МІНІСТЕРСТВО ОСВІТИ ТА НАУКИ УКРАЇНИ
Національний авіаційний університет
Кафедра конструкції літальних апаратів

ДОПУСТИТИ ДО ЗАХИСТУ
Завідувач кафедри, д.т.н., проф.
___________ Сергій ІГНАТОВИЧ
«___» ____________ 2023 р.

КВАЛІФІКАЦІЙНА РОБОТА
ЗДОБУВАЧА ОСВІТНЬОГО СТУПЕНЯ
«БАКАЛАВР»

Тема: «Аванпроект пасажирського середньомагістрального літака місткістю до 170 пасажирів»

Виконав: __________ **Мирослава КИРИК**

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Нормоконтролер: к.т.н, доц. __________ **Володимир КРАСНОПОЛЬСЬКИЙ**

Київ 2023

MINISTRY OF EDUCATION AND SCIENCE OF UKRAINE
National Aviation University
Department of Aircraft Design

PERMISSION TO DEFEND
Head of the department,
Professor, Dr. of Sc.
_________ Sergiy IGNATOVYCH
"___" ____________ 2023

BACHELOR DEGREE THESIS

Topic: "Preliminary design of mid range passenger aircraft with capacity up to 170 passengers"

Fulfilled by:	____________	**Myroslava Kyryk**
Supervisor: **PhD, associate professor**	____________	**Volodymyr KRASNOPOLSKYI**
Standards inspector **PhD, associate professor**	____________	**Volodymyr KRASNOPOLSKYI**

Kyiv 2023

НАЦІОНАЛЬНИЙ АВІАЦІЙНИЙ УНІВЕРСИТЕТ

Аерокосмічний факультет
Кафедра конструкції літальних апаратів
Освітній ступінь «Бакалавр»
Спеціальність 134 «Авіаційна та ракетно-космічна техніка»
Освітньо-професійна програма «Обладнання повітряних суден»

ЗАТВЕРДЖУЮ
Завідувач кафедри, д.т.н, проф.
__________ Сергій ІГНАТОВИЧ
«___» ____________ 2023 р.

ЗАВДАННЯ

на виконання кваліфікаційної роботи здобувача вищої освіти

КИРИК МИРОСЛАВИ ОЛЕГІВНИ

1. Тема роботи: «Аванпроект пасажирського середньомагістрального літака місткістю до 170 пасажирів», затверджена наказом ректора від 1 травня 2023 року № 624/ст.

2. Термін виконання роботи: з 29 травня 2023 р. по 25 червня 2023 р.

3. Вихідні дані до роботи: маса комерційного навантаження 19800 кг, дальність польоту з максимальним комерційним навантаженням 4500 км, крейсерська швидкість польоту 850 км/год, висота польоту 10,5 км, кількість пасажирів 162.

4. Зміст пояснювальної записки: вступ, основна частина, що включає аналіз літаків-прототипів і короткий опис проектованого літака, обґрунтування вихідних даних для розрахунку, розрахунок основних льотно-технічних та геометричних параметрів літака, компонування пасажирської кабіни, розрахунок центрування літака, спеціальна частина, яка містить аналіз конструкції стрінгерів літака прототипу, проектування стрінгера з урахуванням втрати стійкості для забезпечення більшої міцності без збільшення маси.

5. Перелік обов’язкового графічного (ілюстративного) матеріалу: загальний вигляд літака (А1×1), компонувальне креслення фюзеляжу (А1×1).

6. Календарний план-графік:

№	Завдання	Термін виконання	Відмітка про виконання
1	Вибір вихідних даних, аналіз льотно-технічних характеристик літаків-прототипів.	29.05.2023 – 31.05.2023	
2	Вибір та розрахунок параметрів проєктованого літака.	01.06.2023 – 03.06.2023	
3	Виконання компонування літака та розрахунок його центрування.	04.06.2023 – 05.06.2023	
4	Розробка креслень по основній частині дипломної роботи.	06.06.2023 – 07.06.2023	
5	Огляд літератури за проблематикою роботи. Аналіз варіантів завантаження ближньомагістральних літаків.	08.06.2023 – 09.06.2023	
6	Розробка механізму для завантаження негабаритних вантажів.	10.06.2023 – 11.06.2023	
7	Оформлення пояснювальної записки та графічної частини роботи.	12.06.2023 – 14.06.2023	
8	Подача роботи для перевірки на плагіат.	15.06.2023 – 18.06.2023	
9	Попередній захист кваліфікаційної роботи.	19.06.2023	
10	Виправлення зауважень. Підготовка супровідних документів та презентації доповіді.	20.06.2023 – 22.06.2023	
11	Захист дипломної роботи.	23.06.2023 – 25.06.2023	

7. Дата видачі завдання: 29 травня 2023 року

Керівник кваліфікаційної роботи ____________ Володимир КРАСНОПОЛЬСЬКИЙ

Завдання прийняв до виконання ____________ Мирослава КИРИК

NATIONAL AVIATION UNIVERSITY

Aerospace Faculty
Department of Aircraft Design
Educational Degree "Bachelor"
Specialty 134 "Aviation and Aerospace Technologies"
Educational Professional Program "Aircraft Equipment"

APPROVED BY
Head of Department,
Professor Dr. of Sc.
________ Sergiy IGNATOVYCH
‘___’ ____________ 2023

TASK

for the bachelor degree thesis

Myroslava KYRYK

1. Topic: "Preliminary design of mid range passenger aircraft with capacity up to 170 passengers", approved by the Rector’s order № 624/ст from 1 May 2023.

2. Period of work: since 29 May 2023 till 25 June 2023.

3. Initial data: payload 19.8 tons, flight range with maximum capacity 4500 km, cruise speed 850 km/h, flight altitude 10.5 km, passenger number 162.

4. Content (list of topics to be developed): introduction, main part: analysis of prototypes and brief description of designing aircraft, selection of initial data, wing geometry calculation and aircraft layout, landing gear design, engine selection, center of gravity calculation, special part: analyze of structure of the stringer of prototype, designing of stringer taking into account buckling.

5. Required material: general view of the airplane (A1×1), layout of the airplane (A1×1)

6. Thesis schedule:

№	Task	Time limits	Done
1	Selection of initial data, analysis of flight technical characteristics of prototypes aircrafts.	29.05.2023 – 31.05.2023	
2	Selection and calculation of the aircraft designed parameters.	01.06.2023 – 03.06.2023	
3	Performing of aircraft layout and centering calculation.	04.06.2023 – 05.06.2023	
4	Development of drawings on the thesis main part.	06.06.2023 – 07.06.2023	
5	Cargo loading planning analysis for short range aircraft.	08.06.2023 – 09.06.2023	
6	Development of a mechanism for loading of oversized cargo.	10.06.2023 – 11.06.2023	
7	Explanatory note checking, editing, preparation of the diploma work graphic part.	12.06.2023 – 14.06.2023	
8	Submission of the work to plagiarism check.	15.06.2023 – 18.06.2023	
9	Preliminary defense of the thesis.	19.06.2023	
10	Making corrections, preparation of documentation and presentation.	20.06.2023 – 22.06.2023	
11	Defense of the diploma work.	23.06.2023 – 25.06.2023	

7. Date of the task issue: 29 May 2023

Supervisor: ____________ Volodymyr KRASNOPOLSKYI

Student: ____________ Myroslava KYRYK

РЕФЕРАТ

Пояснювальна записка кваліфікаційної роботи бакалавра «Аванпроект пасажирського середньомагістрального літака місткістю до 170 пасажирів»:

66 с., 6 рис., 8 табл., 11 джерел

Дана кваліфікаційна робота присвячена розробці аванпроекту вантажного літака для середньо магістральних авіаліній з можливістю перевезення пасажирів, що відповідає міжнародним стандартам польотів, нормам безпеки, економічності та надійності, а також проектування силового елемента крила з урахуванням втрати стійкості.

В роботі було використано методи аналітичного розрахунку, компьютерного проєктування за допомогою CAD/CAM/CAE систем, розрахунок МКЕ за допомогою пограмного забезпечення.

Практичне значення результату кваліфікаційної роботи полягає в підвищенні ефективності використання маси конструкціїі, шляхом проектування нового перерізу стрінгера, який забезпечить вищу міцність.

Матеріали кваліфікаційної роботи можуть бути використані в навчальному процесі та в практичній діяльності конструкторів спеціалізованих проєктних установ.

Дипломна робота, аванпроєкт літака, компонування, центрування, перепроектування стрінгера, розрахунок напружено-деформованого стану

ABSTRACT

Bachelor degree thesis "Preliminary design of mid range passenger aircraft with capacity up to 170 passengers"

66 pages, 6 figures, 8 tables, 11 references

This thesis is dedicated to preliminary design of mid-range airplane for transportation of passengers and estimation its flight performances as well as designing of structural element of the wing, taking into account buckling.

The design methodology is based on prototype analysis to select the most advanced technical decisions, engineering calculations to get the technical data of designed aircraft and computer based design using CAD/CAM/CAE and FEM systems. In special part the stress analysis is used to estimate allowable stresses of the structural elements.

Practical value of the work is improving the efficiency of using the mass of the structure by designing a new section of the stringer, which provided higher strength. The materials of the bachelor's thesis can be used in the aviation industry and in the educational process of aviation specialties.

Bachelor thesis, preliminary design, cabin layout, center of gravity calculation, stringer redesigning, stress-strain calculation

Format	№	Designation	Name	Quantity	Notes
			<u>General documents</u>		
A4	1	NAU 23 04K 00 00 00 24 TW	Task for work	1	
	2	NAU 23 04K 00 00 00 24	Mid-range passenger aircraft	2	
A1		Sheet 1	General view		
A1		Sheet 2	Fuselage layout		
				1	

	Sh.	№ doc.	Sign	Date	NAU 23 04K 00 00 00 24 EN			
Done by		Kyryk M. O.			Preliminary design of mid range passenger aircraft with capacity up to 170 passengers	list	sheet	sheets
Supervisor		Krasnopolskyi V.S.				Q	9	52
St.control.		Krasnopolskyi V.S.				402 ASF 134		
Head of dep.		Ignatovich S.R.						

CONTENT

INTRODUCTION

Aviation is one of most perspective technical industry which is widely used in different spheres of life: for transportation of passengers and cargo, for military and special purposes, ets. Building of aircrafts is hard and complex process what include many stages and nuances. Important stage of creating is preliminary design of the aircraft.

To make all necessary calculations for preliminary design was chosen narrow body middle range passenger aircraft with capacity 168 people. Future aircraft will have low position of wing, classic tail unit, two engines and tricycle landing gear. Speed of vechicle will be subsonic and distance of flight around 4500 kilometers. To make best construction decisions were chosen three prototypes of designing aircraft, such as Boeing 737-800, Airbus 321-100 and B 757-200.

Main idea of creating passenger aircraft is make profit from selling of it to airlines. To provide good demand, airplane must meet modern requirements, such as: low weight of vehicle, high safety level, easy in operation, comfortable for passengers and relatively cheap.

1. PRELIMINARY DESIGN OF MID-RANGE AIRCRAFT

1.1 Analysis of prototypes and short description of designed aircraft

Aircraft design is an important and complex process that requires a lot of resources, including large investments, a lot of time, and a large number of skilled engineers. Therefore, the design of the aircraft must be justified by the feasibility of the aircraft and the ability to meet all the objectives and modern requirements.

Due of this, the design of each aircraft begins with the collection of the necessary statistical data, which include market dynamics, the purchasing power of airlines, and the most popular aircraft modules. From the point of view of design, it is necessary to examine all aircraft of a similar configuration in order to prevent the most frequent design errors, assess the viability of design solutions, and understand the basic requirements for operation. Also, the evaluation of prototypes will help to improve the efficiency of the design based on the already known characteristics of the aircraft.

