^3}\right) = 0.079, \tag{93}$$

23. The **COEFFICIENT OF THE A/C INDUCTIVE RESISTANCE** $C_{x_{a_i}}$:

$$C_{x_{a_i}} = \frac{C_{y_a}^2(1+\delta)}{\pi\lambda_{ef}} = \frac{C_{y_a}^2(\alpha)(1+0.079)}{\pi \cdot 4.414} = 0.078 C_{y_a}^2(\alpha) = C_{x_{a_i}}(\alpha), \tag{94}$$

(see (§ 7, Fig. 1, 89, 93, 91)).

§ 10. Determination of the correction coefficient ΔC_{x_p} taking into account the alteration of the profile resistance due to the pressures redistribution over the A/C wing surface at $\alpha \neq \alpha_0$

The **RESISTANCE COEFFICIENT** of ΔC_{x_p} at $C_{y_a} > 0$ is a function of the **RELATIVE COEFFICIENT OF THE LIFT FORCE**, [1, p. 13]:

$$\overline{C}_{y_a} = \frac{C_{y_a}}{C_{y_a \max}}; \tag{95}$$

and it is determined by the formula of [1, p. 13]:

$$\Delta C_{x_p} = \left(\overline{C}_{y_a}\right)^4 \left\{1 - \exp\left[-0.1\left(\overline{C}_{y_a} - 0.4\right)^2\right]\right\}. \tag{96}$$

<u>**Thus, calculation:**</u>

<u>**24.** The **RESISTANCE COEFFICIENT** of ΔC_{x_p} :</u>

$$\overline{C}_{y_a}(\alpha) = \frac{C_{y_a}(\alpha)}{C_{y_a \max}}, \tag{97}$$

(see (§ 7, Fig. 1, 95, 96)),

$$\Delta C_{x_p}(\alpha) = \left[\overline{C}_{y_a}(\alpha)\right]^4 \left\{1 - \exp\left[-0.1\left[\overline{C}_{y_a}(\alpha) - 0.4\right]^2\right]\right\}. \tag{98}$$

§ 11. Determination of the A/C drag force coefficient upon the angle of attack

Thus, since (94, 98), the **COEFFICIENT** of the **DRAG FORCE** C_{x_a} of an A/C became a function of the **ANGLES OF ATTACK** values of α.

<u>**Thus, calculation:**</u>

<u>**25.** The **DRAG COEFFICIENT**</u>

At:

$$C_{x_{a0}} = 0.021\,, \qquad C_{x_a} = C_{x_{a0}} + C_{x_{a_i}} + \Delta C_{x_p}\,, \tag{99}$$

(see (88, 75, § 7, Fig. 1, 94, 98)),

$$C_{x_a}(\alpha) = 0.021 + C_{x_{a_i}}(\alpha) + \Delta C_{x_p}(\alpha). \tag{100}$$

§ 12. Plotting the diagram for the dependence of the drag force coefficient upon the angle of attack

Plotting the diagram for the dependence of the **DRAG FORCE COEFFICIENT** of C_{x_a} upon the **ANGLE OF ATTACK** value α: $C_{x_a}(\alpha)$, is implemented (realized) in the following way:

1. In the coordinate (reference) system of the abscise axis of α and the ordinate axis of C_{x_a} respectively, it is put the points (dots) with the coordinates of the **ANGLES OF ATTACK** values of α_j against of the coordinates of the **DRAG FORCE COEFFICIENT** of $C_{x_a j}$ values in correspondence.
2. The points with the coordinates of $(\alpha_j, C_{x_a j})$ are connected by drawing a smooth curve line.

Thus, plotting:

2. The diagram of the dependence of the **DRAG FORCE COEFFICIENT** C_{x_a} upon the **ANGLE OF ATTACK** value α

The diagram of the dependence of the **DRAG FORCE COEFFICIENT** C_{x_a} upon the **ANGLE OF ATTACK** value α, plotted by the calculated data of the **CALCULATION SECTION ITEMS OF 21-25**, in corresponding scales is presented in Fig. 2.

