МІНІСТЕРСТВО ОСВІТИ ТА НАУКИ УКРАЇНИ
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Кафедра конструкції літальних апаратів

ДОПУСТИТИ ДО ЗАХИСТУ
Завідувач кафедри, д.т.н., проф.
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КВАЛІФІКАЦІЙНА РОБОТА
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Тема: «Аванпроект пасажирського літака із застосуванням композитних матеріалів»

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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 passenger aircraft with application of composite materials"

Fulfilled by:	____________	**Pavlo STOVBUN**
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

СТОВБУНА ПАВЛА ВОЛОДИМИРОВИЧА

1. Тема роботи: «Аванпроект пасажирського літака із застосуванням композитних матеріалів», 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 20000 кг, дальність польоту з максимальним комерційним навантаженням 7200 км, крейсерська швидкість польоту 850 км/год, висота польоту 11 км.

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 застосування композиційних матеріалів у обшивці крила літака.

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

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 проектованого 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 застосування композиційних матеріалів.	08.06.2023 – 09.06.2023	
6	3D моделювання кесонної частини крила з композиційною обшивкою.	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 ____________ Павло СТОВБУН

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

Pavlo STOVBUN

1. Topic: "Preliminary design of passenger aircraft with application of composite materials", 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 20 tons, flight range with maximum capacity 7200 km, cruise speed 850 km/h, flight altitude 11 km.
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 composite materials application in the wing structure.
5. Required material: general view of the airplane (A1×1), layout of the airplane (A1×1), SolidWorks model and diagrams.

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	Composite materials application analysis for wing torsion-box.	08.06.2023 – 09.06.2023	
6	3D modeling of wing torsion-box with composite skin.	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: ____________ Pavlo STOVBUN

РЕФЕРАТ

Пояснювальна записка кваліфікаційної роботи бакалавра «Аванпроект пасажирського літака із застосуванням композитних матеріалів»:

56 с., 13 рис., 9 табл., 9 джерел

Дана кваліфікаційна робота присвячена розробці аванпроєкту пасажирського літака для дальньомагістральних авіаліній з можливістю транспортування вантажів, що відповідає міжнародним стандартам польотів, нормам безпеки, економічності та надійності, а також аналіз застосування композиційних матеріалів в обшивці кесонної частини крила.

В роботі було використано методи аналітичного розрахунку, компьютерного проєктування за допомогою CAD/CAM/CAE систем, чисельного моделювання і розрахунку на міцність обшивки крила.

Практичне значення результату кваліфікаційної роботи полягає в зменшенні маси та збільшенні міцності конструкції крила літака, що покращує його льотно-технічні характеристики.

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

Дипломна робота, аванпроект літака, компонування, центрування, обшивка крила, розрахунок на міцність

ABSTRACT

Bachelor degree thesis "Preliminary design of passenger aircraft with application of composite materials"

56 pages, 13 figures, 9 tables, 9 references

This thesis is dedicated to preliminary design of a passenger plane for long-haul airlines with the possibility of transporting cargo, which meets international flight standards, safety, economy and reliability standards, as well as analysis of the composite materials use in the skin of the wing torsion-box.

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 systems. In special part the numerical modeling and calculation of the strength is used to estimate stress state of the wing skin.

Practical value of the work is to reduce the mass and increase the strength of the aircraft wing structure, which improves its flight and technical performances.

The materials of the qualification work can be used in the aviation industry and educational process of aviation specialties.

Bachelor thesis, preliminary design, cabin layout, center of gravity calculation, wing skin, strength calculation

Format	№	Designation	Name	Quantity	Notes
			General documents		
A4	1	NAU 23 10S 00 00 00 86 TW	Task for work	1	
	2	NAU 23 10S 00 00 00 86	Long-range passenger aircraft	2	
A1		Sheet 1	General view		
A1		Sheet 2	Fuselage layout		
A4	3	NAU 23 10S 00 00 00 86 EN	Long-range passenger aircraft	56	
			Explanatory note		
			Documentation for assembly units		
A1	4	NAU 23 10S 00 00 00 86	Design of wing skin	1	

	Sh.	№ doc.	Sign	Date	NAU 23 10S 00 00 00 86 EN			
Done by		Stovbun P.V.			Preliminary design of passenger aircraft with application of composite materials (List of diploma work)	list	sheet	sheets
Supervisor		Krasnopolskyi V.S.				Q	9	56
St.control.		Krasnopolskyi V.S.				402 ASF 134		
Head of dep.		Ignatovich S.R.						

CONTENT

INTRODUCTION

In modern aircraft design, one of the main requirements for aircraft from a commercial point of view is a reduced weight of the aircraft, which leads to a reduction in fuel consumption, and makes it cheaper and more competitive. The next key points are sufficient passenger capacity and ease of maintenance as much as possible.

The production of structural elements from composite materials is gaining more and more popularity. The field of use of these materials is quite extensive, starting with fairings and ending with highly loaded elements of the aircraft structure, such as spars, stringers, ribs, skins, etc. These materials have much less weight and are not inferior in strength to aluminum alloys used in aircraft construction. In addition, they are quite resistant to corrosion and allow reducing the number of parts. Due to such characteristics, fuel consumption is reduced. Therefore, one of the important tasks in aircraft design, such as reducing its cost, is to a certain extent solved by the use of modern composite materials.

1. PRELIMINARY DESIGN OF MID RANGE AIRCRAFT

1.1 Analysis of prototypes and short description of designed aircraft

According to the statistical analysis of the prototypes' characteristics it is possible to choose correctly and constructively the parameters and layout of the designed aircraft. This analysis is based on the experience of previous designers, and prevention of all imperfections of previous versions. To obtain the final appearance of the aircraft, it is necessary to determine the structural parameters of the vehicle, which include its weight, the geometric parameters of its individual components, the version of the engine, its weight and power. Also, if the aircraft series do not significantly differ in parameters, then it could be possible to use the data of the aircraft family.

The application of these data makes it possible to study the requirements for the design, their transformation, and further determination of the aircraft final appearance.

A long-range aircraft with a capacity of up to 220 passengers manufactured by Boeing – B737 MAX 9 was chosen as the prototype of the aircraft developed in this work. The performances of the B737 MAX 9 prototype are given below in table 1.1.

Table 1.1

Performances of prototypes

Parameter	B737 MAX 9	A 320-200	B737-800	Designed aircraft
1	2	3	4	5
Max. payload, kg	20050	16600	20540	20050
Crew, number of pilots	6/2	5/2	5/2	6/2
Passengers	220	150	189	220
Wing loading, kN/m^2	7.07	4.28	6.19	5.557
Flight range with max. payload, km	7200	5000	5665	7200
Cruise speed, km/h	850	829	828	850
Cruise altitudes, km	11	11.27	12.5	11
Thrust/weight ratio, N/kg	3.27	2.91	2.79	3.1

Ending of the table 1.1

1	2	3	4	5
Approach speed, km/h	150	134	130	246
Landing speed, km/h	270	240	270	231
Take-off speed, km/h	270	240	270	282
Take-off run distance, m	2500	2590	2400	1311
Landing run distance, m	1500	1440	1630	714
Take-off distance, m	2600	2058	2652	1890
Landing distance, m	1700	1243	1636	1230
Maximum take-off mass, kg	88314	78000	79010	116422
Landing mass, kg	74343	64500	66361	86041
Empty weight kg	44676	42400	41413	77220
Fuel fraction, %	29.2%	38%	32.9%	34%
Payload fraction, %	22.7%	21.3%	25.9%	17.2%
Wing span, m	35.9	34,1	34.3	43.95
Sweepback angle at ¼ ochord, °	25	28	25.02	25
Wing aspect ratio	9.4	9.37	9.45	9.4
Wing taper ratio	4.1	4.11	4.5	4.11
Fuselage length, m	41.76	37.57	32.18	44.48
Fuselage diameter, m	3.95	3.95	3.76	3.95
Fuselage fineness ratio	10.91	9.51	10.21	11.26
Passenger cabin width, m	3.53	3.63	3.54	3.78
Passenger cabin length, m	35.76	27.5	29.95	27.15
Cabin height, m	2.5	2.0	2.02	2.12
Aisle width, m	0.51	0.69	0.51	0.58
Horizontal tail span, m	14.35	13.4	14.35	14.35
Horizontal tail sweepback angle, °	29	33	30	29
Horizontal tail aspect ratio	6.16	4.41	6.16	6.16
Horizontal tail taper ratio	4.92	2.15	4.92	4.92
Vertical tail height, m	7.16	5.46	7.16	7.16
Vertical tail sweepback angle, °	34	38	35	34
Vertical tail aspect ratio	1.94	1.8	1.91	1.94
Vertical tail taper ratio	3.69	4	3.69	3.69
Landing gear wheel base, m	17.17	12.315	15.6	17.24
Landing gear wheel track, m	5.72	9.85	5.76	5.70

1.2 Brief description of the main parts of the aircraft

1.2.1 Wing

The wing of the designed aircraft is a low-wing configuration, meaning it is attached near the lower part of the fuselage. It has a swept-wing design, which means that the wing sweeps backward from the fuselage. This design helps to reduce drag and improve aerodynamic efficiency.

The wing is made primarily of lightweight aluminum alloys, which provide strength while keeping the overall weight of the aircraft low. It consists of multiple structural components, including spars, ribs, and stringers, that work together to distribute the aerodynamic loads across the wing and ensure its structural integrity.

One notable feature of the designed aircraft is the addition of advanced technology called "Advanced Technology Winglets" or "AT Winglets." These winglets are vertically oriented extensions at the tips of the wings, designed to reduce drag and improve fuel efficiency by reducing the vortices at the wingtips. The winglets on the designed aircraft are slightly larger compared to the prototype, further enhancing the aircraft's performance.

Overall, the wing construction of the Boeing designed aircraft combines lightweight materials, aerodynamic design, and advanced winglets to optimize fuel efficiency, improve performance, and enhance the aircraft's overall capabilities.

1.2.2 Fuselage

The fuselage of the designed aircraft is constructed using advanced materials and manufacturing techniques to ensure strength, durability, and weight efficiency. It is primarily made of lightweight aluminum alloys, which provide a balance between strength and weight.

The fuselage is divided into several sections, including the forward section, midsection, and aft section. These sections are typically manufactured separately and then joined together during the assembly process.

The main structural framework of the fuselage consists of longitudinal stringers and circumferential frames. The stringers are long, thin beams that run along the length of the fuselage, while the frames are curved structures that encircle the fuselage. These components work together to distribute the structural loads evenly and maintain the fuselage's shape and integrity.

To further enhance the strength and rigidity of the fuselage, it also incorporates various reinforcing elements such as bulkheads and floor beams. Bulkheads are partitions that separate different sections of the aircraft, while floor beams provide support for the cabin floor and cargo compartments.

1.2.3 Tail unit

The vertical stabilizer is the upright fin located at the rear of the aircraft. It is responsible for providing stability in the yaw axis, which controls the left and right movement of the aircraft's nose. The vertical stabilizer of the designed aircraft is constructed using lightweight materials, typically composite materials, to reduce weight and improve fuel efficiency. It is attached to the fuselage and incorporates a rudder, which is a movable control surface used to control the aircraft's yaw.

The horizontal stabilizer is positioned at the rear of the fuselage, near the vertical stabilizer. Its main function is to provide stability in the pitch axis, controlling the up and down movement of the aircraft's nose. The horizontal stabilizer of the designed aircraft is also constructed using lightweight materials, often composites, to minimize weight and optimize performance. It includes elevators, that are movable surfaces attached to the trailing edge of the stabilizer, used to control the aircraft's pitch.