As prototypes for future aircraft were chosen three middle range passenger airplanes Boeing 737-800, Airbus 321-100 and Boeing 757-200. Necessary parameters are shown in table 1.1

Table 1.1

Statistic data of prototype

Name and dimensionality	B 737-800	A 321-100	B 757-200	Deseighned aircraft
1	2	3	4	5
Max payload, [kg]	20 540	21 200	25 970	19800
Crew, number of pilot	4/2	2/2	2/2	2
Passengers (max)	189	220 - 236	224	162

Continuation of table 1.1

1	2	3	4	5
Wing loading, [kN/m^2]	6.32	7.638	5.33	5.402
Flight range with max paylod, [km]	5400	5900	5500	4500
Cruise Altitude, [km]	12.497	11.900	12800	10500
Thrust/weight ratio, [kN/kg]	3	3.2	3.9	3.3
Number of engines and their type	2 ×CFMI CFM56-7B24/26,	2×CFMI CFM56 – 5A/5B, 2×IAE V2500 – A5	2 × P&W PW2040, 2 × R-R RB211-535E4B,	2× RB211
Pressure ratio	32.3	35.4	31.2	32.30
Take off distance, [m]	2550	2180	2350	2550
Landing distance, [m]	1190	1580	1550	1287
MTOW Maximum Take Off Mass, [kg]	78 240 - 79 000	89 000 - 93 500	98 880	96090
Landing mass, [kg]	63 320	75 500 - 77 800	89 900	77985
Empty weight, [kg]	41 140	48 100	58 390	50593
Fuel fraction, % Total fuel/MTOW	33%	32%	44%	27%
Payload fraction, % Maximum payload/MTOW	26%	22%	26%	20%
Wing span, [m]	34.3	34.1 - 35.8	38.0	43.77
Sweepback angle at ¼ of the chord, [°]	25	25	25	25
Fuselage length, [m]	39.5	44.51	47.3	43.355

Ending of the table 1.1

1	2	3	4	5
Fuselage diameter, [m]	3.76	3.95	3.76	3.77
Vertical tail height, [m]	12.57	11.76	13.6	8.7

1.2. Classification of designed aircraft according to the flight performances and layout.

Before start of the work on design of aircraft need to be classified. (table 1.2)

Table 1.2

Classification of aircraft

Aircraft classification according to the	
1	2
1. Purpose of the aircraft	passenger commercial aircraft
2. Speed of flight (Much number)	subsonic aircraft
3. Range of flight	Middle range
4. ICAO category	transport
5. FAA Airplane Design Group	III
6. ICAO / EASA Aerodrome Reference Code	5D
7. Aircraft Approach Category	C category
8. Maximum take-off mass	Medium
Describe the aerodynamic scheme of the aircraft	
Monoplane or biplane	Monoplane
High/mid/low wing position	Low wing position
Swept or straight wing	Swept wing
Cantilever or braced	Cantilever

Ending of the table 1.2

1	2
With or without winglets	Without winglets
Type of the fuselage cross-section	Circular
Wide-body or narrow-body	Narrow body
T-type or conventional type of the tail unit	Conventional
What type of stabilizer: fixed or adjustable	Fixed
Type of landing gear scheme: tricycle with nose wheel or multi bogie	tricycle
How many engines?	2
What type of engines?	Turbojet
Where the engines are located?	Under the wing

1.3 Brief description of the aircraft main parts

Desighned aircraft consist of fuselage with circular cross section, cantilever low-wing, conventional tail unit, turbofan engines and tricycle landing gear with a front single-strut landing gear and two main gears. Wing and empennage are equipped with primary control surfaces.

1.3.1. Wing

The wing is the main plane of the aircraft, which is designed to create the aerodynamic lifting force necessary in flight. It`s important to choose correct geometrical characteristic of the wing such as shape of the wing, type of airfoil, thickness of the wing, because that characteristics have influence on control, stability and maneuverability of the aircraft. For project was chosen sweep wing, because this type of shape gives such advantages as: reduction of parasitic drag, possibility to increase speed of the aircraft, better maneuverability The wing is equipped by ailerons, double slotted flaps and slats. [1]

The structure of the wing inclides rids, spars, stringers and skin. Correct type of internal design must be provided to meet present day requirements to safety of aircraft. Wing must withstand loads during flight, standing on the ground, taxing and emergency situation without failure. Structural elements of the wing perceive bending and torsion caused by lift force Torsion box with two spars was chosen, because such type of construction provide necessary strength and in the same time is most economic effective.

1.3.2. Fuselage

The fuselage has semimonocoque desighn with circular shape of the cross-section. The wing, horizonlat stabilizer and fin are attached to the fuselage. Nose gear of the landing gear is hiding in the fuselage during take-off.

The fuselage frame consist of formers, stringets, skin and keel beam. All structural elements are manufactured from extruded profiles, formed sections and rolled sheats. Frame of the fuselage is made from aluminum alloy. Formers and keel beam are made from B95 aluminum alloy and stringers and skin are made from D16 alluminum alloy.

The fuselage houses the cockpit, passenger cabin, some systems of the airplanes and baggage. Baggage compartment will be located under floor of passenger cabin.

1.3.3 Tail unit

Tail unit includes vertical and horizontal stabilizers. Vertical stabilizer consist of two parts: movable and unmovable. Unmovable part of the vertical stabilizer is also called fin. Movable part of the vertical stabilizer is also called rudder. Horizontal stabilizer consist of two parts: movable and unmovable. Movable part of the vertical stabilizer is also called elevator.

The airfoil of vertical stabilizer is symmetrical. The airfoil of vertical stabilizer is unsymmetrical with inverse camber. Type of airfoil of the horizontal stabilizer couses opposite directed lift forse to provide lateral stability of the airplane during flight.

The tail unit also houses de-icing system, sensors, measurement and communication systems that transmit critical information to pilots and aircraft control systems

1.3.4 Landing gear

The designed airplane is equipped with a tricycle-type landing gear. Such configuration of undercarriage consists of two main landing gear assemblies located beneath the wings and a nose landing gear assembly positioned at the front of the aircraft. Each main landing gear assembly includes two struts, one outboard and one inboard, with wheels attached at the ends. Landing gear are designed to absorb shocks during take-off and landing.

Landing gear of designed airplane is retractable. Such design prevents creation of parasitic drag during flight because arter tale-off pilot initiates the retraction process and gears don`t create no more resistance for airflow move below airplane during flight.

1.3.5. Power plant

The power plane of the designed airplane is Rolls Royce RB211-524H. The RB211 is a high-bypass turbofan engine with a three-shaft design. An axial compressor consists of multiple stages of rotating airfoils, known as compressor blades or rotor blades, and stationary airfoils called stator blades. The rotor blades rotate and draw in air, while the stator blades help guide and direct the airflow. As the air passes through each stage, its pressure and velocity increase, resulting in compressed air.

The RB211 engines are typically certified to operate on Jet A/A-1 and Jet B fuels, which are commonly used in commercial aviation.The RB211 engine series employs a multi-spool configuration, meaning that each set of compressor and turbine stages operates independently on separate shafts.

Table 1.3

Examples of application RB211

Engine model	Overal Length (mm)	Overal Width (mm)	Dry Spec. Weight (kg)	Maximum Take-off Power		Normal Take-off Power		Maximum Continuous Power	
				Shaft Power (kW)	Maximum Air Temp for Rated Power (°C)	Shaft Power (kW)	Maximum Air Temp for Rated Power (°C)	Shaft Power (kW)	Maximum Air Temp for Rated Power (°C)
RB211	4500	2650	5500	44700	20	31000	28	28300	35

Conclusions to the analytical part:

Selection and analysis of prototypes is a mandatory step when designing a new aircraft. Because a competent choice of prototypes can greatly facilitate further design. Based on the prototypes, the main characteristics of the aircraft were chosen: the shape of the wing, the type of chassis, the shape of the cross-section of the fuselage, the type of tail unit and the power plant

2. AIRCRAFT GEOMETRY CALCULATION

Preliminary design is necessary from a technical and economic point of view.

During the preliminary design of the airplane, the basic geometric parameters are laid down: shape and proportions of the wings, fuselage length and size its cross-section, layout of the passenger cabin, type and dimensions of the tail unit, area of the main control surfaces and high lift devices.

To provide expedient configuration of all parts of designing aircraft in the next paragraph will be shown calculation of the wing design and high lift devices calculations, the fuselage geometry and cabin layout, landing gear design, tail unit design.

2.1. Wing geometry calculation

For the designing aircraft, the initial data have been calculated by special computer program designed at the Aircraft Design Department of NAU. The data are presented in Appendix A.

1. Wing airfoil: For designing aircraft supercritical airfoil was taken.
2. Reletive thickness of the airfoil is 0.120.
3. Location of the wing on fuselage: low wing.
4. Aspect ratio of the wing λ_w=9.45
5. Taper ratio of the wing η_w=3.49
6. Sweep back angle of a wing is 25 degrees.
7. Wing area:

$$S_{wing} = \frac{m_0 \cdot g}{P_o} = \frac{96096 \cdot 9.8}{4650} = 202.7,\ [\mathrm{m}^2]$$

where m_o – take-off mass of the aircraft;

g –acceleration of gravity;

P_o – wing loading at cruise regime of flight.

After the calculation of wing area, value of it was compared with area of prototype.

So, the wing area S_{wing} is taken 202.7m^2.

8. Wing span is:

$$l = \sqrt{S_{wing} \cdot \lambda_w} = \sqrt{202.7 \cdot 9.45} = 43.77,\ [\text{m}]$$

9. Root chord is:

$$C_{root} = \frac{2 \cdot S_{wing} \cdot \eta_w}{(1+\eta_w) \cdot l} = \frac{2 \cdot 202.7 \cdot 3.49}{(1+3.49) \cdot 43.77} = 7.2,\ [\text{m}]$$

10. Tip chord is:

$$C_{tip} = \frac{C_{root}}{\eta_w} = \frac{7.2}{3.49} = 2.06,\ [\text{m}]$$

11. On board chord for trapezoidal shaped wing is:

$$C_{board} = C_{root} \cdot (1 - \frac{(\eta_w - 1) \cdot D_f}{\eta_w \cdot l}) = 7.2 \cdot (1 - \frac{(3.49-1) \cdot 5.64}{3.49 \cdot 43.77}) = 6.54,\ [\text{m}]$$

12. Wing construction and spars position.

For a wing with two spars relative coordination of the spar's position is equal to:

x_{1spar}=0.2·6.54=1.4 and x_{2spar}=0.6·6.56=4.3 from the leading edge of current chord in the wing cross-section. The spars are shown at the drawing (appendix B).

13. Determination of mean aerodynamic chord.

Mean aerodynamic chord is specific chord of the wing about which the aerodynamic moment is constant and independent angle of attack.[2] Mean aerodynamic chord is used to determine of center of pressure of the wing, stability and controllability scheme.

The geometrical method of mean aerodynamic chord determination has been used, which is presented at Appendix B and at the fig.2.1.

