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

ДОПУСТИТИ ДО ЗАХИСТУ
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«БАКАЛАВР»

Тема: «Оптимізація силових елементів пасажирської кабіни середньомагістрального вузькофюзеляжного літака»

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Київ 2024

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

PERMISSION TO DEFEND
Head of the department,
Associate Professor
__________ Sviatoslav YUTSKEVYCH
"___"   _____________ 2024

BACHELOR DEGREE THESIS

Topic: "Optimization of passenger cabin structural elements of middle range narrow-body airplane"

Fulfilled by:	____________	**Alina LANOVENKO**
Supervisor: **PhD, associate professor**	____________	**Volodymyr KRASNOPOLSKYI**
Standards inspector **PhD, associate professor**	____________	**Volodymyr KRASNOPOLSKYI**

Kyiv 2024

НАЦІОНАЛЬНИЙ АВІАЦІЙНИЙ УНІВЕРСИТЕТ

Аерокосмічний факультет
Кафедра конструкції літальних апаратів
Освітній ступінь «Бакалавр»
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Освітньо-професійна програма «Обладнання повітряних суден»

ЗАТВЕРДЖУЮ
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«___» ____________ 2024 р.

ЗАВДАННЯ

на виконання кваліфікаційної роботи здобувача вищої освіти

ЛАНОВЕНКО АЛІНИ АНДРІЇВНИ

1. Тема роботи: «Оптимізація силових елементів пасажирської кабіни середньомагістрального вузькофюзеляжного літака», затверджена наказом ректора від 15 травня 2024 року № 794/ст.

2. Термін виконання роботи: з 20 травня 2024 р. по 16 червня 2024 р.

3. Вихідні дані до роботи: маса комерційного навантаження 20229.3 кг, дальність польоту з максимальним комерційним навантаженням 6400 км, крейсерська швидкість польоту 825 км/год, висота польоту 11,5 км, кількість пасажирів 190.

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 оптимізацію силового елементу пасажирської кабіни.

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.	20.05.2024 – 21.05.2024	
2	Вибір та розрахунок параметрів проєктованого літака.	22.05.2024 – 23.05.2024	
3	Виконання компонування літака та розрахунок його центрування.	24.05.2024 – 25.05.2024	
4	Розробка креслень по основній частині дипломної роботи.	26.05.2024 – 27.05.2024	
5	Огляд літератури за проблематикою роботи. Оптимізація силового елементу пасажирської кабіни.	28.05.2024 – 29.05.2024	
6	Проведення розрахунку на міцність та аналізу для початкового та оптимізованого елементу.	30.05.2024 – 31.05.2024	
7	Оформлення пояснювальної записки та графічної частини роботи.	01.06.2024 – 02.06.2024	
8	Подача роботи для перевірки на плагіат.	03.06.2024 – 06.06.2024	
9	Попередній захист кваліфікаційної роботи.	07.06.2024	
10	Виправлення зауважень. Підготовка супровідних документів та презентації доповіді.	08.06.2024 – 10.06.2024	
11	Захист дипломної роботи.	11.06.2024 – 16.06.2024	

7. Дата видачі завдання: 20 травня 2024 року

Керівник кваліфікаційної роботи ____________ Володимир КРАСНОПОЛЬСЬКИЙ

Завдання прийняв до виконання ____________ Аліна ЛАНОВЕНКО

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,
Associate Professor
_________ Sviatoslav YUTSKEVYCH
"___" _____________ 2024