In Fig. 2 the values of the allowable (acceptable) and critical (allowable maximal) angles of attack, $\alpha\mathrm{dop}$ and $\alpha\mathrm{cr}$, are shown (the lines are drawn) as well as in Fig. 1.

The calculated above values of $\alpha dop = 12.086$, formula (74), and $\alpha cr = 18.027$, (69).

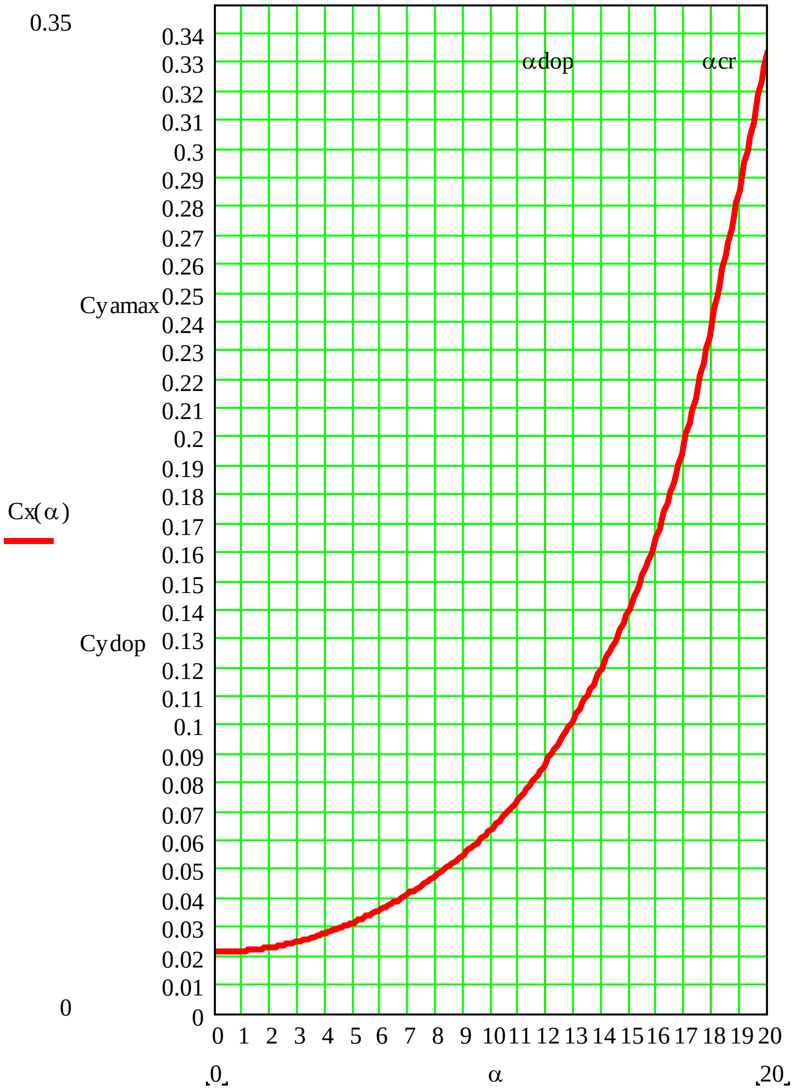

Fig. 2 – The diagram of the dependence of the **DRAG FORCE COEFFICIENT** C_{x_a} upon the **ANGLE OF ATTACK** value α

III. PLOTTING THE A/C POLAR $C_{y_a} = f(C_{x_a})$ FOR THE SMALL NUMBERS OF M FOR THE FLIGHT AT THE ALTITUDE OF $H = 0$

The principal theoretical provisions can be found out in references [1-7].

For the considered case, the dependencies of $C_{y_a} = f(\alpha)$ and $C_{x_a} = f(\alpha)$ make the parametric curve of $C_{y_a} = f(C_{x_a})$. Both dependencies of $C_{y_a} = f(\alpha)$ and $C_{x_a} = f(\alpha)$ plotted in corresponding scales are demonstrated in Fig. 3.