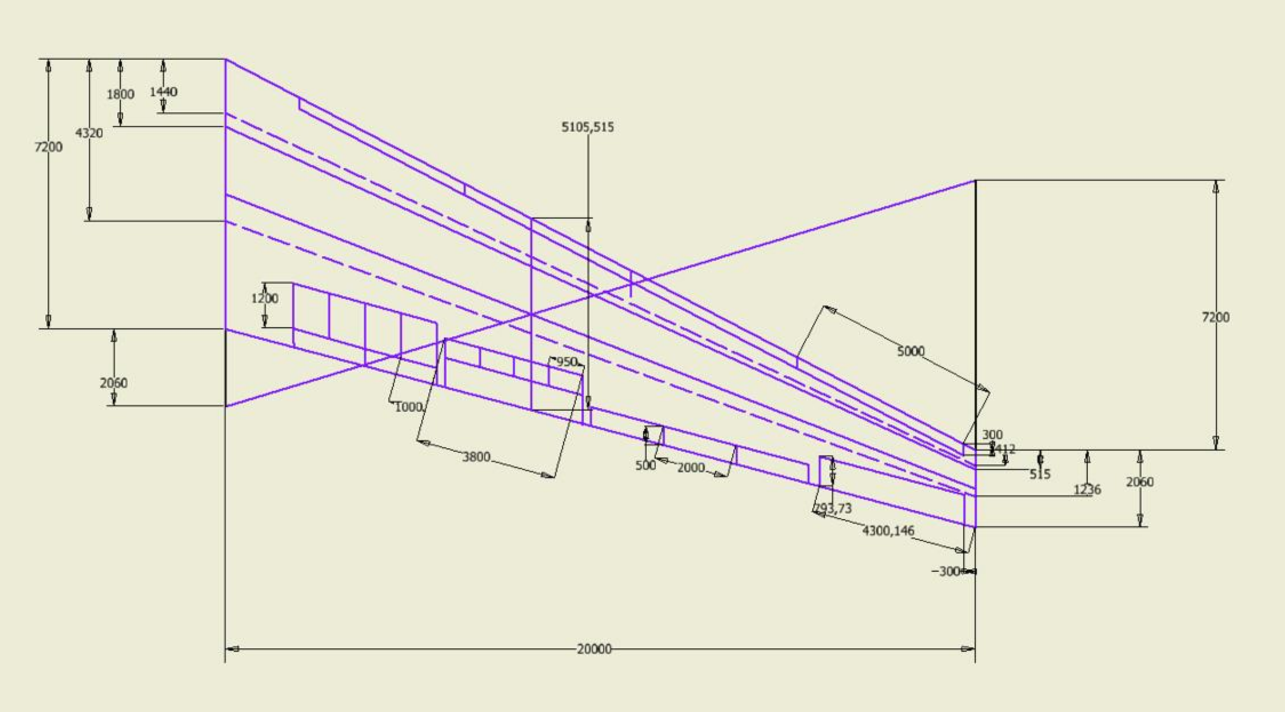


Figure 2.1 Geometrical method of determination mean aerodynamic chord

Mean aerodynamic chord is equal $b_{\max} = 5.106$, [m].

Also to provide more accuracy the MAC is calculated by the approximately formula for trapezoidal wing shape to provide checking:

$$b_{MAC} = \frac{2}{3}\frac{C_{root}^2 + C_{root}C_{tip} + C_{tip}^2}{C_{root} + C_{tip}} = \frac{2}{3}\cdot\frac{7.2^2 + 7.2\cdot 2.06 + 2.06^2}{7.2 + 2.06} = \frac{2}{3}\cdot\frac{70.9}{9.26} = 5.104, \text{ [m]}$$

Modern aircraft needs to provide controllability during flight, so next step of calculation will be determination of design of primary control surface of wing.

14. Ailerons design.

Ailerons are primary control surface of the aticraft, which is created to provide roll moment about longitudinal axis of the aircraft. Ailerons are used to provide contolability of aircraft in lateral direction.

Geometrical characteristic will be determined by next formulas:

Ailerons span: $l_{aileron} = 0.35\cdot\frac{l}{2} = 0.35\cdot\frac{43.77}{2} = 7.66$, [m] Aileron area: $S_{aileron} = 0.06 \cdot \frac{S_w}{2} = 0.06 \cdot \frac{202.7}{2} = 6.08,\ [\text{m}]$

Aerodynamic compensation is necessary to improve controllability of the aircraft. Aileron trim tab provide aerodynamic compensation by creating a rotation moment about the hinge, which helps reduce air pressure on the control surface and increase the controllability of the aircraft.

Area of aileron's trim tab for two engine airplane:

$$S_{trim\ tabs} = 0.06 \cdot S_{aileron} = 0.06 \cdot 6.08 = 0.36,\ [\text{m}^2]$$

15. High lift device of a wing: Double slotted flaps together with slats.

Slats and flaps are high lift devices. Slats are designed to increase maximum angle of attack that allow increase lift force on the wing without stalling.

Flaps are designed to increase lift force on a wing during landing and take-off, because during low speed flight wing generate not enough lift force, and to provide stable flight pilot need to increase area and camber of the wing to compensate decreasing of speed. Double slotted flaps, located at the trailing edge of the wing. It is designed to increase area of a wing and air flow from down surface of a wing come through slots in a flap on upper surface and reinforce upper air flow to reduce possibility of stalling.

From the list of possible coefficient of high lift devices was chosen one which corresponds to high lift devises of designed aircraft.

1.05 – double slotted Fayler flaps in combination with slats;

The relative coordination of high-lift devices on the wing chord are:

For double slotted flaps:

$C_f = 0.3 \cdot 5.106 = 1.532$

For slats:

$C_s = 0.1 \cdot 5.106 = 0.51$

2.2 Fuselage layout

Fuselage is main body of an aircraft which provide space for cargo and passengers. Cross-section of fuselage of designed airplane will be circular which has better resistance from torque which created by deflection of rudder, gusts of side wind ets. Such type of cross-section will be realized by formers, stringers and skin. Formers carry shear forces caused by vertical loads transmitted from the wing, pressurization and cargo. Skin carries tension and compression from bending, tension caused by pressurization and shear caused by bending and torsion. Stringers undergo bending from tail unit and reinforce skin to avoid buckling.

The designed aircraft will be passenger one, necessary passenger layout must be provided. Fuselage layout consist of a comfortable seats for passengers, lavatories, galleys and space for baggage.

In this part will be calculated all necessary geometrical characteristics for passenger cabin to provide comfortable flight and to meet present day requirements of safety and economical profits.

Such geometrical parameters as diameter of fuselage, length of nose and tail pat, length of the cabin, fineness ratio will be calculated.

1. Length of the aircraft fuselage:

1. $$L_{fus} = FR \cdot D_{fus} = 11.5 \cdot 3.77 = 43.355, \text{ [m]}$$

FR – fineness ratio of the fuselage.

D_{fus} – diameter of the fuselage

2. Length of aircraft fuselage forward part:

Necessary fineness ration of nose and tail part are taken from prototype.

$$L_{tail} = FR_{tu} \cdot D_{fus} = 1.27 \cdot 3.76 = 4.7, \text{ [m]}$$

FR_{nose} – finesses ratio of nose part

2. Length of the fuselage tail part:

$$L_{tail} = FR_{tu} \cdot D_{fus} = 1.6 \cdot 3.76 = 6.02, \text{ [m]}$$

FR_{tu} – finesses ratio of tail part

3. Cabin width.

For economic class cabin was chosen the passenger seat as 3 + 3 each row.

$$B_{cab} = n_3 \cdot b_3 + n_{aisle} \cdot b_{aisle} + 2 \cdot \delta + 2 \cdot \delta_{wall} =$$
$$= 2 \cdot 1450 + 1 \cdot 460 + 2 \cdot 50 + 2 \cdot 100 = 3660, \text{[mm]}$$

where n_3 – number of 3 chair block;

b_3 – width of 3 chairs; ;

b_{aisle} – width of aisle;

δ – distance between external armrests to the decorative panels, mm;

δ_{wall} – width of the wall (fuselage structure, insolation, decorative panels), mm.

For the business class was chosen the passenger seat as 2 + 2 each row.

The appropriate width of business class cabin:

$$B_{cab} = n_2 \cdot b_2 + n_{aisle} \cdot b_{aisle} + 2 \cdot \delta + 2 \cdot \delta_{wall} =$$
$$= 2 \cdot 1340 + 1 \cdot 600 + 2 \cdot 90 + 2 \cdot 100 = 3660, \text{[mm]}$$

where n_2 – number of 2 chair block;

b_2 – width of 2 chairs.

4. Cabin height.

To comfortable flight of passengers need calculate necessary height of cabin.

$$H_{cab} = 1.48 + 0.17 \cdot B_{cab} = 1.48 + 0.17 \cdot 3.76 = 2.12, \text{[m]}$$

where B_{cab} - width of the cabin.

For economic profit aircraft must be designed not only comfortable but also to provide good impression for passengers during flight, that`s why windows was placed in one row to give possibility for passengers to see the view during flight. Windows will be rectangular shape with rounded corners to reduce stress concentrators and fatique propagation. Windows located between formers at distance about 550 mm.

5. Length of the cabin

To organize passenger cabin is necessary to choose correct seat pitch relatively to class of the cabin and flight range.

Cabin length $L_{cab.}$ for typical accommodation with constant seat pitch L_{seat}

$$L_{cab} = L_1 + (N-1)L_{seat} + L_2$$

N – number of the seats rows;

L_{seat} – length of the seat pitch;

L_1 – distance from the wall to the back of the seat in first row, mm;

L_2 – distance from the back of the seat in the last row to the wall, mm.

The length of economic passenger cabin is:

$$L_{econ} = L_1 + (N-1)\cdot L_{seat} + L_2 =$$
$$= 1200 + (23-1)\cdot 800 + 300 = 19100, \text{[mm]}$$

In economic class cabin will be 23 passengers rows.

The length of business passenger cabin is equal to:

$$L_{busi} = L_1 + (N-1)\cdot L_{seat} + L_2 =$$
$$= 1200 + (5-1)\cdot 860 + 300 = 4940, \text{[mm]}$$

In business class cabin will be 5 rows.

6. Baggage compartment

As designed aircraft belong to transport category airplane must be provided place for baggage of passengers. Weight and location of baggage influence on gravity center position.

Unit of load on floor is $K = 400\ldots 600$, $[\frac{kg}{m^2}]$

The area of cargo compartment is determined as

$$S_{cargo} = \frac{M_{bag}}{0.4K} + \frac{M_{cargo\&mail}}{0.6K} = \frac{20\cdot 168}{0.4\cdot 600} + \frac{15\cdot 168}{0.6\cdot 600} = 21, \text{[m}^2\text{]}$$

M_{bag} – mass of baggages of all passengers;

M_{bag}=m n_{pass};

m – mass of baggage for one passenger for free;

n_{pass} – number of passengers;

$M_{cargo\ \&mail}$ – mass of additional cargo and mails on the board of aircraft, approximately 15 kilograms for each passenger.

Cargo compartment volume is equal to:

$$V_{cargo} = \nu \cdot n_{pass} = 0.2 \cdot 168 = 33.6,\ [\mathrm{m}^3]$$

where n_{pass} – number of the passenger

Baggage compartment is similar to prototype.

7. Galleys and buffets

As designed aircraft will be middle range one, airlines need provide nutrition for passengers so galleys and buffets take a place in airplane. Aircraft must be designed with accounting of galley and buffets.

To make easy delivery and access to food for flight attend buffets was placed near to economic class and business class.

According to international standards, the volume of the galleys should be about 0.1 cubic meter per passenger, so the volume of galleys should be:

$$V_{galley} = 0.1 \cdot n_{pass} = 0.1 \cdot 168 = 16.8,\ [\mathrm{m}^3]$$

The total area of galley floor:

$$S_{galley} = \frac{V_{galley}}{H_{cab}} = \frac{20}{2.12} = 9.43,\ [\mathrm{m}^2]$$

8. Lavatories

Determination of the time of flight

$$t = \frac{Rangeflight}{V_{cruise}} + 0.5 = \frac{5200}{828} + 0.5 = 6.78,\ [\mathrm{h}]$$

where V_{cruise} – speed of flight of airplane

The number of lavatories is chosen according to the calculated value.

$$N_{lavatory} = \frac{N_{passenger}}{40} = \frac{162}{40} > 4$$

Area of lavatory:

$$S_{lav} = 1.5,\ [\mathrm{m}^2]$$

Width of lavatory: 1m. Toilets design is similar to the prototype.

On designed aircraft was placed 4 galleys and 5 lavatories.