TASK

for the bachelor degree thesis

Alina LANOVENKO

1. Topic: "Optimization of passenger cabin structural elements of middle range narrow-body airplane", approved by the Rector's order № 794/ст from 15 May 2024.
2. Period of work: since 20 May 2024 till 16 June 2024.
3. Initial data: payload 20.3 tons, flight range with maximum capacity 6400 km, cruise speed 825 km/h, flight altitude 11.5 km, passenger number 190.
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: optimization of passenger cabin structural elements.
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.	20.05.2024 – 21.05.2024	
2	Selection and calculation of the aircraft designed parameters.	22.05.2024 – 23.05.2024	
3	Performing of aircraft layout and centering calculation.	24.05.2024 – 25.05.2024	
4	Development of drawings on the thesis main part.	26.05.2024 – 27.05.2024	
5	Optimization of passenger cabin structural elements.	28.05.2024 – 29.05.2024	
6	Performing strength calculations and analysis for the initial and optimized element.	30.05.2024 – 31.05.2024	
7	Explanatory note checking, editing, preparation of the diploma work graphic part.	01.06.2024 – 02.06.2024	
8	Submission of the work to plagiarism check.	03.06.2024 – 06.06.2024	
9	Preliminary defense of the thesis.	07.06.2024	
10	Making corrections, preparation of documentation and presentation.	08.06.2024 – 10.06.2024	
11	Defense of the diploma work.	11.06.2024 – 16.06.2024	

7. Date of the task issue: 20 May 2024

Supervisor: ___________ Volodymyr KRASNOPOLSKYI

Student: ___________ Alina LANOVENKO

РЕФЕРАТ

Пояснювальна записка кваліфікаційної роботи бакалавра «Оптимізація силових елементів пасажирської кабіни середньомагістрального вузькофюзеляжного літака»:

78 с., 20 рис., 23 табл., 4 джерел

Дана кваліфікаційна робота присвячена розробці аванпроекту пасажирського літака для середньо магістральних авіаліній з можливістю перевезення вантажів, що відповідає міжнародним стандартам польотів, нормам безпеки, економічності та надійності, а також оптимізація силового елемента пасажирської кабіни.

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 зменшені маси конструкціїі, шляхом проектування нового перерізу балки підлоги, який забезпечить кращу ремонтно здатність та тріщиностійкість.

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

Дипломна робота, аванпроєкт літака, компонування,центрування, перепроектування балки підлоги, розрахунок на міцність

ABSTRACT

Bachelor degree thesis "Optimization of passenger cabin structural elements of middle range narrow-body airplane"

78 pages, 20 figures, 23 tables, 4 references

This thesis is dedicated to preliminary design of mid-range airplane for transportation of passengers and estimation its flight performances as well as optimization of passenger cabin structural elements.

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 reduced weight of the structure by redesigning a floor beam cross-section that will provide better repair capability and crack resistance. 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, floor beam redesigning, strength calculation.

Format	№	Designation	Name	Quantity	Notes
			General documents		
A4	1	NAU 24 02L 00 00 00 07 TW	Task for work	1	
	2	NAU 24 02L 00 00 00 07	Mid-range passenger aircraft	2	
A1		Sheet 1	General view	1	
A1		Sheet 2	Fuselage layout	1	
A4	3	NAU 24 02L 00 00 00 07 EN	Middle-range passenger aircraft	78	
			Explanatory note		

Sh.	№ doc.	Sign	Date	NAU 24 02L 00 00 00 07 EN			
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Supervisor	Krasnopolskyi V.S.				Q	9	78
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St.control.	Krasnopolskyi V.S.						
Head of dep.	Yutskevych S.S.						

CONTENT

INTRODUCTION

In the ever-evolving field of aviation, the pursuit of innovation and efficiency in the design of passenger aircraft is a profound indicator of human development. Thus, the choice of the topic "Optimization of passenger cabin structural elements of middle range narrow-body airplane" was driven not only by a personal interest in aircraft design, but also by a desire to make a small contribution to aviation.

The data for this project was mainly collected from existing aircraft, which serve as prototypes and benchmarks for this project. It is noteworthy that the main source of inspiration was the Boeing 737-800 series aircraft. Known for its versatility and reliability, the Boeing 737-800 is an iconic aircraft in the aviation industry. Accordingly, by studying and comparing new design to such an aircraft appears the opportunity to gain valuable insights into potential improvements and innovations.

Once realized, the new passenger aircraft can be used in a variety of areas, such as commercial airlines. This aircraft has the potential to become the preferred choice for commercial airlines, allowing them to efficiently serve medium-haul routes, reduce fuel costs and improve passenger comfort.

