МІНІСТЕРСТВО ОСВІТИ ТА НАУКИ УКРАЇНИ
НАЦІОНАЛЬНИЙ АВІАЦІЙНИЙ УНІВЕРСИТЕТ
Кафедра конструкції літальних апаратів

ДОПУСТИТИ ДО ЗАХИСТУ
Завідувач кафедри
д-р техн. наук, проф.
_____________ С. Р. Ігнатович
«_____» ___________ 2020 р.

ДИПЛОМНА РОБОТА
(ПОЯСНЮВАЛЬНА ЗАПИСКА)
ЗДОБУВАЧА ОСВІТНЬОГО СТУПЕНЯ
"БАКАЛАВР"

Тема: «Аванпроект надлегкого навчально тренувального літака взлітною вагою до 500кг»

Виконав: __________ **В.О.Матвієнко**

Керівник: старший викладач ___________ **В.С.Краснопольський**

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Київ 2020

MINISRY OF EDUCATION AND SCIENCE OF UKRAINE
NATIONAL AVIATION UNIVERSITY
Department of Aircraft Design

AGREED
Head of the Department
Professor, Dr. of Sc.
_________S.R. Ignatovych
«___» _______ 2020 year.

DIPLOMA WORK

(EXPLANATORY NOTE)
OF ACADEMIC DEGREE
«BACHELOR»

Theme: «Preliminary design of light training aircraft with take-off mass up to 500 kilograms»

Performed by: __________ **V.O.Matviienko**

Supervisor: senior teacher __________ **V.S.Krasnopolskii**

Standard controller: __________ **S. V. Khizhnynak**

Kyiv 2020

NATIONAL AVIATION UNIVERSITY
Aerospace Faculty
Department of Aircraft Design
Academic Degree «Bachelor»
Speciality: 134 "Aviation and Rocket-Space Engineering"

APPROVED
Head of the Department
Professor, Dr. of Sc.
_________S.R. Ignatovych
«___» _______ 2020 year

TASK
for bachelor diploma work
MATVIIENKO VALENTYN

1. Theme: «Preliminary design of light training aircraft with take-off mass up to 500 kilograms »

Confirmed by Rector's order from 05.06.2020 year № 801/ст
2. Period of work execution: from 25.05.2020 year to 21.06.2020 year.
3. Work initial data: cruise speed V_{cr}=110 km/h, flight range L=400 km, take-off distance $L_{\text{take-off}}$ = 80 m, 1 passenger capacity or 90 kg cargo weight.
4. Explanation note argument (list of topics to be developed): choice and substantiations of the airplane scheme, choice of initial data; engine selection, center of gravity calculation, aircraft layout, aerodynamic calculation, pilot seat design.
5. List of the graphical materials: general view of the airplane (A1×1); layout of the airplane (A2×4), seat view(A1x1).
Graphical materials are performed in SolidWorks and are illustrated as drawings.

6. Calendar Plan

Task	Execution period	Signature
Task receiving, processing of statistical data		
Aircraft take-off mass determination		
Aircraft layout		

Aircraft centering determination		
Graphical design of the parts		
Completion of the explanation note		
Preliminary examination and defence of the diploma work		

7. Task issuance date: 25.05.2020.

Supervisor of diploma work __________ V.S.Krasnopolskii

Task for execution is given for __________ V.O.Matviienko

ABSTRACT

Explanatory note to the diploma work «Preliminary design of light training aircraft with take-off mass up to 500 kilograms» contains:

sheets , figures , tables , references ,drawings

Object of the design is development of ultralight training aircraft, which will be simple for manufacturing.

The aim of the diploma work is the preliminary design of the aircraft and its design characteristic estimation.

The method of design is analysis of the prototypes and selections of the most advanced technical decisions, analysis of center of gravity position.

The diploma work contains drawings of the light aircraft, calculations and drawings of the aircraft layout and pilot seat.

The results of the diploma work can be implemented to the academic education and also it can be used for the design bureaus.

PRELIMINARY DESIGN, ULTRALIGHT AIRCRAFT, PILOT SEAT

List of diploma work

Format	№	Designation	Name	Quantity	Notes
			General documents		
A4	1	NAU 20 10 M TP	Task of project	1	
			Graphic documentation		
		NAU 20 10 M 00 00 00 26 GV	Ultralight aircraft	2	
A1	2	Sheet 1	General view	1	
A2x4	3	Sheet 2	Aircraft layout	1	
A4	4	NAU 20 10 M 00 00 00 26 EN	Ultralight aircraft	66	
			Explanatory note		
			Special part documentation		
	5	NAU 20 10 M 00 00 00 26 AD	Pilot's seat design	1	
A1	6	Sheet 1	General view	1	

Aircraft design department	NAU 20 10M 00 00 00 81 EN		
Done by	Matviienko V.O.	List of diploma work	list / sheet 6 / sheets 66
Tutor	Krasnopolskii V.S.		
Instructor			AF 402 134
N. contr.	Khizhnyak S.V.		
Head. of d.	Ignatovich S.R.		

Content

Introduction

Ultralight aircraft, also known as the microlight aircraft, are lightweight fixed-wing aircraft with capacity of one or two people.

- Rising interest in air sports such as aerial acrobatics has increased the adoption of the ultralight aircraft across the world. This has boomed the ultralight and light aircraft market trend.
- Furthermore, ultralight aircraft has been widely used in military as well as civil & commercial operations due to their role in investigation procedures on the front lines.
- Increasing importance by market players on aircraft regulations and technological advances is estimated to enhance the ultralight and light aircraft market during the forecast period.

In general, the aim of this diploma paper is to develop an ultralight aircraft with a priority for the following characteristics:

- Flight safety;
- Reliability and ease of operation;
- Easy to manufacture;
- Good aerodynamic characteristics (especially high rate of climb);
- To perform takeoff and landing on unequipped unpaved runways;
- Operation in a wide temperature range.

The aircraft can also be converted to deliver small loads to a planned place, for ultra-low-volume pollination of fields, resettlement of biological plant protection agents, initial training of pilots for ultralight aviation, transport-related operations, aerial imaging observations, aerial photography and patrolling.

1. PROJECT PART. PRELIMINARY DESIGN OF LIGHT AIRCRAFT

1.1 Analysis of prototypes

1.1.1 Overview general performances

The selecting of the optimum design parameters of the aircraft is the multidimensional optimization task, aimed at forming a "look" promising aircraft. In its configuration mean the whole complex flight-technical, weight, geometrical, aerodynamic and economic characteristics. In forming the "Appearance of the plane" in the first stage is widely used statistics methods transfers, approximate aerodynamic and statistical dependence. The second stage uses a full aerodynamic calculation, aircraft specified formulas of aggregates weight calculations, experimental data.

For designed aircraft there were chosen the prototypes with take-off mass up to 500kg with capacity of 2 people. Such aircraft like Hi-MAX, Challenger II, Aeroprakt 20.

1) The Team Mini-Max Hi-MAX is a single-seat, high wing, strut-braced, single engine aircraft. It first flew in 1987.

Here are some of the standard features of the Hi-Max:

- High wing;
- Enclosed cabin;
- Truss landing gear standard; Optional spring steel landing gear;
- Easily removable wings;
- One 6 gallon wing tank (optional 2nd wing tank);
- Electronic elevator trim;
- Outstanding short field performance.

One of the main priority of this diploma paper is the simplicity of design and this aircraft is very simple for manufacturing, unfortunately it has a number of drawbacks, such as not high aerodynamic coefficient, it also required special storage conditions. General view of Hi-MAX prototype are presented in Fig. 1.1 and it’s performances are presented in table 1.1.

Fig. 1.1 The Team Mini-Max Hi-MAX

Table 1.1

Hi-MAX performances

PARAMETER	VALUE
Maximum speed, km/h	129
Stall speed, km/h	50
Cruise speed, km/h	113
Climb rate, m/s	4.6
Minimum rate of decline, m/s	1.8
Take-off Roll, m	31
Landing Roll, m	64
Wingspan, m	7.62
Length, m	4.88
Height, m	1.68
Wing area, m^2	10.45
Empty weight, kg	149
Gross weight, kg	254
Overload	+4.4/-1.8

2) The Challenger (Challenger II) ultralight is a high wing, tricycle gear kit aircraft with a frame structure built from 6061-T6 aluminum alloy tubing fastened with aircraft grade AN bolts and rivets and covered with either pre sewn Dacron envelopes or standard aircraft fabric. The engine is mounted in pusher configuration and turns the propeller through a reduction drive that uses a cogged tooth rubber belt. The aircraft has the ability to soar with its motor switched off. The Challenger design has been criticized by reviewers for its landing gear, which is a rigid cable-braced type and is subject to being bent during hard landings. A number of after-market suppliers have designed steel gear legs as replacements for the stock landing gear in an attempt to rectify this problem. The improved factory-designed Light Sport Special (LSS) model incorporates revised landing gear to address this deficiency.

A very light airframe, built by the factory using triangulated 6061-T6 aircraft-grade aluminum, is the secret to the Challenger's unusually low weight and high payload. A low weight benefits all aspects of performance, not just payload but also takeoff, climb, cruise and landing.

The Challenger uses tandem seating rather than side-by-side to achieve less drag for higher speed as well as to place the pilot ahead of the wing for magnificent helicopter style visibility in all directions - even in turns.

The Challenger employs a fully triangulated truss design which is not just lighter but is significantly stronger than alternatives. Since construction of such a design is beyond the capabilities of most homebuilders, every Challenger airframe is built in jigs at the factory. This ensures consistent quality and integrity of critical components.

The Challenger II has good aerodynamic characteristics, good visibility from the cabin, not demanding in maintenance, unfortunately the design is quite complex and require a lot of technical equipment for manufacturing, which makes the final cost of the product high. General view of Challenger II prototype are presented in Fig. 1.2 and it’s performances are presented in table 1.2.

Fig. 1.2 Challenger II

Table 1.2

Challenger II performances

PARAMETER	VALUE
Maximum speed (Rotax 503), km/h	156
Stall speed, km/h	45
Cruise speed, km/h	138
Maximum flight range, km	324
Climb rate, m/s	3.8
Take-off Roll, m	61 m
Landing Roll, m	70 m
Wingspan, m	9.6 m
Length, m	4.88 m
Height, m	1.83 m
Wing area, m^2	$16.47m^2$
Empty weight, kg	140 kg
Maximum take-off weight, kg	435 kg
Overload	+6/-3

3) Aeroprakt A20. Design of the A-20 began in 1990, with the first prototype making its maiden flight on 5 August 1991, with the first production aircraft flying on 15 August 1993.

The A-20 is constructed with a fiberglass forward fuselage and cockpit with aluminum wings and tail surfaces covered in doped aircraft fabric. The wing is fitted with half-span ailerons and flaps. The flaps are quite effective and lower the landing speed to 30 mph (48 km/h). Flaperons are available on some models. The conventional landing gear has steel sprung main gear legs.

The A-20 was originally designed for the 50 hp (37 kW) Rotax 503 two-stroke aircraft engine. The low drag airframe produces acceptable performance on this low power output. Optional engines include the 64 hp (48 kW) Rotax 582 and 100 hp (75 kW) Rotax 912ULS.

Aeroprakt A20 was selected as a prototype, because of it’s good aerodynamic characteristics and simplicity of design. General view of Aeroprakt A20 prototype are presented in Fig. 1.3 and it’s performances are presented in table 1.3.

Table 1.3

Aeroprakt A20 performances

PARAMETER	VALUE
Maximum speed, km/h	140
Stall speed, km/h	46
Maximum flight range (fuel 38L, no wind), km/h	400 km
Maximum flight time, hours	4.5 h
Climb rate, m/s	3 m/s
Minimum rate of decline, m/s	1.8 m/s
Take off distance, m	80 m
Wingspan, m	11.4 m
Length, m	6.7 m
Height, m	1.8 m
Wing area, m^2	15.7 m^2
Maximum take-off weight, kg	450 kg
Empty weight, kg	218 kg
Overload	+4/-2

Fig. 1.3 Aeroprakt A20

1.1.2 Brief description of the main parts of the aircraft

The plane consists of a fuselage to which the wing, tail strut, landing gear and power plant are attached. The fuselage contains a cockpit made of composite materials with buoyancy and low electrical conductivity. Flap control in three fixed positions: 0 degrees, 20 degrees and 30 degrees.

A wing with a high aspect ratio, which is based on R-III airfoil with good lift coefficient characteristics. Tail unit has a T-type construction, due to low location of tail boom.

1.1.2.1 Fuselage

The fuselage is designed to accommodate crew, cargo, equipment and fuel. From the point of view of structural mechanics, it is a structural element that take loads from all parts of the aircraft. A high wing scheme of aircraft was chosen. The advantages of the upper location of the wing is:

- reduction of aerodynamic drag from interference of the “wing - fuselage” combination;
- improved ground visibility from the cockpit;
- providing the ability to install engines on the wing and the use of a tray wing scheme.

The disadvantages of the upper location of the wing include a decrease in the efficiency of vertical stabilizer at high angles of attack, when the stabilizer enters at turbulent stream from the wing. To provide a good view from the cockpit and to increase efficiency of propeller, engine is located in the fuselage rear part. A high-tech monocoque fuselage are made from sandwich panels (three-layers panels, fiberglass – filler – fiberglass), which allows to provide high stability and strength with minimum weight. Also, sandwich panels can significantly increase sound insulation in the cabin. The entire fuselage has a minimum double curvature parts in order to minimize the complexity of production, with a minimum of technological equipment. Cockpit canopy made of 3 sheets of plexiglass, without double curvature, connected by aluminum tubes, which eliminates a lot of complex technological operations and increases manufacturing speed.

To control an ultralight aircraft, the main and auxiliary mechanical control systems were used. The main control systems include elevator control, ruder control and ailerons control. Auxiliary mechanical control systems include flap control, engine control, wheel brake control and rescue parachute drive control. Since flaperons are installed on the plane, the flap control lever and aileron control are adds in flaperons mixer, which is installed in the rear fuselage part.

1.1.2.2 Wing

The wing of the aircraft based on R-III-15.5 airfoil, consists of a center section, two detachable consoles, and a system of struts. The wing consoles are connected to the fuselage by torqueless nodes and a strut system. The wing is equipped with slotted flaperons. In addition to controlling the roll or bank of an aircraft, as do conventional ailerons, both flaperons can be lowered together to function similarly to a set of flaps. Pilot still has the standard separate controls for ailerons and flaps, but the flap control also varies the flaperons range of movement. A mechanical device called a "mixer" is used to combine the pilot's input into the flaperons.

The console frame consists of a longitudinal and transverse elements. The longitudinal set includes the front and rear spars, perceiving bending, torques and cutting forces. The transverse set includes ribs, on each console.

The wing struts are streamlined rods in cross section made of pipes. Composite wing, made from three-layer panels, which prevents buckling and significantly reduces the number of ribs and weight of the wing.

1.1.2.3 Tail unit

The are made according to the T-shaped scheme and consists of a keel, rudder, stabilizer and elevator. Airfoil for horizontal and vertical stabilizer is NACA0009 (modified).

The fin is attached to the tail boom with brackets. Elevator control cables redirected by rollers and pass inside the horizontal stabilizer.

Ruder control are also use cables. The rudder is controlled by pedals. The pedals are mounted on an axis that is mounted on the fuselage. To adjust the angle of deviation of the pedals and, accordingly, the ruder on the beam opposite the pedals mounted stops with adjusting screws. The pedals are connected to the cables through the thunder to adjust the relative position of the pedals and rudder and to adjust the tension of the cables. With double control of the ruder, pedals of the second pilot are installed, similar in design to the pedals of the first. The pedals of the first and second pilot are interconnected by an adjustable thrust.
Elevator control system is also use cable to transmit movement from the control leaver to elevator.

1.1.2.4 Landing gear

The chassis is a support system that provides the required position of the aircraft during parking and its movement during takeoff, landing and taxiing on the aerodrome.

The main landing gear is located in front of the aircraft center of gravity, an additional - in the tail. Since the additional strut at the tail of the aircraft is far from the center of gravity, it has a small weight, the landing gear may be smaller than if it were located in the bow, which improves the aerodynamic characteristics of the aircraft. Since a small additional strut is located behind the aircraft, the risk of its breakdown when traveling on unpaved and unprepared airfields is significantly reduced. Failure of an additional strut in the event of a hard landing leads to less serious consequences than a breakdown of the nose landing gear. Otherwise such scheme complicates the controllability of the aircraft on the ground.

Rear landing gear controlled by pedals. Main LG have a hydraulic breaking system (hydraulic cylinder are installed on control leaver). Spring (рессора) of the main landing gear, produced from composite material (fiberglass and carbon), absorbs shock waves during landing.

1.2 Aircraft layout and center of gravity calculation

1.2.1 Geometry and mass calculations for the aircraft principles structural units

Aircraft layout calculation is based on the selection of the purpose of the designed aircraft, its main dimensions, and operational requirements.

Layout consists of geometry calculation of large constructional modules as wing, fuselage, tail unit, and landing gear. Besides all above mentioned, this analytical part includes choice of power plant and interior scheme. The interior scheme estimation includes dimensional calculation based on aircraft capacity requirements.

This layout was implemented in line with both modern standards and well-established calculation methods.

1.2.1.1 Preliminary mass calculation

Let maximum take-off mass of aircraft m_{max} = 450kg, The aircraft mass can be written as the sum of its separate units:

$$m_0 = m_{wing} + m_f + m_{tail} + m_{pp} + m_{lg} + m_{cs} + m_{eq} + m_{fuel} + m_{em}$$

$$cm_{wing} + cm_f + cm_{tail} + cm_{pp} + cm_{lg} + cm_{cs} + cm_{eq} + cm_{fuel} + cm_{em} = 1$$

For first approach we will use statistical data about the relative mass of main aircraft units, table 1.4

Table 1.4

Relative mass of some aircraft units

Unit	Relative mass
Wing cm_{wing}	0.16
Fuselage cm_f	0.14
Tail unit cm_{tail}	0.04
Power plant cm_{pp}	0.11
Landing gear cm_{lg}	0.06
Control system cm_{cs}	0.06
Fuel mass cm_{fuel}	0.075
Payload mass cm_{pm}	0.43

Let time of flight t_{flight} = 2.5 hours, than relative mass of fuel can be find as:

$$cm_{fuel} = 0.3 \cdot t_{flight} \cdot cm_{pp} = 0.3 \cdot 2.5 \cdot 0.11 = 0.075$$

Relative payload mass is:

$$cm_{pm} = 1 - 0.16 - 0.14 - 0.04 - 0.11 - 0.06 - 0.18 - 0.075 = 0.43$$

Payload mass in first approach equal 193.5 kg.

1.2.1.2 Engine selection and calculation of power plant mass

Rotax 503 (or RMZ 500) has established itself as high reliable and simple engine. It has power 37kW at 6800rpm or 36 kW at 6500rpm, engine performance are shown in Fig. 1.4. Fuel consumption performances are shown in table 1.5. Engine can be find as a sum of its main components table 1.6.