2.3 Layout and calculation of basic parameters of tail unit.

Tail unit is part of airplane created to provide stability and controllability during flight. So choice correct layout and position of empennage is important part of preliminary design.

For designed airplane was chosen conventional tail unit configuration with sweepback angle 25 degrees. Tail unit equipped with such primary control surfaces as rudder and elevator. Rudder mounted on fin to create rotation moment about vertical exes to make yaw. Elevator used to provide pitch and descend aircraft during landing.

Area of horizontal tail unit is:

$$S_{HTU}=(0.18..0.25)S = 36.486 \ldots 50.675,\ [\mathrm{m}^2]$$

Area of vertical tail unit is*:*

$$S_{VTU}=(0.12..0.20)S = 24.324 \ldots 40.54,\ [\mathrm{m}^2]$$

For more exact:

$$S_{HTU} = \frac{b_{mac} \cdot S}{L_{htu}} \cdot A_{HTU} = \frac{5.106 \cdot 202.7}{19.2} \cdot 0.8 = 42.03,\ [\mathrm{m}^2]$$

where L_{HTU} – length of horizontal tail unit;

A_{HTU} – coefficient of static momentum of horizontal tail unit;

l, S – wing span and wing area.

$$S_{VTU} = \frac{l \cdot S}{L_{vyu}} \cdot A_{VTU} = \frac{43.77 \cdot 202.7}{19.2} \cdot 0.1 = 46.2,\ [\mathrm{m}^2]$$

where L_{VTU} – length of vertical tail unit;

A_{VTU} – coefficient of static momentum of vertical tail unit.

Values L_{HTU} and L_{VTU} depend on some factors. First of all their value are influenced by: the length of the nose part and tail part of the fuselage, sweptback and wing location, and also from the conditions of stability and control of the airplane.

In the first iteration $L_{HTU} \approx L_{VTU}$ may be supposed and it may be found from the dependences:

Trapezoidal scheme, normal scheme L_{VTU} = (0.2..3.5) b_{mac}

$$L_{VTU} = 3\cdot5.106=15.32, \text{[m]}$$

Determination of the elevator and rudder area:

Elevator area is:

$$S_{el}=(0.3..0.4)\ S_{HTU=}42.03\cdot0.35=14.71, [\text{m}^2]$$

Rudder area is:

$$S_{rudder}=(0.2..0.22)\ S_{VTU}=0.2\cdot46.2=9.24, [\text{m}^2]$$

Choice of aerodynamic balance area.

$$0.3\leq M\leq0.75$$

$$S_{ab\ el}=(0.22..0.25)\cdot S_{el}=0.23\cdot17.71=3.38, [\text{m}^2]$$

$$S_{ab\ rudder}=(0.2..0.22)\cdot S_{rudder=}0.2\cdot9.24=1.85, [\text{m}^2]$$

Determination of the tail nit span.

Tail Unit span is related to the following dependence:

$$l_{HTU}=(0.32..0.5)\cdot l_{wing=}0.5\cdot43.77=21.885, \text{[m]}$$

In this dependence the lower limit corresponds to the turbo jet engine aircraft, equipped with all-moving stabilization.

The height of the vertical TU h_{vtu} is determined according to the location of the engines. Taking it into account we assume:

Low wing, EonW, M<1

$$h_{vtu}=(0.14..0.2)\cdot l_w=0.16\cdot 43.77=7, \text{ [m]}$$

For low wing airplanes upper limit must be setted.

Tapper ratio of horizontal and vertical TU must be chosen:

For planes $M<1$

$\eta_{htu}=2.5$

$\eta_{vtu}=1.25$

TU aspect ratio

For transonic planes $\lambda_{vtu}=1$, $\lambda_{htu}=4$

Determination of TU chords b_{end}, b_{CAX}, b_{root}:

$$b_{tip}=\frac{2\cdot S_{htu}}{(\eta_{htu}+1)\cdot l_{htu}}=\frac{2\cdot 42.03}{(2.5+1)\cdot 21.885}=1.1, \text{ [m]}$$

$$b_{mac}=0.66\cdot\frac{\eta_{htu}^2+\eta_{htu}+1}{\eta_{htu}+1}\cdot b_{htutip}=2.02, \text{ [m]}$$

$$b_{root}=b_{tip}\cdot\eta_{htu}=2.75, \text{ [m]}$$

TU sweptback.

TU sweptback is taken in the range 3..5°, and not more than wing sweptback. It is necessary to provide the control of the airplane in shock stall on the wing.

2.4. Calculation of basic parameters and layout of landing gear.

Landing gear is one of the main part of aircraft which designed to absorb shocks during landing and take-off, provide stability during taxi on aerodrome, integrity of structure during stand. For designed aircraft was chosen tricycle type of landing gear to prevent nose-over during landing, improve braking process and prevent rotation during taxing.

To accurately estimate the outline of the landing gear in this project, it is essential to calculate the relative positions of each strut in relation to one another.

This calculation helps determine the loads that the landing gear system will experience and its optimal location with respect to the airplane's center of gravity. The layout of the landing gear in this project closely follows the prototype data, ensuring that the fundamental design of the landing gear system is based on proven and reliable information. Parameters of aircraft landing gear is shown on figure 2.4 [3]

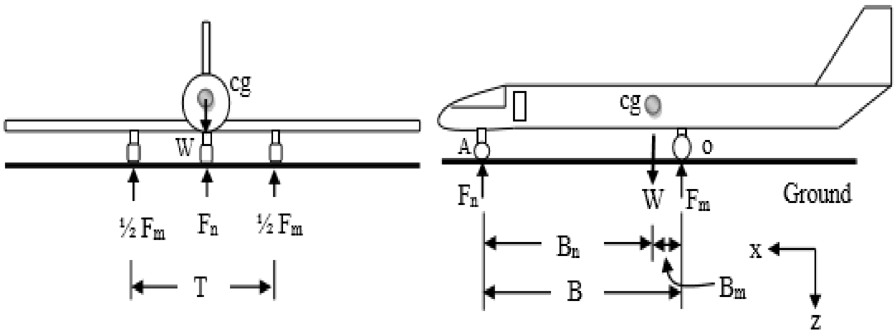


Figure 2.2. Landing gear parameters

The distance from the centre of gravity to the main LG

$$B_m = (0.15..0.20)\cdot b_{MAC}=0,2\cdot 5,106=1,\ [m]$$

Landing gear wheel base comes from the expression:

$$B = (0.3..0.4)\cdot l_f = (6..10)\cdot Bm=43,355\cdot 0.4=19.16,\ [m]$$

Distance from nose wheel to center of gravity may be find as

$$Bn = B\text{-}Bm = 19.16\text{-}1=18.16,\ [m]$$

Wheel track is:

$$T = (0.7..1.2)\cdot B \leq 12,\ [m]$$

$$T=0.7\cdot 11.3=7.9,\ [m]$$

Wheels is important part of landing gear, because it must withstand loads during as usual take-off and landing as emergency situation and not disturb integrity of airplane.

Tires are chosen from the list of all possible tires according to the landing speed and maximum load.

The load on the wheel is determined:

$$F_{main} = \frac{(B - B_m) \cdot m_0 \cdot 9.81}{B \cdot n \cdot z} = \frac{(15.1745 - 0.77) \cdot 96090 \cdot 9.81}{15.1745 \cdot 2 \cdot 2} =$$
$$= 223361.1, \text{[N]} = 50213.55, \text{[lbs]}$$

$$F_{nose} = \frac{B_m \cdot m_0 \cdot 9.81 \cdot K_g}{B \cdot z} = \frac{0.77 \cdot 96090 \cdot 9.81 \cdot 2}{15.1745 \cdot 2} =$$
$$= 44278.63, \text{[N]} = 9954, \text{[lbs]}$$

where n, and z – is the quantity of the supports and wheels on the one gear.

K_g = 1.5..2.0 – dynamics coefficient.

Tires are chosen according to the maximum load and landing speed from the Michelin data table. In table 2.4.1, table 2.4.2 and table 2.4.3 shown all characteristics of tires.

Table 2.1

Tire Discription

	Tire discription					Application rating			Qualification standard
	Size			Ply rating	Speed index	Max loading	Inflation pressure (unloaded)	Approx. bottoming load	
	M	N	D						
Nose gear	34X	14	12	24	174	17300	155	51900	MIL-T-5041
Base gear	49 X	17		32	235	50400	210	151.2	TSO-C62

2.5 Determination of centers of gravity of masses of the desighned aircraft

1. Determination of centers of gravity of masses of the equipped wing

To make designed aircraft stable in air during flight and on the ground correct centering position must be provide. Mass of aircraft consist of mass of structure, mass of systems, mass of passengers and its equipment.

In table 2.2 is shown all masses that located in the wing, in table 2.5.2 will be shown all masses in fuselage.

Coordinates of the center of gravity for the equipped wing are defined by the formulas:

$$X_w^{/} = \frac{\Sigma m_i^{/} x_i^{/}}{\Sigma m_i^{/}}$$

Table 2.2

Trim sheet of equipped wing masses

N	Object name	Mass		C.G coordinates Xi, м	Moment of mass
		units	total mass m(i)		
1	Wing	0.11177	10740	1.880	20192.24
2	Fuel system, 80%	0.0063	607	1.504	913.41
3	Flight control system, 30%	0.00177	170	2.507	426.35
4	Electrical equipment, 20%	0.0063	607	0.418	253.73
5	Anti-ice system , 70%	0.0160	1540	0.418	643.55
6	Hydraulic systems , 30%	0.0050	479	2.507	1199.57
7	Engines (-fuel system)	0.09602	9227	-1.6	-14762.5
	Equipped wing without landing gear and fuel	0.2432	23370	0.379	8866.35
8	Nose landing gear (20%)	0.0078	751	-12.86	-9653,42
9	Main landing gear (80%)	0.0312	3003	0.82	2462,15
10	Fuel for flight	0.2307	22166	1.253	27782.92
	Reserve fuel	0.0368	3531	1.462	5163.83
	Total	0.54789	52646.7501	0.655	34621.82

2. Determination of the centre of gravity of the equipped fuselage:

As origin of coordinate is chosen nose of fuselage to provide easy determination of location of fuselage parts and equipment. The C.G. coordinates of the fully equipped fuselage are determined by formulas:

$$X_f = \frac{\Sigma m_i^{/} X_i^{/}}{\Sigma m_i^{/}};$$

Table 2.3

Trim sheet of equipped fuselage masses

N	Objects names	Mass		C.G coordinates Xi, м	Moment of mass
		units	total mass		
1	2	3	4	5	6
1	Fuselage	0.09643	9266	21.678	200862.82
2	Horizontal tail	0.01066	1024	41.3	42312
3	Vertical tail	0.01051	1010	39.55	39942
4	Radar	0.003	288	1	288.27
5	Radio equipment	0.0023	221	1	221.007
6	Instrument panel	0.0053	509	2	1018.55
7	Navigation equipment	0.0045	432	2	864.81
8	Flight control system 70%	0.00413	397	23.845	9463
9	Hydraulic system 70%	0,0116	1117	23.695	26457
10	Electrical equipment, 80%	0.0253	2429	21.6775	59240.26333
11	Not typical equipment	0.0031	298	3	893.637
12	Lining and insulation	0.0068	653	19.510	12747.9
13	Anti ice system, 30%	0.0069	660	40.7	26868.95

Continuation of table 2.3

1	2	3	4	5	6
14	Airconditioning system	0.0015	139	19.510	2718.3
15	Fuel system, 20%	0.00158	152	22.5	3416
16	Passenger seats (business class)	0.0114	1094	5.925	6481.95
17	Passenger seats (economy class)	0.0528	5075	24	121800
18	Seats of flight attendances	0.0006	56	1.820	101.92
19	Seats of pilots	0.0004	42	2.25	94.05
20	Emergency equipment	0.001147	1102	8.1	8927.43
21	Lavatory 1	0.0057	550	7.75	4262.5
22	Lavatory 2	0.0057	550	32	13440
23	Galley 1	0.0044	420	32	17600
24	Galley 2	0.0057	550	33	9259.2324
25	Wardrobe	0.0052	500	35	17500
26	Tools	0.0057	550	5	2750