1. PRELIMINARY DESIGN OF MID-RANGE AIRCRAFT

1.1. Analysis of prototypes and short description of designed aircraft

The systematic study of the prototype's characteristics allows us to select the design parameters and layout of the aircraft rationally. This analysis process corrects and eliminates the shortcomings identified in previous versions based on the combined experience of previous aircraft designers. An essential step in shaping the final version of the aircraft is to determine the design parameters, including elements such as weight, geometric characteristics of individual components, engine specifications, and related weight and performance aspects. If aircraft in a series have similar parameters, utilizing data from the more prominent family of aircraft is a viable strategy. Using this rich data facilitates the detailed study of design requirements and further modifications of the aircraft.

As a case in point, the Boeing B737-800 is a successful prototype for the aircraft considered in this study. The long-range aircraft can accommodate up to 189 passengers and embodies Boeing's commitment to innovation and efficiency.

Table 1.1 shows the performance indicators of the B737-800 prototype. This provides the basis for a comprehensive analysis and further design process improvement.

Table 1.1

Performances of prototypes

Parameter	A 320-200	B737-800	Designed aircraft
1	2	3	4
Max. payload (kg)	16600	20540	20229.3
Crew/number of pilots	5/2	5/2	5/2
Passengers	150	189	190
Wing loading (kN/m^2)	4.28	6.25	5.173
Flight range with max. payload (km)	5000	5400	6400
Cruise speed (km/h)	829	828	825

Ending of the table 1.1

1	2	3	4
Cruise altitudes (km)	11.27	12.5	11.5
Thrust/weight ratio (N/kg)	2.91	2.79	3.5
Approach speed (km/h)	134	130	247.3
Landing speed (km/h)	240	270	232.3
Take-off speed (km/h)	240	270	271
Take-off run distance (m)	2590	2400	1554
Landing run distance (m)	1440	1630	748
Take-off distance (m)	2058	2652	2132
Landing distance (m)	1243	1636	1264
Maximum take-off mass (kg)	78000	79010	94203
Landing mass (kg)	64500	66361	75067
Empty weight (kg)	42400	41413	48213
Fuel fraction (%)	38	32.9	27.34
Payload fraction (%)	21.3	25.9	21.47
Wing span (m)	34.1	34.3	43.3
Sweepback angle at ¼ chord (°)	28	25.02	29
Wing aspect ratio	9.37	9.45	10.5
Wing taper ratio	4.11	4.5	3.4
Fuselage length (m)	37.57	32.18	39.05
Fuselage diameter (m)	3.95	3.76	3.55
Fuselage fineness ratio	9.51	10.21	11
Passenger cabin width (m)	3.63	3.54	3.55
Passenger cabin length (m)	27.5	29.95	25.49
Cabin height (m)	2.0	2.02	2.08
Aisle width (m)	0.69	0.51	0.5
Horizontal tail span (m)	13.4	14.35	17.32
Horizontal tail sweepback angle (°)	33	30	32
Horizontal tail aspect ratio	4.41	6.16	4
Horizontal tail taper ratio	2.15	4.92	2.6
Vertical tail height (m)	5.46	7.16	7.79
Vertical tail sweepback angle (°)	38	35	35
Vertical tail aspect ratio	1.8	1.91	1.15
Vertical tail taper ratio	4	3.69	1.2
Landing gear wheel base (m)	12.315	15.6	13.67
Landing gear wheel track (m)	9.85	5.76	10.9

1.2. Brief description of the main parts of the aircraft

1.2.1. Wing

A wing is an airplane's load-bearing surface designed to generate the aerodynamic lift required for flight. In addition, the wing provides lateral stability and controllability of the aircraft and can be used to mount landing gear, engines, etc.

As in other passenger aircraft, the wing passes through the lower half of the fuselage (monoplane). The advantages of this solution are easy maintenance, less weight (landing gear struts can be made shorter and, as a result, more robust and lighter), and more passenger safety in case of emergency compared to other types of wing arrangement.

The plane's wing shape significantly impacts the aircraft's aerodynamic, weight, structural, and technological characteristics. This airplane has a straight-swept wing, which has become widespread due to various modifications and design solutions. Even though this configuration increases weight and decreases wing stiffness, it is used because of the slow growth of lift depending on the angle of attack and, therefore, better resistance to atmospheric turbulence. In addition, the weight increase is compensated for by the torsion box of wing because the elements that work in compression and tension caused bending are sufficiently far from the section's neutral axis, the material is used rationally, and the wing's weight is not too large. In addition, the internal volume of the wing structure under consideration is free and can be well used for fuel. Supercritical type of airfoil is usually recommended for modern passenger and transport aircraft. It is characterized by a large nose radius, an almost flat upper surface, a convex lower surface, and a thin, curved tail.