The tail unit construction incorporates structural components such as spars, ribs, and stringers to ensure strength and rigidity. These components are typically made of aluminum alloys or composites, depending on the specific design requirements.

Advanced design techniques, such as computational fluid dynamics and wind tunnel testing, are employed to optimize the shape and aerodynamic performance of the tail unit. This ensures efficient airflow and minimizes drag, contributing to the overall performance and fuel efficiency of the aircraft.

1.2.4 Landing gear

The landing gear system of the designed aircraft comprises the main landing gear and the nose landing gear. The main landing gear consists of two sets of wheels located under the wings near the aircraft's center of gravity. The nose landing gear is positioned under the aircraft's forward fuselage and includes a single set of wheels.

The construction of the landing gear involves a combination of high-strength steel, aluminum alloys, and titanium, depending on the specific components and their requirements. These materials are chosen for their strength, durability, and resistance to fatigue and corrosion.

The landing gear system is designed to withstand the forces experienced during landing, including the weight of the aircraft, impact loads, and braking forces. It incorporates shock absorbers, commonly known as struts, to absorb the energy from landing impacts and provide a smooth ride for passengers. These struts contain hydraulic systems that dampen the forces during touchdown.

The wheels and tires of the landing gear are designed to withstand high loads and extreme conditions. They are made from robust materials and are equipped with reliable braking systems to ensure effective deceleration during landing and ground operations.

The landing gear construction also includes various mechanisms, such as retraction and extension systems, which allow the landing gear to be retracted in the aircraft's fuselage during flight and deployed for landing. These mechanisms are typically hydraulic or electrically operated and undergo rigorous testing to ensure their reliability and safety.

Additionally, the landing gear is equipped with various sensors and indicators to provide feedback to the flight crew regarding the position, status, and performance of the landing gear system.

1.2.5 Cockpit

Two-person "glass" cockpit is equipped by controls of all the aircraft systems located on the top panel. The display system belongs to the second generation, which consists of three display management computers and two system data acquisition devices. It adopts the second-generation digital automatic flight system integrated with the fly-by-wire control system. Therefore, there is no special guidance computer and engine thrust control computer on the aircraft and there is no independent servo mechanism for autopilot and auto throttle. These functions are all included in the flight management computer. The command signal of the computer is provided to the pitch through the fly-by-wire control computer and the roll control surface is controlled by the flight stabilization computer to control the yaw. Thrust control is part of the engine's full-function electronic control system. There are only six computers for such automatic flight control systems. The second-generation digital automatic flight system with a higher degree of comprehensiveness is adopted which improves safety and reliability, simplifies the system and reduces the cost and weight of the aircraft. A centralized fault display system is adopted. When the aircraft system fails, the two cathode-ray tubes in the center of the instrument panel respectively display warning signals and system movement. The controller and display device automatically analyze the cause of the failure through the integrated failure display system to avoid unnecessary reading of manuals and documents.

Each cabin crew member place is equipped with a communication system, including a telephone and a flight attendant panel. The cabin door has a single-channel evacuation slide which is equipped with a lighting system. Oxygen masks are located above the passengers and each cabin crew member including some other emergency equipment.

1.2.6 Control system

The control system of the designed aircraft consists of various components and mechanisms that enable the pilots to control the aircraft's flight path, attitude, and performance. It includes both primary and secondary control systems. Here's a short description of its construction.

The primary control surfaces of the designed aircraft include the ailerons, elevators, and rudder. These control surfaces are located on the wings and tail and are directly manipulated by the pilot's inputs through the control stick or rudder pedals. The ailerons control roll by moving in opposite directions, the elevators control pitch by moving together, and the rudder controls yaw movement.

The designed aircraft utilizes a fly-by-wire system, which replaces traditional mechanical linkages with electronic signals. It consists of electronic sensors that detect the pilot's input and transmit the information to the flight control computers.

The computers then interpret these inputs and send appropriate commands to actuators located near the control surfaces, which move the surfaces accordingly.

The flight control computers in the designed aircraft receive inputs from various sources, including the pilot's controls, sensors, and other systems. They continuously monitor the aircraft's flight parameters, such as airspeed, altitude, and attitude, and process this information to determine the appropriate control commands. These computers also provide envelope protection features to enhance the aircraft's safety and stability.

The control laws in the designed aircraft define the response characteristics and limitations of the fly-by-wire system. These laws ensure smooth and predictable handling by automatically adjusting control inputs based on flight conditions and aircraft configuration. The control laws help maintain stability, prevent excessive maneuvers, and enhance the aircraft's overall performance.

In addition to the primary control surfaces, the designed aircraft has secondary control systems that assist in controlling the aircraft. These systems include spoilers, flaps, slats, and leading-edge devices. Spoilers are deflected to reduce lift and assist in descent and braking. Flaps and slats increase lift during take-off and landing, while leading-edge devices enhance the aircraft's performance at low speeds.

1.2.7 Onboard equipment

The onboard equipment of the designed aircraft consists of a range of systems and equipment designed to enhance the safety, efficiency, and comfort of passengers and crew during flight. Here's a short description of some onboard equipment.

The cockpit of the designed aircraft is equipped with modern avionics and instruments, including primary flight displays and multifunction displays.

The flight management system is a computer-based system that assists the pilots in planning and managing the aircraft's flight. It calculates optimal flight routes, controls the autopilot, and manages various navigation functions.

The aircraft's navigation systems include inertial navigation systems and GPS, which provide accurate positioning and guidance information to the flight crew.

The designed aircraft may be equipped with in-flight entertainment systems to enhance the passenger experience. These systems typically include seatback screens or overhead displays that offer a variety of entertainment options, such as movies, TV shows, music, and games.

The cabin management systems control various aspects of the passenger cabin, including lighting, temperature, and audio systems. These systems allow for customized cabin environments, ensuring passenger comfort throughout the flight.

The designed aircraft incorporates numerous safety systems, such as Traffic Collision Avoidance System, Terrain Awareness and Warning System, and Enhanced Ground Proximity Warning System. These systems provide warnings and alerts to the flight crew regarding potential conflicts, terrain hazards, and other information.

The aircraft is equipped with emergency equipment, including life rafts, life vests, oxygen masks, and fire suppression systems. These equipment and systems are critical for ensuring the safety and well-being of passengers and crew in case of emergencies.

1.2.8 Choice and description of power plant

The engine used for the designed aircraft is turbofan Rolls-Royce RB211 due to its take-off thrust that should be bigger then obtained one in the calculations. The RB211 engine differs from other engines in its class by having three instead of two shafts. Each shaft has a compressor and turbine section on its ends. This design permitted each compressor to run nearer its optimum speed and efficiency. It also reduces the number of blades and other parts, required for the engine. RB211 utilizes advanced materials, such as lightweight composites, to reduce the engine’s weight. The data of this engine are presented in table 1.2.

Table 1.2

Engine performances

Model	Dry weight	Take-off thrust	Bypass ratio	Pressure ratio	Temperature in combustion chamber	Thrust-specific fuel consumption
RB211	5411 kg	182.5 kN	5	24.7	1850 °C	10.8 g/kN·s

Conclusions to the analytical part

Based on the analysis of choused prototypes, the long-range aircraft with a capacity of up to 220 passengers was designed. The Boeing 737 MAX 9 was chosen as the main prototype for the one, developed in this work, because of its novelty (entered the service in March 2018) and wide usage of high-tech technologies. This model attracts with its improved performances, longer flight range, bigger fuselage and wide usage of composite materials in its design. In this part also was performed an engine choice for the aircraft and made a technical description of its principal structural units.

2. AIRCRAFT MAIN PARTS CALCULATIONS

2.1 Geometry calculations for the main parts of the aircraft

The layout of the aircraft is determined by the relative location of the units, their number and shape. Its aerodynamic and operational properties depend on the scheme and aerodynamic layout of the aircraft. A well-chosen scheme makes it possible to increase the safety and regularity of flights, and the economic efficiency of the aircraft. The selection of the scheme of the designed aircraft is preceded by the study and analysis of the schemes of aircrafts accepted as prototypes.

2.1.1 Wing geometry calculation

Full wing area is:

$$S_w = \frac{m_0 \cdot g}{P_0} = \frac{116422 \cdot 9.81}{5557} = 205.45 \text{ m}^2,$$

where m_0 – take-off weight, kg; g – gravity acceleration, m/s^2; P_0 – specific wing load, N/m^2. Relative wing extensions area is 0.01.

Wing span is:

$$l_w = \sqrt{S_w \cdot \lambda_w} = \sqrt{205.45 \cdot 9.4} = 43.95 \text{ m},$$

where λ_w – wing aspect ratio.

Root chord is:

$$b_0 = \frac{2S_w \cdot \eta_w}{(1+\eta_w)\cdot l_w} = \frac{2 \cdot 205.45 \cdot 4.11}{(1+4.11)\cdot 43.95} = 7.51 \text{ m},$$

where η_w – wing taper ratio.

Tip chord is:

$$b_t = \frac{b_0}{\eta_w} = \frac{7.51}{4.11} = 1.83 \text{ m},$$

Maximum wing thickness is:

$$c_{\max} = c_w \cdot b_t = 0.12 \cdot 1.83 = 0.22 \text{ m},$$

where c_w – medium wing relative thickness.

On board chord is:

$$b_b = b_0 \cdot \left(1 - \frac{(\eta_w - 1) \cdot D_f}{\eta_w \cdot l_w}\right) = 7.51 \cdot \left(1 - \frac{(4.11 - 1) \cdot 3.95}{4.11 \cdot 43.95}\right) = 7.0 \text{ m},$$

where D_f – fuselage diameter.

For mean aerodynamic chord determination the geometrical method was used (fig. 2.1). The geometrical method implies the measuring of parallel to the chords line which lies on the intersection of the section connecting the middles of tip and root chords with another section connecting the upper end of tip chord extension (which is equal to the length of root chord) with lower end of root chord extension (which is equal to the length of the tip chord). This method was chosen due to accuracy and simplicity in performance.

Thus, the mean aerodynamic chord is equal to 5.25 m.

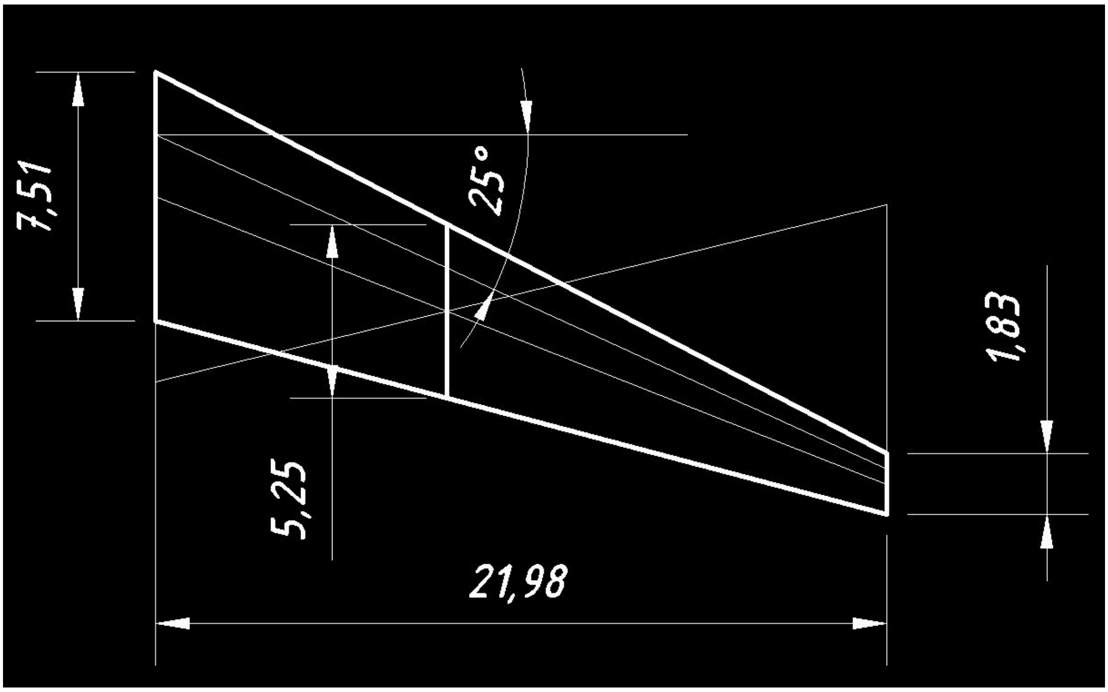

Fig. 2.1. Geometrical method of determination of mean aerodynamic chord.