Ending of table 2.3

1	2	3	4	5	6
27	Water and liquid	0.0035	340	35	11900
28	Baggage equipment	0.0094	900	25	22500
29	Additional equipment	0.01147	1102	5	5510.8
30	**equipped fuselage without payload**	**0.2192**	**30325**	**21.652**	**656615.54**
31	Passengers (economy class)	0.1358	13050	20	261000
32	Passengers (business class)	0.0215	2070	5.925	12265
33	On board meal	0.004	386	30	11580
34	Baggage	0.0402	3864	18	69552
35	Cargo, mail	0.0262	2520	18	45360
36	Flight attend	0.0017	160	1.820	291.2

2.3 Calculation of center of gravity positioning variants

After determination of location of gravity center of loaded and equipped wing and fuselage need to construct equation of moment equilibrium:

$$m_f x_f + m_w (x_{MAC} + x_w^{/}) = m_0 (x_{MAC} + C)$$

From here we determined the wing MAC leading edge position relative to fuselage, means X_{MAC} value by formula:

$$X_{MAC} = \frac{m_f x_f + m_w \cdot x_w^{/} - m_0 C}{m_0 - m_w} = $$
$$= \frac{434433.22 \cdot 19.96 + 52646.751 \cdot 17 - 96096 \cdot 1.27}{96096 - 52646.75} = 18.78, \text{ [m]}$$

where m_0 – aircraft takeoff mass, kg;

m_f – mass of fully equipped fuselage, kg;

X_f – coordination of fully equipped fuselage,

m_w – mass of fully equipped wing, kg;

X_w – coordination of equipped wing

C – distance from MAC leading edge to the C.G. point.

$$C = (0,22...0,25)\ B_{MAC} = 0.25 \cdot 5.106 = 1.27, \text{ [m]}$$

Knowing the wing position relatively to fuselage on the layout drawing, the wing mass positions and the fuselage mass positions may be connected. After the wings and fuselage arrangement a C.G. calculation takes place. C.G. positioning is called the relative position of centre of masses relatively to MAC leading edge, presented percents:

$$\bar{X}_T = \bar{X}_C = \frac{X_{C.G.} - X_{MAC}}{b_{MAC}} \cdot 100\% = \frac{C}{b_{MAC}} \cdot 100\% = \frac{1.27}{14.8} \cdot 100\% = 18.78, \text{ [m]}$$

Calculation of C.G. positioning variants

The list of mass objects for C.G. variant calculation given in Table 3.3.1, completes on the base of both previous tables. In table 2.5.3 is shown position of gravity center in different variants.

Table 2.4.

Calculation of the C.G. positioning variants

Name	Mass, Kg	Coordinate	Mass moment
1	2	3	4
Object	m_i	C.G., M	Kg.m
Equipped wing (without fuel and landing gear)	23370	23.475	548612.58
Nose landing gear (extended)	751	10.236	7683.44
Main landing gear (extended)	3003	23.916	71809.59
Fuel for flight	22166	24.349	539721.82
Reserve fuel	3531	24.558	86721.64
Equipped fuselage (without payload)	30325	20.121	610165.82
Passengers of economy class	13050	20.00	261000.00
Passengers(bussiness)	2070	5.93	12264.96
Baggage	3864	18.000	69552.00
Cargo, mail	2520	18.000	45360.00
On board meal	386	30.000	11580.00
Flight attendants	160	1.820	291.20
Crew	160	2.400	384
Nose landing gear (retracted)	751	10.174	7637.19
Main landing gear (retracted)	3003	23.916	71809.59

Table 2.5

Aircraft's center of gravity position variants

Variant of loading	Mass, kg	Moment of mass, kg·m	CG coordinates, m	Centering, %
Take-off mass (landing gear extended)	105356	2265147.05	21.500	-38.19
Take-off mass (landing gear retracted)	105356	2265100.81	21.500	-38.20
Landing variant (landing gear extended)	82804	1713845.22	20.698	-57.40
Transportation variant (without payload)	83306	1865052.64	22.388	-16.94
Parking variant (without fuel and payload)	57449	1238271.43	21.554	-36.89

Conclusion to the project part:

Preliminary design is important part of aircraft creation. It is complicate process determination of all necessary parameters of aircraft parts such as wing shape and configuration, fuselage diameters, length and finesses ratio, geometry of tail unit and type, landing gear configuration and tires type. Selecting and changing each parameter entails changes in the configuration of the aircraft, the shape and dimensions of other components and parts. Therefore, it is necessary that the selection of initial data be carried out carefully, because even a small error will entail a number of requisite corrections

All this parameters influence on the mass centering. Mass centering affects on stability and controllability of the aircraft. If the location of the center of mass does not correspond to the given aerodynamic scheme, then an additional recalculation of the stabilizer and control surfaces must be carried out. Since the location and arrangement of the tail unit is chosen from the condition of the mutual location of the center of pressure on the wing and the center of mass.

3. DESIGHNING OF THE STRINGER OT THE WING UPPER PANEL

3.1 Justification of the task

Creating a new aircraft is a time-consuming, complex and expensive process. However, this is fully justified if the sale of the aircraft brings a significant profit to the airlines. For this, it is necessary to create an attractive aircraft for buyers. This can be achieved in several ways: create a unique product, create a cheap product, or create a more cost-effective product.

As a medium-haul passenger plane, it was designed taking into account the following prototypes: Boeing 737-800, Airbus 321-100 and Boeing 757-200. Since there is no fundamental difference between the prototypes and the designed aircraft, the uniqueness of the aircraft cannot be considered the main characteristic that will attract the attention of buyers.

Creating a cheaper product is also not considered, because creating a new aircraft is a complex process and cheapening the process can negatively affect the quality and the aircraft will not be able to meet modern requirements.

Therefore, to ensure the commercial success of the aircraft, it is necessary to create an economically profitable product. This can be achieved in several ways: facilitate and speed up operation, in this case the aircraft will pass all the necessary checks sooner and will not be used for its intended purpose - the transportation of passengers and cargo. It is also possible to lighten the structure, which will lead to an increase in the payload and allow to increase the amount of fuel used, which will also have a positive effect on the flight range. Lightening the design can reduce fuel consumption, which will also significantly save money for airlines and make the designed aircraft more attractive to buyers.

A variant of the advantage of the aircraft in a competitive environment is also better strength characteristics at the same weight of the structure. Because this will help not only to use the material from which the part is made more efficiently, but also to ensure greater safety in an emergency situation.

Therefore, it was decided to improve the strength characteristics without increasing the weight of the structure. In order to better assess the advantage of the efficiency of the designed aircraft, it is necessary to compare the allowable stresses of the same structural element. Boeing 737-800 was chosen for comparison.

3.2 Technical description

A stringer is a power element of a semi-monocoque wing structure that reinforces the skin. The stringer is placed along the wing from the root rib to the wing tips. Stringers are usually made of aluminum and titanium alloys and composite materials. In a conventional aircraft, the stringers on the upper panel and the lower panel of the wing usually differ in both shape and type of alloy. The stringers on the lower wing panel are loaded in tension, so their critical design is fatigue. The stringers of the upper wing panel experience compressive stresses, so their dangerous calculation is a buckling. Shear and tensile stresses in the upper and lower wing panels are the result of transverse lift force applied to the wing, which causes bending in the wing structure during flight. When the aircraft is parked on the ground, the wing is also loaded under its own weight by a bending moment, but it has the opposite direction, so in the case of parking, the upper panel is stretched and the lower panel is compressed. But since during flight the lift force is greater than the weight of the empty wing, therefore it is necessary to carry out calculations specifically for the case of flight.

Buckling is a sudden change in the deformed state of the part without changing the type of loading. The general buckling of compressed rods is a sudden transition of compression of the column to its bending. With a general buckling, the axis of the column is bent. The general bukling of a compressed column occurs due to the presence of initial eccentricities, the appearance of which is caused by the inaccuracy of force application or the non-ideality of the design. During the application of the load, the line of action of the resultant of all internal forces does not coincide with the line of action of the applied force, so there is a moment that tends to bend the rod. As the applied force increases, the bending moment will grow and at some point the flexural stiffness of the rod will not be enough to continue resisting the bending of the beam, then the rod will begin to bend, thereby increasing the shoulder between the applied

force and the resultant of all forces, that is the moment increases. So, after reaching the critical loads in the beam, with an increase in the load, the beam will begin to deform non-linearly, because the bending moment will increase due to the increase in force and due to the increase in the shoulder between the forces. At some point, the secondary bending stresses will exceed the compressive stresses and the column will reach breaking loads and the part will collapse. When calculating the compressive strength of the bars, it is necessary to take into account the secondary stresses from bending to prevent premature collapse of the columns.[6]

The stringers on the upper panel of the wing can be subjected to a general buckling, so to avoid this it is necessary to ensure the minimum moment of inertia of the cross section of the stringer is as much as possible. The moment of inertia of the section shows how easy it is to bend the body relative to some axis.

Moment of inertia formula:

$$I=\int c^2 dA$$

Where:

c - distance to the axis,

A – area of considered section.

From the formula of the moment of inertia, it follows that to ensure maximum flexural stiffness with minimal use of material, it is necessary to carry the material to the greatest distance from the axis, since the squared distance has a greater influence on the moment of inertia. Therefore, when designing, it is worth paying attention to the distribution of material relative to the main central axes of the beam. Principal axes are the axes relative to which there are the largest and smallest moments of inertia of the section. [7]

Therefore, it is worth making the stringer thin-walled and carry the material as much as possible relative to the axes. However, the following problem arises here.

Thin-walled structures are very often subject to local buckling. The local buckling is characterized by the distortion of the section under the action of compression without distortion of the axis of the rod. Therefore, it is also necessary to check the strength of thin-walled structures under the action of compression.

As a deck, the main design cases of a stringer are: compressive strength checking, general buckling checkling and local buckling checking.

Since the design calculation for the stringer will be carried out, it is necessary to know the load on the stringer. And it is also necessary to compare the weight of the stringer of the designed aircraft and the prototype. Therefore, it was decided to design the stringer based on the critical loads for the prototype aircraft.

3.3 Calculation

3.3.1 Calculation of critical stresses of general buckling.

Critical stresses of general buckling is calculated by Euler's formula in the elastic region.

$$\sigma_g = \frac{\pi^2 \cdot E}{\lambda^2} \qquad (1)$$

where σ_g – the critical stress of general buckling of the compressed column

π – the Pi constant which characterizes the shape of the rod's curvature - a sinusoid

E – Young's modulus of the material, which characterizes the stiffness of the material.