The wing's structural elements are the skin – a structural element that forms the bearing surface of the airfoil, participates in the perception of bending and torsional moments, and is the wall of the sealed fuel compartments; stringers – longitudinal force elements that participate in the perception of bending moments and also reinforce the skin against buckling; spars – longitudinal force elements that participate in the perception of bending and torsional moments, shear forces, and also serve as the walls of sealed fuel compartments. They consist of flange that absorb most of the normal bending stresses and a web that absorbs most of the tangential stresses from wing torsion and shear forces; and ribs – transverse structure members designed to maintain the shape of the wing airfoils, transfer loads from the skin and stringers to the webs of the spars, reinforce the skin and stringers against buckling, and serve as the walls of the sealed fuel compartments. Reinforced ribs at the powerplant hinge transmit forces from the powerplant to the spars.

External shapes, area, materials, structural schemes, weight, and other parameters of the wing are determined based on relevant calculations during the aircraft design.

1.2.2. Fuselage

The part of the airplane that carries the payload is known as the fuselage, more commonly referred to as the body. Passengers must be transported safely and comfortably, which means the interior of the fuselage has to be environmentally controlled for temperature and pressure. In addition to accommodating payloads, the fuselage houses the landing gears, a multiplicity of electrical/control systems, and environmental control equipment.

The circular cross-section is not considered the best regarding crew, passenger, and cargo accommodation. However, it is the most rational for fuselages with pressurized compartments subjected to significant loads when flying at higher altitudes. Under the influence of internal pressure, the fuselage skin of a circular cross-section works in tension.

A typical airplane fuselage is a semi-monocoque structure with skin reinforced by longitudinal stringers and circumferential frames.

Frames are transverse structure elements designed to maintain the shape of the fuselage airfoils and participate in the perception of skin stretching from cabin pressurization, transfer loads from floor beams and other components to the skin by a flow of tangential stresses, and support the skin and stringers against buckling.

Reinforced frames have attachment points to the wing, fuselage, and front landing gear. Each frame of the pressurized compartment has a transverse horizontal beam that supports the cabin floor.

Stringers are longitudinal structure elements that take part in the perception of bending moments and reinforce the skin against buckling. The stringers are pressed profiles with an L-angle cross-section and a partially tapered section in the lower part. The skin is a structure element that forms a hermetic shell and aerodynamic flow surface and is involved in the perception of bending and torsional moments and cabin pressurization. The skin is attached to the stringers and frames with rivets, and in the sealed fuselage compartment, the sealing tapes are laid between the elements to be riveted. In addition, a layer of sealant is applied to the riveted seams from the inside of the fuselage.

In addition to the main structural elements listed above, the fuselage contains various auxiliary structural elements, such as cabin floors, brackets, doors, airtight and fireproof partitions, all possible hatch covers, etc.

1.2.3. Tail unit

Aircraft fins are load-bearing surfaces designed to provide stability, controllability, and balance of the aircraft in flight.

The empennage consists of the vertical fin or stabilizer, rudder and dorsal fin, and the horizontal stabilizers, including the center section inside the fuselage and elevators.

The vertical fin is a symmetrical airfoil rigidly mounted on the vertical centerline of the fuselage, behind of the pressure bulkhead, and the dorsal fin fairs the leading edge of the fin into the fuselage and extends forward of the pressure bulkhead. The fin and rudder provide lateral stability and control to the airplane about the vertical yaw axis.

The horizontal stabilizer is an inverted airfoil section that passes through the unpressurized fuselage behind the pressure bulkhead. The outboard sections are joined at the side of the fuselage to a separate center section. The horizontal stabilizer and elevators provide balancing loads and control about the lateral pitch axis. Unlike the fin, the stabilizer is pivoted behind of the rear spar to enable the stabilizer angle of attack to be changed using a jack screw that is attached to the center section front spar to trim and balance the airplane.