To choose the structural scheme of the wing it is necessary to determine the type of its internal design. The torsion-box type with two spars was chosen to meet the requirements of strength and at the same time to make the structure comparatively light.

For wing geometry estimation it is necessary to determine and calculate the main parameters of control surfaces.

Ailerons geometrical parameters are determined in the next order:

Ailerons span:

$$l_{ail} = (0.3...0.4) \cdot \frac{l_w}{2} = 0.3 \cdot \frac{43.95}{2} = 6.59 \text{ m}.$$

Aileron chord:

$$b_{ail} = (0.2...0.26) \cdot b_t = 0.25 \cdot 1.83 = 0.46 \text{ m}.$$

Aileron area:

$$S_{ail} = (0.05...0.08) \cdot \frac{S_w}{2} = 0.05 \cdot \frac{205.45}{2} = 5.14 \text{ m}^2.$$

The calculated above values are recommended. Increasing of aileron span and chord more than these values are not convenient because with the increase of aileron span the aileron’s coefficient decreases, as well as the high-lift devices span. In case of aileron chord, increasing of its value lead to the decreasing of wing’s box width.

Aerodynamic compensation of the aileron:

$$\text{Axial } S_{ax.ail} \le (0.25...0.28) \cdot S_{ail},$$

$$S_{ax.ail} = 0.25 \cdot 5.14 = 1.28 \text{ m}^2.$$

Area of ailerons trim tab. For two engine airplane:

$$S_{tt} = (0.04...0.06) \cdot S_{ail} = 0.05 \cdot 5.14 = 0.257 \text{ m}^2.$$

Range of aileron deflection for upward is 25 degrees, downward is 15 degrees.

2.1.2 Fuselage layout

Generally, the fuselage layout estimation consists of main geometrical dimensions calculation and interior scheme creation.

In case of geometrical calculation, it is necessary to take into account the expected aerodynamic characteristics of designed airplane, typical resistances during normal and extreme flight conditions in accordance with estimated purpose. Airplane's fuselage geometry should allow to avoid high values of parasitic, skin friction and wave drags, withstand the aerodynamic loads and have as greater as possible safety factor value. To decrease form and wave drug and to provide necessary strength characteristics avoiding the stress concentrators in fuselage cross-section the round shape was chosen.

Another part of fuselage calculation as interior scheme creation is based on the required capacity of designed aircraft. Besides that, the requirements of ergonomics and sanitary standards must be considered for passenger aircrafts.

In the next steps it is necessary to calculate the main geometrical characteristics of the fuselage and consequently to obtain its outline.

Nose part length is:

$$L_{np} = (2...3) \cdot D_f = 2 \cdot 3.95 = 7.9 \text{ m}.$$

Fuselage length is:

$$L_f = \lambda_f \cdot D_f = 11.26 \cdot 3.95 = 44.48 \text{ m}.$$

where λ_f – fuselage fineness ratio.

Fuselage nose part fineness ratio is:

$$\lambda_{np} = \frac{L_{np}}{D_f} = \frac{7.9}{3.95} = 2.$$

Length of the fuselage rear part is:

$$L_{rp} = \lambda_{rp} \cdot D_f = 3.1 \cdot 3.95 = 12.24 \text{ m},$$

where λ_{rp} – fuselage rear part fineness ratio.

During the determination of fuselage length it is necessary to get minimum mid-section on the one hand and meet layout demands on the other.

For passenger airplanes fuselage mid-section first of all depends on the size of passenger cabin. One of the main parameter that determines the mid-section of passenger airplane is the height of the passenger cabin.

Cabin height is:

$$H_{cab} = 1.48 + 0.17 B_{cab} = 1.48 + 0.17 \cdot 3.78 = 2.12 \text{ m},$$

where B_{cab} – width of the cabin, m.

For economic class passenger cabin the location of seats in one row (3+3) determine the next parameter:

$$B_{cab} = n_{3chblock} \cdot b_{3chblock} + b_{aisle} + 2 \cdot \delta = 2 \cdot 1.48 + 0.58 + 2 \cdot 0.12 = 3.78 \text{ m},$$

where $n_{3chblock}$ – width of 3 chairs; $b_{3chblock}$ – number of 3 chair block, m; b_{aisle} – width of aisle, m; δ – distance between external armrests to the decorative panels, m.

The length of passenger cabin is:

$$L_{cab} = L_1 + (n_1 - 1) \cdot L_{sp} + L_2 = 1.25 + (33 - 1) \cdot 0.8 + 0.3 = 27.15 \text{ m},$$

where L_1 – distance between the wall and the back of first seat, m; n_1 – number of rows; L_{sp} – seat pitch, m; L_2 – distance between the back of last seat and the wall, m.

2.1.3 Luggage compartment

Given the fact that the unit load on floor K = 400…600 kg/m^2.

The area of cargo compartment is:

$$S_{cargo} = \frac{M_{bag}}{0.4 \cdot K} + \frac{M_{c\&m}}{0.6 \cdot K} = \frac{220 \cdot 20}{0.4 \cdot 600} + \frac{220 \cdot 15}{0.6 \cdot 600} = 27.5 \text{ m}^2,$$

where M_{bag} – mass of the baggage, kg; $M_{c\&m}$ – mass of the cargo and mail, kg.

Cargo compartment volume is:

$$V_{cargo} = \nu \cdot n_{pass} = 0.23 \cdot 220 = 50.6 \text{ m}^3,$$

where ν – relative mass of baggage (0.2…0.4 for $D_f \le 4$ m and 0.36…0.38 for $D_f > 4$ m); n_{pass} – number of passengers.

Luggage compartment design is similar to the prototype.

2.1.4 Galleys and buffets

International standards require two dishes for airplanes with mixed layout. Kitchen cupboards must be placed between the cockpit and passenger cabin. Refreshment and food can not be placed near the toilets and wardrobes. Volume of buffets and galleys is equal to:

$$V_{galley} = (0.1...0.12) \cdot n_{pass} = 0.1 \cdot 220 = 22 \text{ m}^3,$$

where V_{galley} – volume of buffets, m^3; n_{pass} – number of passengers.

Area of buffets (galleys) is:

$$S_{galley} = \frac{V_{galley}}{H_{cab}} = \frac{22}{2.12} = 10.38 \text{ m}^2.$$

Number of meals per passenger breakfast, lunch and dinner – 0.8 kg, tea and water – 0.45 kg. Passengers are fed every 3.5…4 hour of flight. Buffet design is similar to prototype.

2.1.5 Lavatories

Number of toilet facilities is determined by the number of passengers and flight duration: with t > 4 hours should be one toilet for 40 passengers. The number of lavatories is equal to:

$$N_{lav} = \frac{n_{pass}}{40} = \frac{220}{40} = 5.5.$$

So the chosen number of lavatories is 6. Area of each lavatory is 1.6 m^2 and width – 1 m. Toilets design is similar to the prototype.

2.1.6 Layout and calculation of basic parameters of tail unit

The chosen tail unit scheme is conventional. This choice is based on all three prototypes empennage schemes.

To estimate the general tail unit outlines it is necessary to calculate the geometrical dimensions of vertical and horizontal stabilizers and dimensions of control surfaces. In general tail unit must to meet the requirements of aircraft stability and controllability.

Area of vertical tail unit is:

$$S_{VTU} = \frac{l_w \cdot S_w}{L_{VTU}} \cdot A_{VTU} = \frac{43.95 \cdot 205.45}{9.135} \cdot 0.05 = 49.42 \text{ m}^2,$$

where L_{VTU} – length of vertical tail unit, m; A_{VTU} – coefficient of static momentum of vertical tail unit.

Area of horizontal tail unit is:

$$S_{HTU} = \frac{b_{MAC} \cdot S_w}{L_{HTU}} \cdot A_{HTU} = \frac{5.25 \cdot 205.45}{11.3} \cdot 0.65 = 62.04 \text{ m}^2,$$

where L_{HTU} – length of horizontal tail unit, m; A_{HTU} – coefficient of static momentum of horizontal tail unit.

Determination of the elevator area and rudder area:

Altitude elevator area is:

$$S_{el} = k_{el} \cdot S_{HTU} = 0.2 \cdot 62.04 = 12.41 \text{ m}^2,$$

where k_{el} – relative elevator area coefficient.

Rudder area is:

$$S_{rud} = k_r \cdot S_{VTU} = 0.25 \cdot 49.42 = 12.35 \text{ m}^2,$$

where k_r – relative rudder area coefficient.

Choice of the aerodynamic balance area:

$$S_{eb} = (0.22...0.25) \cdot S_{el} = 0.22 \cdot 12.41 = 2.73 \text{ m}^2,$$

$$S_{rb} = (0.2...0.22) \cdot S_{rud} = 0.2 \cdot 12.35 = 2.47 \text{ m}^2,$$

where k_{eb} – relative elevator balance area coefficient; k_{rb} – relative rudder balance area coefficient.

The area of altitude elevator trim tab is:

$$S_{te} = k_{te} \cdot S_{el} = 0.08 \cdot 12.41 = 0.99 \text{ m}^2,$$

where k_{te} – relative elevator trim tab area coefficient (k_{te} = 0.08…0.12).

Area of rudder trim tab is:

$$S_{tr} = k_{tr} \cdot S_{rud} = 0.06 \cdot 12.35 = 0.741 \text{ m}^2,$$

where k_{tr} – relative trim tab area coefficient (k_{tr} = 0.04…0.06 for airplanes with 2 engines and k_{tr} = 0.06…0.1 for airplanes with 4 engines).

Root chord of horizontal stabilizer is:

$$b_{0HTU} = \frac{2 \cdot S_{HTU} \cdot \eta_{HTU}}{(1+\eta_{HTU}) \cdot L_{HTU}} = \frac{2 \cdot 62.04 \cdot 3.69}{(1+3.69) \cdot 11.3} = 8.64 \text{ m},$$

where η_{HTU} – horizontal tail unit taper ratio; L_{HTU} – horizontal tail unit span, m.

Tip chord of horizontal stabilizer is:

$$b_{tHTU} = \frac{b_{0HTU}}{\eta_{HTU}} = \frac{8.64}{3.69} = 2.34 \text{ m}.$$

Root chord of vertical stabilizer is:

$$b_{0VTU} = \frac{2 \cdot S_{VTU} \cdot \eta_{VTU}}{(1+\eta_{VTU}) \cdot L_{VTU}} = \frac{2 \cdot 49.42 \cdot 4.92}{(1+4.92) \cdot 9.135} = 8.99 \text{ m}.$$

where η_{VTU} – vertical tail unit taper ratio; L_{VTU} – vertical tail unit span, m.

Tip chord of vertical stabilizer is:

$$b_{tVTU} = \frac{b_{0VTU}}{\eta_{VTU}} = \frac{8.99}{4.92} = 1.83 \text{ m}.$$

2.1.7 Landing gear design

To estimate the landing gear outline in this project it is necessary to calculate the location of every strut relatively to each other, to determine the loads on landing gear system, and its location considering centre of gravity of an airplane. In this layout the principal scheme of landing gear is fully based on the prototype data.