$E = 0.7 \cdot 10^5$ MPa

λ – slenderness ratio, which characterizes the efficiency of the rod under compression. The greater the flexibility of the rod, the less effectively it perceives compression and the sooner it loses stability.
The slenderness ratio of the rod is found according to the following formula:

$$\lambda = \frac{\mu \cdot l}{i} \tag{2}$$

μ is the end fixity factor of the rod
The end fixity factor is selected from the conditions of pinching of the rod at the ends. It is most conservative for the stringer to choose hinged operation on both sides, since in this case we underestimate the critical stresses of the rod.
μ= 1
l – the length of the stringer, which buckles. The most conservative option would be to choose a wing span, but in this case the stiffness of the ribs is not taken into account, which does not allow the stringer to lose its stability as a cantilever beam. Therefore, in order not to overload the structure, it is decided to choose the pitch of the ribs for the length of the stringer.
l=1575 mm

Since the rib pitch of the design aircraft and the prototype aircraft are the same, the effective length of the stringer will also be the same.
i – the radius of gyration. The radius of gyration shows how well the area of the rod is spread relative to the central axes.
The formula for the radius of gyration:

$$i = \sqrt{\frac{I_{\min}}{A}} \tag{3}$$

$I_{\min}$ – is the minimum moment of inertia
A – the cross-sectional areaThe geometric dimensions of the cross-section of the stringer of the prototype aircraft are shown in Figure 3.1:

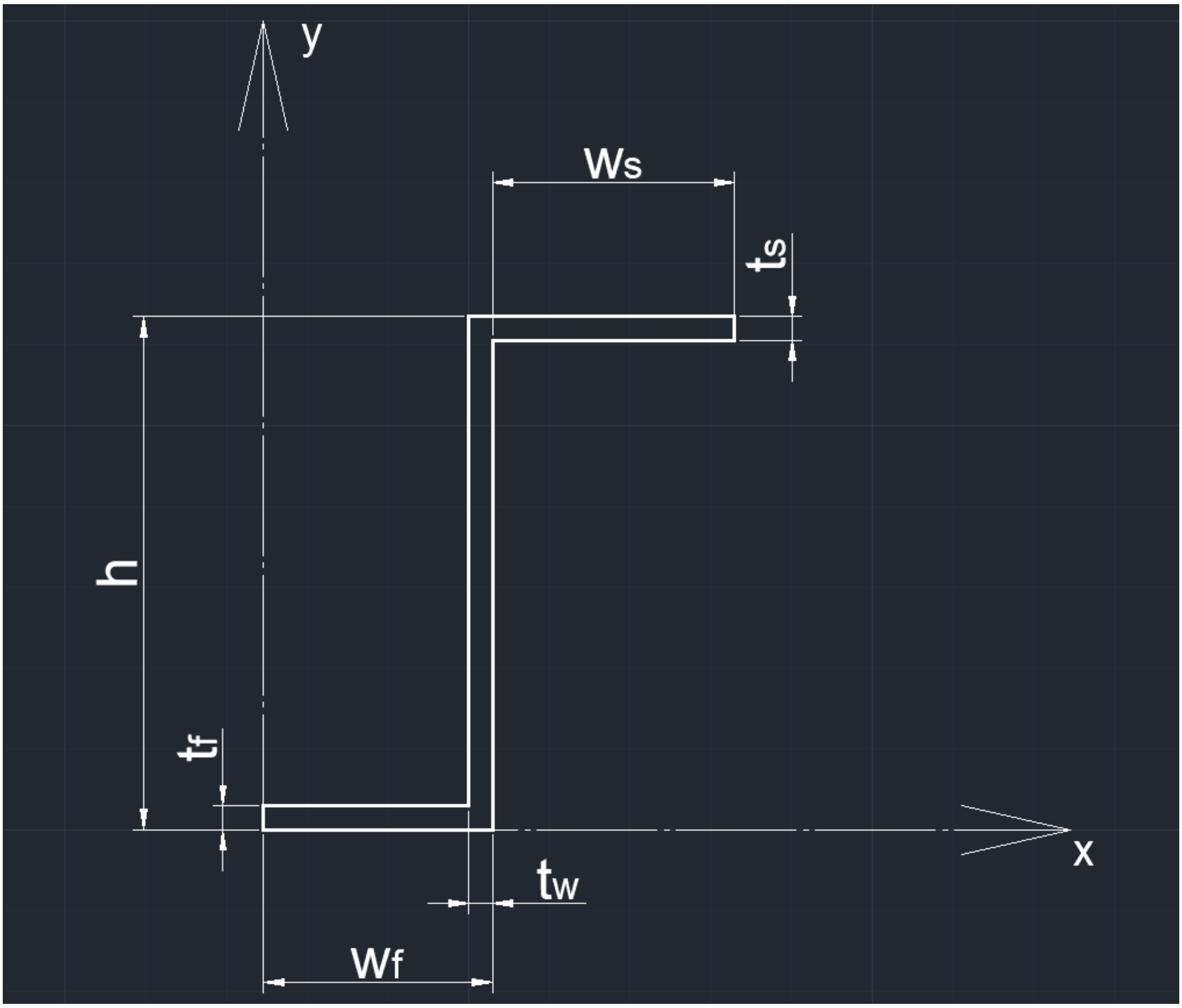


Fig. 3.1. Geometrical characteristic of cross-section of the stringer.w_s=30 mm; w_f=28.5 mm; h=65.5 mm; t_s=3mm; t_f=3 mm; t_w=3mm.

The cross-sectional area of the rod is calculated according to the following formula:

$$A = w_s \cdot t_s + w_f \cdot t_f + t_w \cdot (h - t_s - t_f) = 30 \cdot 3{+}28.8 \cdot 3{+}3 \cdot (65.5{-}3{-}3){=}359.12,\ [\mathrm{mm}^2]$$

The moment of inertia must be found relative to the axes that pass through the center of mass of the section. To find the centroid necessary to find static moments relative to arbitrary axes. First moment of area are relative to the x and y axes shown in Figure 3.1.

The first moment of area about x axis is found by the following formula:

$$S_x = w_s \cdot t_s \cdot (h - \frac{t_s}{2}) + \frac{h^2 \cdot t_w}{2} + \frac{t_f^2 \cdot (w_s - t_w)}{2} =$$

$$30 \cdot 3 \cdot (65.5 - 1.5) + (\frac{65.5^2 \cdot 3}{2}) + \frac{3^2 \cdot (30 - 3)}{2} = 12310.13, \text{ [mm}^3\text{]}$$

The first moment of area about y axis is found by the following formula:

$$S_y = w_s \cdot t_s \cdot (w_f + \frac{w_s}{2}) + h \cdot t_w \cdot (w_f - \frac{t_w}{2}) + t_f \cdot \frac{(w_f - t_w)^2}{2} =$$

$$= 30 \cdot 3 \cdot (28.5 + 15) + 65.5 \cdot 3 \cdot (28.5 - 1.5) + 3 \cdot \frac{(28.5 - 3)^2}{2} = 10195.88, \text{ [mm}^3\text{]}$$

The first moment of area about centroid is zero. Therefore, the position of the center of mass relative to the x axis is determined by the following formula:

$$y = \frac{S_x}{A} = \frac{12310.13}{359.12} = 34.3, \text{ [mm]}$$

The position of the center of mass relative to the y axis is determined by the following formula:

$$x = \frac{S_y}{A} = \frac{10195.88}{359.12} = 28.4, \text{ [mm]}$$

Since the stringer is not symmetrical, the main central axes will be rotated relative to the centroidal axes. However, since the stringer is riveted to the skin, the panel must deform together, so the skin will prevent the stringer from general buckling in the direction of the minimum moment of inertia. In this case, there is no need to calculate the angle of rotation of the main central axes and find the minimum moment of inertia.

If the stringer is riveted to the skin, the stringer loses stability in the direction parallel to the skin. Therefore, the moment of inertia relative to the x_c axis will be used in the calculations.

The moment of inertia of the stringer about the x_c axis:

$$I_x = \frac{w_s \cdot t_s^3}{12} + t_s \cdot w_s \cdot (h - y - \frac{t_s}{2})^2 + h \cdot t_w \cdot (\frac{h}{2} - y)^2 + \frac{t_w \cdot h^3}{12} +$$
$$+ \frac{(w_f - t_w) \cdot t_f^3}{12} + t_f \cdot (w_f - t_f) \cdot (y - \frac{t_w}{2})^2 = \frac{30 \cdot 3^3}{12} + 3 \cdot 30 \cdot (65.5 - 34.3 - \frac{3}{2})^2 +$$
$$+65.5 \cdot 3 \cdot (\frac{65.5}{2} - 34.3)^2 + \frac{65.5^3 \cdot 3}{12} + \frac{(28.5 - 3) \cdot 3^3}{12} +$$
$$+3 \cdot (28.5 - 3) \cdot (34.3 - \frac{3}{2})^2 = 230092,\ [\mathrm{mm}^4]$$

The radius of inertia from the formula (3):

$$i = \sqrt{\frac{230092}{359.12}} = 25.3,\ [\mathrm{mm}]$$

Slenderness ratio from the formula (2):

$$\lambda = \frac{1 \cdot 1575}{25.3} = 62.25$$

Critical stresses of the general buckling from formula (1)

$$\sigma_g = \frac{3.14^2 \cdot 0.7 \cdot 10^5}{62.25} = 1109.8,\ [\mathrm{MPa}]$$

3.3.2 Calculation of critical stresses of local buckling

The formula for calculation of plate modulus

$$D = \frac{E \cdot t^3}{12 \cdot (1 - \upsilon^2)} = \frac{0.7 \cdot 10^5 \cdot 3^3}{12 \cdot (1 - 0.33^2)} = 176747.8$$

where t – thickness of the plate;

υ – Poisson`s ratio of aluminum

As Z section was manufactured by forming, thickness of the element is constant.

The formula for the buckling of plate is calculated by the formula [7]:

$$\sigma_l = k\frac{\pi^2 \cdot D}{b^2 \cdot t} \qquad (4)$$

k is flat plate buckling coefficient

b is the width of the plate

k_c is chosen for two types of fixing plates. The end fixity factor of the web is the same as for the plate hinged on all sides (k_{cw}). The end fixity factor of chord is the same as for a plate that has hinged operations on three sides and a free edge parallel to the load (k_{cc}). [9]

k_w=4

k_c=0.5

The choice of such load models is determined by the fact that the real supports can be attributed to something in the middle between a rigid pinch and a hinge. Therefore, a more conservative option was chosen - a hinge.

Critical stresses of local web buckling from equation 4:

$$\sigma_{lw} = k\frac{\pi^2 \cdot D}{b^2 \cdot t} = \frac{4 \cdot 3.14^2 \cdot 176747.8}{65.5^2 \cdot 3} = 542, \text{ [MPa]}$$

The upper cab of Z-section is attached to the paneling with rivets. During compression, the upper flange will not buckle due to the fact that the skin will not allow the stringer to bend from its plane. Therefore, only the lower cab should be considered for local buckling.

Critical stresses of local cab buckling:

$$\sigma_{lc} = k\frac{\pi^2 \cdot D}{b^2 \cdot t} = 0.5 \cdot \frac{3.14^2 \cdot 176747.8}{28.5^2 \cdot 3} = 358, \text{ [MPa]}$$

Let's compare the values of the critical buckling stresses of the cab and web and the general buckling stresses. To do this, it is first necessary to compare the stress of the local buckling of the chord and the web.

Comparison of local buckling stresses with each other:

$$\sigma_{lw} \text{ and } \sigma_{lc}$$
$$542 \text{ MPa} > 358 \text{ MPa}$$

It is necessary to compare the stresses of general buckling with smaller stresses of local buckling stresses, namely local buckling stresses of flanges.

Comparison of local buckling stresses of chord and general buckling stresses:

$$\sigma_{lc} \text{ and } \sigma_{g}$$
$$358 \text{ MPa} < 1109.8 \text{ MPa}$$

The minimum stress is the local buckling stress of the cab. Now it is necessary to compare the minimum critical stresses with the proportional limit σ_{pr} of the structural alloy. [10]

$$\sigma_{pr} = 400 MPa$$

Clearly the local buckling stress of cab is less than the approximate proportional limit, so the calculations can be considered fair.

On the figure 3.2 is shown deformation state of the stringer loaded by the general buckling load.