1.2.4. Landing gear

The purpose of the landing gear is to absorb landing impact energy, provide ground handling capability, assist in stopping the aircraft after touchdown, provide adequate rotation clearance during take-off and landing, and provide stable support for the airplane when it is on the ground.

The landing gear is built around a single-stage oleo-pneumatic shock strut that absorbs most vertical landing impact loads. The damping characteristic is controlled by a variable-diameter metering pin that travels through a fixed-diameter orifice.

The landing gear retraction axis is called the trunnion axis. The landing gear is attached to the airplane at two support points on the trunnion axis. The trunnion spans load from the outer cylinder to the trunnion supports.

A non-folding brace is usually attached to the bottom of the outer cylinder and the trunnion. This brace reduces the bending loads in the outer cylinder by providing a more direct load path to the trunnion.

To provide stability for the landing gear in the direction perpendicular to the plane of the landing gear, folding struts are used to attach the outer cylinder to support points on the airplane.

The rolling stock (wheels, tires, and brakes) is mounted on axles. On multi-axle landing gear, the axles are attached to a truck beam. The truck beam is then mounted to the inner cylinder with a single pin joint.

Two torque links transfer torque about the shock strut axis from the inner cylinder to the outer cylinder. The lower link is attached to either the inner cylinder or truck beam assembly, and the upper link is attached to the outer cylinder. The two links are then attached at an apex point. This folding arrangement allows the shock strut to stroke freely without hindrance from the torque links.

1.2.5. Power plant

An aircraft power plant is a combination of an engine with its assembly, systems, and devices that ensure the reliable functioning of the engine during its operation. It comprises many components, such as cylinders, pistons, and fans, which help produce the energy needed to propel an aircraft.

The specifics of the powerplants are determined by the engine type, which is selected based on the technical requirements of the aircraft being designed. This aircraft is powered by a turbofan engines CFMI CFM56-7B24.

Turbofan engines are designed to generate additional thrust by redirecting secondary airflow around the combustion chamber. Bypassing air from a jet engine produces increased thrust, cools the engine and helps reduce exhaust noise. This provides jet engine-like cruising speeds and reduced fuel consumption.

The engines located in the wing. This arrangement reduces the wing structure's weight, as the engines' weight loads act in the opposite direction to the aerodynamic loads. Accordingly, the bending moment acting in the root section of the wing is reduced.

Table 1.2

Engine performances

Engine Model	Overall Length	Overall Width	Overall Height	Dry Spec. Weight	Take-off thrust	Bypass ratio	Overall pressure ratio
	mm	mm	mm	kg	kg		
CFM56-7B24	2508	2118	1829	2370	110000	5.3	32.7

1.2.6. Control Surfaces

The primary control surfaces of an airplane are the ailerons, rudder, and elevator. Flight control surfaces, such as the aileron, are responsible for providing roll control of the aircraft and controlling its movement around the longitudinal axis. Typically, they are placed on the wing's trailing edge near the wing tip.

In order to control the ailerons manually the pilot must use balancing mechanisms that allow to compensate any air loads acting on control surfaces during flight. The ailerons in this aircraft exhibit symmetrical growth, meaning they move up the same amount they move down.

The elevator is the control surface that governs the movement (pitch) of the aircraft about the lateral axis. They are attached to hinges of the rear crossmember of a horizontal stabilizer are connected to them. There is a similar structure to other control surfaces, and the design of the elevator may not be aerodynamically or statically balanced.

A flight control surface, known as the rudder, directs the aircraft' movement around its vertical axis. The construction of the rudder is similar to other flight control surfaces, including its spars, ribs and skin. The rudder is typically proportionally balanced between the aerodynamic and statical positions to make it more user-friendly and prevent vibration.

The secondary control surfaces include tabs, flaps, spoilers, and slats. Secondary flight control surfaces, known as tabs, are typically small and located on the primary surface's trailing edges. By "loading" the control surfaces, the pilot can maintain a steady attitude and reduce the amount of work required for this to occur. Additionally, they can be utilized to assist the pilot in positioning the control surface at a center or reduced position. The airplane's balance tab is connected in a way that causes the main rudder to move against the tab. Moving the primary control surface can be facilitated by the balance tab. When it comes to moving control surfaces of a large aircraft with ease, these feet can be particularly useful for leveling.