In Fig. 3 the points of allowable and critical/maximal values with the coordinates of $(\alpha_{allow}, C_{y_a allow})$ and $(\alpha_{cr}, C_{y_a \max})$ are indicated (the lines are drawn) correspondingly as $\alpha\mathrm{dop}$, Cydop and $\alpha\mathrm{cr}$, Cyamax (also see Fig. 1).

The calculated above values of $\alpha\mathrm{dop} = 12.086$, formula (74), $\mathrm{Cydop} = 0.881$, (71); and $\alpha\mathrm{cr} = 18.027$, (69), $\mathrm{Cyamax} = 1.101$, (66), are portrayed in Fig. 1.

The **A/C POLAR** $C_{y_a} = f(C_{x_a})$ for the **SMALL NUMBERS** of M for the **FLIGHT AT THE ALTITUDE** of $H = 0$ is plotted in Fig. 4.

In Fig. 4, actually, it is shown the phase portrait of the dependence of $C_{y_a} = f(\alpha)$ upon the dependence of $C_{x_a} = f(\alpha)$ as the phase variable of the angle of attack of α; i.e. $C_{y_a} = f(C_{x_a})$.

The calculated above values of $\alpha\mathrm{dop} = 12.086$, formula (74), $\mathrm{Cydop} = 0.881$, (71); and $\alpha\mathrm{cr} = 18.027$, (69), $\mathrm{Cyamax} = 1.101$, (66), (see Fig. 1) now correspond to the phase coordinates of $\mathrm{Cx}(\alpha\mathrm{dop}) = 0.088$, $\mathrm{Cydop} = 0.881$; and $\mathrm{Cx}(\alpha\mathrm{cr}) = 0.241$, $\mathrm{Cyamax} = 1.101$ (see both Fig. 1 and Fig. 4; as well as Fig. 2 and Fig. 3; and compare them). Also, there are C_{y_a} values at the different angles of attack denoted as $\mathrm{Cy}(0\ldots18)$ in Fig. 4.