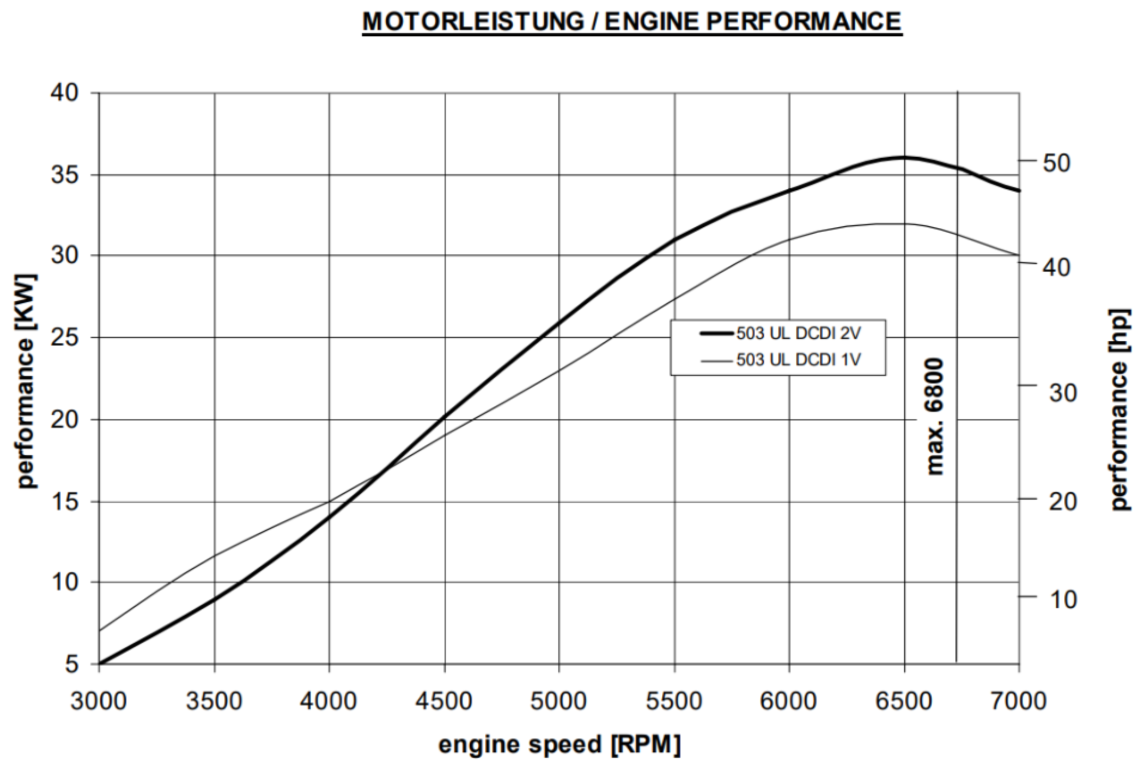


Fig. 1.4 Rotax 503 engine performance

Table 1.5

Rotax 503 fuel consumtion

Period	Fuel consumption
At take-off performance	25 l/h
At 75% continuous performance	15 l/h
Specific fuel consumption	550 g/kWh

Table 1.6

Rotax 503 components mass

Part	Weight (kg)	Info
Rotax 503	31.4	(**without**: exhaust system, radiator)
Exhaust sys	5.1	
Carburator	0.9	
Gear box (type C)	4.5	2.6:1 (for selected propeller)
Electric starter	3.5	Not used at basic conf.
Total	41.9 (38.1 at light conf.)	

Let engine mount parts mass equal 3kg additional elements mass equal 5kg, than total mass equal 45.1 kg or $cm_{pp} = 0.1$.

1.2.1.3 Propeller selection at first approach

Diameter of the propeller is depending on its rotation speed and engine power. At the stage of preliminary design it can approximately be determined from the diagram presented in Fig. 1.5.

$$n = engineRPM \,/\, gearBoxRatio = 6500/2.6 = 2500 \text{ rpm}$$

$$N = \text{power} \cdot gearBoxRatio \cdot gearBoxEfficiency = 36\text{kW} \cdot 2 \cdot 0.85 = 61.2 \text{ kW}$$

Let propeller diameter $D_p = 1.6$ m.

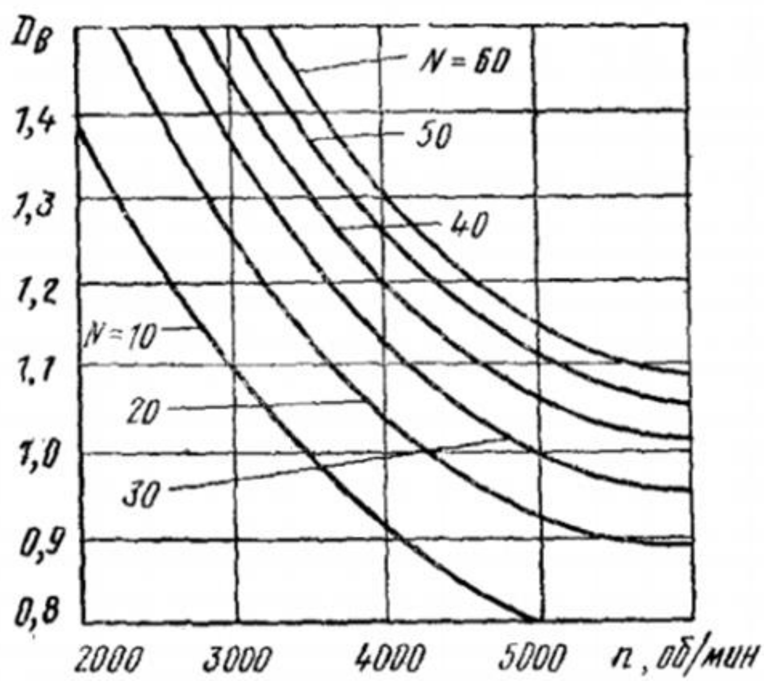


Fig. 1.5 Dependence of the propeller diameter on its rotation speed and engine power

1.2.1.4 Calculation of wing geometry and its aerodynamic characteristics

The wing is the main part of the aircraft, and the parameters of the entire aircraft are highly dependent on the geometric parameters of the wing. The aerodynamic characteristic of the wing are highly depend on the selected airfoil.

The most widely used airfoils for ultralight aircraft are the well-proven profiles such as R-II, R-III and others with a relative thickness 12 ... 20%.

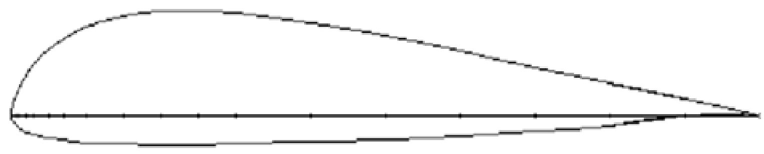
R-III with relative thickness 15.5% (Fig. 1.6) will be a good choice for such type of aircraft, because of its simplicity and good aerodynamic characteristics, table 1.7.

Take off angle of attack equal 15° (from prototype), then C_y for take-off mode can be obtain from wing polar:

$$C_y(15°) = 1.42 \text{ (from } AD \text{ analysis of wing without flaps)}$$

Full wing area is:

$$Y = C_y \cdot \rho \cdot V_{take-off}^2 S/2 \rightarrow S = 2 \cdot G_0/(C_{y\,take-off} \cdot \rho \cdot V_{take-off}^2)$$



Аэродинамические коэффициенты профиля **Р-III (15,5%)**

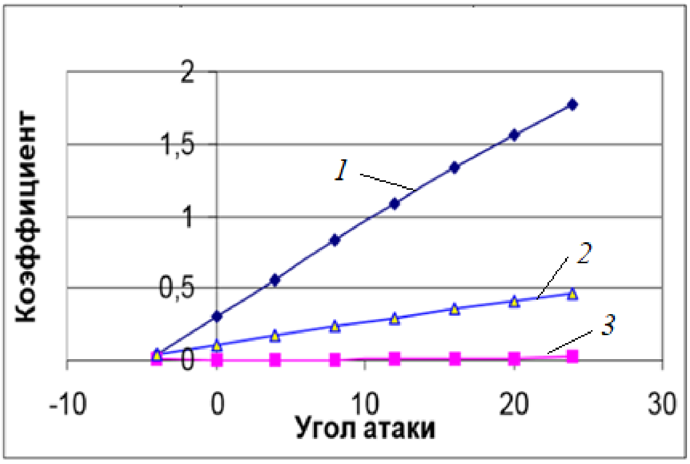


Fig. 1.6 R-III airfoil and its aerodynamic characteristics: *1* – C_y, *2* – C_x, *3* – C_m.

Table 1.6

R-III airfoil aerodynamic characteristics

Аэродинамические характеристики			
α°	C_y	C_x	C_m
-4	0,04	0,0142	0,045
0	0,30	0,0018	0,109
4	0,56	0,0032	0,172
8	0,84	0,0059	0,240
12	1,08	0,0090	0,298
16	1,34	0,0136	0,360
20	1,56	0,0190	0,417
24	1,78	0,0250	0,467

Or:

$$S_{wfull} = \frac{m_0 \cdot g}{P_0} = \frac{450 \cdot 9.8}{280} = 15.75 \text{ m}$$

where m_0 – take-off weight;

g – gravity acceleration;

P_0 – specific wing load.

Wing aspect ratio:

$$\lambda = l^2 / S = l / b_a$$

Wing length:

$$l = \sqrt{S_{w\,full} \cdot \lambda} = \sqrt{15.75 \cdot 8.14} = 11.4 \text{ m}$$

For determination of aerodynamic characteristics of the wing, aerodynamic analysis may be performed, as example, XFLR5 software can be used. XFLR5 is an analysis tool for airfoils, wings and planes operating at low Reynolds Numbers. It also allow us to get a diagram of lift force distribution on wing surface, from which it is possible to calculate the bending moment on the wing with higher accuracy.

It includes:

- XFoil's Direct and Inverse analysis capabilities;
- Wing design and analysis capabilities based on the Lifiting Line Theory, on the Vortex Lattice Method, and on a 3D Panel Method.

Data for analysis (for XFLR5):

- Airfoil – RIII-15.5
- Density of air ρ = 1.225
- Speed V = 33 m/s
- Chord b = 1.4m
- Span l = 11.4

The results (polar) of the wing simulation is shown in Fig. 1.7.

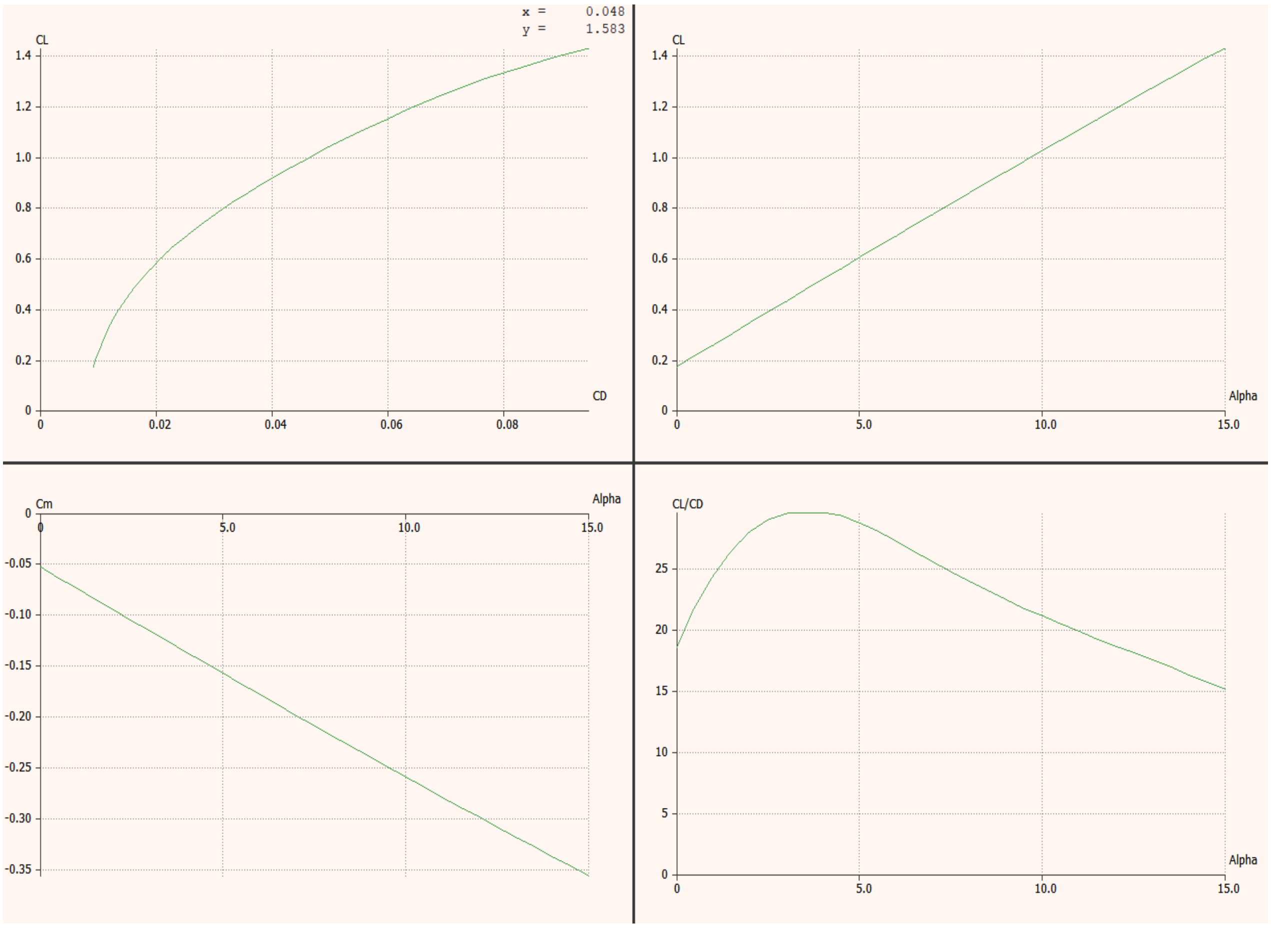


Fig. 1.7 Wing aerodynamic characteristics

Knowing the wing area and lift coefficient, we can find take-off speed without using flaps:

$$V_{take-off} = \sqrt{\frac{G \cdot 2}{C_y \cdot S \ \cdot \rho}} = \sqrt{\frac{450 \cdot 9.8 \ \cdot 2}{1.42 \cdot 15.96 \ \cdot 1.225}} = 17.82 \frac{m}{s} = 64 \text{ km/h}$$

To determine take-off speed when flaps are use, wing tunnel tests or, against, simulation may be used. For this diploma paper ANSYS CFX is used.

For simulation was used K-Omega turbulence model, volumetric polyhedral high quality mesh of 1m wing section. The data for analysis is the same, except speed, for ANSYS simulation speed V = 20 m/s (such value is closer to take-off speed).

The flaperons chord length was obtain analytically. The geometry of wing cross section, that are used for simulation is shown in Fig. 1.8.

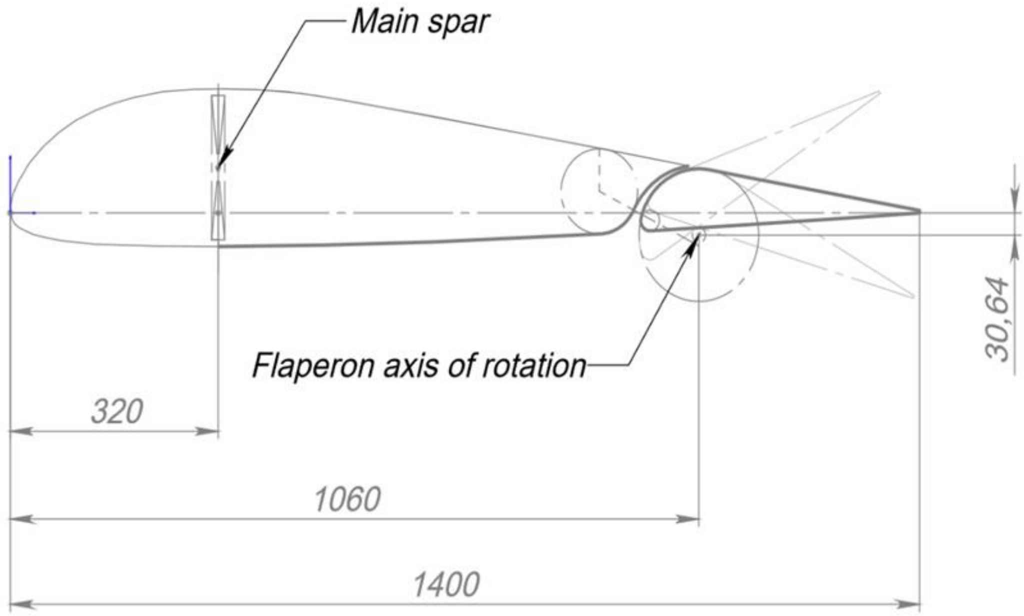


Fig. 1.8 Wing cross-section geometry

Simulation 1.0:
- Angle of attack $\alpha = 15°$;
- No flaps;
- Speed $V = 20$m/s;
- Density $\rho = 1.225$;
- Reynolds number $V \cdot b \cdot 70000 = 20 \cdot 1.4 \cdot 70000 = 1960000$

Results are presented in appendix A

Simulation 1.1:
- Angle of attack $\alpha = 15°$;
- Angle of flap in relation to the wing $\alpha_{flap} = 0°$;
- Speed $V = 20$m/s;
- Density $\rho = 1.225$;
- Reynolds number $V \cdot b \cdot 70000 = 20 \cdot 1.4 \cdot 70000 = 1960000$

Results are presented in appendix B

Simulation 1.2:
- Angle of attack $\alpha = 15°$;
- Angle of flap in relation to the wing $\alpha_{flap} = 20°$;
- Speed $V = 20$m/s;
- Density $\rho = 1.225$;
- Reynolds number $V \cdot b \cdot 70000 = 20 \cdot 1.4 \cdot 70000 = 1960000$

Results are presented in appendix C

Simulation results should be corrected in accordance with experimental data and wing chord length (in CFX analysis, chord length equal 1.4m, that is why C_Y has greater value):

$$C_y^{sim}(\alpha = 15,\ \alpha_{\text{flap}} = 0)\ (b = 1.4) = 1.0844\text{e-01} + 2.1858\text{e+00} = 2.294$$

$$C_y^{sim}(\alpha = 15,\ \alpha_{\text{flap}} = 0)\ (b = 1) = C_y^{sim}(b = 1.4)\ /1.4 = 1.638$$

$$C_y^{sim}(\alpha = 15,\ \alpha_{\text{flap}} = 20)\ (b = 1.4) = 2.9019\text{e-01} + 3.3222\text{e+00} = 3.61$$

$$C_y^{sim}(\alpha = 15,\ \alpha_{\text{flap}} = 20)\ (b = 1) = C_y^{sim}(b = 1.4)\ /1.4 = 2.579$$

Correction coefficient $k_{y\ correction} = C_{y\ \text{actual}}\ /\ C_y^{sim}(b = 1) = 1.42/1.638 = 0.867$, than:

$$C_y\ (\alpha = 15,\ \alpha_{\text{flap}} = 0) = 0.867 \cdot 1.638 = 1.42$$

$$C_y\ (\alpha = 15,\ \alpha_{\text{flap}} = 20) = 0.867 \cdot 2.579 = 2.24$$

Take-off speed, using flaps:

$$V_{take-off} = \sqrt{\frac{G \cdot 2}{C_y \cdot S \cdot \rho}} = \sqrt{\frac{450 \cdot 9.8 \cdot 2}{2{,}24 \cdot 15.96 \cdot 1.225}} = 14.19\frac{\text{m}}{\text{s}} = 51\ \text{km/h}$$

The use of flaps during take-off reduced required speed by 20%.