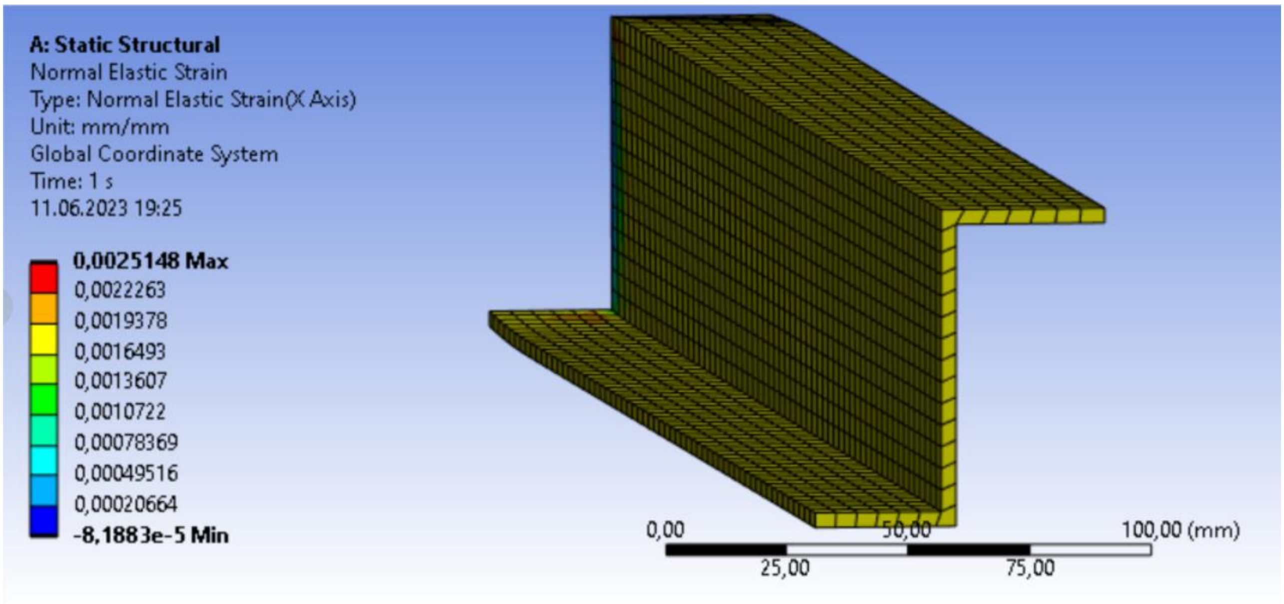


Fig. 3.2. Deformation state of Z-type stringer

3.4. Stringer design

The proposed cross-sectional shape of the stringer was chosen based on the following considerations:

1) The minimum critical stresses are the stresses of the buckling of the cabs, so this part of the cross-section should be redesigned.

2) It is not necessary to reduce the height of the stringer, because this will lead to a decrease in the moment of inertia and reduce the strength to a general buckling

3) When the cab is moved parallel to half its length to the left, the moment of inertia will not change [11]

4) Also, in order to reduce the stress of local buckling, it is necessary to either reduce the width of the plate or its thickness. An increase in thickness will entail an increase in mass, which is undesirable. Therefore, it is necessary to reduce the effective width of the plate.

The modified section is shown in Figure 3.3.

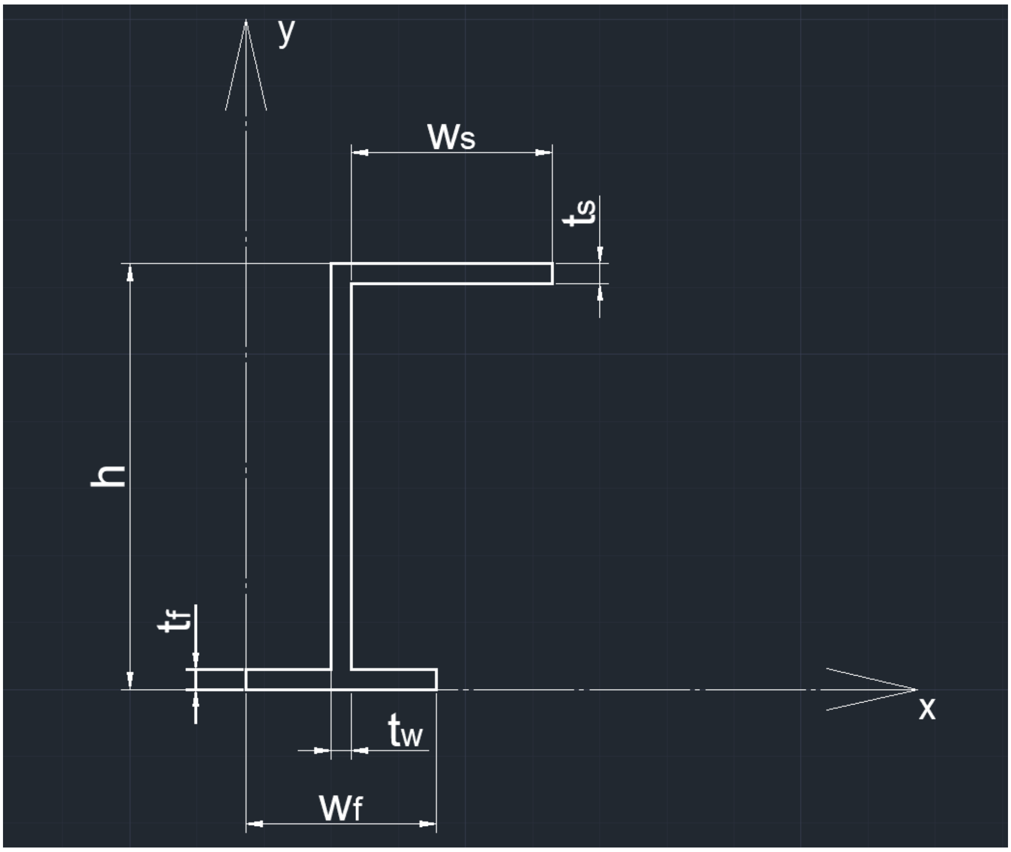


Fig. 3.3. Modified section

Verification calculation for the modified stringer is being conducted. The calculation should be carried out only for the cab local buckling stress.

$$\sigma_{lcm} = k\frac{\pi^2 \cdot D}{b^2 \cdot t} = 0.5 \cdot \frac{3.14^2 \cdot 176747.8}{14.25^2 \cdot 3} = 1432, \text{[MPa]}$$

Let`s compare the local buckling stress of cab of modified stringer and approximately proportional limit.

σ_{lcm} and σ_{prop}

1435 MPa > 400 MPa

Since the stresses of the local bucking of cab exceed the limit of proportionality, the accuracy of the calculations cannot be sufficient. It is necessary to use the plasticity reduction factor. However, a number of experimental data are needed to obtain the plasticity coefficient, so it was decided to accept the stress of the limit of proportionality for the stress of the buckling of chord. This calculation is very conservative, because it does not allow loading beyond the limit of proportionality. Although even after the stress exceeds the limit of proportionality, the material can still carry the load.

Let`s compare the local buckling stress of cab of modified stringer and cab of stringer prototype.

σ_{lcm} and σ_{lc}

400 MPa > 341 MPa

Since the allowable stresses of the modified stringer are greater, the work was carried out successfully.

On the figure 3.4 is shown deformation state of the modified stringer loaded by the general buckling load.

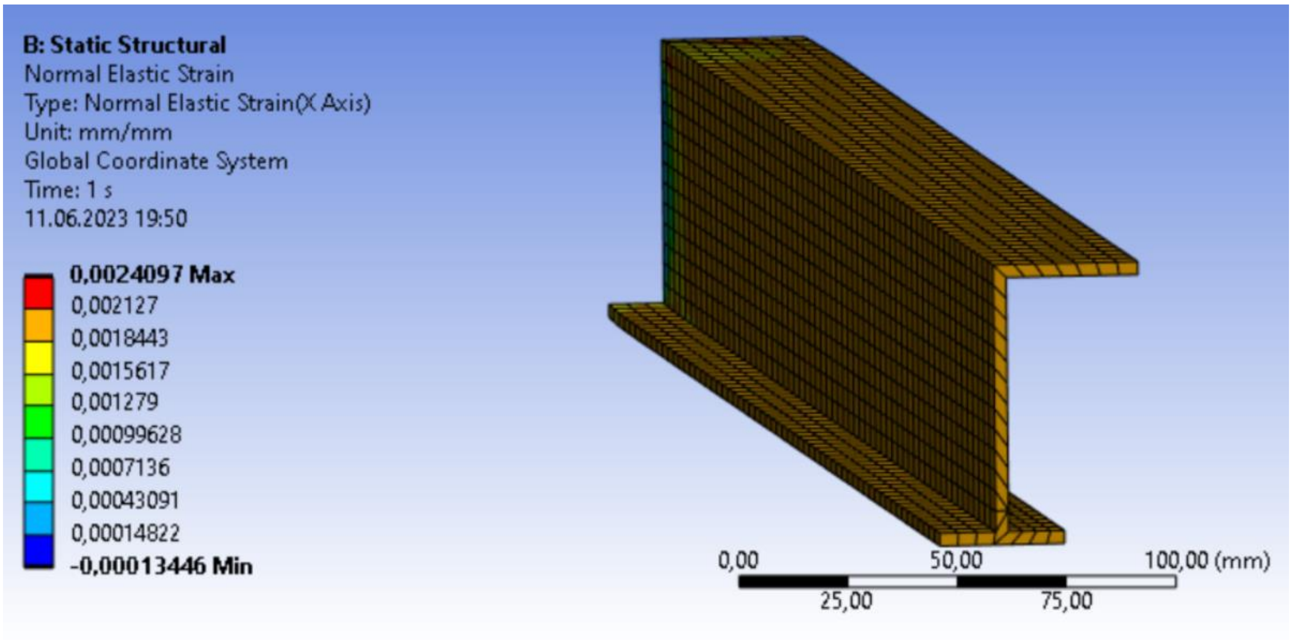


Fig. 3.4. Deformation state of the modified stringer

Conclusion to the special part

During operation, the aircraft stringer was redesigned to account for the buckling. A critical case in the calculation is the local buckling of the lower plate. When designing, it is taken into account that after reaching critical loads, the structure will no longer accept the load and will collapse. However, this is not the case, reaching critical stresses of local loss of stability does not lead to destruction at the moment of reaching these stresses. Therefore, such a calculation is conservative and to some extent overloads the structure. In the further design of the aircraft, it is worth paying attention to the compressed panel and calculating the permissible loads on the wing in the plastic zone.

GENERAL CONCLUSIONS

During this work, a series of aircraft were analyzed in order to start designing the aircraft based on the data obtained. Then the main geometric characteristics of the aircraft and the position of the center of mass of parts and units were calculated. In the next part, the stringer of the upper wing panel was redesigned to account for the general and local buckling. The purpose of this redesign was to increase the efficiency of using the aircraft's mass and provide better compressive strength. The result of the performed work fully meets the set goals.