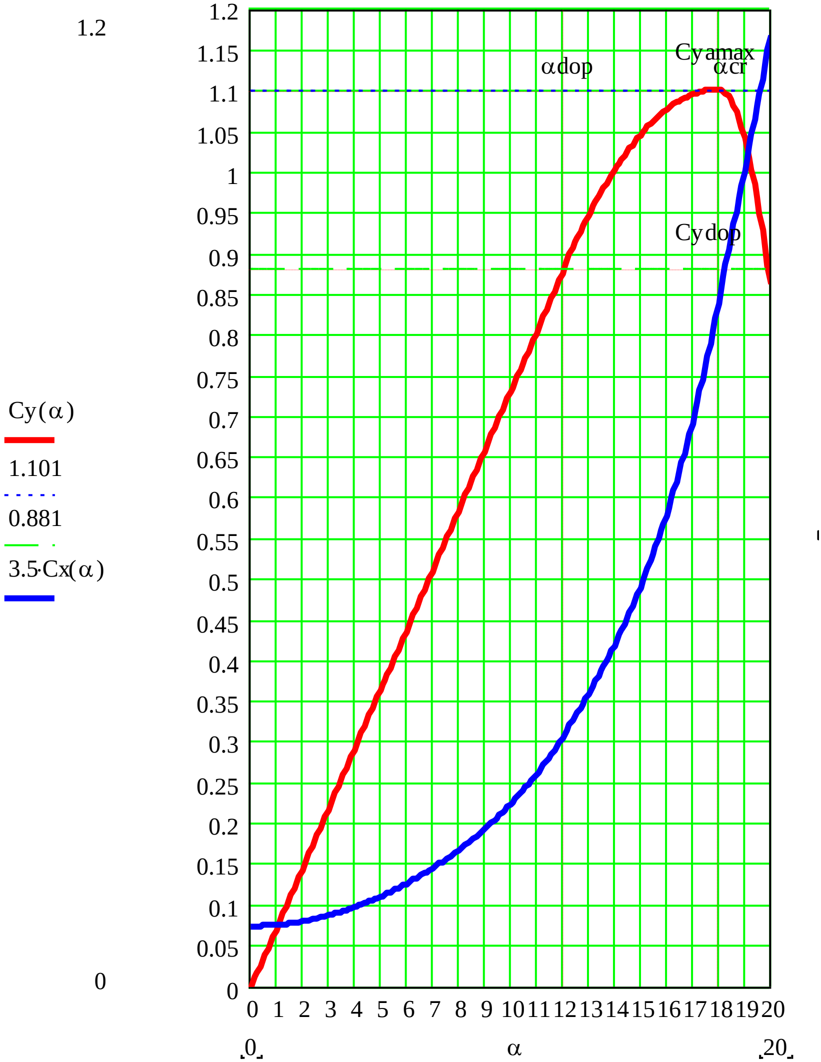

Fig. 3 – The diagrams of the dependences of the **LIFT** and **DRAG FORCE COEFFICIENTS**, C_{y_a} and C_{x_a} in respect, in corresponding scales, upon the **ANGLE OF ATTACK** value α

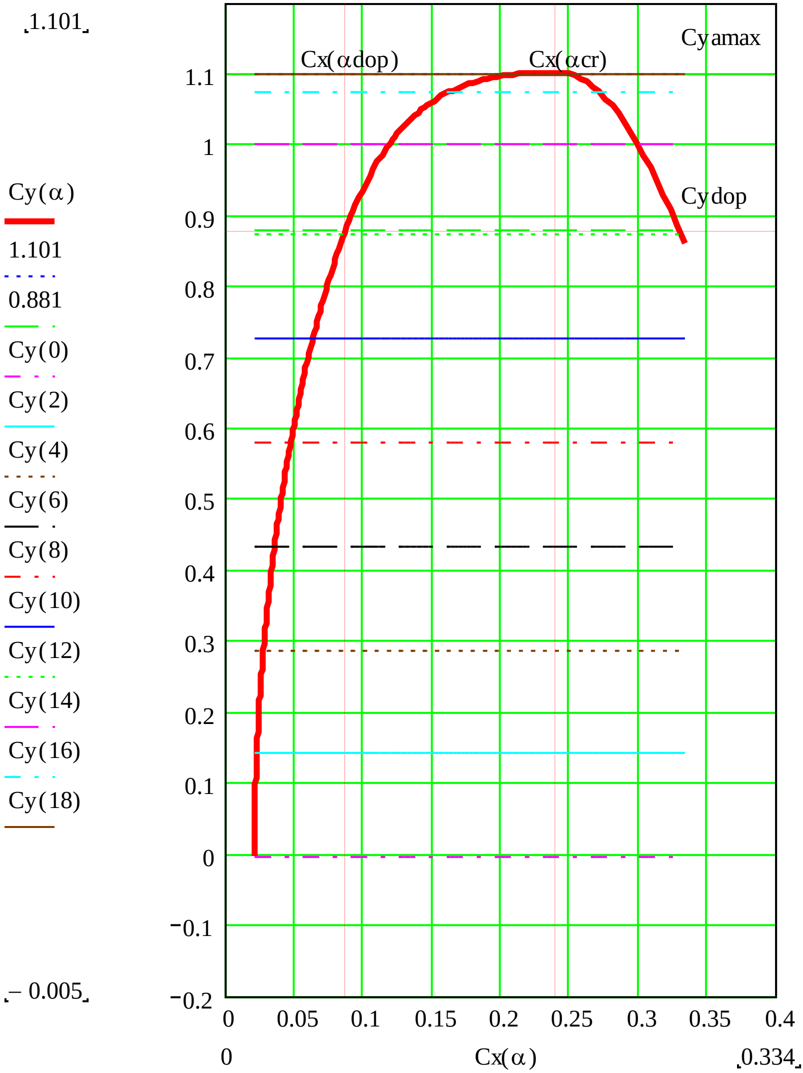

Fig. 4 – The **A/C Polar** $C_{y_a} = f\left(C_{x_a}\right)$ for the **Small Numbers** of M for the **Flight at the Altitude** of $H = 0$

IV. ASSESSMENT OF THE A/C FLIGHT MODES AT THE DESIGNED SPEED AND ALTITUDE OF THE FLIGHT (CRUISING MODE OF FLIGHT)

The principal theoretical provisions can be found out in references [1-7].