It is more convenient to evaluate the effectiveness of ailerons using a value of m_x called the ailerons moment coefficient:

$$m_x = \frac{S_{ail.e} \cdot a_{ail}}{Sl} \cdot \frac{l_{ail}}{l} \times \sqrt{\frac{b_{ail}}{b_{wing}}} = \frac{7.56 \cdot 3}{15.96 \cdot 11.4} \cdot \frac{5.4}{11.4} \cdot \sqrt{\frac{0.4}{1.4}} = 0.0316$$

where $S_{ail.\ e}$ – effective aileron area (wing area);

a_{ail} – the distance between the centers of the aileron effective areas;

l_{ail} – aileron span;

b_{ail} – mean aerodynamic chord of aileron.

The optimal values of m_x can be considered:

- 0.012 ... 0.018 - for non-maneuverable ultralight aircrafts
- 0.018 ... 0.024 - for maneuverable ultralight aircrafts

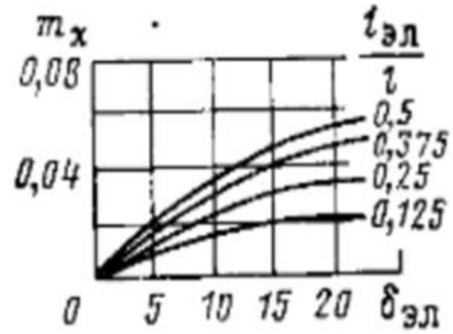


Fig. 1.9 The influence of the aileron deflection angles and the relative aileron span on the transverse moment coefficient

It is advisable to increase the indicated values by 0.003 ... 0.005 for light aircrafts with upper wing position.

Such high coefficient was obtained due to using of flaperons control scheme, so we need to reduce aileron deflection angles.

1.2.1.5 Calculation of tail unit geometry.

Area of horizontal stabilizer:

$$S_{HS} = \frac{A_{HS} \cdot b_{wing} \cdot S_{wing}}{L_{hs}} = \frac{0.49 \cdot 1.4 \cdot 15.75}{3.6} = 3 \text{ m}^2$$

where L_{HS} – distance between CG and 25% of MAC of horizontal stabilizer (determined analytically).

A_{HS} – static moment coefficient of horizontal stabilizer, A_{HS} = 0.45-0.55 for most light aircraft of normal scheme.

Area of vertical stabilizer:

$$S_{VS} = \frac{B_{VS} \cdot l_{wing} \cdot S_{wing}}{L_{vs}} = \frac{0.028 \cdot 11.4 \cdot 15.75}{3.4} = 1.48 \text{ m}^2$$

where L_{VS} – distance between CG and 25% of MAC of vertical stabilizer. (determined analytically) .

B_{VS} – static moment coefficient of VS.

Horizontal stabilizer aspect ratio λ_{HS} = 2 … 5 (for light aircraft of normal scheme). Let λ_{HS} = 4.5.

Horizontal stabilizer length:

$$L_{HS} = \sqrt{S_{HS} \cdot \lambda} = \sqrt{3 \cdot 4.5} = 3.67 \text{ m}$$

Horizontal stabilizer chord:

$$B_{HS} = S/L_{HS} = 0.82 \text{ m}$$

Vertical stabilizer chord:

$$B_{VS} = 0.88 \text{ m}$$

For the vertical and horizontal stabilizer NACA0009 airfoil will be used (Fig. 1.10) its aerodynamic characteristics (for *Re* = 1,000,000) is shown in Fig. 1.11 and fig. 1.12.

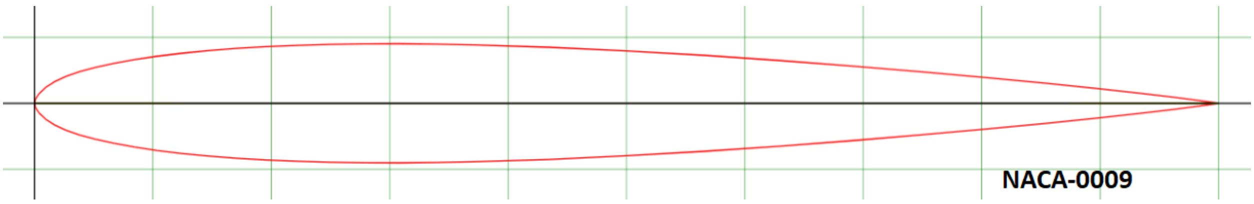


Fig. 1.10 NACA0009 airfoil

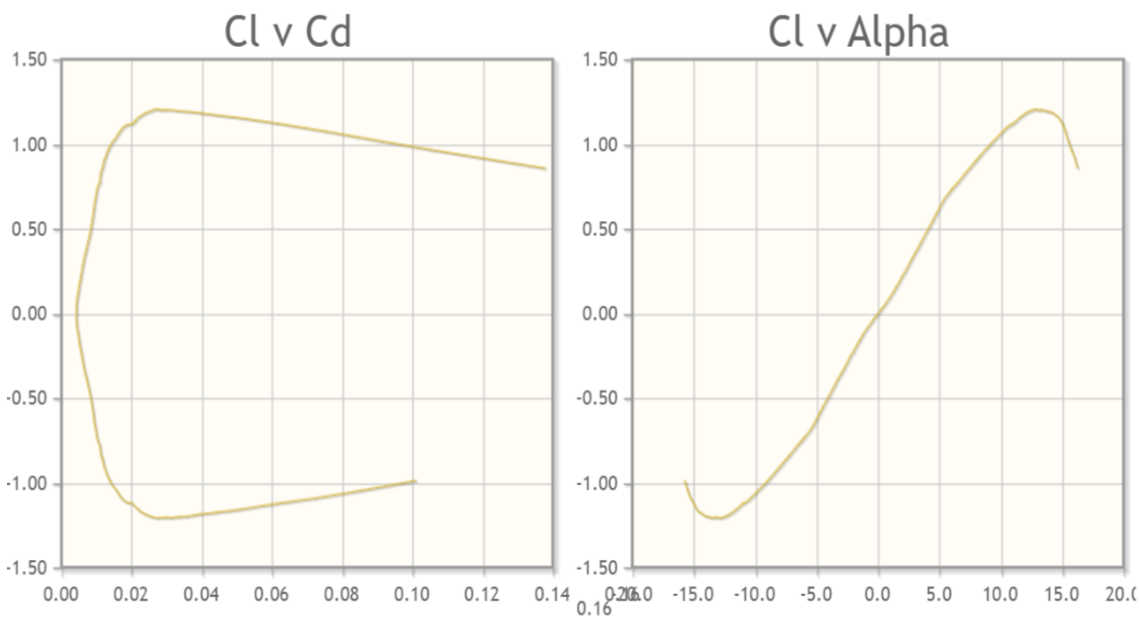


Fig. 1.11 Lift and drag coefficients diagram of NACA009 airfoil.

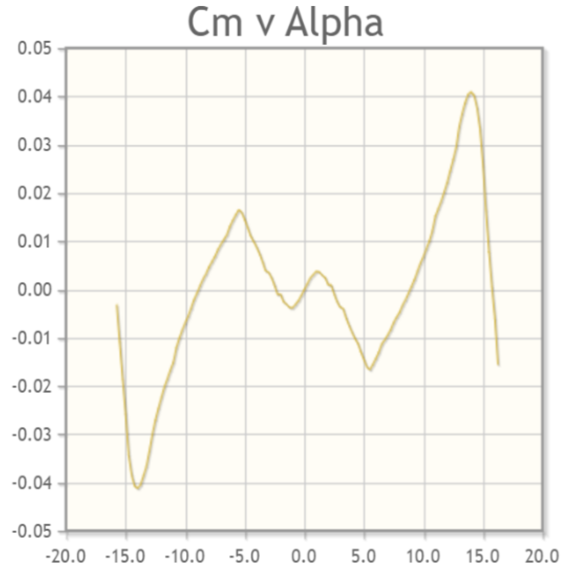


Fig. 1.12 Moment coefficients diagram of NACA009 airfoil.

For rectangular horizontal stabilizer (elevator covers the entire length of horizontal stabilizer), elevator efficiency coefficient k_{elevator} is equal:

$$k_{\text{elevator}} = \sqrt{\frac{S_{elevator}}{S_{HS}}} = \sqrt{\frac{b_{elevator}}{b_{HS}}}$$

where $b_{elevator} \leq 0.3 \ldots 0.4$ for most subsonic light aircrafts.

Such value of b_{elevator} is not always optimal due to very small hinge moment transmitted from the elevator to the aircraft control leaver and in case of high horizontal stabilizer elongation, so we will increase it to 0.45. Also it will be appropriate to include spring or rubber loaders in the control system.

1.2.1.6 Wing mass calculation

Mass of the wing in second approach (without strut) can be calculated as average result of this two formulas:

$$m_{wing\ 1} = 0.002 \cdot k_m \cdot m_0 \cdot n_p[0.6(l/2)^2+1]+3S =$$
$$0.002 \cdot 0.7 \cdot 450 \cdot 4[0.6(11.4/2)^2+1]+3 \cdot 15.7 = 98.74 \text{ kg}$$

$$m_{wing\ 2} = 0.0001 \cdot k_m \cdot m_0 \cdot n_p[\lambda(\eta+3)\sqrt{S/\eta} \cdot \sqrt{c}] =$$
$$0.0001 \cdot 0.7 \cdot 450 \cdot 4[8.28(1+3)\sqrt{15.7/1} \cdot \sqrt{15.5}] = 65 \text{ kg}$$

$$m_{wing} = (98.74+65)/2 = 81.87 \text{ kg}$$

$$cm_{wing} = m_{wing} / m_0 = 0.182$$

k_m – material coefficient (0.8 – for D16T, 0.7 – for composites, carbon, fiberglass)

where m_0 – take-off mass in first approach

n_p – loading (4 g)

l – wing span

S – wing area

η – wing narrowing

c – relative thickness of airfoil

1.2.1.7 Fuselage mass calculation

Approximate mass of the fuselage in second approach can be calculated as:

$$m_f = 2.5 \cdot S_{fuselage} + 4 \cdot S_c + 20 = 2.5 \cdot 7.6 + 4 \cdot 2 + 20 = 47 \text{ kg}$$

$$cm_f = 47/450 = 0.104$$

Correction 1:

The fuselage will be build using sandwich panels Fig. 1.13.

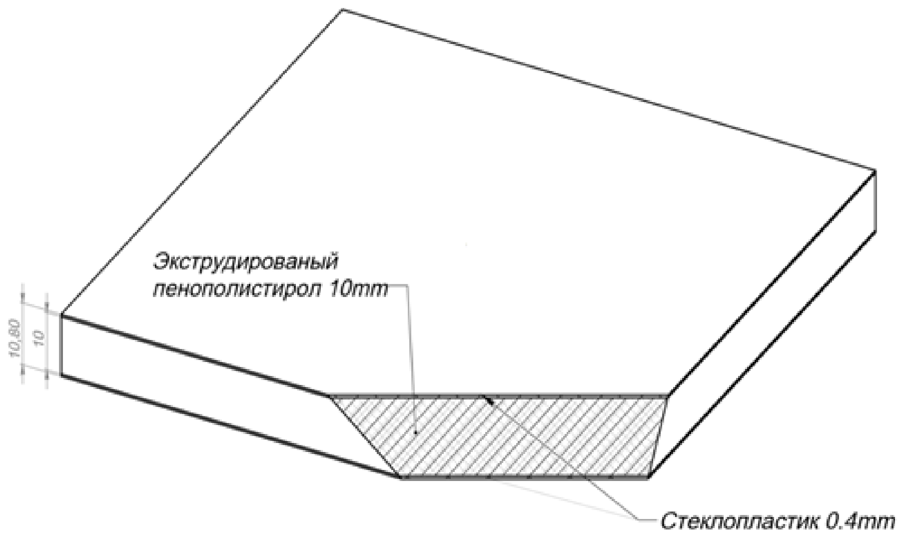


Fig. 1.13 Sandwich panel

Mass of such panel = 1.9 kg/m2 (Let density equal 176 kg/m^3, such density will be used in SolidWorks mass estimation).

Cockpit canopy is produced from 4 mm plexiglass (density 1180 kg/m^3) that are connected and strengthened by duralumin pipes with total length 3m and additional structural elements. Then mass of the canopy is:

$$m_{canopy} = S_{canopy} \cdot t \cdot \rho_{plexi} + l_{tube} \cdot r + m_{additional} = 2 \cdot 0.004 \cdot 1180 + 3 \cdot 0.3 + 2 = 12.34 \text{ kg}$$

where t – thickness of plexiglass;
r – 1m tube mass.

Mass of fuselage skin (Monocoque structure):

$$m_{skin} = S_{skin} \cdot 1.9 + m_{rp} = 16.44 \text{ kg}$$

where m_{rp} – mass of reinforcement pieces (~2 kg).

1.2.1.8 Landing gear design

At the primary stage of design, when the airplane center-of-gravity position is defined and there is no drawing of airplane general view, only the part of landing gear parameters may be determined.

From the airfoil data, let CG located at 0.28 of MAC. Taking into account length from 0.25 of MAC of wing to the 0.25 MAC of horizontal stabilizer (3670mm) plus estimated length to the mount point of the TU landing gear, length from CG to rear landing gear(e) ≈ 3970mm. Let force applied to the main LG = 92.7% than distance to main LG:

$$a = \frac{7.3 \cdot 3970}{92.3} = 314 \text{ mm}$$

Wheel base B = 4284mm.

Wheel track is:

$$T = B \cdot 0.37 = 1585 \text{ mm}$$

We install breaks on the main wheel, without brakes on rear wheel. The load on the nose wheel can be determined as:

$$P_{NLG} = \frac{9.81 \cdot e \cdot K_g \cdot m_0}{B \cdot n \cdot z} = \frac{9.81 \cdot 3.97 \cdot 2 \cdot 450}{4.284 \cdot 2 \cdot 1} = 4091 \text{ N}$$

Rear wheel load is equal:

$$P_{RLG} = \frac{9.81 \cdot a \cdot K_g \cdot m_0}{B \cdot n \cdot z} = \frac{9.81 \cdot 0.314 \cdot 2 \cdot 450}{4.284 \cdot 1 \cdot 1} = 647 \text{ N}$$

where K_g = 1.5...2.0 – dynamics coefficient.

For preliminary calculation spring type grove landing gear Zenair CH701 (Fig. 14) from Aircraft Spruce catalogue can be selected, it has next features:

- Grove PN: 1214-3;
- Aircraft Weight: 1060 lbs or 480 kg;
- Weight: 18.1 lbs or 8.2kg;
- Material: 7075-T6 Aluminum;
- Axle Bolt Hole Pattern: Various;
- Type PN 06-00058.

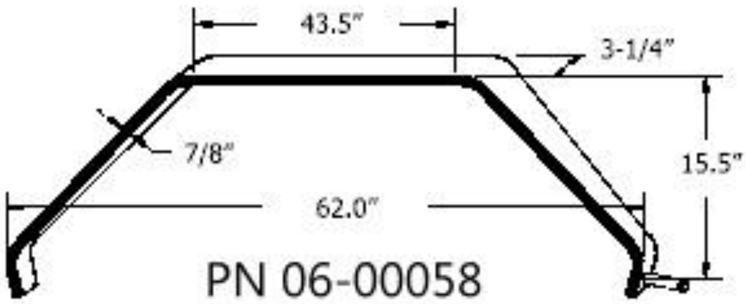


Fig. 1.14 Landing gear geometry

Main wheel GOODYEAR 856T61-1.

Rear wheel VAN'S TAILWHEEL 5-1/2" Diameter X 1-1/2" Tire X 2" Hub Thickness.

1.3 Aerodynamic center calculation

Accurate determination of the aircraft aerodynamic center position is a rather complex task, and in many cases, in addition to theoretical calculations, it requires special tests in aerodynamic tunnel. It can be significantly simplified if we assume that only the wing and the horizontal stabilizer influence on the position of aerodynamic center.

1.3.1 Wing aerodynamic center calculation

The Aerodynamic center point F_K (Fig 1.15) is the point at which the pitching moment coefficient for the airfoil does not vary with lift coefficient, making analysis simpler.

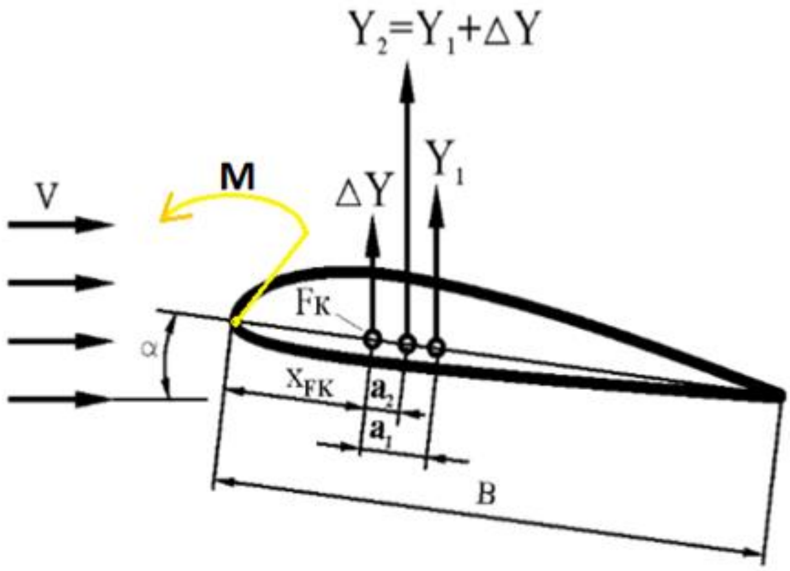


Fig 1.15 Wing aerodynamic center

At point F_K, the moment remains constant:

$$Y_1 \cdot a_1 = Y_2 \cdot a_2$$

To determine distance to F_K we need just two centers of pressure points with different angles of attack. Let $\alpha_1 = 0°(C_y = 0.3)$ and $\alpha_2 = 8°(C_y = 0.84)$. The distance to the center of pressure can be determined using the coefficients C_m obtained from the experimental data:

$$X_{FK} + a = M / R$$

$$M = \frac{\rho V^2}{2} C_y S$$

$$R = \frac{\rho V^2}{2} C_r S$$

Within small angles of attack (0 … 15 °), i.e., those angles with which you have to deal in flight, the value of C_r is not very much differ from C_y.

After division by *(ρ × B × V2 / 2)* we obtain:

$$X_{FK} + a = B \cdot C_m / C_y$$

where B – chord equal 1.4 m.

Results of distances to center of pressure calculation is shown in table 1.7.