Located near the back of the main spoiler, there is a small auxiliary flap called "the trailing edge flap" that can be hinged along if deflected at axis. The aerodynamic properties of the wing are altered by the deflection and a change in its shape. Flaps are designed as a high-lift device only deflect downward.

To decrease a wing's lift, special parts known as spoilers are attached to the control surface. The spoilers are located on the upper surface of the wings. The addition of flight spoilers to wings is intended to decrease their lift and maintain control over altitude while limiting speed.

Slats are extendable, high-lift devices on the leading edge of the wings of some fixed-wing aircraft. They aim to increase lift during low-speed operations such as take-off, initial climb, approach, and landing. They accomplish this by increasing both the surface area and the wing's camber by deploying outwards and drooping downwards from the leading edge. Slats normally have several possible positions and extend progressively in concert with flap extension [1].

Conclusion to the analytic part

An integral part of aircraft design is the selection and analysis of prototypes. This not only facilitates the further design of the aircraft, but also helps to make it more efficient. Therefore, in this section, based on the selected B737-800 and A320 prototypes, the following characteristics were selected: wing shape and type, cross-sectional shape and structural scheme of the fuselage, type of tail unit, landing gear and power plants.

The resulting characteristics satisfied all the requirements and were better in some of the parameters. For example, the designed airplane has more flight range with maximum payload and less take-off and landing distance.

2. AIRCRAFT GEOMETRY CALCULATION

2.1. Wing geometry calculation

The initial data for this airplane were obtained by calculations in a special computer program developed by the Department of Aircraft Design of the National Aviation University. The data are presented in Appendix A. (Aircraft initial data).

Full wing area is:

$$S_{wing} = \frac{m_0 \cdot g}{P_0} = \frac{94203 \cdot 9.8}{5173} = 178.5\ (\text{m}^2),$$

where m_0 – take-off mass of the aircraft, kg; g – gravitational acceleration, m/s^2; P_0 – wing loading at cruise regime of flight, N/m^2.

Wing span is:

$$l_{wing} = \sqrt{S_{wing} \cdot \lambda_w} = \sqrt{178.5 \cdot 10.5} = 43.3\ (\text{m}),$$

where λ_w – wing aspect ratio.

Root chord is:

$$C_{root} = \frac{2 \cdot S_w \cdot \eta_w}{(1+\eta_w) \cdot l_w} = \frac{2 \cdot 178.5 \cdot 3.4}{(1+3.4) \cdot 43.3} = 6.37\ (\text{m}),$$

where η_w – wing taper ratio.

Tip chord is:

$$C_{tip} = \frac{C_{root}}{\eta_w} = \frac{6.37}{3.4} = 1.87\ (\text{m}).$$

On board chord for trapezoidal shaped wing is:

$$C_{board} = C_{root} \cdot (1 - \frac{(\eta_w - 1) \cdot D_f}{\eta_w \cdot l_w}) = 6.37 \cdot (1 - \frac{(3.4-1) \cdot 3.55}{3.4 \cdot 43.3}) = 6.00\ (\text{m}),$$

where D_f – fuselage diameter, m.

Wing construction and spars position. Before choosing a wing structure scheme, it is necessary to determine the type of its internal structure. In order to meet the strength requirements and at the same time make the structure relatively light, a torsion box wing with two spars was chosen. Relative coordination of the spar's position for a wing with two spars is equal:

$$x_{2spar} = 0.6 \cdot C_i = 0.6 \cdot 3.53 = 2.12 \text{ (m)},$$

from the leading edge of current chord in the wing cross-section.

Mean aerodynamic chord definition. The geometrical method of mean aerodynamic chord determination has been taken, due to accuracy and simplicity in performance (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).

The mean aerodynamic chord is equal 4.53 m.