As in the case with the tail unit it is necessary to provide the aircraft with the stable and controllable base during operation on the ground including landing and take-off.

Main wheel axes offset is:

$$e = k_e \cdot b_{MAC} = 0.15 \cdot 5.25 = 0.79 \text{ m},$$

where k_e – coefficient of axes offset (k_e = 0.15…0.3); b_{MAC} – mean aerodynamic chord, m.

With the large wheel axial offset the lift-off of the front gear during take-off is complicated and with small the drop of the airplane on the tail is possible when the loading of the airplane back comes first.

Landing gear wheel base is:

$$B = k_b \cdot L_f = 0.4 \cdot 44.48 = 17.79 \text{ m},$$

where k_b – wheel base calculation coefficient (k_b = 0.3…0.4).

That means that the nose strut holds 5...11% of airplane weight.

Front wheel axial offset is:

$$d_n = B - e = 17.79 - 0.79 = 17 \text{ m}.$$

Wheel track is:

$$T = k_T \cdot B = 0.7 \cdot 17.79 = 12.45 \text{ m}.$$

where k_T – wheel track calculation coefficient (k_T = 0.7…1.2).

Wheels for the landing gear is chosen by the size and run loading caused by the take-off weight. For the front support it is necessary to consider dynamic loading as well. Type of the pneumatics (balloon, half balloon, arched) and the pressure in it is determined by the runway surface, which should be used. There are breaks on the main wheels.

Nose wheel load is:

$$P_n = \frac{9.81 \cdot e \cdot k_d \cdot m_0}{B \cdot i} = \frac{9.81 \cdot 0.79 \cdot 1.5 \cdot 116422}{17.79 \cdot 2} = 38025\ \text{N} = 8548\ \text{lbf},$$

where k_d – dynamics coefficient (k_d = 1.5…2.0); i – number of wheels.

Main wheel load is equal to:

$$P_m = \frac{9.81 \cdot (B - e) \cdot m_0}{B \cdot n \cdot i} = \frac{9.81 \cdot (17.79 - 0.79) \cdot 116422}{17.79 \cdot 2 \cdot 2} = 272752\ \text{N} = 61317\ \text{lbf},$$

where n – number of main landing gear struts.

According the calculated values of wheel loading and take-off speed we can choose the tires for landing gear. From the catalog we got:

- for nose landing gear:

Aircraft Rib 461B-3563-TL with parameters $P_{rated} = 8600\ \text{lbf}$; $V_{rated} = 250\ \text{MPH}$; size 18×5.7-8.

- for main landing gear:

Flight Leader 521K62-3 with parameters $P_{rated} = 62500\ \text{lbf}$; $V_{rated} = 225\ \text{MPH}$; size 52×20.5-20.

The rate of wheel loading is:

for nose wheel $\frac{8600 - 8548}{8600} \cdot 100\% = 0.60\%$

for main wheel $\frac{62500 - 61317}{62500} \cdot 100\% = 1.89\%$

The values are less than 10% so chosen tires can be used for this airplane.

2.2 Determination of the aircraft center of gravity position

The distance from the main aerodynamic chord to the centre of gravity of the airplane is called the centering. Due to changing of the aircraft loading variants or the weight during flight the position of aircraft centre of gravity is changing. The moving of the cargo inside the aircraft leads to changing of centre of mass position too.

The centering is important aircraft characteristic as it affects on the balancing, stability and controllability of the aircraft. That’s why it is necessary to keep it inside strict limits. To calculate the centering, it is necessary to determine the mass of main structural units and devices.

The longitudinal static stability of the aircraft is determined by the location of its centre of mass relatively to the focuses. The closer the centre of mass is to the nose part of the aircraft, the more longitudinal stability the aircraft have.

2.2.1 Determination of centering of the equipped wing

Mass of the equipped wing contains the mass of its structure, mass of the equipment located in the wing and mass of the fuel. Regardless of the place of mounting (to the wing or to the fuselage) the main landing gear and the front gear are included in the mass register of the equipped wing. The mass register includes names of the objects, mass themselves and their center of gravity coordinates. The origin of the given coordinates of the mass centers is chosen by the projection of the nose point of the mean aerodynamic chord (MAC) for the surface XOY. The positive values of the coordinates of the mass centers are accepted for the aft part of the aircraft.

The list of the mass objects for the aircraft, where the engines are located under the wing, included the names is given in the table 2.1. Coordinates of the center of mass for the equipped wing are determined by the formula:

$$X'_w = \frac{\sum m'_i \cdot x_i}{\sum m'_i},$$

where X'_w – center of mass for equipped wing, m; m'_i – mass of a unit, kg; x_i – center of mass of the unit, m.

Table 2.1

List of equipped wing masses

#	Object name	Mass		Center of gravity coordinates, m	Moment of mass, kg·m
		Units	Total mass, kg		
1	2	3	4	5	6
1	Wing (structure)	0.0910	10594	2.363	25029.27
2	Fuel system	0.0103	11.99	1.890	2266.39
3	Flight control system, 30%	0.0015	175	3.150	550.09
4	Electrical equipment, 20%	0.0061	710	0.525	372.84
5	Anti-icing system, 70%	0.0035	407	0.525	213.93

1	2	3	4	5	6
6	Hydraulic system, 30%	0.0044	512	3.150	1613.61
7	Power plant	0.0920	10711	-2.191	-23467.42
	Equipped wing without landing gear and fuel	**0.2088**	**24309**	**0.271**	**6578.72**
8	Nose landing gear	0.0049	570	-13.172	-7514.20
9	Main landing gear	0.0351	4086	2.800	11441.95
10	Fuel for flight	0.3080	35858	1.575	54476.31
11	Reserve fuel	0.0350	4075	1.838	7487.39
	Totally equipped wing	**0.5918**	**68899**	**1.081**	**74470.17**

2.2.2 Determination of the centering of the equipped fuselage

Mass of the equipped fuselage contains the mass of its structure, mass of the equipment located in it, mass of all commercial payload, crew, attendants and masses of special systems. The mass register includes names of the objects, mass themselves and their center of gravity coordinates. Origin of the coordinates is chosen in the projection of the fuselage nose on the horizontal axis. The list of the objects for the equipped fuselage, with engines are mounted under the wing, is given in table 2.2. The center of gravity coordinates of the equipped fuselage are determined by formula:

$$X'_f = \frac{\sum m'_i \cdot x_i}{\sum m'_i},$$

where X'_f – center of mass for equipped fuselage, m; m'_i – mass of a unit, kg; x_i – center of mass of the unit, m.

Table 2.2

List of equipped fuselage masses

#	Object name	Mass		Center of gravity coordinates, m	Moment of mass, kg·m
		Units	Total mass, kg		
1	2	3	4	5	6
1	Fuselage	0.0980	11409	22.240	253744.08
2	Horizontal tail unit	0.0094	1094	39.841	43600.45
3	Vertical tail unit	0.0093	1083	32.390	35069.45
4	Radiolocation equipment	0.0029	338	1.161	391.98
5	Dashboard and instrument equipment	0.0050	582	2.119	1233.49
6	Aero navigation equipment	0.0043	501	2.119	1060.80

Ending of the table 2.2

1	2	3	4	5	6
7	Radio equipment	0.0022	256	2.119	542.74
8	Flight control system, 70%	0.0036	419	24.464	10253.33
9	Electrical equipment, 80%	0.0246	2864	22.240	63694.94
10	Hydraulic system, 70%	0.0102	1188	21.031	21974.41
11	Anti icing system, 30%	0.0015	175	38.988	6808.63
12	Air-conditioning system	0.0151	1758	20.016	35187.57
13	Emergency equipment	0.0026	300	19.405	5821.50
14	Tools	0.0000	4	4.126	14.86
15	Water and liquid	0.0013	150	19.405	2910.75
16	Lavatory 1	0.0017	200	7.470	1494.00
17	Lavatory 2	0.0017	200	12.661	2532.20
18	Lavatory 3	0.0017	200	32.705	6541.00
19	Wardrobe 1	0.0001	10	7.470	74.70
20	Wardrobe 2	0.0001	10	12.661	126.61
21	Wardrobe 3	0.0001	10	32.705	327.05
22	Galley 1	0.0017	200	4.520	904.00
23	Galley 2	0.0017	200	35.210	7042.00
24	Baggage equipment	0.0019	220	19.405	4269.10
25	Interior panels, lining and insulation	0.0058	675	20.016	13515.76
26	Passengers' seats 1 (business class)	0.0013	150	9.881	1482.15
27	Passengers' seats 2 (economic class)	0.0086	1005	23.577	23694.89
28	Pilots' seats	0.0002	20	3.511	70.22
29	Flight attendants' seats	0.0003	30	19.405	582.15
30	Non-typical equipment	0.0035	407	8.522	3472.52
	Equipped fuselage without commercial load	**0.2204**	**25657**	**21.492**	**551437.31**
31	Passengers 1(business class)	0.0129	1500	9.881	14821.50
32	Passengers 2(economic class)	0.1288	15000	23.577	353655.00
33	Passengers' baggage	0.0283	3300	20.450	67485.00
34	Cargo, mail	0.0120	1400	20.450	28630.00
35	On board meal	0.0015	176	19.405	3415.28
36	Flight attendants	0.0028	330	19.405	6403.65
37	Crew	0.0014	160	3.511	561.76
	Totally equipped fuselage	**0.4082**	**47523**	**21.598**	**1026409.50**

2.2.3 Calculation of center of gravity positioning variants

After the center of gravity of fully equipped wing and fuselage is determined, the moment equilibrium equation relatively to the fuselage nose can been made:

$$m_f \cdot X'_f + m_w(X_{MAC} + X'_w) = m_0(X_{MAC} + C).$$

where m_0 – aircraft take-off mass, kg; m_f – mass of fully equipped fuselage, kg; m_w – mass of fully equipped wing, kg; C – distance from MAC leading edge to the center of gravity point determined by the designer.

From here it is possible to determine the wing MAC leading edge position relative to fuselage, means X_{MAC} value by the formula:

$$X_{MAC} = \frac{m_f \cdot X'_f + m_w \cdot X'_w - m_0 \cdot C \cdot b_{MAC}}{m_0 - m_w},$$

$$X_{MAC} = \frac{47523 \cdot 21.598 + 68899 \cdot 1.081 - 116422 \cdot 0.22 \cdot 5.25}{116422 - 68899} = 20.336\ \text{m}.$$

The list of mass objects for center of gravity variants calculation is given in table 2.3 and center of gravity calculation options are given in table 2.4 completed on the base of both previous tables.

Table 2.3

Calculation of center of gravity position variants

#	Object name	Mass, kg	Center of gravity coordinates, m	Moment of mass, kg·m
1	2	3	4	5
1	Equipped wing without landing gear and fuel	24309	20.606	500912.08
2	Nose landing gear (extended)	570	7.163	4086.53
3	Main landing gear (extended)	4086	23.135	94541.10
4	Fuel for flight	35858	21.910	785665.38
5	Reserve fuel	4075	22.173	90349.78
6	Equipped fuselage without commercial load	25657	21.598	554148.65
7	Passengers 1 (business class)	1500	9.881	14821.50
8	Passengers 2 (economic class)	15000	23.577	353655.00
9	Baggage of passengers	3300	20.450	67485.00
10	Cargo, mail	1400	20.450	28630.00
11	On board meal	176	19.405	3415.28
12	Flight attendants	330	19.405	6403.65
14	Crew	160	3.511	561.76
15	Nose landing gear (retracted)	570	5.762	3287.14
16	Main landing gear (retracted)	4086	23.135	94541.10

Table 2.4

Aircraft's center of gravity position variants

#	Variant of loading	Mass, kg	Moment of mass, kg·m	Center of gravity coordinates, m	Centering, %
1	2	3	4	5	6
1	Take-off mass (landing gear extended)	116422	2504675.71	21.514	22.44
2	Take-off mass (landing gear retracted)	116422	2503876.32	21.507	22.31
3	Landing variant (landing gear extended)	80388	1715595.05	21.341	19.16
4	Transportation variant (without payload)	94716	2029465.89	21.427	20.79
5	Parking variant (without fuel and payload)	54623	1153688.36	21.121	14.96

Conclusions to the project part

In this part, the main geometric dimensions and centering of the designed aircraft were calculated.