Table 1.7

Center of pressure location

Angle of attack α	C_y	C_m	X_{FK} + a (m)
0°	0.3	0.109	$X_{FK} + a_1 = 0.509$
8°	0.84	0.24	$X_{FK} + a_2 = 0.4$

In mechanics, force Y_2 can be replaced by a pair of forces equivalent to it:

$$Y_2=Y_1 + \Delta Y \rightarrow \Delta Y = Y_2 - Y_1 \rightarrow \Delta C_Y = C_{Y2} - C_{Y1} = 0.54$$

Now we can determine distance from Y_2 to F_K:

$$\frac{\Delta Y}{Y2} = \frac{a_2}{a_1} \rightarrow a_2 = \frac{\Delta Y\ a_1}{Y2} = \frac{0.54 \cdot 0.109}{0.84} = 0.07 \text{ m}$$

Than distance from the wing nose to aerodynamic center F_K equal:

$$X_{FK} = 0.4 - 0.07 = 0.333\text{m} = 23.6\% \text{ of MAC}$$

To check the result, we will add one more point α_3 = 12°. Results is shown in table 1.8

Table 1.8

Aerodynamic center of wing

Angle of attack α	C_y	C_m	X_{FK} + a	Momentum at F_K / (ρ • B • V2 / 2) (For checking)
0°	0.3	0.109	$X_{FK} + a_1 = 0.509$	$a1 \cdot C_y = 0.0528$
8°	0.84	0.24	$X_{FK} + a_2 = 0.4$	$a2 \cdot C_y = 0.0563$
12°	1.08	0.298	$X_{FK} + a_2 = 0.3863$	$a3 \cdot C_y = 0.0576$

1.3.2 Horizontal stabilizer aerodynamic center calculation

For our aircraft we will use NACA-0009 airfoil, as we know, for symmetrical airfoils, the center of pressure application does not change with a change of α, so the HS aerodynamic center coincides with the center of pressure. Therefore, the aerodynamic center of the horizontal stabilizer can be determined as:

$$X_{FS} = B \cdot C_m / C_y$$

where B – chord equal 0.82 m.

Results of distances to aerodynamic center calculation is shown in table 1.9.

Table 1.8

Horizontal stabilizer aerodynamic center location

Angle of attack α	C_y	C_m	X_{FK} (m)
4°	0.3	0.072	0.197
6°	0.45	0.108	0.197
8°	0.6	0.150	0.205
12°	0.9	0.216	0.197
14°	1.05	0.252	0.197

As we can see from the calculation, the distance to the center of pressure will really remain the same with different angles of attack.

1.3.3 Aircraft aerodynamic center calculation

When the angle of attack changes, increments of the lifting force occur both on the wing and on the horizontal stabilizer (Fig 1.16). They are applied, respectively, in the aerodynamic centers of the wing F_w and horizontal stabilizer F_{HS}. In accordance with the definition of the aircraft aerodynamic center, the moment relative to point F should not change. So the force moment ΔY_w on the distance ΔX_F should be balanced by the moment ΔY_{HS} on the distance L_{HS}-ΔX_F:

$$\Delta Y_w \cdot \Delta X_F = \Delta Y_{HS}(L_{HS} - \Delta X_F) \rightarrow$$

$$\Delta C_Y \frac{\rho V^2}{2} S \cdot \Delta X_F = \Delta C_{Y\,HS} \frac{\rho V^2}{2} S_{HS}(L_{HS} - \Delta X_F)$$

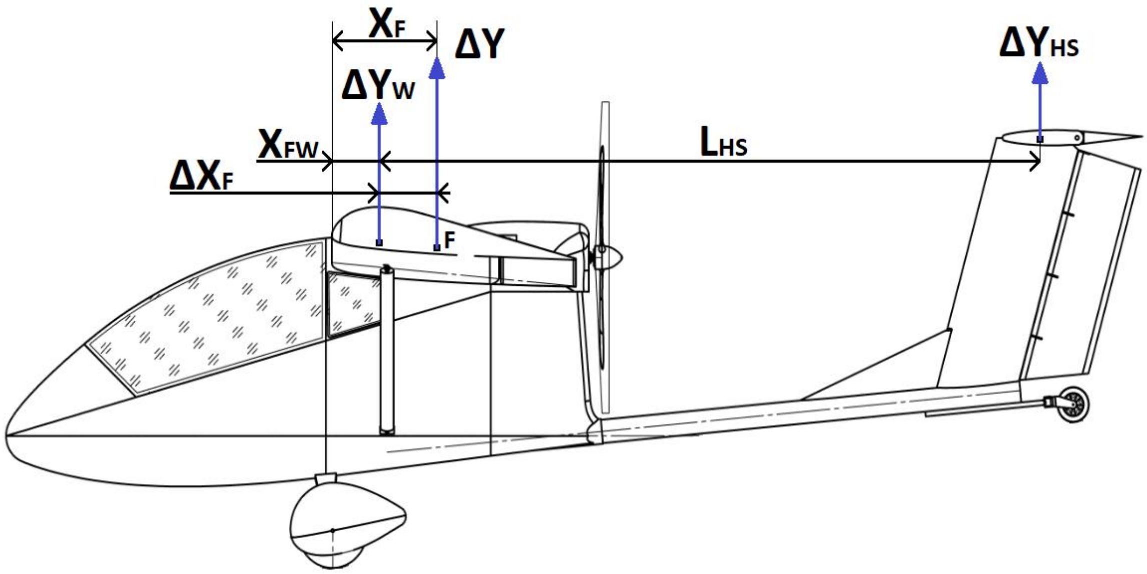


Fig 1.16 Aircraft aerodynamic center

Taking into account that $\Delta C_Y = C_Y^{\alpha} \cdot \alpha$ and $\Delta C_{Y\,HS} = C_{Y\,HS}^{\alpha} \cdot \alpha$ than:

$$C_Y^{\alpha} S\, \Delta X_F = C_{Y\,HS}^{\alpha} S_{HS} (L_{HS} - \Delta X_F) \rightarrow$$

$$\Delta X_F = \overline{S_{HS}}\, L_{HS} \frac{C_{Y\,HS}^{\alpha}}{C_Y^{\alpha} + C_{Y\,HS}^{\alpha}}$$

where $\overline{S_{HS}} = S_{HS}/S$

Derivative C_Y^{α} is taking from the chart $C_Y = f(\alpha)$ of the selected airfoil and represents the increment of the lift coefficient with an increase in the angle of attack by one degree. The approximate value of the coefficient C_Y^{α} can be taken from the Fig. 1.17.

In our case, C_Y^{α} coefficient was found during the flow simulation of the wing from which $C_Y^{\alpha} \approx 0.08$.

The value of $C_{Y\,HS}^{\alpha}$ of non-arrow-shaped horizontal stabilizer or arrow-shaped stabilizer with a small sweep angle mainly depends on the elongation λ_{HS} and just a little depends on the airfoil type. So it can be expressed by the formula:

$$C_{Y\,HS}^{\alpha} = \frac{0.085\lambda_{HS}}{1.73+\lambda_{HS}} = \frac{0.085 \cdot 4.5}{1.73+4.5} = 0.0614$$

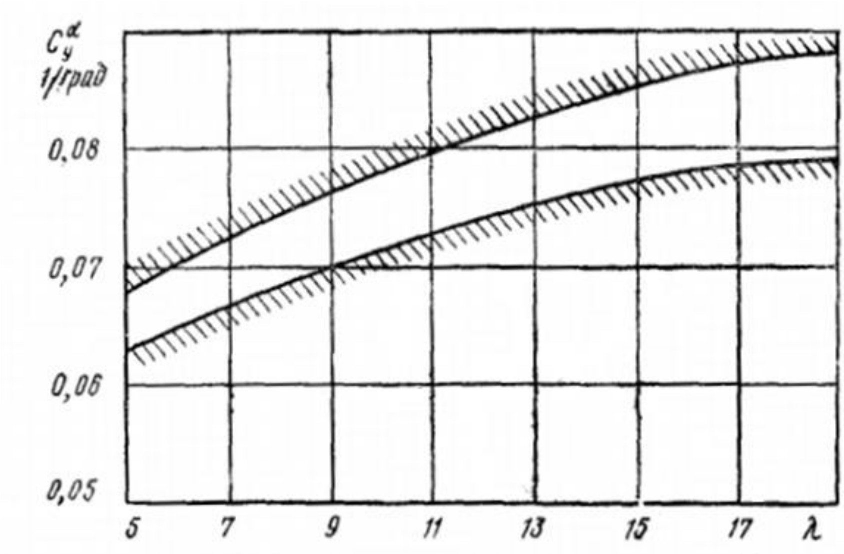


Fig 1.17 Derivative C_Y^α from wing aspect ratio

Distance from aerodynamic center of wing to aircraft aerodynamic center:

$$\Delta X_F = \overline{S_{HS}} \cdot L_{HS} \frac{C^{\alpha}_{Y\,HS}}{C_Y^{\alpha}+C^{\alpha}_{Y\,HS}} = 0.19 \cdot 3.6 \frac{0.0614}{0.08+0.0614} = 0.297 \text{ m}$$

Finally the distance from the root of aerodynamic chord to aircraft aerodynamic center is:

$$\Delta X_F = X_{FW} + \Delta X_F = 0.333 + 0.297 = 0.63 = 45\% \text{ of MAC}$$

1.4 Aircraft center of gravity position calculation

The distance from the main aerodynamic chord to the center of mass of the aircraft is called the centering determination of the aircraft. During the changing of the variants of aircraft loading or during the changing of weight in flight because of fuel burning, consequently the aircraft center of mass is changing. The moving of the cargo inside the aircraft lead to the center of mass change too. The centering is important aircraft characteristic as it affects on the balancing, stability and controllability of the aircraft. That is why it is necessary to keep it in strict limits, which is can be calculated.

It is desirable that the permissible range of centering (from the frontal to the backward) is at least 20% of the MAC. To provide aircraft stability during overload, it is necessary that the center of mass of the aircraft for all flight regimes locate ahead of its aerodynamic center.

When the stability margin is less than 5 ... 7% of the average aerodynamic chord, due to the high sensitivity to deviation of the elevator, the aircraft becomes "strict in control", requiring increased attention and accuracy from the pilot in controlling the leaver deviation. For ultralight aircrafts, in which the center of mass is significantly shifted even when the tilt of the pilot's body changes, this value at the most rear operational center of mass should be at least 10% of the MAC.

The most forward center of mass position is determined from the condition of ensuring the controllability and balancing of the aircraft in all its flight regimes. Mass of some units is shown in table 1.9.

Table 1.9

Mass of aircraft units

№	Objects	Total (kg)
1	Fuselage walls (empty)	16.4
2	Frames	8
3	Wing	81.9
4	Horizontal stabilizer	15
5	Vertical stabilizer	7.4
6	Main landing gear	13.2
7	Rear landing gear	3
8	Instrumental panel	3
9	Pilot 1	70
10	Pilot 2	70
11	Tail tube	7.4
12	Power plant	45.1
13	Engine cowl	3.5
14	Canopy assembly	11.1
15	Fuel tank	3.2
16	Fuel max	30L ≈ 21.3kg
17	Struts	6
18	Battery	5
19	Control system	20

Next cases of centering for such aircraft should be considered:

a) Take-off configuration. For such configuration mass of fuel is maximum, 2 peoples in a cabin (Let pilots weigh is maximum and equal 86 kg).
b) Landing configuration (most forward centering). For such configuration mass of fuel is minimum, 2 peoples in a cabin (Let pilots weigh is maximum and equal 86 kg).

c) Most backward configuration. For such configuration mass of fuel is minimum, rear seat is not occupied, pilot weight is minimum (60 kg).

For our case aircraft layout has already prepared in CAD SolidWorks and may be used to find the center of gravity. For this purpose, the mass characteristics of all main components of the aircraft were set designing (for example Fig. 1.18).

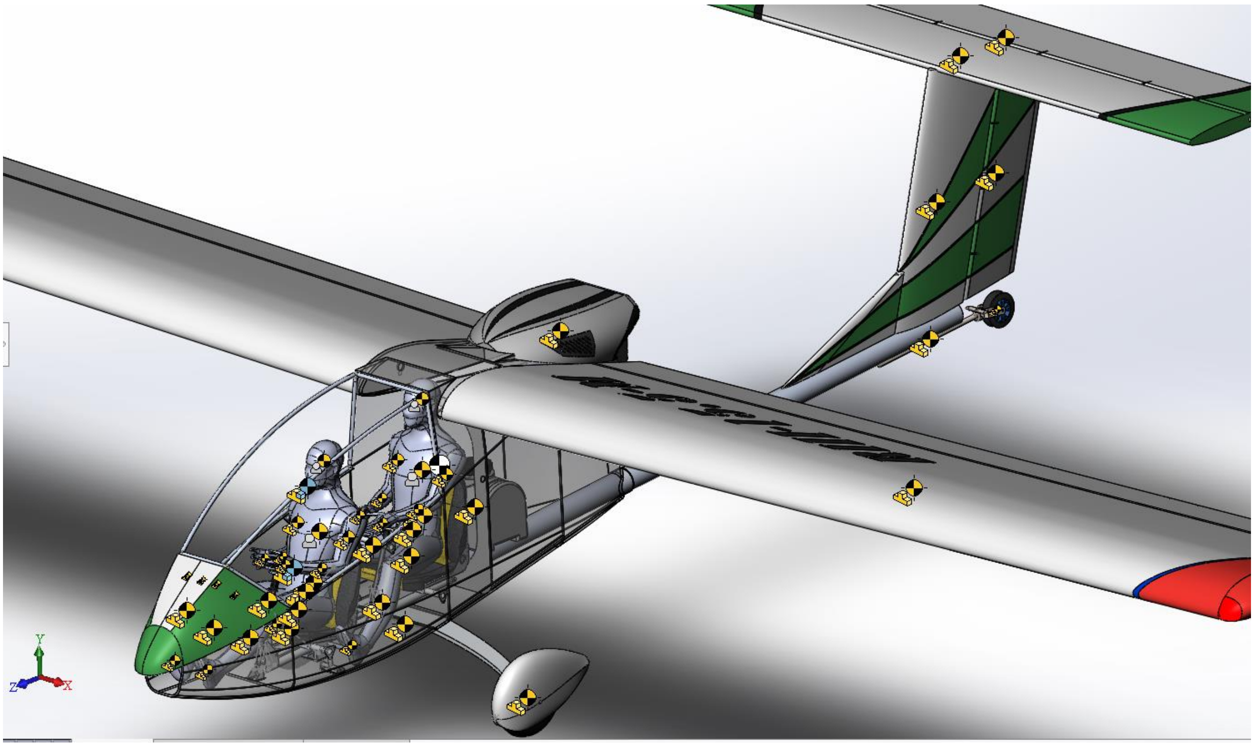


Fig 1.18 Aircraft center of mass location

For take-off configuration, distance to CG is equal 0.27 m of MAC.
For landing configuration, distance to CG is equal 0.268 m of MAC.
For most backward configuration, distance to CG is equal 0.38m of MAC.
As we can seen from software calculation, center of gravity in all flight regimes is within acceptable limits.

1.5 Aerodynamic drag coefficient calculation

1.5.1 Wing drag coefficient calculation

Wing drag coefficient at zero angle of attack(without inductive drag) is:

$$C_{X\,wing} = C_{X\,airfoil}(1-k_{AI} \cdot S_{WF}/S) + \Sigma\,\Delta C_X = 0.0104\,(1-0.95 \cdot 0.87/15.7) + (0.0013 + 0.00161) = 0.01276$$

Where:

Airfoil drag is:

$$C_{X\ airfoil} = 1.85 \cdot C_f \cdot \eta_C = 1.85 \cdot 0.00375 \cdot 1.5 = 0.0104$$

C_f – flat plane friction coefficient, is shown on the graph (Fig. 1.19 a). and depends on the Reynolds number(1960000 in our case*)*. The coefficient C_f also depends on the position of the transition point x_t of the laminar boundary layer to the turbulent, for most airfoils (the exception is laminar airfoils) x_t <0.15. Let x_t equal 0.1;

η_C – coefficient that takes into account the transition from a flat plate to a selected wing profile. Coefficient η_C, depends on the average value of the relative thickness of the wing profile $\bar{c}$ and the position of the transition point x_t, is shown in the graph (Fig. 1.19 b);

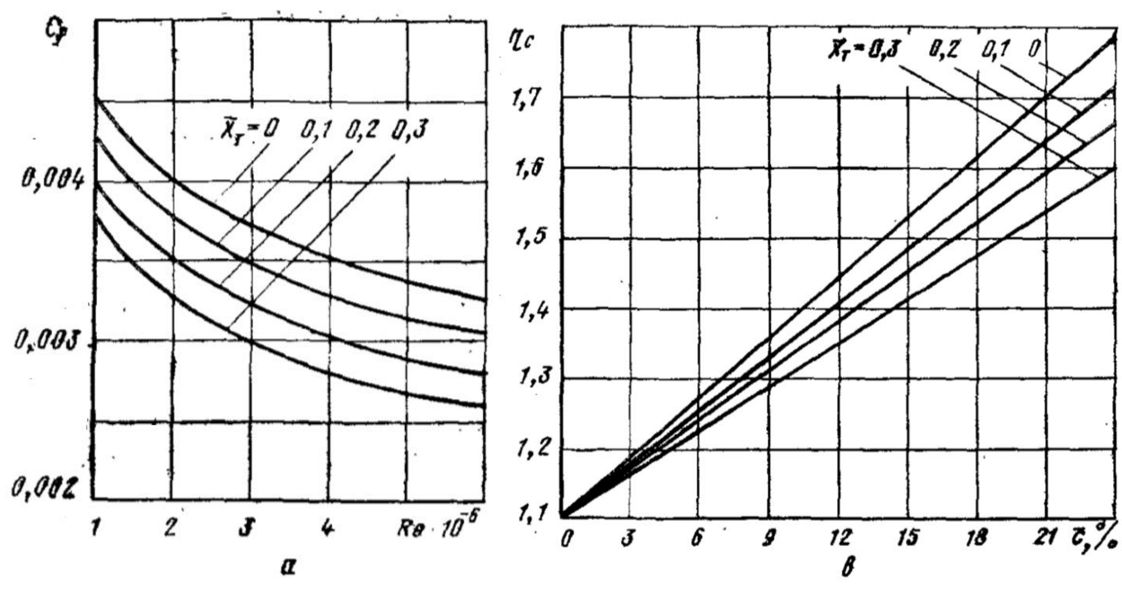


Fig 1.19 a – plate friction coefficient, b – airfoil transition coefficient.

k_{AI} – aerodynamic interaction (interference) coefficient, depends on the scheme of the aircraft and the shape of the fuselage cross section (Table 1.10).

S_{WF} – fuselage wing area.

$\Sigma\Delta C_X$ – sum of the coefficients of additional drag to account the clearance of the wing surface, slots in it and add-ons.

If 20% of the airfoil from the nose is clear from the protruding rivet heads, then the ΔC_X can be equal to 0.0013. If the entire wing has protruding rivets, then the value of the additional resistance increases can be equal to 0.0020. Our wing has, conceivably, composite skin, without rivets in nose part, let ΔC_X equal 0.0013.

Table 1.10

Interference coefficient

Wing position	Interference coefficient
High wing	0.95
Mid wing	0.85
Low wing with round fuselage cross section	0.25
Low wing with oval fuselage cross section	0.50
Low wing with rectangular fuselage cross section	0.60

The slots drag between the wing and the aileron or mechanization can be find, using the following approximate formula:

$$\Delta C_{X\,slots} = 0.0017 \cdot l_S / l = 0.0017 \cdot 10.8/11.4 = 0.00161$$

1.5.2 Tail unit drag coefficient calculation

Drag coefficients of horizontal and vertical stabilizer are determined similarly to the coefficient of the wing. Coefficient x_t is equal to zero, since the empennage of the aircraft, that are made according to the normal scheme, is always locate in a disturbed flow generated by the wing, fuselage and propeller.