The methodic for the **A/C CRUISING MODES OF FLIGHT POLARS** (as well as other variants) calculations depends upon: the M **NUMBERS RANGE** (diapason): **SUBSONIC** or **TRANSONIC; GROUND INFLUENCE; A/C FLIGHT CONFIGURATIONS; PROPELLER SELECTION** etc.: [1, pp. 16-52, Sections 3-7, (12-15), Fig. 7-21, including Appendixes 1-4]. And those **TASKS** are suggested to be accomplished, but as an **EXTRA** and **INDEPENDENT STUDY** for CGW carrying out.

§ 13. For CP and TP

For plotting the **A/C POLARS** at other **FLIGHT MODES** and **A/C CONFIGURATIONS** in case of CP and TP execution it is recommended to use the provisions and data of the corresponding **SECTIONS** (Chapters) of the reference for CP and TP: [1, pp. 16-52, Sections 3-7, (12-15), Fig. 7-21, including Appendixes 1-4].

§ 14. For CGW

In case of CGW performance (execution, carrying out, completion), for academic purposes, the **SIMPLIFIED METHODS** are allowed. It is permitted and acceptable in view of the rather plane studying due to the academic hours constrains. It is basically plotting the **A/C POLAR DIAGRAMS** at other **FLIGHT MODES** and **A/C CONFIGURATIONS** in a proportional manner in order to demonstrate the principal (major, main) character of the **FLIGHT MODES** dependencies.

Modifications to **AERODYNAMIC COEFFICIENTS** therefore **AERODYNAMIC CHARACTERISTICS** can be made following [1, pp. 16-52, Sections 3-7, (12-15), Fig. 7-21, including Appendixes 1-4].

In case of CGW completion the **FIGURES** and **PARAMETERS** of [1, pp. 16-52, Sections 3-7, (12-15), Fig. 7-21, including Appendixes 1-4] can be used as an **ORIENTEER** of **VISUAL REFERENCE**. Especially that deals with [1, p. 34, Fig. 20, 21] and the **CALCULATED AND PLOTTED POLARS, CURVES, PARAMETERS, AND CHARACTERISTICS INTERPRETATIONS.**

§ 15. For computerized computational techniques

The **COMPUTERIZED COMPUTATIONAL TECHNIQUES** are so popular nowadays that we cannot evade them in our consideration.

In case of CGW performance (execution, carrying out, completion), for academic purposes, the simplified methods are allowed. In principle, it is recommended in some parts of the CGW carrying out for the approximation and conditional fragmentation objectives.

§ 16. For flight dynamics

The **SECOND PART**, **PART II**, of the **SS METHOD GUIDE** is envisaged for the **FLIGHT DYNAMICS PROBLEMS**. Therefore, the results of the presented CGW are considered as the initial data and conditions for the **SECOND PART** of the **ACADEMIC SUBJECT OF AERO-HYDRO-GAS-DYNAMICS AND FLIGHT DYNAMICS.**

§ 17. For further graduation papers and prospective research

The **TASKS** of the presented CGW are not restricted just to the set problems and demonstrated calculations. They are intended to urge and instigate the **STUDENTS'** further development.

For **FURTHER GRADUATION PAPERS** it is recommended for the **STUDENTS'** to formulate their **THEMES** (at least the field, area or sphere) the **STUDENTS** are the most interested in. The role of the **SUPERVISOR** at this is very important.

All mentioned above is even more important for **FURTHER PROSPECTIVE RESEARCH**, likewise **PH.D. STUDIES.** If the **CONTENDERS** are ready and distinguished seriously about the career of a **UNIVERSITY PROFESSOR** and/or **SCIENTIST**.

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