During the calculation, the main geometric parameters were determined by taking into account the operational purpose, the planned number of passengers, the speed and height of the flight, the landing and take-off conditions. All obtained values meet the requirements for long-range passenger aircraft.

Centering of the designed plane is performed. The most forward position of the center of gravity of the equipped aircraft is 19.16% of the main aerodynamic chord during flight. The most aft position of the aircraft center of gravity is 22.44% of the main aerodynamic chord. Between these values, the plane is centered.

As a result an analysis and substantiation of the designed aircraft layout was carried out in accordance with the issued technical task, computer calculations and drawings based on the B737 MAX 9 prototype were performed. The geometric parameters almost coincide with the chosen prototype.

3. THE COMPOSITE UPPER WING SKIN

3.1 Introduction

Composite materials are increasingly being used in the aerospace industry due to their unique properties and advantages over traditional materials such as aluminum. The wing skin of an aircraft is one of the key areas where composites are used extensively.

A composite material is a material made up of two or more constituent materials with different physical or chemical properties that, when combined, produce a material with enhanced properties that would not be achievable with any of the individual materials alone. The constituents can be in the form of fibers, particles, or matrices.

The fibers are typically made of materials such as glass, carbon, or aramid, while the matrix materials are often polymers, metals, or ceramics. The fibers provide strength and stiffness to the composite, while the matrix material holds the fibers in place and transfers loads between them.

The resulting composite material can have a range of properties, such as increased strength, stiffness, toughness, corrosion resistance, and thermal stability, making it suitable for a wide range of applications in various industries, including aerospace, automotive, construction, and sports equipment. Composite materials are also known for their ability to be tailored to specific design requirements, allowing for greater flexibility in their use.

The use of composite materials for the wing skin of an aircraft offers several advantages such as increased strength and stiffness, reduced weight, improved fatigue resistance, and enhanced corrosion resistance. These properties allow for greater fuel efficiency, increased range and payload capacity, and lower maintenance costs.

The main goal of providing composite materials for the wing skin of an aircraft is to improve the performance and efficiency of the aircraft while reducing its weight. This is particularly important in the aerospace industry, where even small reductions in weight can have a significant impact on an aircraft's performance and fuel efficiency.

The object of study is the upper part of the wing skin. Modeling of a part of the skin was carried out in the SolidWorks 2021 software. The goal of the study is to compare the physical, technological, and mechanical properties of the same skin made of aluminum alloy and composite material. Also, it is important to achieve better mechanical properties of the composite wing skin under the same loads, as well as to reduce the technological complexity of production and maintenance of the wing skin's airworthiness.

3.2 Comparison of aviation-grade aluminum and carbon fiber reinforced plastic

The selected materials for the work are aluminum alloy (2024-T4) and composite material made of epoxy as a matrix and carbon fibers as a reinforcing material. The data of materials are given in tables 3.1-3.2.

Table 3.1

Alloy 2024-T4 properties

Physical Properties	Value	Comments
Density	2.78 g/cc	
Hardness, Brinell	120	500 g load, 10 mm ball
Ultimate tensile strength	469 MPa	
Tensile yield strength	324 MPa	
Elongation at break	19%	12.7 diameter
Elongation at break	20%	1.6 thickness
Modulus of elasticity	73.1 GPa	Average of tension and compression
Poisson's ratio	0.33	
Fatigue strength	138 MPa	500,000,000 cycles reversed stress
Machinability	70%	
Shear modulus	28 GPa	
Shear strength	283 MPa	

Table 3.2

CFRP properties (HEXCEL prepreg)

Physical Properties	Value	Comments
Density	1.6 g/cc	Average
Hardness, Mohs	2	
Ultimate tensile strength	800-750 MPa	Plain weave fabric
Tensile yield strength	720 MPa	
Elongation at break	0.8%	
Modulus of elasticity	65 GPa	High strength carbon
Poisson’s ratio	0.05	
Fatigue strength	600 MPa	500 000 000 cycles reversed stress
Machinability	Easy	Dust and anti-abrasive protection
Shear modulus	30 GPa	
Shear strength	70 MPa	

Using sandwich construction, stiffness and flexural strength are incredibly increased (fig. 3.1).

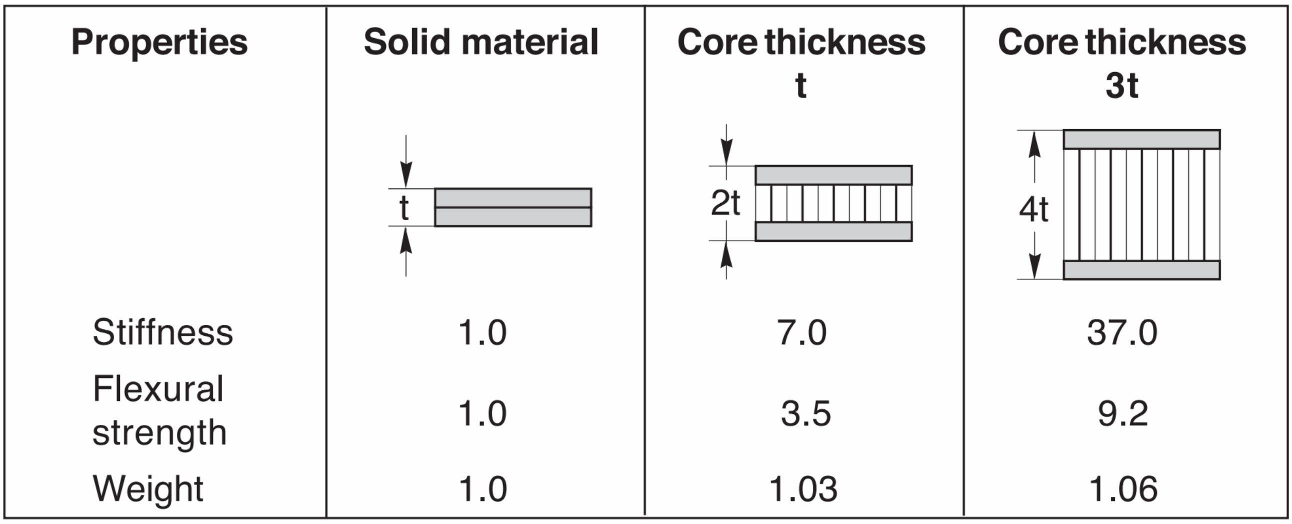

Fig. 3.1. Properties of sandwich structures.

Comparing the tables of material properties, it can be concluded that the carbon fiber construction will be 42% lighter, able to withstand higher stresses under the same service life, but 11% less rigid, which can be compensated by adding a core.

3.3. Determination of the forces and moments acting on the aircraft wing skin during operation.

During operation, the aircraft wing skin is subjected to various forces and moments (fig. 3.2). The major ones are:

- Lift force: this is the force that lifts the aircraft into the air, generated by the wing as it moves through the air;

- Weight force: this is the force due to the weight of the aircraft acting downward;

- Drag force: this is the force that resists the forward motion of the aircraft as it moves through the air;

- Shear force: this is the force acting parallel to the surface of the wing skin, which tends to cause the material to slide or deform.

- Bending moment: this is the moment that causes the wing skin to bend, due to the lift and weight forces acting at different points along the wing span;

- Torsion moment: this is the moment that causes the wing skin to twist, due to the difference in lift generated at the wing root and wing tip.

The wing skin material should be able to withstand these forces and moments without undergoing excessive deformation or failure.

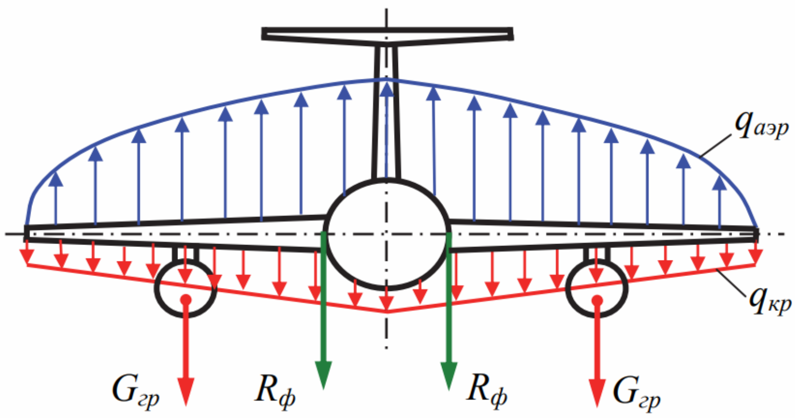

Fig. 3.2. The flight load case for the aircraft wing.

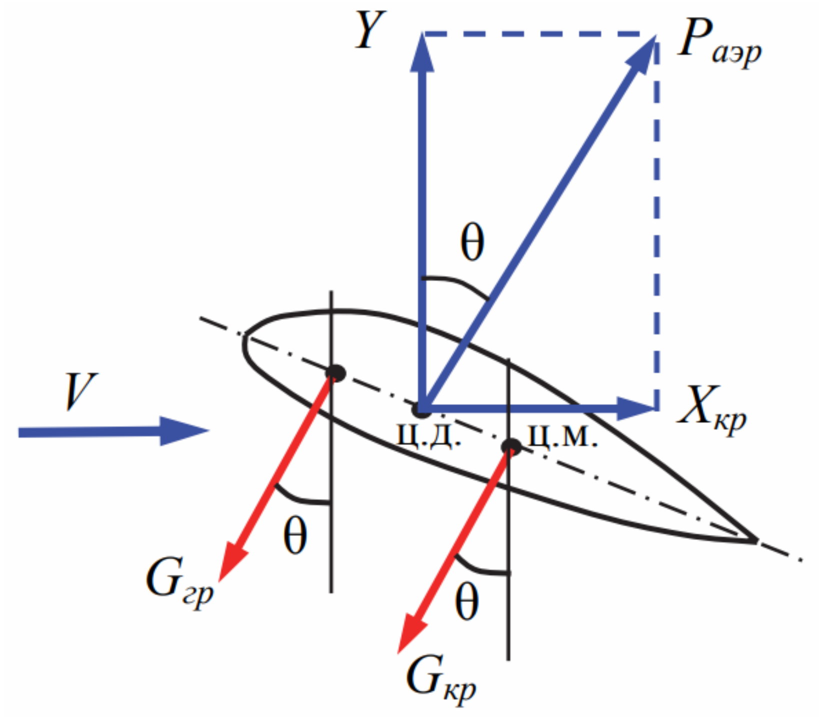

Fig. 3.3. The resultant of aerodynamic and inertial forces.

During flight, forces continuously change in magnitude and direction. The resultant of the aerodynamic forces on the wing P_{aer} (fig. 3.3), applied at the center of pressure, can be decomposed into the wing's drag force X_{kr}, directed along the flow (along the flight speed vector V), and the lift force Y, directed perpendicular to the flow. Aerodynamical force P_{aer} is determined as:

$$P_{aer} = \frac{Y}{\cos\Theta}, \tag{3.1}$$

where $\Theta = \tan^{-1}\left(\frac{X_{kr}}{Y}\right) = \tan^{-1}\left(\frac{C_{xkr}}{C_{ykr}}\right)$.