Reynolds number for horizontal stabilizer is:

$$Re_{HS} = v \cdot b_{HS} \cdot 70000 = 30 \cdot 0.82 \cdot 70000 = 1722000$$

Reynolds number for vertical stabilizer is:

$$Re_{VS} = v \cdot b_{HS} \cdot 70000 = 30 \cdot 0.88 \cdot 70000 = 1848000$$

Airfoil drag coefficient of horizontal stabilizer is:

$$C_{X\,HS\,airfoil} = 1.85 \cdot C_f \cdot \eta_C = 1.85 \cdot 0.0042 \cdot 1.36 = 0.0106$$

Airfoil drag coefficient of vertical stabilizer is:

$$C_{X\,HS\,airfoil} = 1.85 \cdot C_f \cdot \eta_C = 0.0106$$

Additional Drag coefficient is:

$$\Delta C_{X\,HS} = \Delta C_{X\,VS} = 0.0013$$
$$\Delta C_{X\,HS\,slots} = \Delta C_{X\,HS\,slots} = 0.0017$$

Horizontal stabilizer drag coefficient is:

$$\mathbf{C_{X\ HS}} = \mathbf{C_{X\ HS\ airfoil}} + \Sigma\, \Delta C_X = 0.0106 + 0.0013 + 0.0017 = 0.0136$$

Vertical stabilizer drag coefficient is:

$$\mathbf{C_{X\ VS}} = 0.0135$$

1.5.3 Fuselage drag coefficient calculation

To determine approximate drag coefficient, we can make flow simulation of the fuselage. For this project simplified fuselage model (Fig. 1.20) was prepared for simulation in SolidWorks Flow.

Let V = 30m/s (approximate cruise speed)

Fuselage midsection $C_{M\ fuselage} = 0.895m^2$

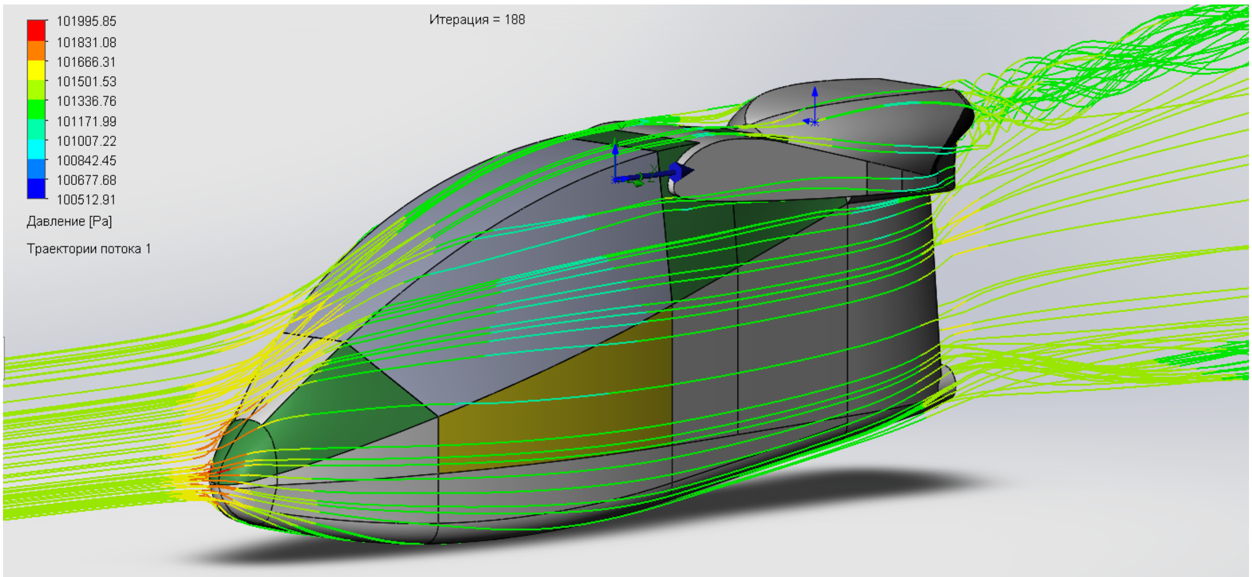


Fig 1.19 Simplified fuselage model fluid flow analysis

From simulation we determine drag force that is equal 55N, then we can calculate drag coefficient of fuselage:

$$C_{X\,fuselage} = \frac{2D}{\rho A V^2} = \frac{2 \cdot 55}{1.225 \cdot 0.895 \cdot 30^2} = 0.111$$

Wing Struts drag coefficient is:

$$C_{D\ struts} = 2(C_{X\ strut\ main} \cdot S_{M\ strut\ main} + C_{X\ strut\ add} \cdot S_{M\ strut\ add}) = 2(0.8 \cdot 0.081 + 1.04 \cdot 0.008) = 0.146$$

1.5.4 Landing gear drag coefficient calculation

Drag coefficient of landing gear can be calculated as:

$$C_{D\ LG} = 2 \cdot C_{X\ wheel\ main} \cdot S_{M\ wheel\ main} + C_{X\ wheel\ back} \cdot S_{M\ wheel\ back} +$$
$$C_{X\ strut\ main} \cdot S_{M\ strut\ main} + C_{X\ strut\ back} \cdot S_{M\ strut\ back} =$$
$$2 \cdot 0.35 \cdot 0.03785 + 0.5 \cdot 0.01 + 1.2 \cdot 0.04 + 1.2 \cdot 0.0033 = 0.0835$$

Drag coefficient for specified wheel can be taken from the table 1.11

Table 1.11

Wheel drag coefficient

Diametric section of wheel	Drag coefficient
Elliptical	0.25
Rectangular with rounded edges	0.35
Rectangular	0.50

Using the fairings, drag of the wheels can be reduced by 2 ... 3 times.

1.5.5 Aircraft drag coefficient calculation

Aircraft drag coefficient at zero lift is:

$$C_{X0} = 1.1(C_{X\ win} + C_{X\ HS} \cdot \overline{S_{HS}} + C_{X\ VS} \cdot \overline{S_{VS}} + C_{X\ fuselage} \cdot C_{M\ fuselage}/S + C_{D\ LG} \times$$
$$S_{M\ LG}/S + \Sigma C_{X\ additional} \cdot C_{M\ additional}/S) =$$
$$1.1(0.01276 + 0.0106 \cdot 0.19 + 0.0105 \cdot 0.094 + 0.111 \cdot 0.895/15.75 + 0.0835 \cdot$$
$$0.129/15.75 + 0.146\ 0.178/15.75) = 1.1 \cdot 0.0244 = 0.0268$$

Inductive drag is:

$$C_{Xi} = A \cdot C_Y^2$$

Where:

$$A = (1+\delta)/\pi \cdot \lambda_{effective} = (1+0.075)/(\pi \cdot 6.941) = 0.0493$$

where δ – coefficient that takes into account elongation and narrow of the wing. The value of coefficient δ can be taken from the graph (Fig. 1.20).

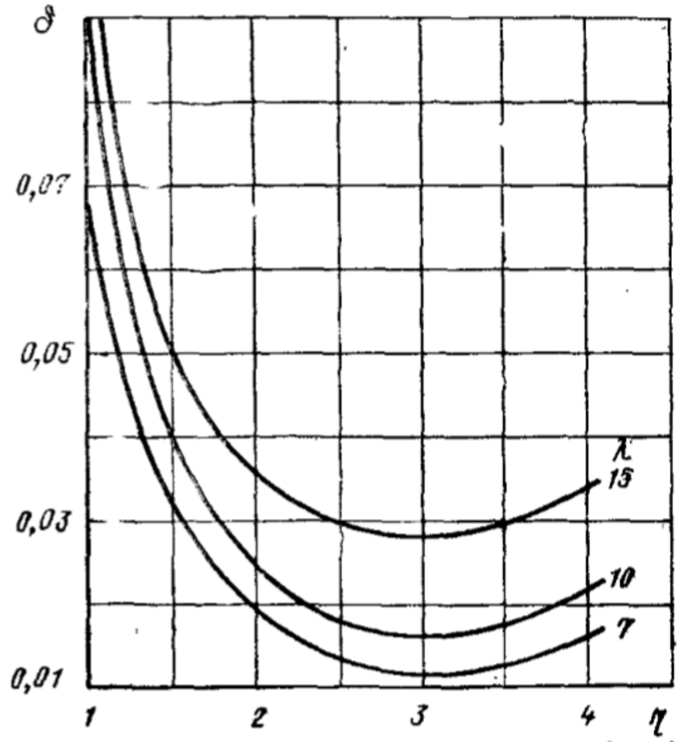


Fig. 1.20

$\lambda_{effective}$ – effective wing elongation. To determine the effective wing elongation of the ultralight aircraft, approximate formula may be used:

$$\lambda_{effective} = 0.9\lambda/(1 + S_{WF}/S) = 0.9 \cdot 8.14/(1 + 0.87/15.7) = 6.941$$

where S_{WF} – fuselage wing area

Final result is shown in table 1.12.

Table 1.12

Data for polar construction

Angle of attack	C_Y	C_{X0}	C_{Xi}	C_X
-4	0.04		0	0.0268
0	0.30		0.00444	0.03124
4	0.56		0.01546	0.04226
8	0.84		0.03479	0.06159
12	1.08	0.0268	0.0575	0.0843
16	1.34		0.08852	0.11532
20	1.56		0.11997	0.14677
24	1.78		0.1562	0.183

Aircraft polar is shown in Fig 1.21.

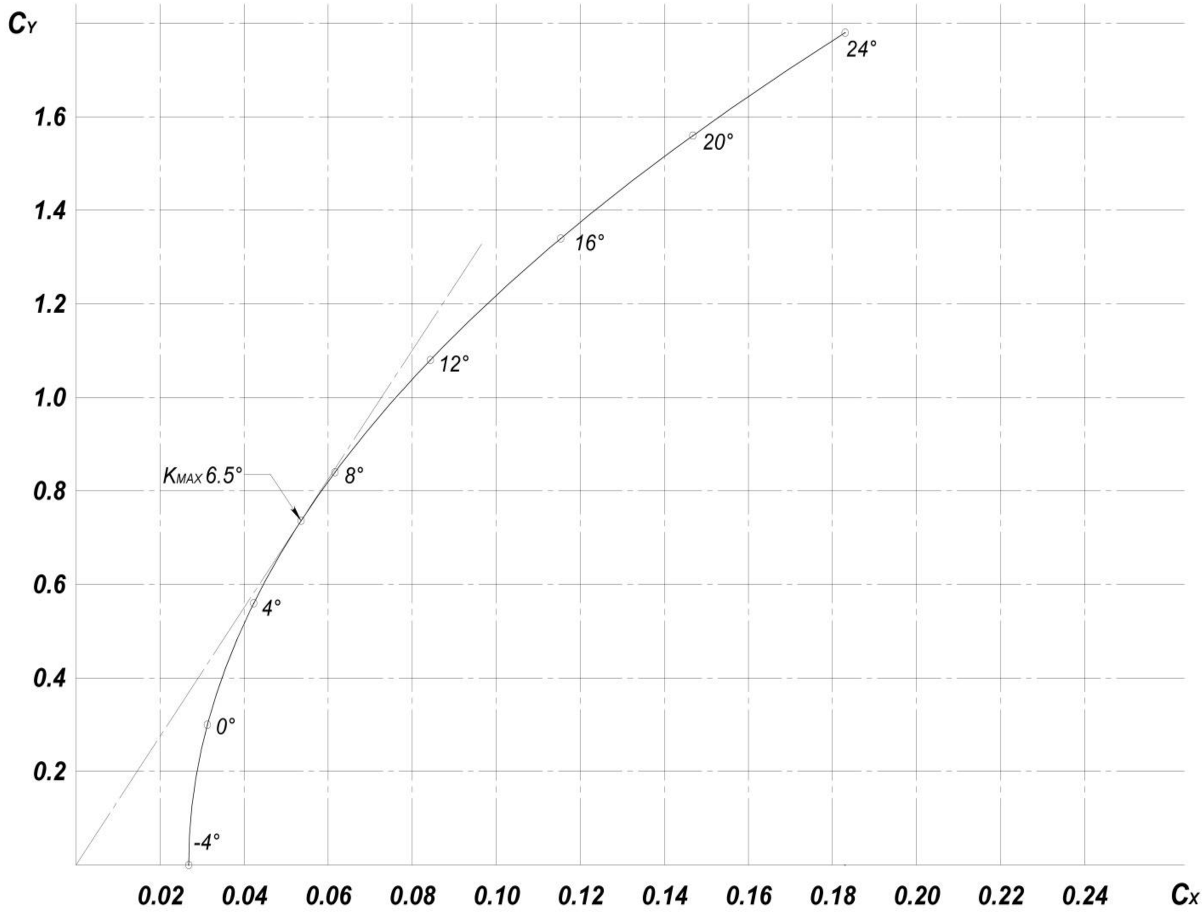


Fig. 1.20 Aircraft polar

1.6 Calculation of the required and available power for steady level flight

The diagram of required power or, as it is also called, the Zhukovsky diagram, is constructed in the coordinates of power and speed. Each point of this curve corresponds to a well-defined angle of attack, for which the values of V_{SLF} and N_{SLF} are calculated.

Required speed for steady level flight is:

$$V_{SLF} = \sqrt{\frac{2G}{C_y \rho S}}$$

Required power for steady level flight is:

$$N = \frac{\rho V^3}{2} C_x S$$

The results of calculation is shown in table 1.13.

Table 1.13

Calculation of required power for steady level flight

Angle of attack	C_Y	C_X	V_{SLF} steady level flight m/s	V_{SLF} steady level flight km/hour	*N* KW
-4	0.04	0.0268	-	-	-
0	0.30	0.03124	39.06	140.62	17.960
2	0.45	0.037	31.94	114.98	11.630
4	0.56	0.04226	28.59	102.92	9.527
8	0.84	0.06159	23.34	84.02	7.554
12	1.08	0.0843	20.58	74.09	7.088
16	1.34	0.11532	18.48	66.53	7.021
20	1.56	0.14677	17.13	61.67	7.117
24	1.78	0.183	16.03	57.71	7.272

Available power is part of the engine power N that is used to move the aircraft. For this calculation computer software were used, it allows to get approximate results of propeller efficiency in different flight speeds.

Propeller data:

- Fixed pitch of the propeller 37” (optimal)
- Propeller diameter 1.6m
- Max RPM 2500

Results of calculation is shown in table 1.14.

Table 1.14

Calculation of available power for steady level flight

Speed of flight V	Thrust N	Power Absorbed W	Power Output W	Efficiency %
16.03	1104.2	35782	17700	49.47
17.13	1081.2	35508	18521	52.16
18.48	1051.3	35127	19428	55.31
20.58	1001.2	34425	20605	59.85
23.34	928.9	33286	21682	65.14
28.59	771.8	30353	22065	72.69
31.94	658.4	27890	21028	75.40
39.06	384.6	20905	15022	71.86

Diagram of required and available power for steady level flight is shown in Fig. 1.21.

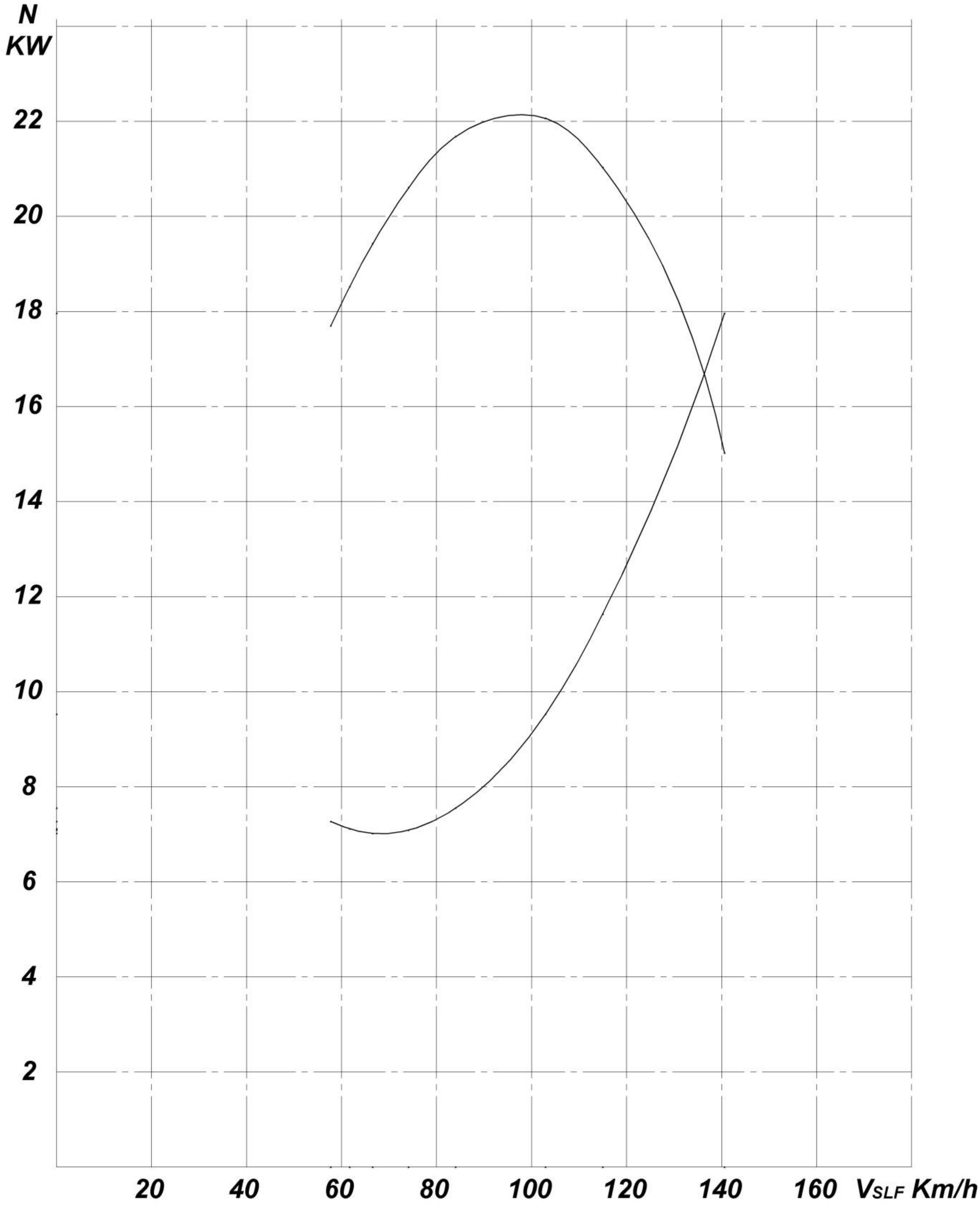


Fig. 1.21 Diagram of required and available power for steady level flight

Conclusion to the project part

In this part the main geometric dimensions, aerodynamic characteristics and centering of designed aircraft were determined. All these data allows to proceed to a more accurate design of aircraft units.