REFERENCES

1. Mohammad H Sadraey. (2013) Aircraft Design. p. 161
2. Barnes McCormick, Barnes (1995) Aerodynamics, Aeronautics, aand flight mechanics. p.126
3. Mohammad H Sadraey. (2013) Aircraft Design. p. 479
4. John D. Anderson, Jr. (2010). Aircraft performance and design. p.257
5. John D. Anderson, Jr. (1990). Introduction to fkight. p. 259
6. T. H. G. Megson. (2010). Aircraft Structural Analysis. p. 253
7. Niu.(1999). Airframe stress analysis and sizing. p. 458
8. T. H. G. Megson. (2010). Aircraft Structural Analysis. p. 292
9. Niu.(1999). Airframe stress analysis and sizing. p. 460
10. Federal Aviation Administration. (2003). Metallic Materials Properties Development and Standardization. p. 3-370
11. Ferdinand P. Beer (2009). Static and mechanics of materials. E. Russell Johnston, Jr., John T. DeWolf, David F. Mazurek. p. 444

APPENDIX A

INITIAL DATA AND SELECTED PARAMETERS

Passenger Number	162
Flight Crew Number	2
Flight Attendant or Load Master Number	5
Mass of Operational Items	1780.06 kg
Payload Mass	19800.00 kg
Cruising Speed	850 km/h
Cruising Mach Number	0.7924
Design Altitude	10.50 km
Flight Range with Maximum Payload	4500 km
Runway Length for the Base Aerodrome	2.55 km
Engine Number	2
Thrust-to-weight Ratio in N/kg	3
Pressure Ratio	32.30
Assumed Bypass Ratio	4.5
Optimal Bypass Ratio	4.5
Fuel-to-weight Ratio	0.33
Aspect Ratio	9.45
Taper Ratio	3.49
Mean Thickness Ratio	0.12

Wing Sweepback at Quarter Chord	25 deg
High-lift Device Coefficient	1.1
Relative Area of Wing Extensions	0.01
Wing Airfoil Type supercritical	
Winglets -	
Spoilers +	
Fuselage Diameter	3.77
Finess Ratio	11.50
Horizontal Tail Sweep Angle	30 deg
Vertical Tail Sweep Angle	35 deg

CALCULATION RESULTS

Optimal Lift Coefficient in the Design Cruising Flight Point	0.43016
Induce Drag Coefficient	0.00913

ESTIMATION OF THE COEFFICIENT $D_m = M_{critical} - M_{cruise}$

Cruising Mach Number	0.79237
Wave Drag Mach Number	0.80057
Calculated Parameter Dm	0.00820

Wing Loading in kPa (for Gross Wing Area):

At Takeoff	5.402
At Middle of Cruising Flight	4.650
At the Beginning of Cruising Flight	5.211

Drag Coefficient of the Fuselage and Nacelles	0.00855
Drag Coefficient of the Wing and Tail Unit	0.00914

Drag Coefficient of the Airplane:

At the Beginning of Cruising Flight	0,02892
At Middle of Cruising Flight	0,02782
Mean Lift Coefficient for the Ceiling Flight	15,46083
Mean Lift-to-drag Ratio	15,46083
Landing Lift Coefficient	1,615
Landing Lift Coefficient (at Stall Speed)	2.422
Takeoff Lift Coefficient (at Stall Speed)	1.986
Lift-off Lift Coefficient	1.450
Thrust-to-weight Ratio at the Beginning of Cruising Flight	0.600
Start Thrust-to-weight Ratio for Cruising Flight	2.365
Start Thrust-to-weight Ratio for Safe Takeoff	3.055

Design Thrust-to-weight Ratio	3.177
Ratio $D_r = R_{cruise} / R_{takeoff}$	0.774

SPECIFIC FUEL CONSUMPTIONS (in kg/kN·h):

Takeoff	37.5576
Cruising Flight	59.6314
Mean cruising for Given Range	63.1389

FUEL WEIGHT FRACTIONS:

Fuel Reserve	0.03675
Block Fuel	0.23068

WEIGHT FRACTIONS FOR PRINCIPAL ITEMS:

Wing	0.11177
Horizontal Tail	0.01066
Vertical Tail	0.01051
Landing Gear	0.03906
Power Plant	0.09602
Fuselage	0.09643
Equipment and Flight Control	0.13197
Additional Equipment	0.01147

Operational Items	0.01852
Fuel	0.26743
Payload	0.20606

Airplane Takeoff Weight	96090 kG
Takeoff Thrust Required of the Engine	152.64 kH

0,7Air Conditioning and Anti-icing Equipment Weight Fraction	0.0229
Passenger Equipment Weight Fraction	
(or Cargo Cabin Equipment)	0.0168
Interior Panels and Thermal/Acoustic Blanketing Weight Fraction	0.0068
Furnishing Equipment Weight Fraction	0.0146
0,3Flight Control Weight Fraction	0.0059
0,3Hydraulic System Weight Fraction	0.0166
0,2Electrical Equipment Weight Fraction	0.0316
Radar Weight Fraction	0.0030
Navigation Equipment Weight Fraction	0.0045
Radio Communication Equipment Weight Fraction	0.0023
Instrument Equipment Weight Fraction	0.0053
0,8Fuel System Weight Fraction	0.0079

Additional Equipment:

Equipment for Container Loading	0.0083
No typical Equipment Weight Fraction	0.0031
(Build-in Test Equipment for Fault Diagnosis,	
Additional Equipment of Passenger Cabin)	

TAKEOFF DISTANCE PARAMETERS

Airplane Lift-off Speed	277.83km/h
Acceleration during Takeoff Run	2.45 m/s^2
Airplane Takeoff Run Distance	1214 m
Airborne Takeoff Distance	578 m
Takeoff Distance	1792 m

CONTINUED TAKEOFF DISTANCE PARAMETERS

Decision Speed	263.94 km/h
Mean Acceleration for Continued Takeoff on Wet Runway	0.28 m/s^2
Takeoff Run Distance for Continued Takeoff on Wet Runway	2120.41 m
Continued Takeoff Distance	2698.79 m
Runway Length Required for Rejected Takeoff	2794.76 m

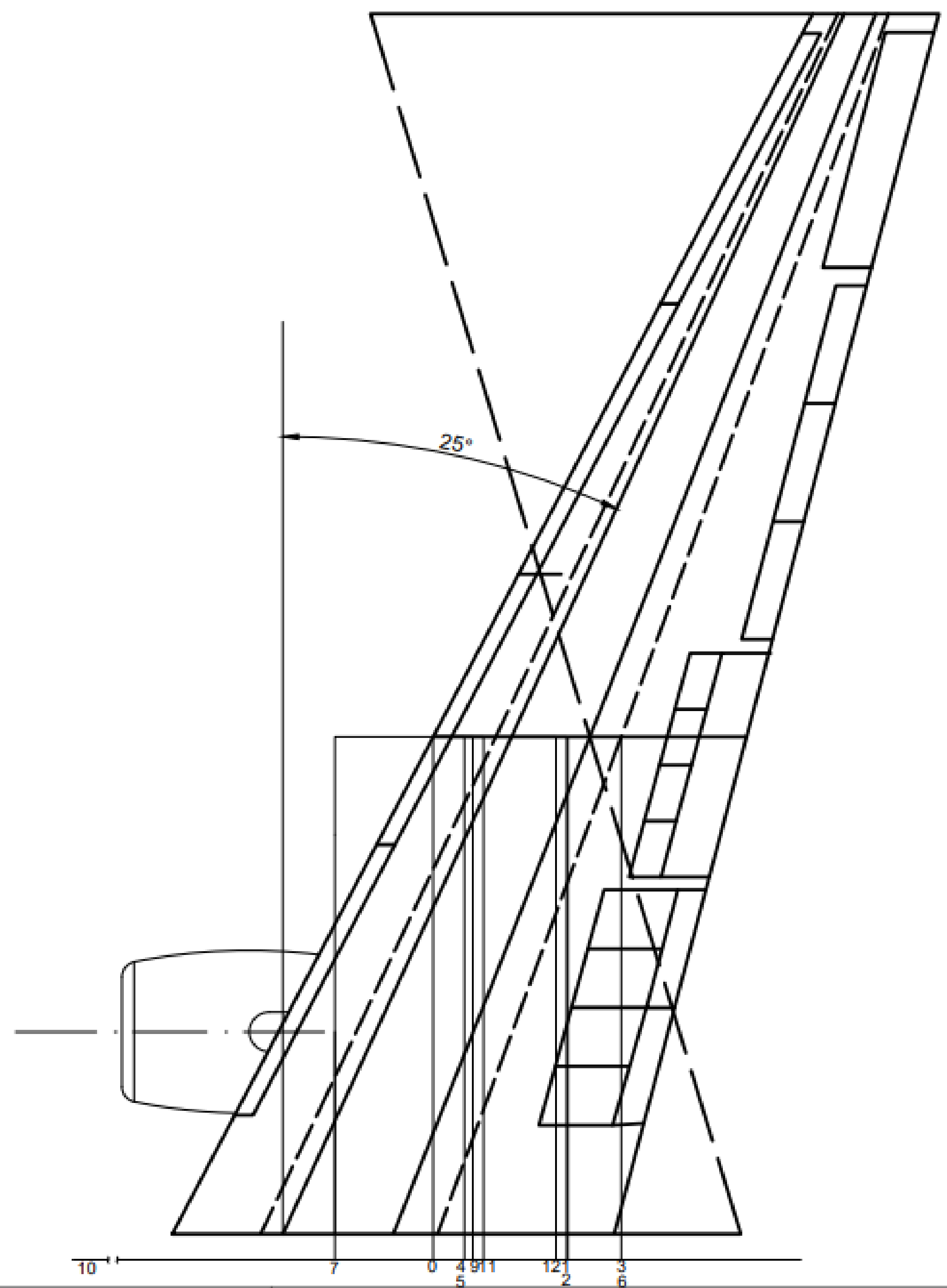
LANDING DISTANCE PARAMETERS

Airplane Maximum Landing Weight	77985 kg
Time for Descent from Flight Level till Aerodrome Traffic Circuit Flight	20.8 min
Descent Distance	49.19 km
Approach Speed	255.06 km/h
Mean Vertical Speed	2.05 m/s
Airborne Landing Distance	519 m
Landing Speed	240.06 km/h
Landing run distance	768 m
Landing Distance	1287 m
Runway Length Required for Regular Aerodrome	2150 m
Runway Length Required for Alternate Aerodrome	1828 m

ECONOMICAL EFFICIENCY

THESE PARAMETERS ARE NOT USED IN THE PROJECT

APPENDIX B



Center of gravity of wing

NAU 23 04K 00 00 00 06 GV

Ch./Sheet		Sign.	Date
Performed	Myroslava Kyryk		
Checked	Krasnopolskiy V.S.		

Middle range passenger aircraft

Letter	Weight	Scale
		1:100

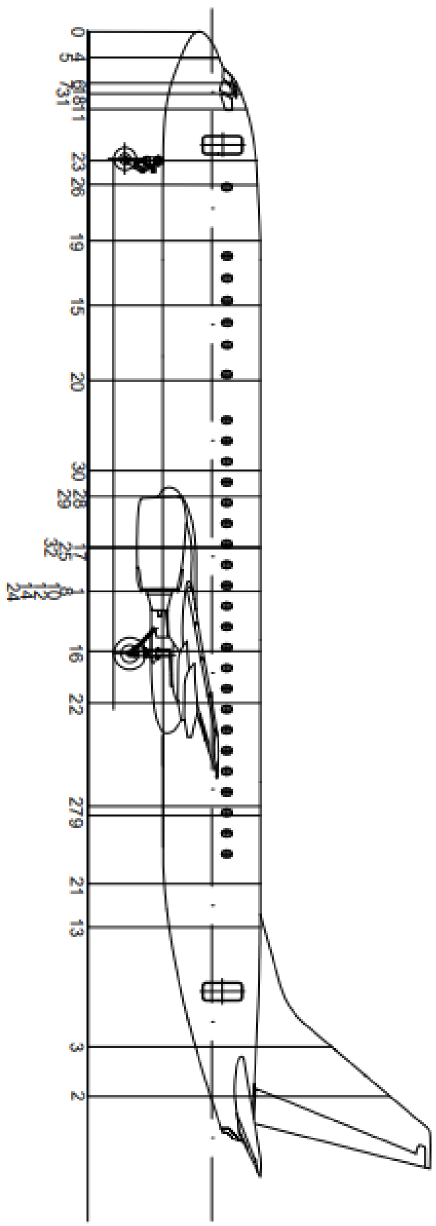
Sheet 1 Sheet 2

Appendix B

402 AKF 134

APPENDIX C

Appendix C



Center of gravity of wing	NAU 23 04K 00 00 00 02 P3			
	Middle range passenger aircraft	Letter	Weight	Scale
				1:250
		Sheet 1	Sheet 2	
	Appendix C	402 AKF 134		

Ch./Sheet		Sign.	Date
Performed	Myroslava Kyryk		
Checked	Krasnopolskiy V.S.		