The coefficients of aerodynamic drag C_{xkr} and lift C_{ykr} are determined from the wing polar diagram. In the operational range, the angle Θ is relatively small, so in practical engineering calculations $\cos\Theta \approx 1$ is usually assumed. So, $P_{aer} \approx Y$.

The calculated (destructive) lift force of the wing is equal to:

$$Y^P = Y^E \cdot f = G \cdot n^E = G \cdot n^P,$$

where f – safety coefficient; n^E – operational load factor; n^P – destructive load factor.

The distribution of the total aerodynamic load along the wing span can be approximated by the distribution law of lift force according to (3.1), although there is some deviation due to different distribution laws of X_{kr} and Y along the span (fig. 3.4). Then, the linear aerodynamic load at any cross-section with a coordinate z will be equal to:

$$q_{aer}^{P} \approx q_{y}^{P} = q_{y}^{E} \cdot f = C_{ysec} \cdot \left(\frac{\rho \cdot V^{2}}{2} \right) \cdot b_{sec} \cdot f, \tag{3.2}$$

where C_{ysec} – is the coefficient of wing lift at the selected section; b_{sec} – is the chord of the wing at the given section.

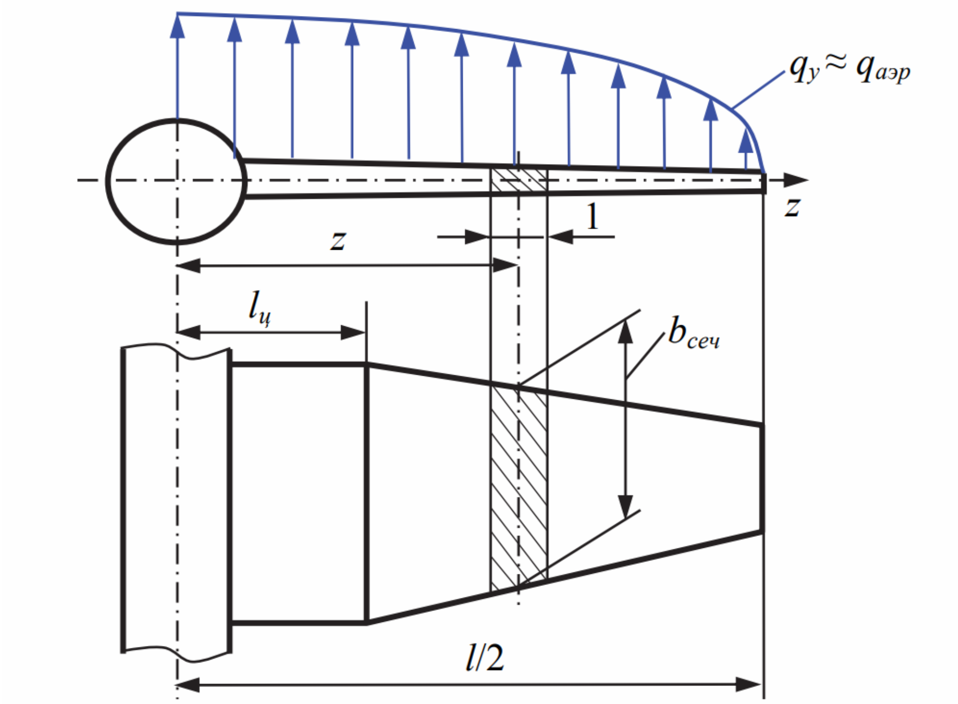

Fig. 3.4. Lift force distribution along the wing span.

$$Y^{E} = C_{ykr} \cdot \left(\frac{\rho \cdot V^{2}}{2} \right) \cdot S = G \cdot n^{E} \Rightarrow \frac{G \cdot n^{E}}{C_{ykr} \cdot S} \tag{3.3}$$

Combining the (3.2) and (3.3) it can be got:

$$q_{aer}^{P} = C_{ysec} \cdot \frac{G \cdot n^{E}}{C_{ykr} \cdot S} \cdot b_{sec} \cdot f = \frac{G \cdot n^{E} \cdot b_{mgc} \cdot f}{S} \cdot \frac{C_{ysec} \cdot b_{sec}}{C_{ykr} \cdot b_{mgc}},$$

where b_{mgc} – mean geometric chord.

Having determined the loads that act on the wing, it is possible to consider in what sequence and exactly how these loads are perceived by the structural elements of the wing and transmitted to the fastening joints. The latter will allows better understand the purpose of the structural elements of the wing that are the interest of the investigation – skin and stringers.

The skin, relying on stringers and ribs as linear supports, directly perceives the air load in the form of pressure and rarefaction forces. At the same time, the skin element works in tension (if the skin is thick, then also in bending). This load, in the form of transverse distributed forces is transmitted to the stringers and ribs through the rivets, which work for separation. The stringers, which attached to the top of the ribs, as multi-support beams, perceive the distributed load from the skin and transfer it in the form of small concentrated forces to the ribs.

The skin forms the surface of the wing, gives it a streamlined shape in accordance with the selected airfoil, directly perceives the aerodynamic load and transfers it to the elements of the longitudinal and transverse sets of the wing, works on the shear from the torsion of the wing, participates in the perception of the bending moment of the wing, while working together with these stringers for tension or compression.

The ratio of the weight of the skin to the weight of the wing is approximately 0.25...0.4, depending on the structural scheme of the wing.

The simplest in design and the most widespread is a skin made of duralumin sheets, for aircraft with a high supersonic speed – a skin made of steel or titanium alloys. From composite materials, carbon and boroplastics began to be used for skin. The thickness of the skin δ can be from a few tenths of a millimeter to 10 mm or more. Skin sheets are butt-jointed or overlapped with the help of hidden rivets, glued or welded seams. From an aerodynamic point of view, the butt connection is the best, which is used in this work. Joints are usually placed on frame elements. Recently, three-layer skin, wafer-type skin and monolithic panels have become widespread.

Work of the cladding in tension, compression and shear: in the case of tension, the skin is destroyed at stresses $\sigma_{ten} = k \cdot \sigma$, where k = 0.8...1.0 is a coefficient that takes into account the stress concentration at the joints. The value of k depends on the type of connection, for example, for a riveted connection $k = 0.95$.

When compressed and sheared, the skin loses its stability. For the skin element – a plate that attached to the ribs and stringers – the buckling critical stress σ_{kr} and τ_{kr} within the limits of elasticity are determined by the formulas:

- during compression $\sigma_{kr} = k_\sigma \cdot \sigma_0$;
- during shear $\tau_{kr} = k_\sigma \cdot \tau_0$;

where $\sigma_0 = \dfrac{0.9 \cdot E \cdot \delta^2}{b^2}$; E – elasticity modulus, MPa; δ – skin thickness, mm; b – skin width, mm.

k_σ and k_τ are coefficients that depend on the conditions of fixing the plate and the ratio of its sides a/b. In the case of hinged support of a compressed plate on four faces with $a/b \geq 1$, the coefficient k_σ = 4, and in shear $k_\tau = 5.35 + 4 \cdot \left(\dfrac{b}{a}\right)^2$. After buckling under compression, the skin near the stringers works quite effectively.

3.4 Upper wing skins comparison

The following simulation conditions are established for the calculation of the action of aerodynamic forces on the aircraft skin using the above-mentioned formulas:

1. Size: chord 4300 mm × span 1100 mm;
2. Since the dimensions are small, bending and torsion were not taken into account;
3. Pressure force is 6200 Pa, as skin is from upper surface, direction of pressure is perpendicular to curvature of skin;
4. Compression force is 2000 N due to chord length.

Results of simulation are shown in the fig. 3.5-3.12.

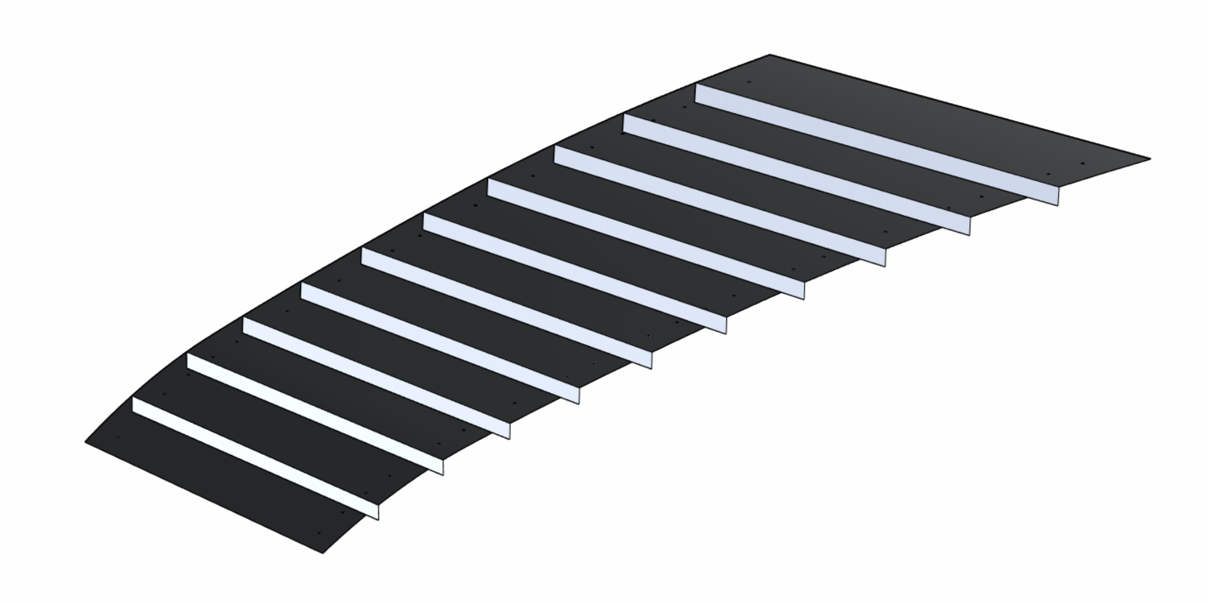

Fig. 3.5. Upper wing skin part in SolidWorks 2021.

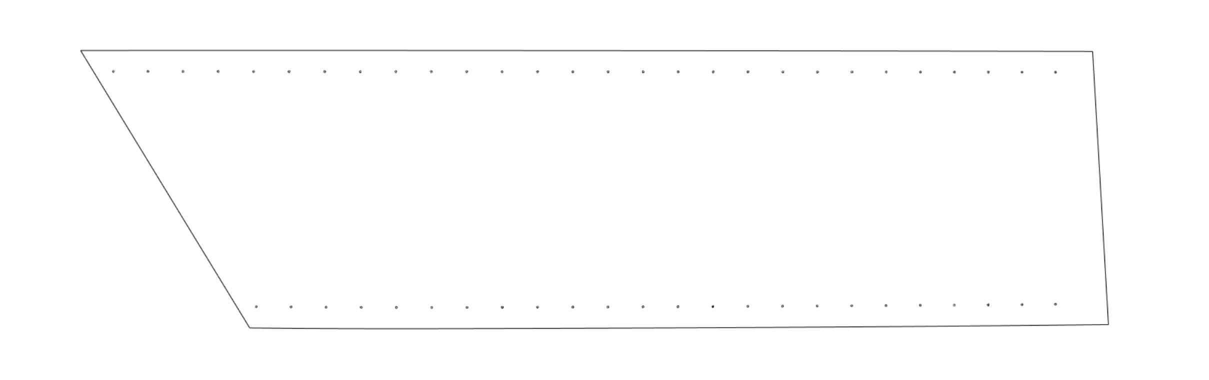

Fig. 3.6. Top view of the skin.