During the calculation the main geometrical parameters caused by operational purpose, planned take-off weight, speed, conditions of landing and take-off, were considered. All obtained values meet requirements for the ultralight training aircraft.

The centering of the designed aircraft was performed using computer method for determination of center of gravity location. The most forward center of gravity position is 19% from the origin of the leading edge of MAC and the most aft is 27% of MAC, which makes the plane quite easy for controlling in pitch.

From calculations we can see that flight performances is close to the selected prototype. Fixed pitch propeller is used for such airplane, but flight performance can be improved by using a variable pitch propeller. Also aerodynamic drag can be reduced, for such purpose fairings for landing gear and another strut design may be considered.

2. SPECIAL PART. DESIGN OF THE PILOT SEAT FOR ULTRALIGHT AIRCRAFT

In a modern passenger airplane relative pilot seat mass is only 0.001 ... 0.002 and does not have a significant impact on the overall mass balance, but on the ultralight aircrafts the relative mass of the same seat is already 0.04 ... 0.06, so it becomes comparable with the relative mass of tail unit, chassis, engine or fuel.

2.1 Analysis of ultralight aircraft seat market

A modern aviation components market has a huge number of seats, but often the layout of the aircraft or other reasons forces designers to create their own seat design for specified plane. Let us consider some of the models available on the market of ultralight aircraft seats.

1) Piper J-3 as well as selected prototype, Aeroprakt A20, has tandem scheme (Fig 2.1 a). One of the variants of its seat frame is shown in Fig 2.1 b. It has a simple design and is produced in different configurations. The disadvantages include: high market cost, difficult mounting for defined fuselage layout.

a *b*

Fig. 2.1 *a* – Piper J-3 pilot seat location, *b* – seat frame

2) Black Max seat(Fig. 2.2). This light weight ulralight aircraft seat weighs under 6 pounds and constructed from 6061-T6 aluminum and hardware. The sling back seat is made from a durable Cordura Nylon and molds to the shape of body making it very comfortable. There is an optional seat cushion that can be attached to seat bottom for additional support. Includes a large storage pocket in the back, additional storage between pan and seat pivots down to make it easier to get in behind the seat on a tandem.

The disadvantages include: a big height of frame bottom part that are made from sheet metal, so it is necessary to increase distance between fuselage flor and seat coating, which is fraught with an increasing in height of fuselage canopy and other structural elements. Also many additional mounting elements are required for such type of seat.

After studying of the ultralight aircraft seats market, it was decided to design a new seat construction that will takes into account all the features of specified aircraft layout.

Fig. 2.2 Back Max seat design

2.2 Strength calculation of the pilot seat structure

As a rule, pilot's seat of light aircrafts is included in the structural scheme of the fuselage, that is why all of its main elements should be made, based on the most severe case of loading. Also the pilot's seat must be equipped with safety belts that can withstand an overload of at least 8 ... 10G.

2.2.1 Pilots center of gravity determination

In super light aircraft, the mass of the pilot has a significant effect on centering. There are huge amount of possible fuselage design schemes and accordingly, the position of the pilot in them will differ. For example, on large passenger aircrafts the pilot usually occupies a sitting position but on ultralight aircraft it may be advisable to place the pilot in a reclined or even lying position. Knowing pilots center of mass location and having the layout of the aircraft, it is possible to calculate strength of structural elements of the aircraft and especially the pilot seat with higher accuracy.

Let assume that the average pilot mass is 70kg. For our calculation, we can use experimental data (Table 2.1) that describes relative mass of body parts and its location.

Table 1.14

Relative mass of body parts

Part	Relative mass
Head	0.07
Torso	0.43
Shoulder	0.03
Forearm	0.02
Brush	0.01
Hip	0.12
Shin	0.05
Feet	0.02

For the calculation, layout of the aircraft is already prepared and we know how the pilot will be placed (Fig. 2.2). To find the center of mass of the whole body of specified position, CAD can be used. For my case I prepared a manikin model (Fig. 2.3) in CAD/CAE SolidWorks, set the mass and its position for each part, according to the experimental data.

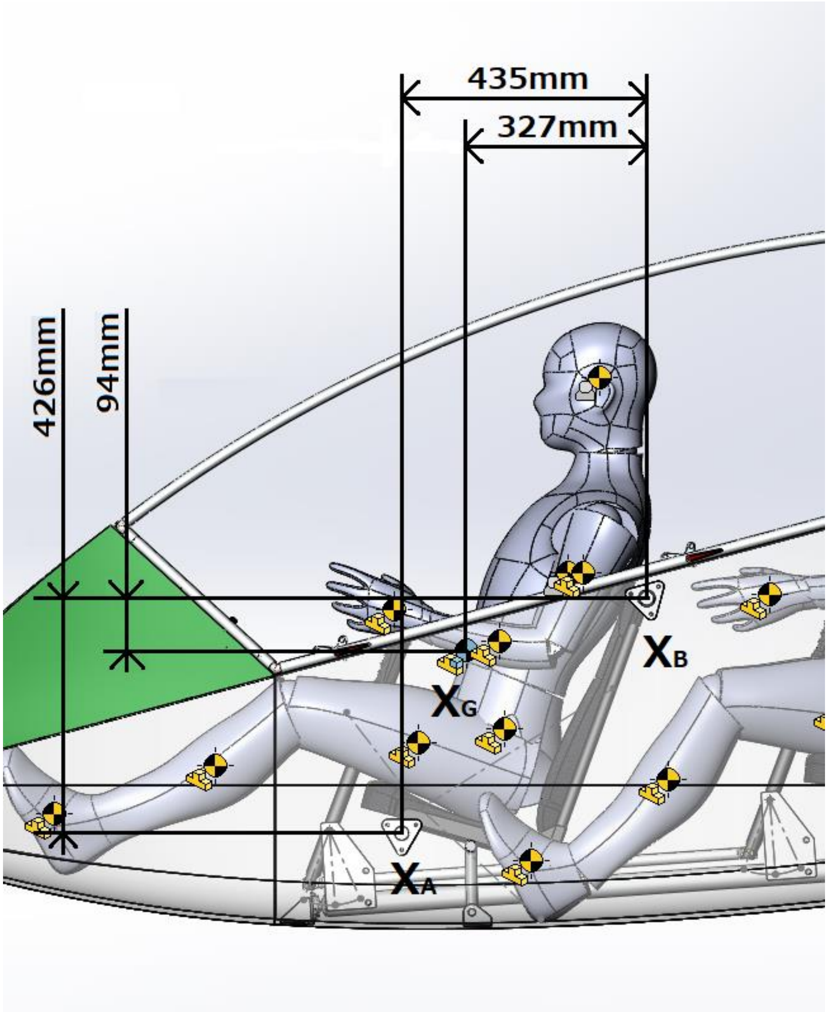


Fig. 2.3 Pilots position in aircraft layout

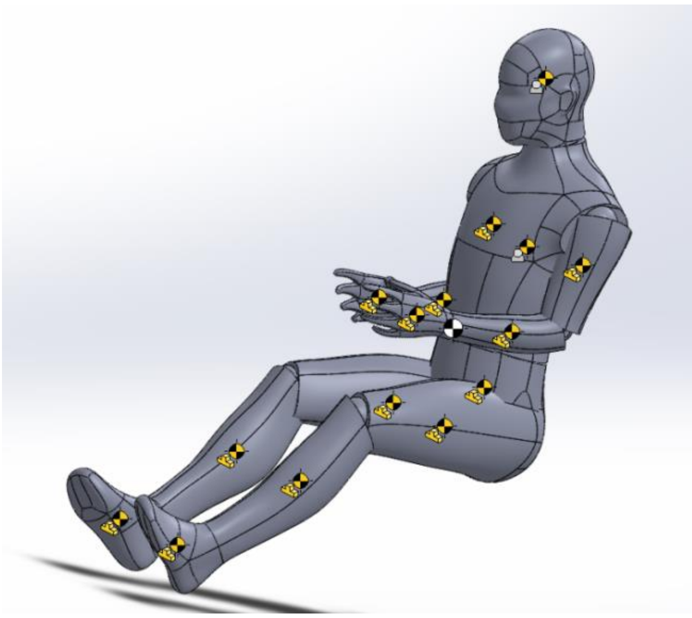


Fig. 2.4 Pilots center of gravity.

2.2.2 Seat strength calculation

Pilot's seat is attached to the pipes, which in turn are attached to the side walls of fuselage. For simplification, we assume that the center of mass that acts on the seat to fuselage fastening (X_A, X_B) is in the same place as the pilot center of gravity (X_G).

All possible cases of loading and emergency landing must be considered during strength calculation of a seat.

From CS-VLA 561 (Emergency landing conditions):

The occupant experiences the ultimate inertia forces listed below:

Ultimate Inertia Load Factors

- Upward 3.0 g
- Forward 9.0 g
- Sideward 1.5 g.

To simplify our calculation we will consider fastening pipe as a simple beam. I think such allowance is permissible, since fuselage walls are not absolutely rigid.

First let consider loads in vertical direction (+4, -2). In our case, for vertical direction, it is enough to calculate tubes just in case, when load factor is equal +4G.

Load that acts on tube X_A:

$$\text{Load } X_A = b/l \cdot k \cdot n \cdot G = 327/425 \cdot 1.5 \cdot 4 \cdot 843.66 = 3895 \text{ N}$$

Load that acts on tube X_B:

$$\text{Load } X_B = a/l \cdot k \cdot n \cdot G = 108/425 \cdot 1.5 \cdot 4 \cdot 843.66 = 1257 \text{ N}$$

where k – safety factor (1.5)

n – loading(+4g)

G – pilot mass (consider not less than 86 kg from CS-VLA)

From construction of the seat we assume that the load on the lower tube is evenly distributed (Fig. 2.5). Computer software methods may be used to calculate bending moment.

Having a value of maximum bending moment (435.28 N/m), we can check strength of the pipe. Material of the tubes is D16T (or 2024) for which [σ] equal 200MPa.

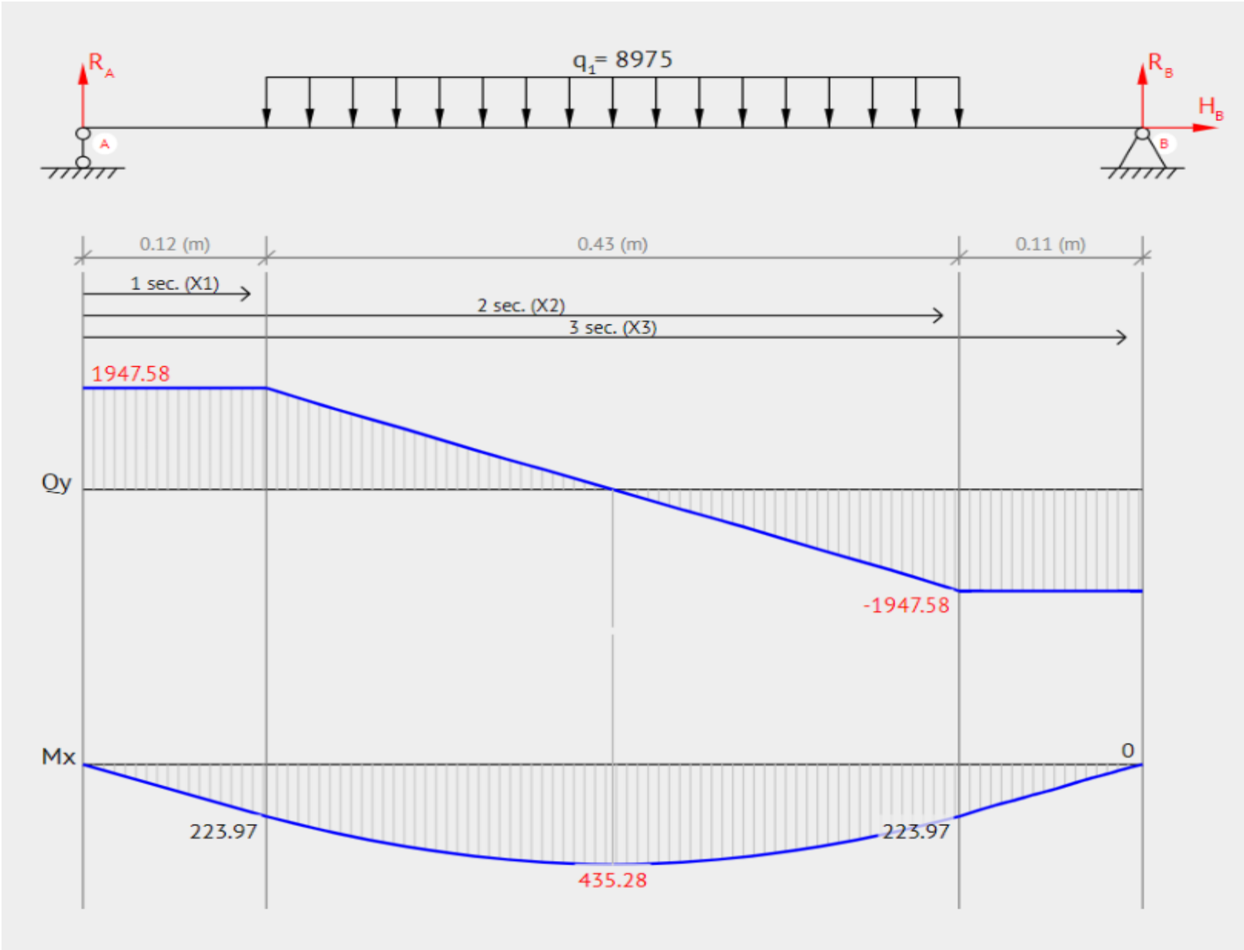


Fig. 2.5 Lower tube bending moment diagram.

Required moment of resistance is:

$$\sigma_{mas} = \frac{M_{max}}{W_z} \leq [\sigma] \rightarrow W_{z\ rq} \geq \frac{M_{max}}{[\sigma]} = \frac{435}{200 \cdot 10^6} = 2.175\ \text{cm}^3$$

Inertia moment for selected tube is:

$$I_z = I_y = \frac{\pi D^4 - d^4}{64} = \frac{\pi \cdot 0.028^4 - 0.024^4}{64} = 1.389 \cdot 10^{-8}\ \text{m}^4$$

Moment of resistance is:

$$W_{z\ tube} = W_y = \frac{2I_z}{D} = 9.918\ \text{cm}^3$$

$W_{z\ tube} > W_{z\ rq}$, the tube has sufficient strength to withstand maximum vertical overload.

For the upper pipe, it makes sense to consider only an emergency case of landing (9g, forward direction), since pilot center of mass is closer to upper tube, and safety belts are attached to it. The seat is mounted on 2 bolts to the upper tube. This tube also can be considered as a simple beam (Fig. 2.6).

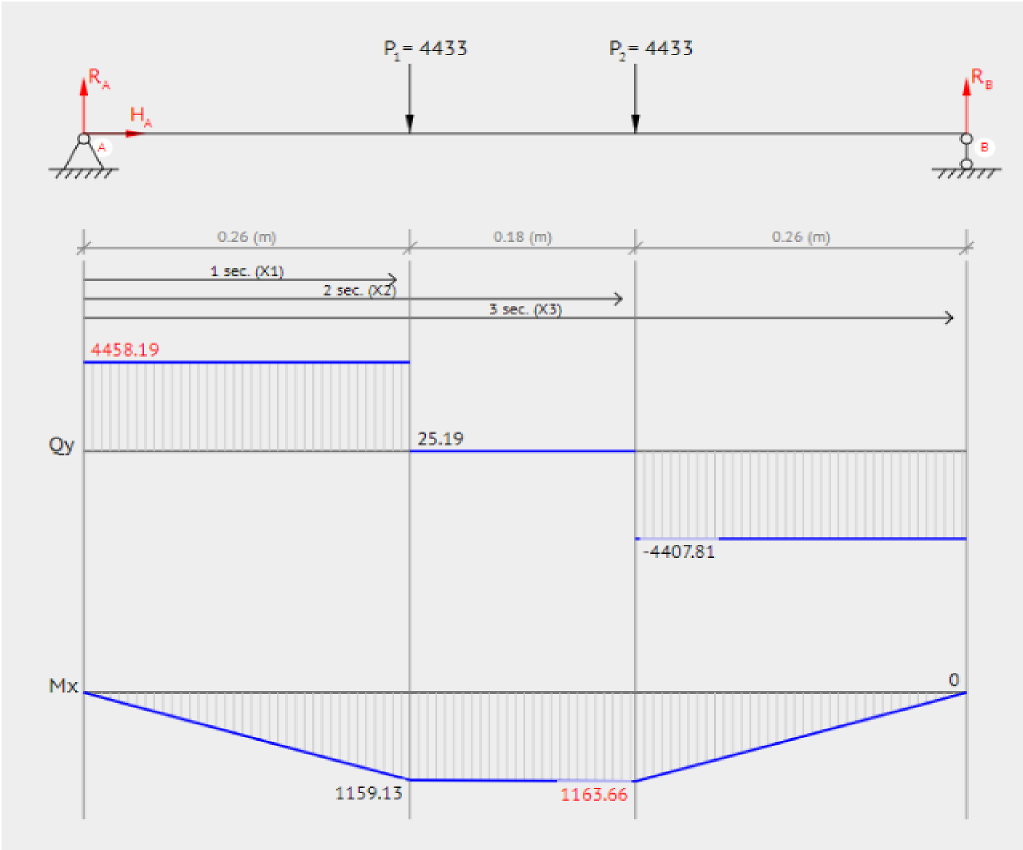


Fig. 2.6 Upper Lower bending moment diagram.

Required moment of resistance is:

$$\sigma_{mas} = \frac{M_{max}}{W_z} \leq [\sigma] \rightarrow W_{z\ rq} \geq \frac{M_{max}}{[\sigma]} = \frac{1163.66}{200 \cdot 10^6} = 5.82\ \text{cm}^3$$

$W_{z\ tube} > W_{z\ rq}$ the tube has sufficient strength to withstand maximum forward overload.

Seat structure includes part that are produced from tube using bending method and sheet metal part. Let estimate that this parts is produced from D16AT. This parts have a complex shape and it can be difficult to calculate them using standard methods. SolidWorks simulation may used to calculate strength of specified parts. In simulation, let assume that the force evenly distributed on surface of sheet metal part, pipes have a hinged type of fastening and the bottom part is firmly fixed (Fig. 2.7 and Fig. 2.8). Surface force is equal 5500 N.