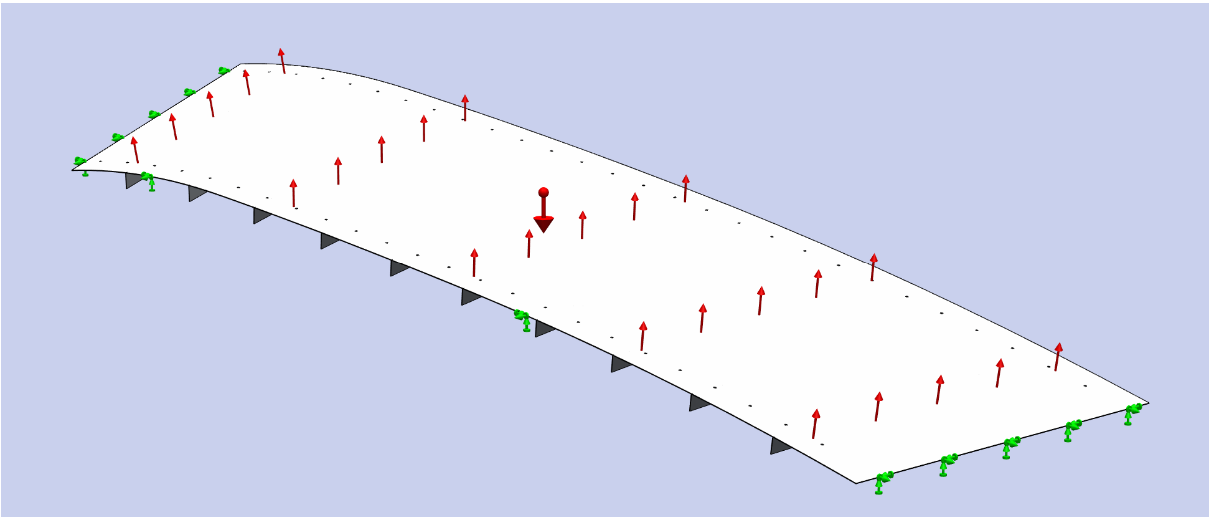

Fig. 3.7. Loading scheme of the skin.

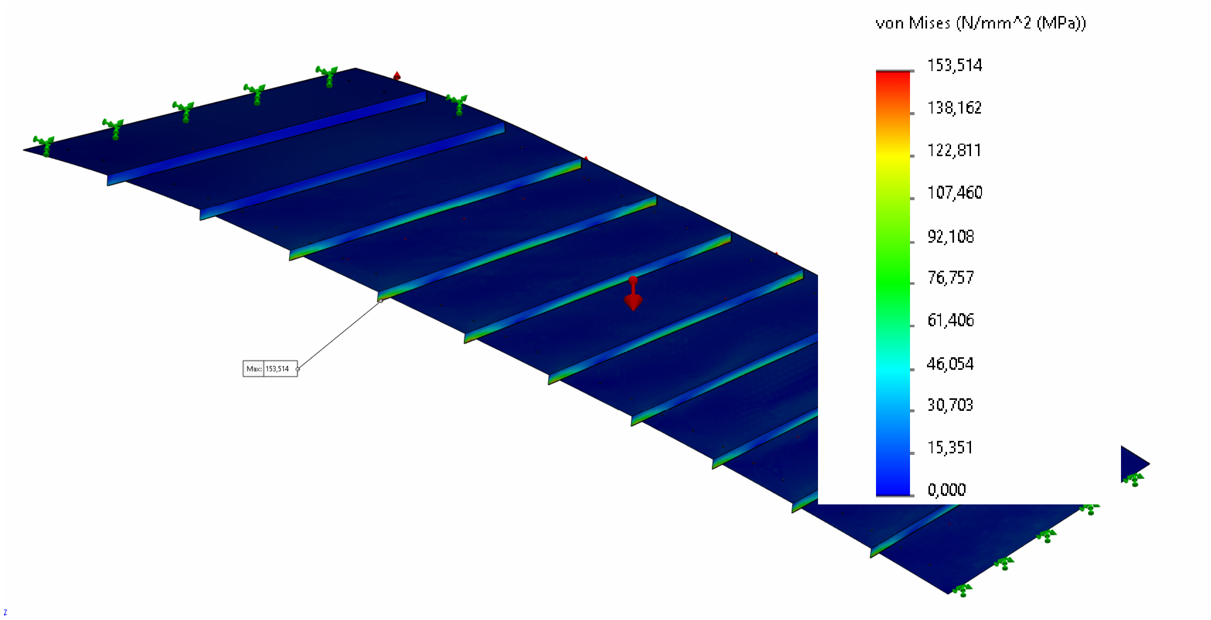

Fig. 3.8. Stresses in 2024-T4 aluminum skin.

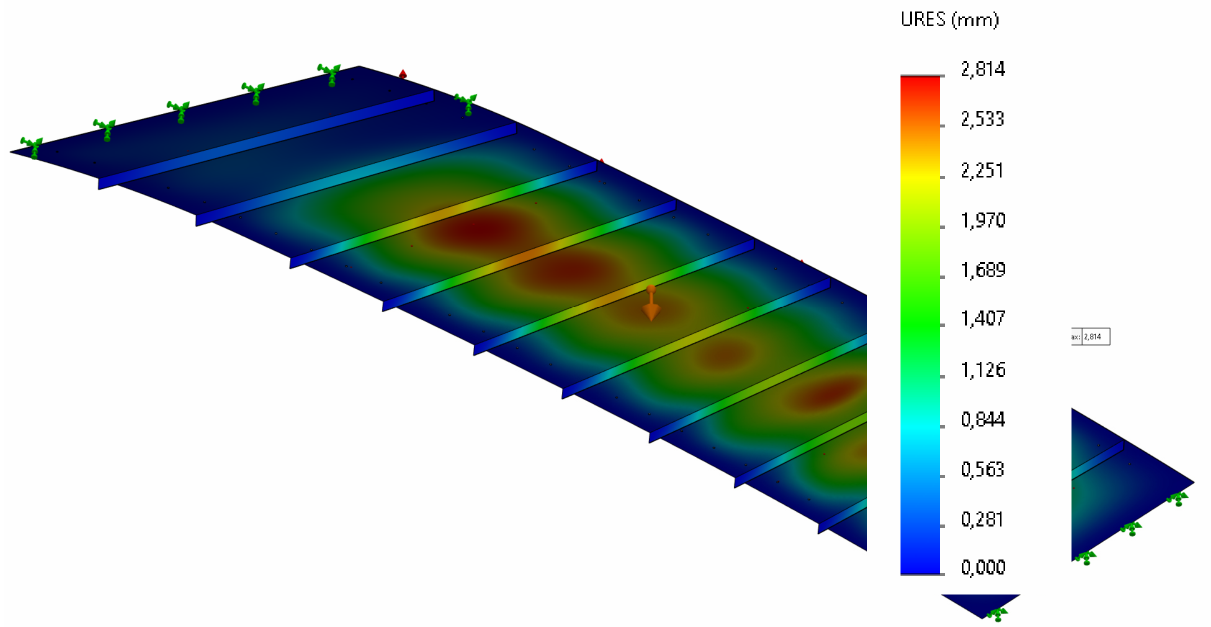

Fig. 3.9. Displacement in 2024-T4 aluminum skin.

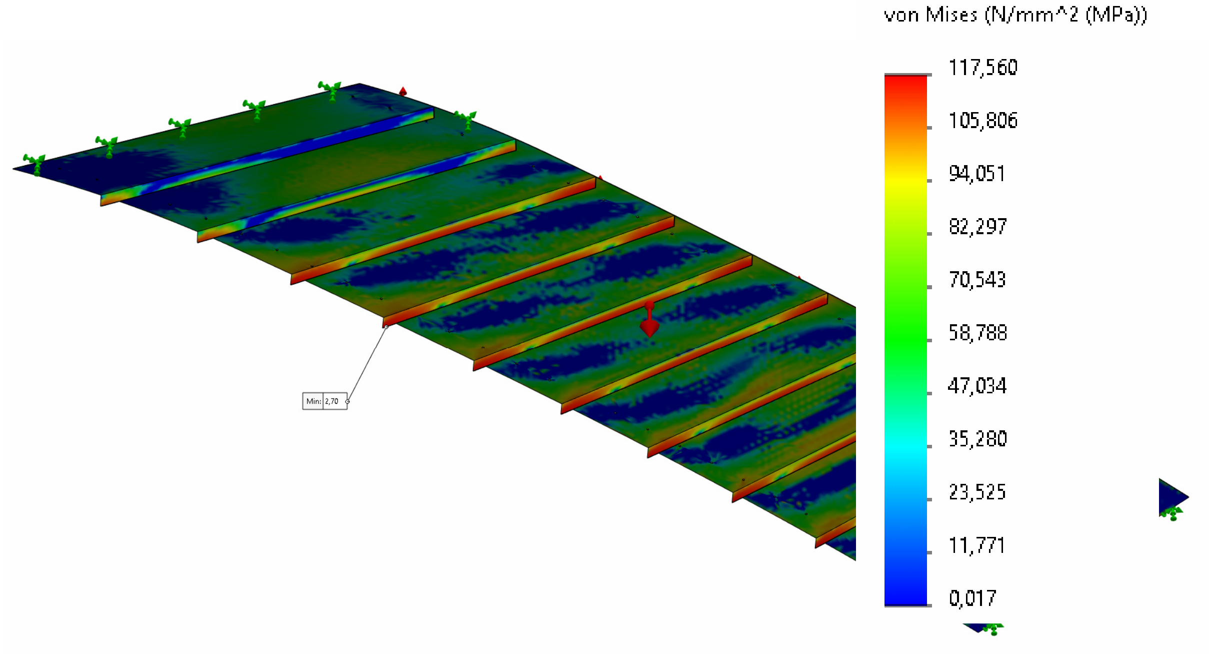

Fig. 3.10. Safety factor in 2024-T4 aluminum skin.

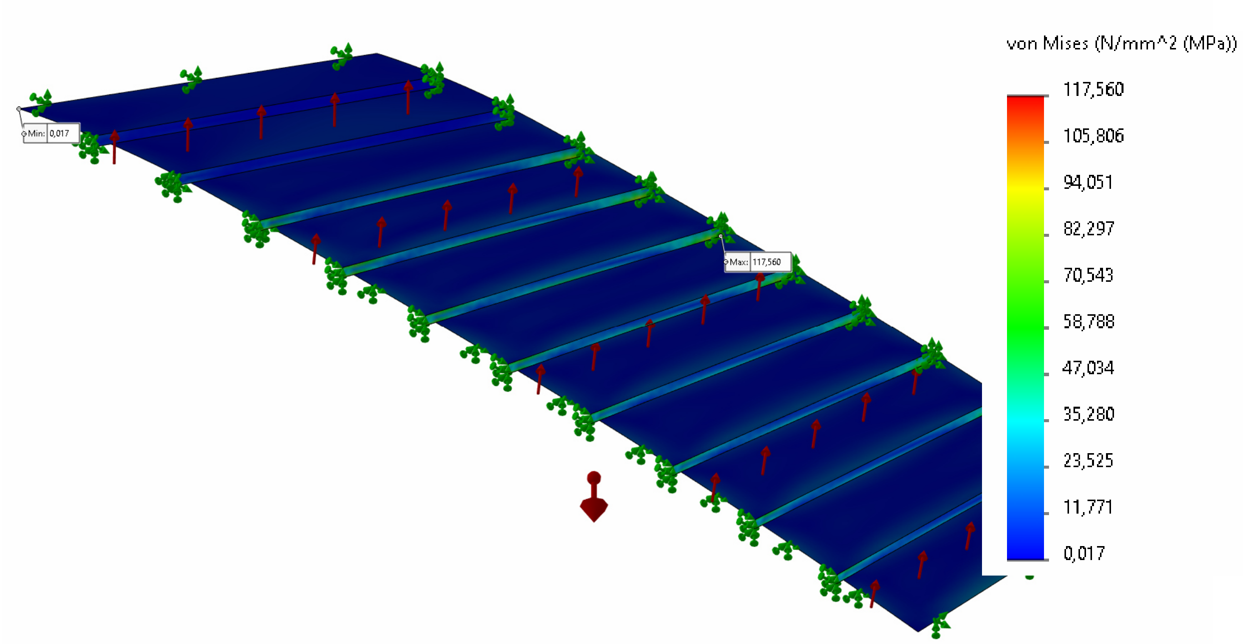

Fig. 3.11. Stresses in CFRP skin.