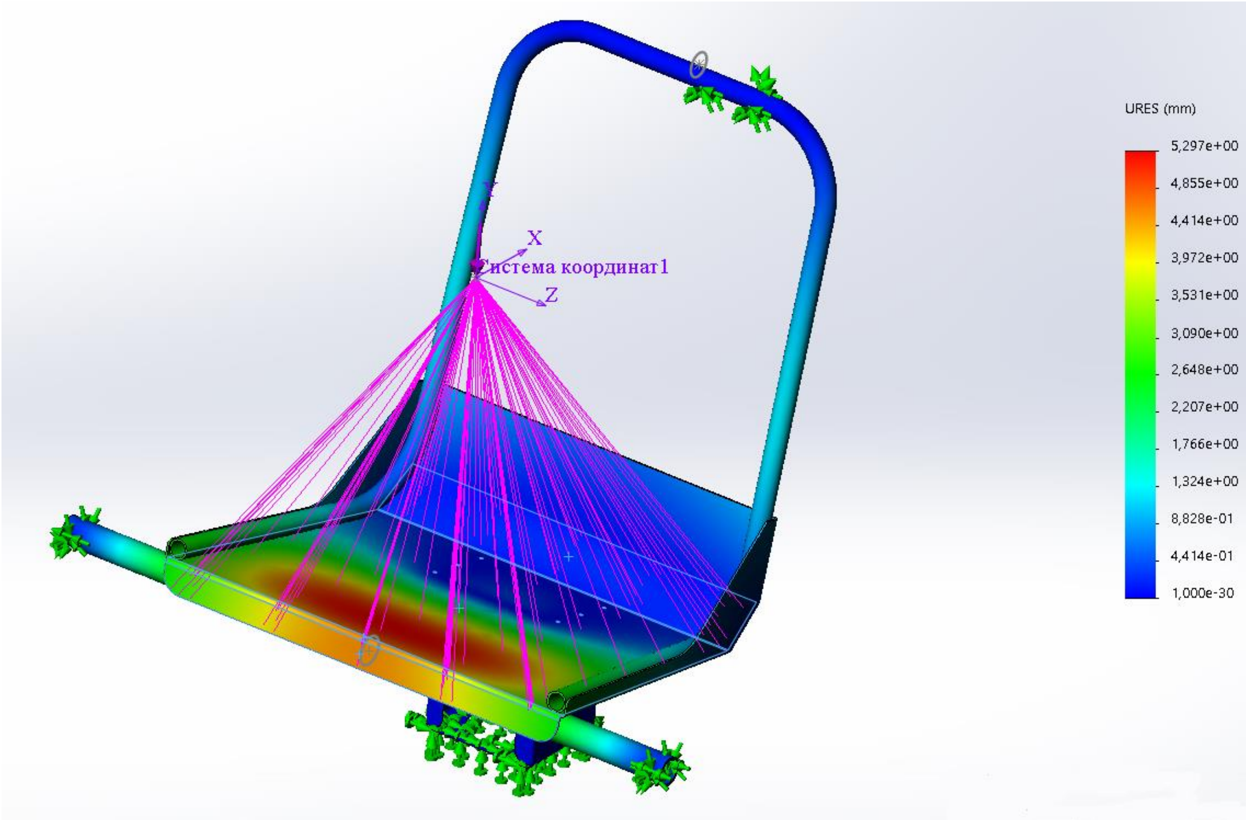


Fig. 2.6 Seat loading simulation under vertical acceleration.

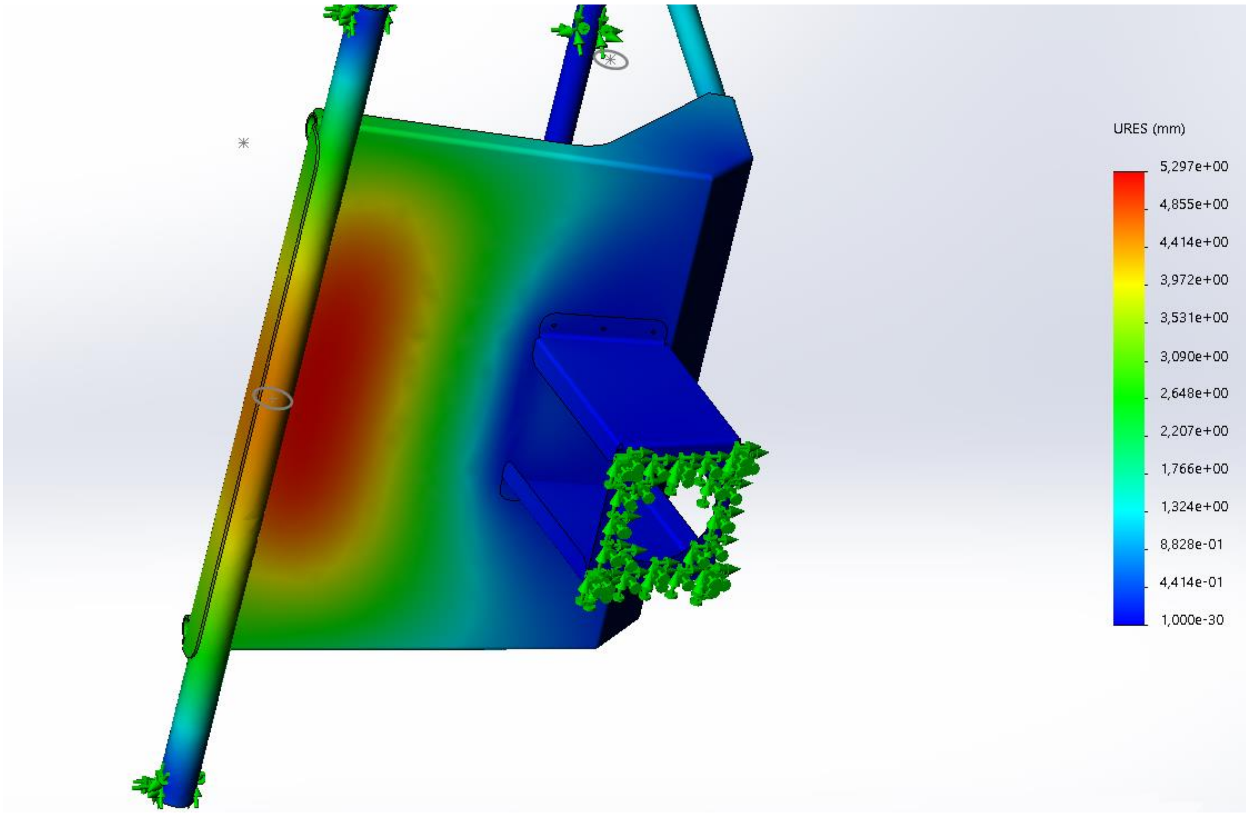


Fig. 2.7 Seat loading simulation under vertical acceleration

Conclusion to the special part

In this part the main cases of loading of pilot seat were considered and the strength calculation of the basic elements of the pilots seat was made.

To calculate the strength of sheet metal parts, a load simulation was performed for maximum vertical acceleration that is equal 6g, the safety factor was more than one, respectively, the strength of the seat frame meets the requirements.

The frame design was developed for the standard Oregon Aero seat coating, but also some other types of coating may be used.

General conclusion

In the course of this diploma paper, the following results were obtained:

- preliminary design **o**f light training aircraft with take-off mass up to 500 kilograms;
- the schematic design of layout light training aircraft;
- the center of gravity of the airplane position;
- the calculation of aerodynamic performances of aircraft;
- the design of pilot seat for ultralight aircraft

Design aircraft satisfies the planned aim of usage, its simple design allows to produce it with a minimum resources and special equipment. Its geometrical characteristics will provide the necessary aerodynamic performances, which makes this aircraft economically efficient and simple in operation.

References :

1. Чепурных И. В. Прочность конструкций летательных аппаратов. Комсомольск-на-Амуре, 2013.

2. Чумак П.И., Крывокрысенко В.Ф. Расчет, Проектирование и постройка сверхлегких самолетов. Патриот, 1991.

3. Одиноков Ю. Г. Расчет самолета на прочность. Машиностроение, 1973.

4. Болонкин А. Теория полета летающих моделей. ДОСААФ, 1962.

5. Балакин В. Л., Лазарев Ю. Н. Динамика полета самолета. Самара, 2011.

6. Патент РФ 2336200.

7. CS-VLA / Amendment 1, 5 March 2009.

8. Aircraftspruce. Электронный ресурс: https://www.aircraftspruce.com/

Appendix

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	Sheets	№ doc.	Sign.	Date				
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						AF 402 134		
St.control.		Khyzhniak S.V.						
Head of dep.		Ignatovich S.R.						

Appendix A

Simulation 1.0

AOA = 15° , NO FLAP, V = 20m/s, ρ = 1.225, Re = v × l × 70000 = 20 × 1.4 × 70000 = 1960000

Warning: results bellow for chord 1.4m

Results:

continuity	x-velocity	y-velocity	energy	k	epsilon	lift_foil	drag_foil
1.1007e-02	3.4720e-06	4.1903e-06	3.1309e-09	8.1910e-05	9.2621e-05	2.5608e+00	6.8832e-02

Forces - Direction Vector (1 0 0)

Forces (n) Coefficients

Zone	Pressure	Viscous	Total	Pressure	Viscous	Total
foil	13.457095	2.7447382	16.201834	0.057182777	0.011663122	0.068845899

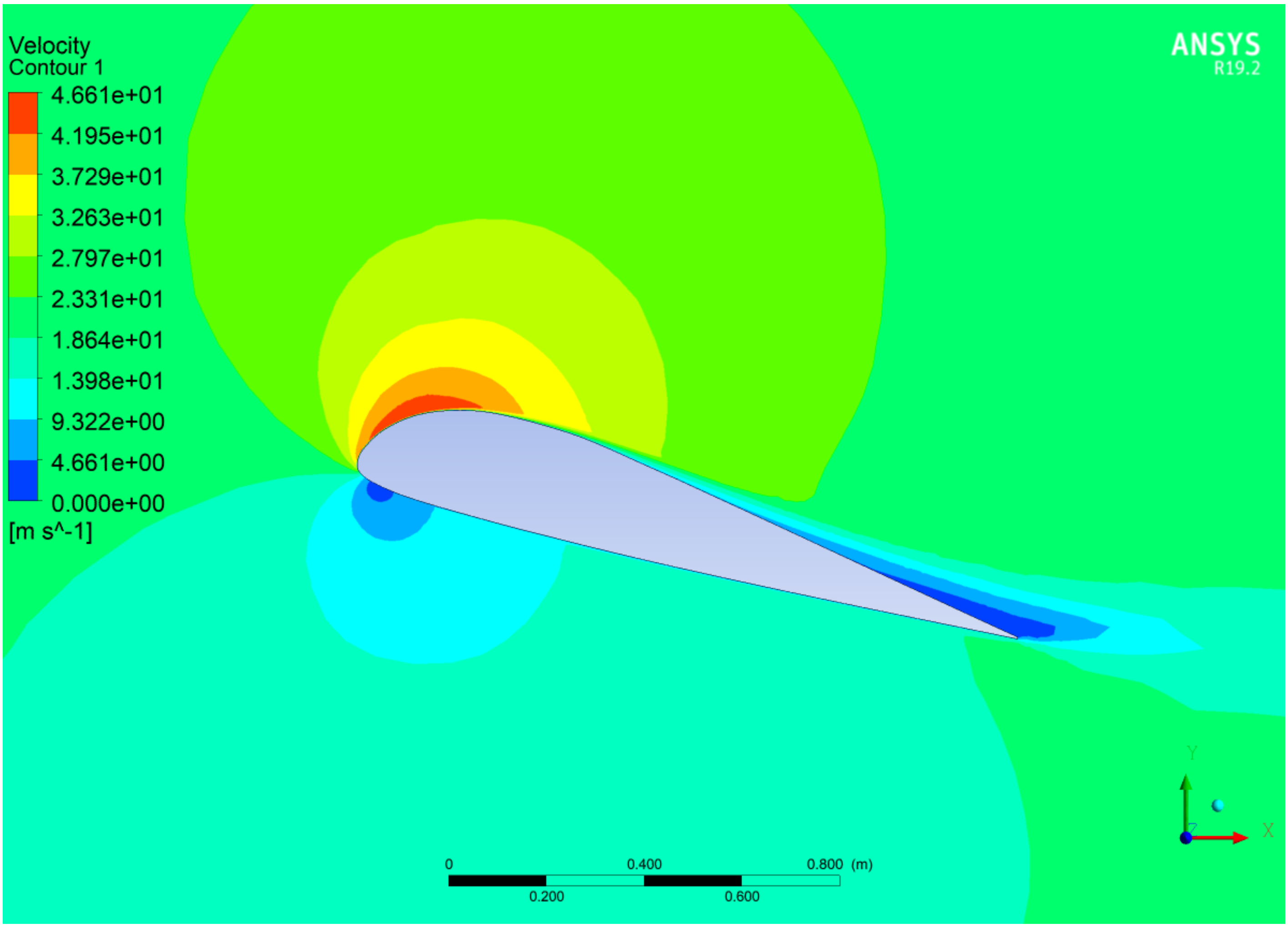
Forces - Direction Vector (0 1 0)

Forces (n) Coefficients

Zone	Pressure	Viscous	Total	Pressure	Viscous	Total
Foil	602.66435	0.17387542	602.83823	2.5608811	0.00073884	2.5616199

Center of Pressure - Set Coordinate y = 0 (m)

Zone	x
foil	0.36186548



Appendix B

Simulation 1.1

AOA = 15°, AOF = 0°, V = 20m/s, ρ = 1.225, Re = v × l × 70000 = 20 × 1.4 × 70000 = 1960000

Results:

continuity	x-velocity	y-velocity	energy	k	epsilon	drag_flap	lift_flap	lift_foil	drag_foil
9.9621e-04	4.8635e-06	2.3662e-06	7.6349e-10	1.0759e-04	9.7295e-05	6.8968e-02	1.0844e-01	2.1858e+00	2.4500e-02

Forces - Direction Vector (1 0 0)
Forces (n) Coefficients

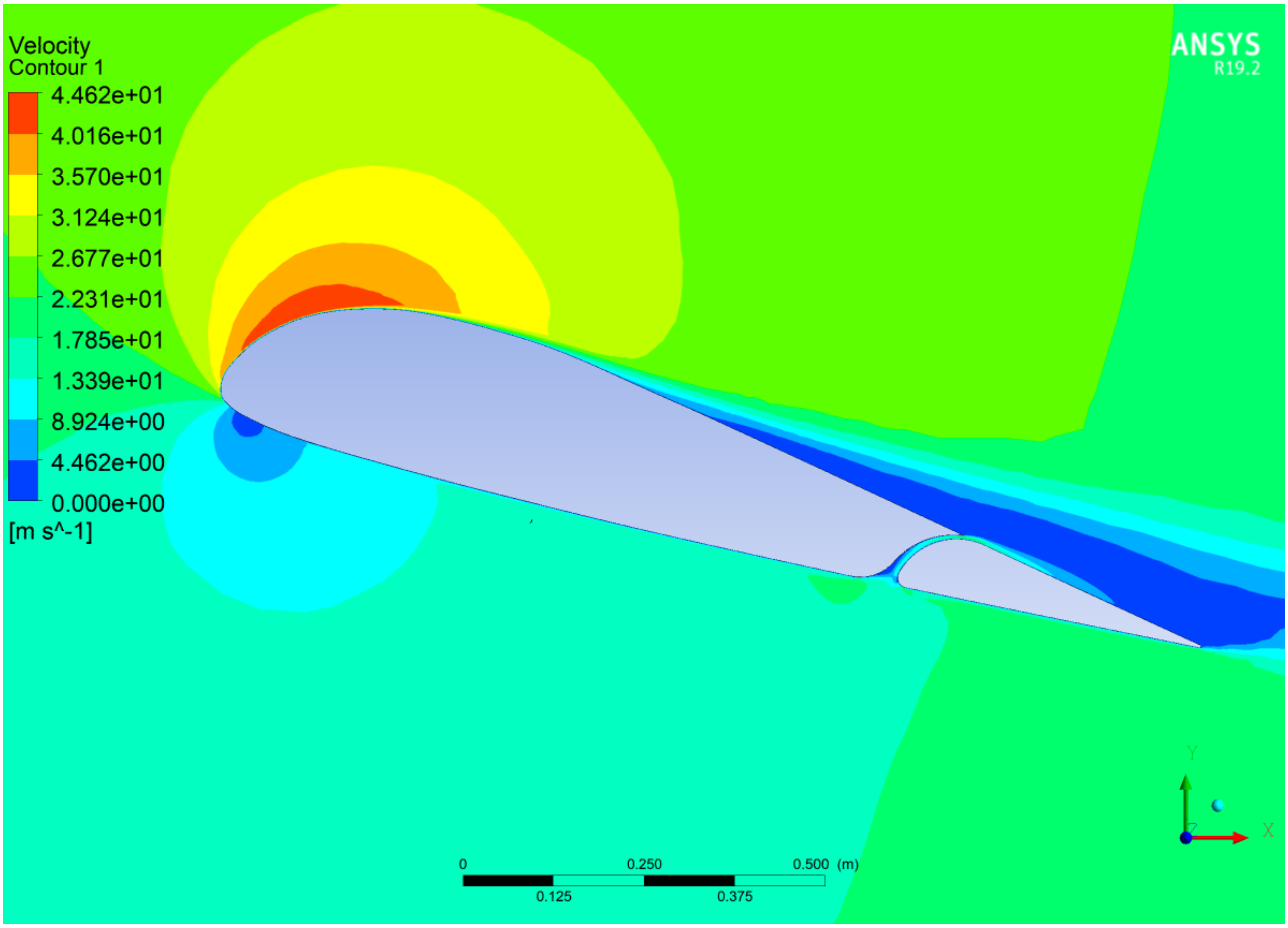
Zone	Pressure	Viscous	Total	Pressure	Viscous	Total
flap	15.524699	0.70426758	16.228967	0.065968574	0.0029926202	0.068961194
foil	3.5855352	2.1757004	5.7612356	0.015235893	0.0092451293	0.024481022
Net	19.110234	2.8799679	21.990202	0.081204467	0.01223775	0.093442216

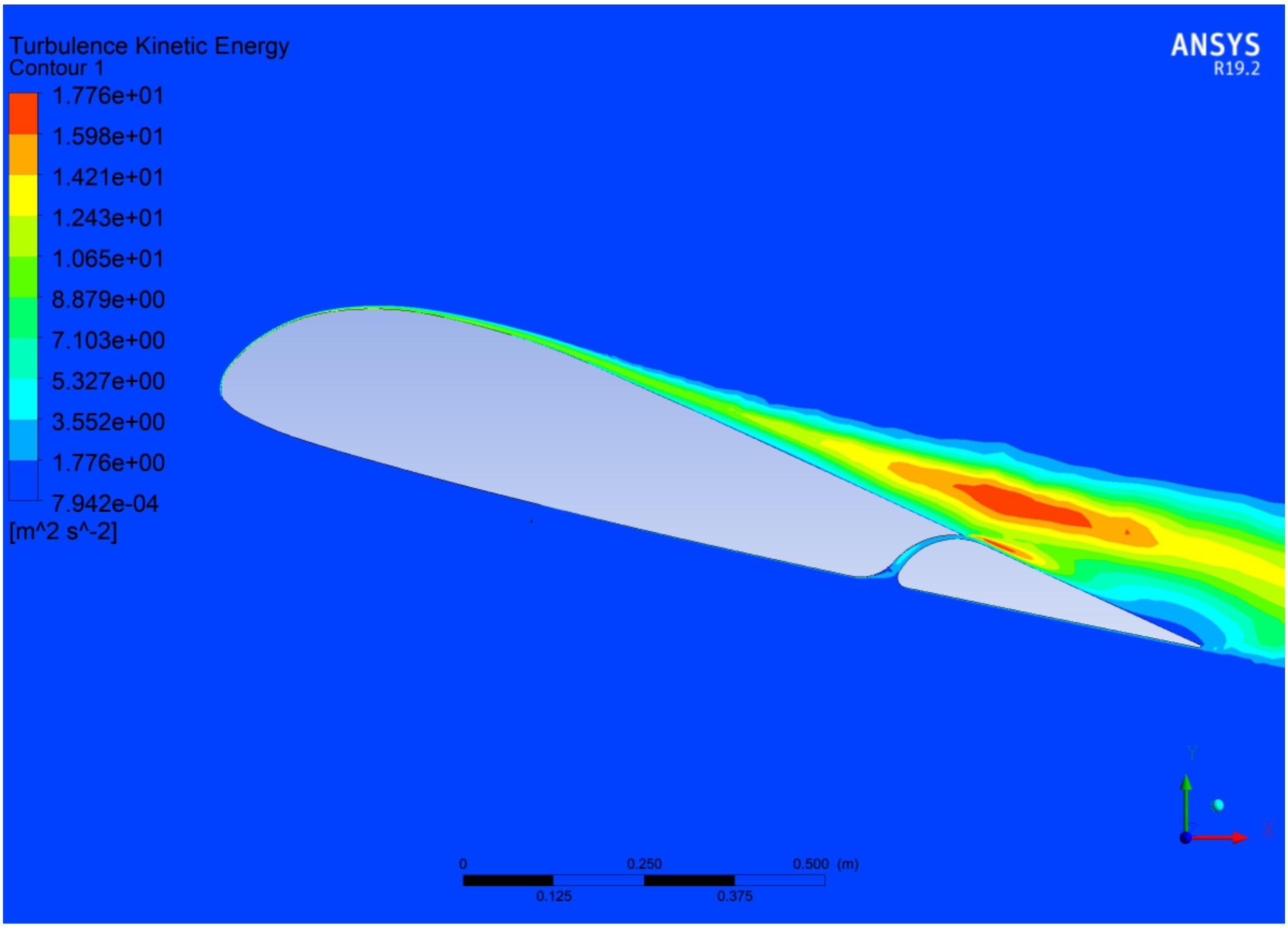
Forces - Direction Vector (0 1 0)
Forces (n) Coefficients

Zone	Pressure	Viscous	Total	Pressure	Viscous	Total
flap	25.640025	-0.12267213	25.517352	0.10895128	-0.00052126	0.10843001
foil	514.11308	0.33045772	514.44354	2.1846032	0.001404202	2.1860074
Net	539.7531	0.20778559	539.96089	2.2935544	0.000882936	2.2944374

Center of Pressure - Set Coordinate y = 0 (m)

Zone	x
Flap	1.2952882
Foil	0.32684114
Net	0.37284552




Turbulence Kinetic Energy
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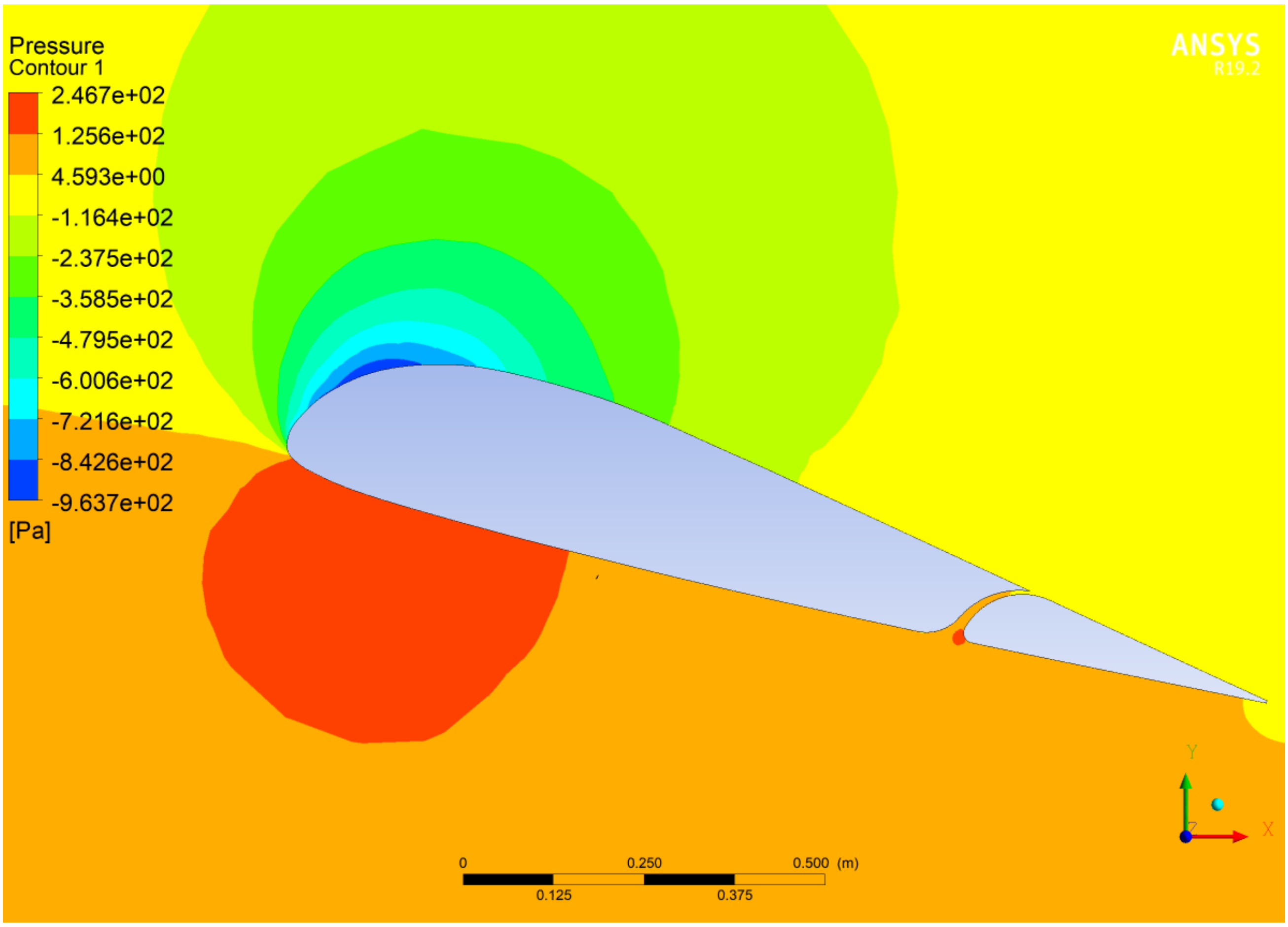
1.598e+01
1.421e+01
1.243e+01
1.065e+01
8.879e+00
7.103e+00
5.327e+00
3.552e+00
1.776e+00
7.942e-04
[m^2 s^-2]
ANSYS
R19.2
0
0.250
0.500 (m)
0.125
0.375


Pressure
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4.593e+00
-1.164e+02
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-6.006e+02
-7.216e+02
-8.426e+02
-9.637e+02
[Pa]
ANSYS
R19.2
0
0.250
0.500 (m)
0.125
0.375

Appendix C

Simulation 1.2

AOA = 15°, AOF = 20°, V = 20m/s, ρ = 1.225, Re = v × l × 70000 = 20 × 1.4 × 70000 = 1960000

Results:

continuity	x-velocity	y-velocity	energy	k	epsilon	drag_flap	lift_flap	lift_foil	drag_foil
9.9746e-04	5.7333e-06	3.2379e-06	1.0997e-09	6.8234e-05	6.8671e-05	2.2073e-01	2.9019e-01	3.3222e+00	-7.7963e-02

Forces - Direction Vector (1 0 0)
Forces (n) Coefficients

Zone	Pressure	Viscous	Total	Pressure	Viscous	Total
flap	51.34151	0.60349069	51.945001	0.21816373	0.002564392	0.22072812
foil	-20.893317	2.5460094	-18.347307	-0.088781257	0.010818671	-0.077962586
Net	30.448193	3.1495001	33.597693	0.12938247	0.013383063	0.14276553

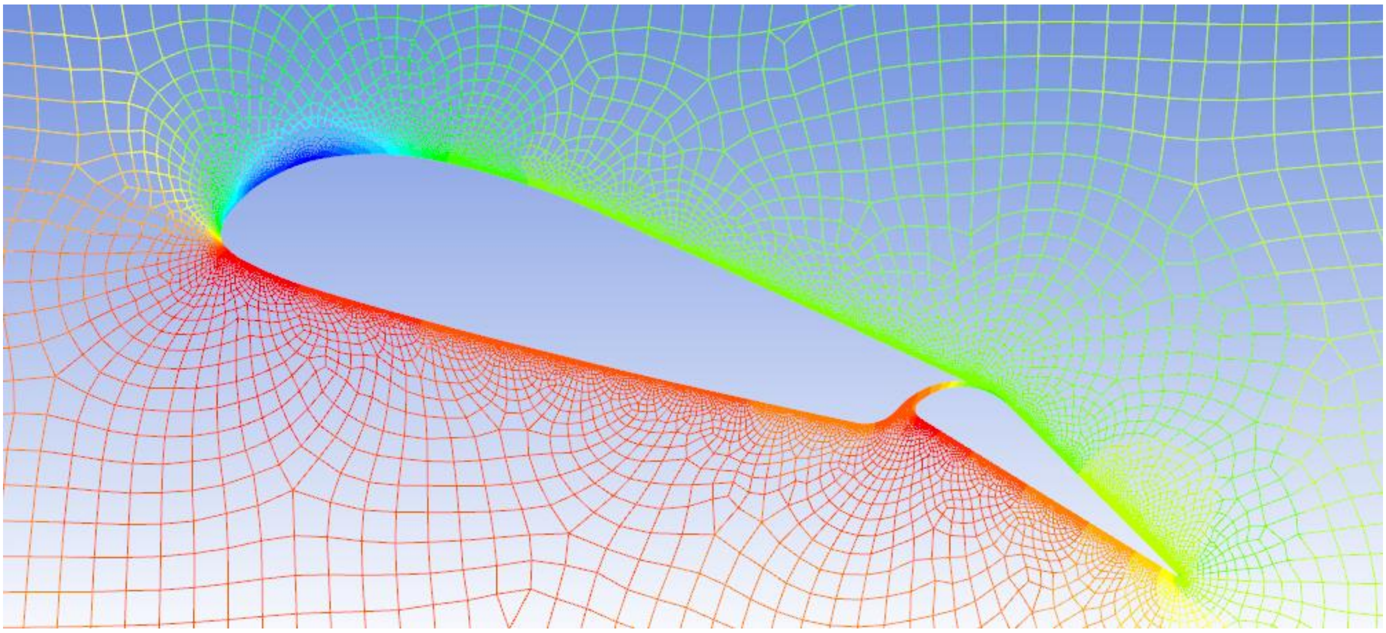
Forces - Direction Vector (0 1 0)
Forces (n) Coefficients

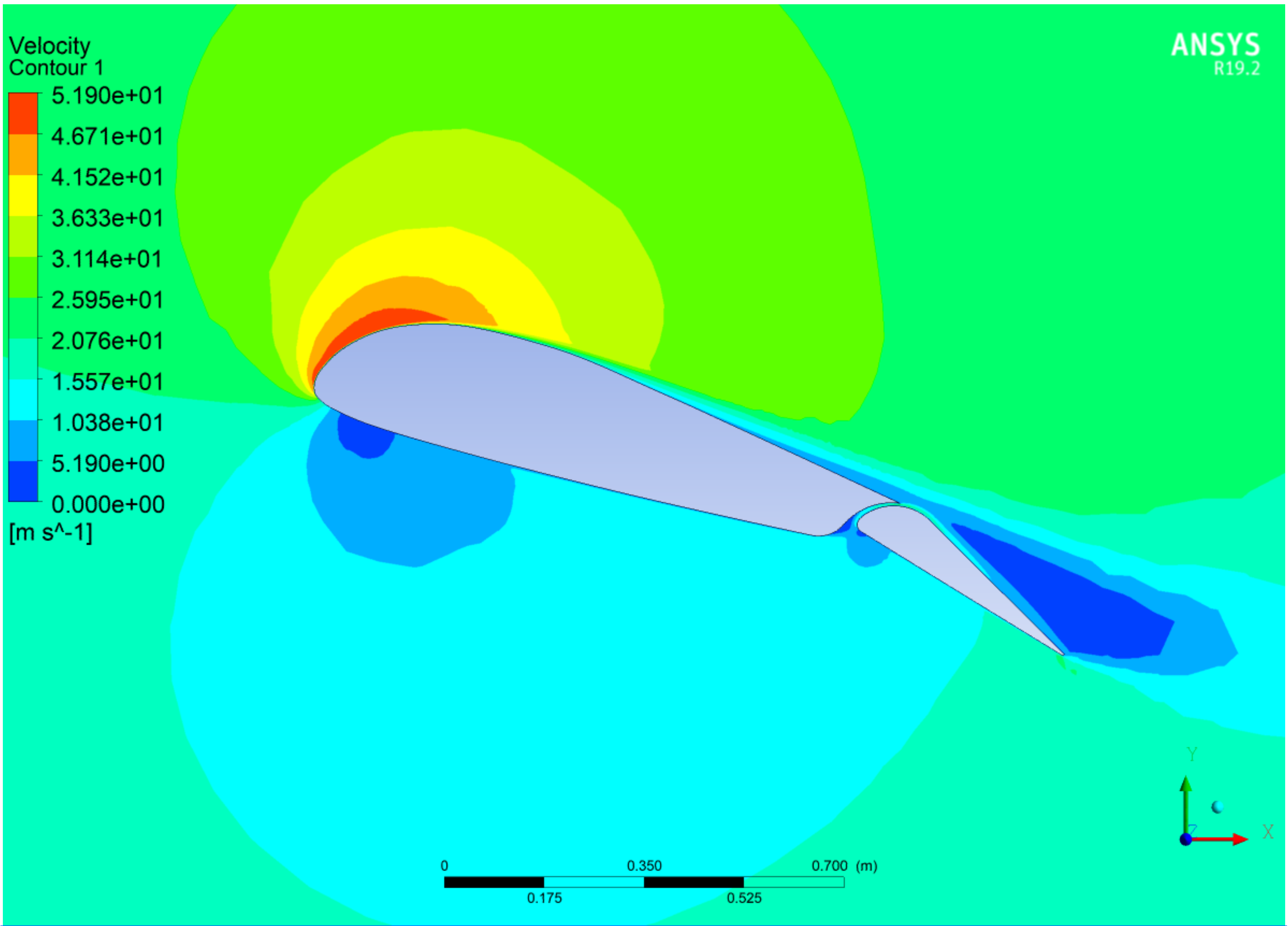
Zone	Pressure	Viscous	Total	Pressure	Viscous	Total
Flap	68.614481	-0.32326207	68.291219	0.29156118	-0.001373626	0.29018755
foil	781.30167	0.5196595	781.82133	3.3199585	0.002208171	3.3221667
Net	849.91616	0.19639743	850.11255	3.6115197	0.000834544	3.6123543

Center of Pressure - Set Coordinate y = 0 (m)

Zone	x
flap	1.339602

Meshing:

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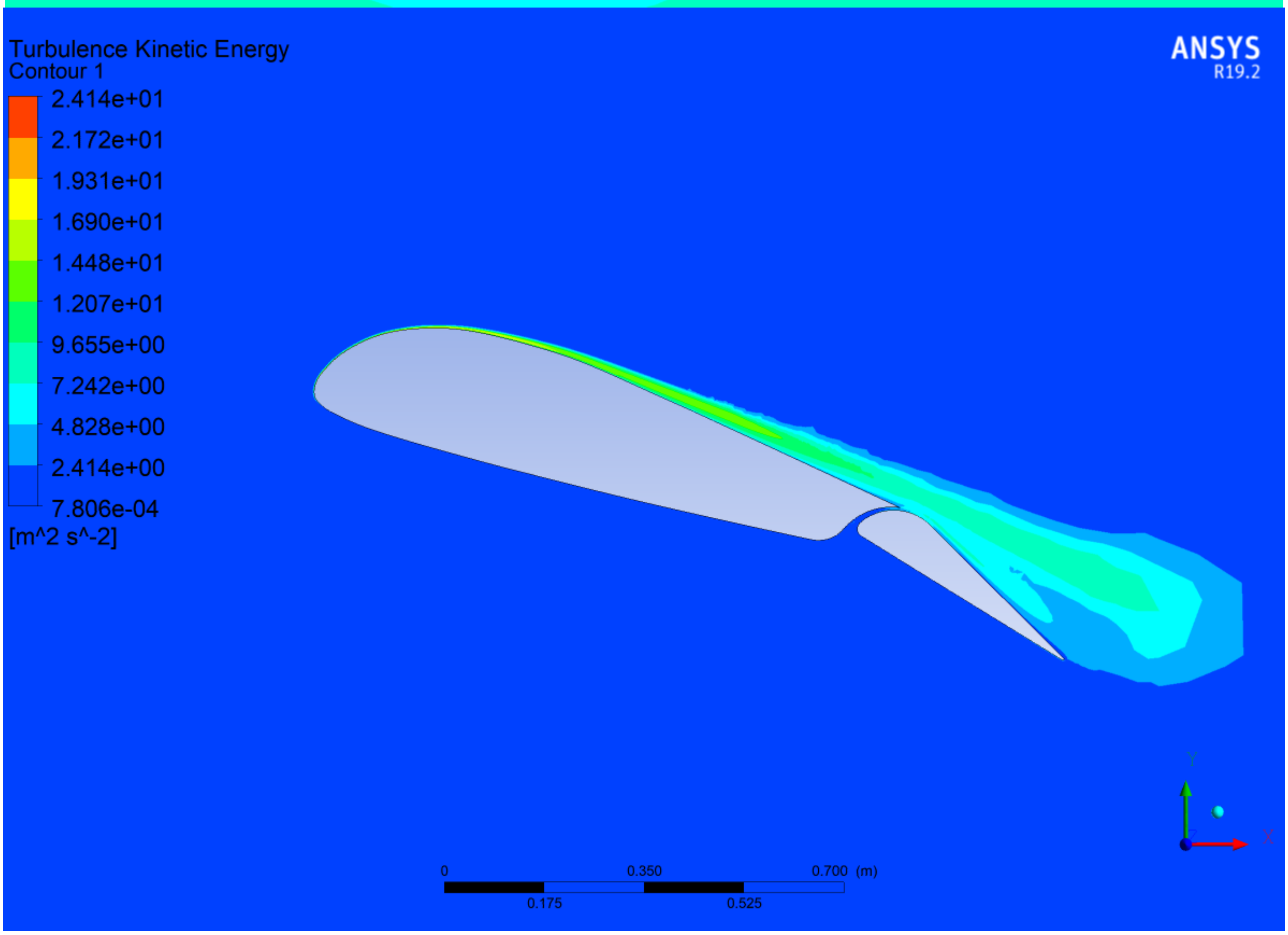
4.152e+01
3.633e+01
3.114e+01
2.595e+01
2.076e+01
1.557e+01
1.038e+01
5.190e+00
0.000e+00
[m s^-1]
ANSYS
R19.2
0
0.350
0.700 (m)
0.175
0.525


Turbulence Kinetic Energy
Contour 1
2.414e+01
2.172e+01
1.931e+01
1.690e+01
1.448e+01
1.207e+01
9.655e+00
7.242e+00
4.828e+00
2.414e+00
7.806e-04
[m^2 s^-2]
ANSYS
R19.2
0
0.350
0.700 (m)
0.175
0.525

Appendix D