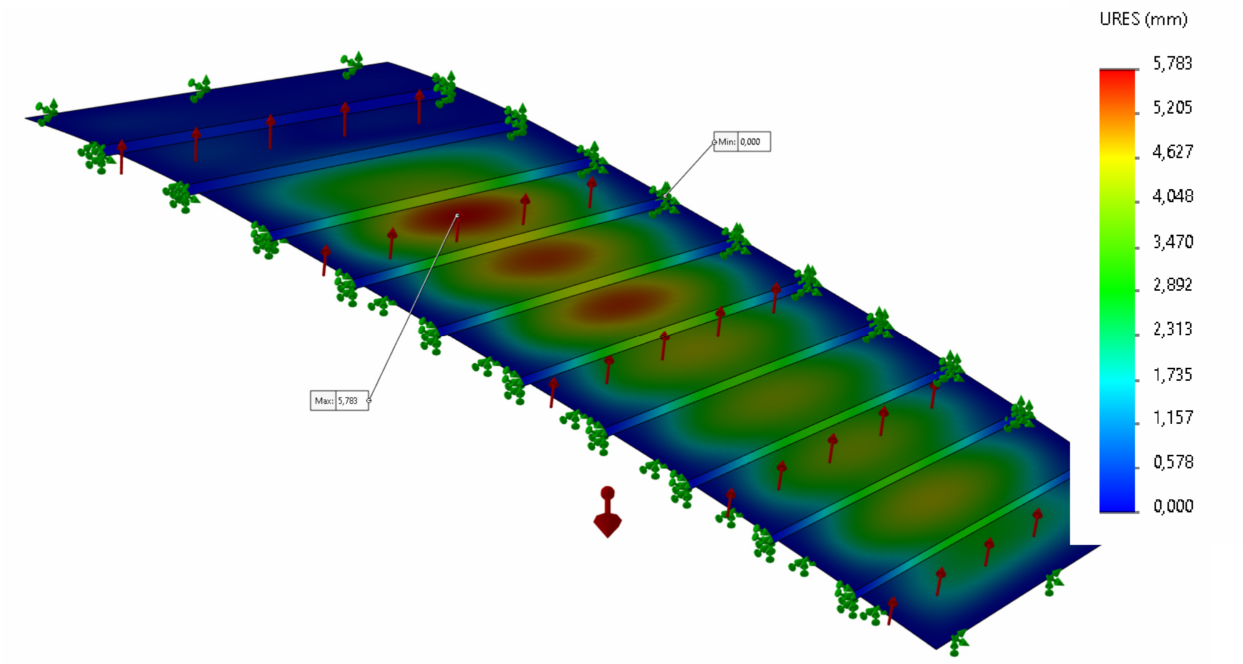

Fig. 3.12. Displacement in CFRP skin.

Table 3.3

Skins comparison

Material	**2024-T6**	**CFRP**
Maximum stress, MPa	151.51	117.56
Maximum displacement, mm	2.8	5.7
Safety factor	2.7	6.8
Mass, kg	30.48	20.68

Conclusions to the special part

In conclusion, use of composites presents significant advantages in various aspects of structural design. By incorporating composites, specifically CFRP (Carbon Fiber Reinforced Polymer), the mass of the structure can be reduced by 32%. This reduction in weight not only enhances the overall performance of the structure but also contributes to fuel efficiency, reducing operational costs, and environmental impact.

Furthermore, the introduction of composites results in a notable decrease in the maximum stress experienced by the skin of the structure. This reduction in stress levels significantly extends the service life of the skin, leading to a substantial decrease in maintenance costs associated with ensuring airworthiness. The improved durability and longevity offered by composites ultimately contribute to enhanced operational efficiency and reduced expenses over the lifespan of the structure.

GENERAL CONCLUSIONS

1. In this project, a preliminary design of the long-range passenger aircraft with a capacity of up to 220 passengers was completed, in accordance with the task. The main geometric dimensions and centering of the designed aircraft are determined, taking into account the main geometric parameters, operational purpose, the planned number of passengers, the speed and altitude of the flight, the landing and take-off conditions. All values obtained meet the requirements.

2. Centering of the designed aircraft was carried out, analysis and justification of the layout of the designed aircraft was carried out in accordance with the issued technical task, drawings were made based on the Boeing 737 MAX 9 prototype. The geometrical parameters almost coincide with the selected prototype.

3. A comparative analysis of the physical, technological and mechanical properties of the same aluminum alloy skin and composite material on the upper part of the aircraft wing skin was carried out.

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Appendix

Appendix A

Performed by: Stovbun Pavlo
Supervisor: Krasnopolskyi Volodymyr

PRELIMINARY DESIGN OF THE AIRCRAFT
INITIAL DATA AND SELECTED PARAMETERS

Passenger Number	220.
Flight Crew Number	2.
Flight Attendant or Load Master Number	6.
Mass of Operational Items	2162.06 kg
Payload Mass	20050.80 kg
Cruising Speed	850. km/h
Cruising Mach Number	0.7966
Design Altitude	11.00 km
Flight Range with Maximum Payload	7200. km
Runway Length for the Base Aerodrome	2.95 km
Engine Number	2.
Thrust-to-weight Ratio in N/kg	3.1000
Pressure Ratio	30.00
Assumed Bypass Ratio	6.00
Optimal Bypass Ratio	6.00
Fuel-to-weight Ratio	0.1700
Aspect Ratio	9.40
Taper Ratio	4.11
Mean Thickness Ratio	0.120
Wing Sweepback at Quarter Chord	25.0 deg
High-lift Device Coefficient	1.16
Relative Area of Wing Extensions	0.050

Wing Airfoil Type – Supercritical
Winglets – No
Spoilers – Yes

Fuselage Diameter	3.95 m
Fineness Ratio	11.26
Horizontal Tail Sweep Angle	29.0 deg
Vertical Tail Sweep Angle	34.0 deg

CALCULATION RESULTS

Optimal Lift Coefficient in the Design Cruising Flight Point	0.45135
Induce Drag Coefficient	0.00911

ESTIMATION OF THE COEFFICIENT Dm = Mcritical – Mcruise

Cruising Mach Number	0.79660
Wave Drag Mach Number	0.80698
Calculated Parameter Dm	0.01038

Wing Loading in kPa (for Full Wing Area):

At Takeoff	5.557
At Middle of Cruising Flight	4.562
At the Beginning of Cruising Flight	5.352

Drag Coefficient of the Fuselage and Nacelles	0.00847
Drag Coefficient of the Wing and Tail Unit	0.00915
Drag Coefficient of the Airplane:	
At the Beginning of Cruising Flight	0.02927
At Middle of Cruising Flight	0.02769
Mean Lift Coefficient for the Ceiling Flight	0.45135
Mean Lift-to-drag Ratio	16.29968
Landing Lift Coefficient	1.623
Landing Lift Coefficient (at Stall Speed)	2.435
Takeoff Lift Coefficient (at Stall Speed)	1.984
Lift-off Lift Coefficient	1.448
Thrust-to-weight Ratio at the Beginning of Cruising Flight	0.553
Start Thrust-to-weight Ratio for Cruising Flight	2.342
Start Thrust-to-weight Ratio for Safe Takeoff	2.847
Design Thrust-to-weight Ratio	2.989
Ratio Dr = Rcruise / Rtake-off	0.823

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

Takeoff	34.8865
Cruising Flight	57.8883
Mean cruising for Given Range	63.2099

FUEL WEIGHT FRACTIONS:

Fuel Reserve	0.03490
Block Fuel	0.30804

WEIGHT FRACTIONS FOR PRINCIPAL ITEMS:

Wing	0.09100
Horizontal Tail	0.00941
Vertical Tail	0.00934
Landing Gear	0.03995
Power Plant	0.09195
Fuselage	0.09802
Equipment and Flight Control	0.11679
Additional Equipment	0.00986
Operational Items	0.01857
Fuel	0.34294
Payload	0.17223

Airplane Takeoff Weight	116442. kgf
Takeoff Thrust Required of the Engine	173.99 kN

Air Conditioning and Anti-icing Equipment Weight Fraction	0.0201
Passenger Equipment Weight Fraction (or Cargo Cabin Equipment)	0.0129
Interior Panels and Thermal/Acoustic Blanketing Weight Fraction	0.0058
Furnishing Equipment Weight Fraction	0.0117
Flight Control Weight Fraction	0.0051
Hydraulic System Weight Fraction	0.0146

Electrical Equipment Weight Fraction	0.0307
Radar Weight Fraction	0.0029
Navigation Equipment Weight Fraction	0.0043
Radio Communication Equipment Weight Fraction	0.0022
Instrument Equipment Weight Fraction	0.0050
Fuel System Weight Fraction	0.0103

Additional Equipment:

Equipment for Container Loading	0.0064
No typical Equipment Weight Fraction (Build-in Test Equipment for Fault Diagnosis, Additional Equipment of Passenger Cabin)	0.0035

TAKE-OFF DISTANCE PARAMETERS

Airplane Lift-off Speed	281.95	km/h
Acceleration during Takeoff Run	2.33	m/s*s
Airplane Take-off Run Distance	1311.	m
Airborne Take-off Distance	578.	m
Take-off Distance	1890.	m

CONTINUED TAKE-OFF DISTANCE PARAMETERS

Decision Speed	267.85	km/h
Mean Acceleration for Continued Take-off on Wet Runway	0.30	m/s*s
Take-off Run Distance for Continued Take-off on Wet Runway	2149.39	m
Continued Take-off Distance	2727.77	m
Runway Length Required for Rejected Take-off	2826.12	m

LANDING DISTANCE PARAMETERS

Airplane Maximum Landing Weight	86041.	kg
Time for Descent from Flight Level till Aerodrome Traffic Circuit Flight	21.8	min
Descent Distance	51.38	km
Approach Speed	246.19	km/h
Mean Vertical Speed	1.99	m/s
Airborne Landing Distance	515.	m
Landing Speed	231.19	km/h
Landing run distance	714.	m
Landing Distance	1230.	m
Runway Length Required for Regular Aerodrome	2053.	m
Runway Length Required for Alternate Aerodrome	1746.	m

ECONOMICAL EFFICIENCY

The equipped aircraft mass to payload mass ratio	2.7582	
The mass of empty equipped aircraft per 1 passenger	297.33	kg/p
Relative performance with full load	437.89	km/h
Aircraft performance with maximum payload	16460.2	kg*km/h
Average time fuel consumption	4088.977	kg/h
Average distance fuel consumption	4.98	kg/km
Average fuel consumption for ton-kilometer	248.416	g/t*km
Average fuel consumption for passenger-kilometer	23.6287	g/p*km
Approximate evaluation of relative expenses for ton-km	0.3637	$/t*km