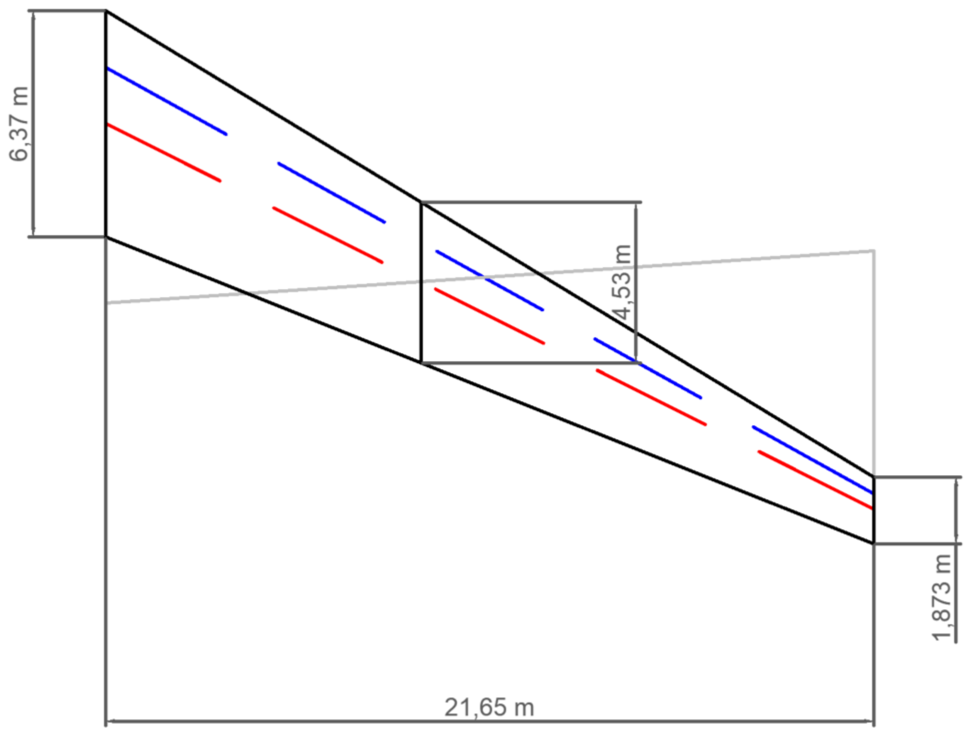


Fig. 2.1. Geometrical method of determination of mean aerodynamic chord.

Also, for self-testing we could calculate the MAC by the approximately formulas for trapezoidal wing shape:

$$C_{MAC} = \frac{2}{3} \cdot \frac{C_{root}{}^2 + C_{root} \cdot C_{tip} + C_{tip}{}^2}{C_{root} + C_{tip}} = \frac{2}{3} \cdot \frac{6.37^2 + 6.37 \cdot 1.87 + 1.87^2}{6.37 + 1.87} = 4.53\,(\text{m}).$$

Since the definition of the geometric characteristics of the wing has already been completed, we can come to the assessment of the geometry of the aileron and means of lifting force.

Ailerons design. The main purpose of the ailerons is to create rolling moment and provide adequate rate of roll. Ailerons geometrical parameters are determined by the next formulas:

Aileron's span*:*

$$l_{aileron} = \frac{0.35 \cdot l_{wing}}{2} = \frac{0.35 \cdot 43.3}{2} = 7.58\,(\text{m}).$$

Aileron's chord*:*

$$C_{aileron} = 0.24 \cdot C_i = 0.24 \cdot 3.53 = 0.85\,(\text{m}).$$

Aileron area*:*

$$S_{aileron} = \frac{0.06 \cdot S_{wing}}{2} = \frac{0.06 \cdot 178.5}{2} = 5.35\,(\text{m}^2).$$

Inner axial balance:

$$S_{in\ axail} = 0.31 \cdot S_{aileron} = 0.31 \cdot 5.35 = 1.66\,(\text{m}^2).$$

Area of aileron's trim tabs (for the aircraft with two engines) *:*

$$S_{trim\ tabs} = 0.06 \cdot S_{aileron} = 0.06 \cdot 5.35 = 0.32\,(\text{m}^2).$$

Range of aileron deflection: upward $\delta_{aileron} \geq 25^\circ$ downward $\delta_{aileron} \geq 15^\circ$. So, the results are:

Ailerons' span:

$$l_{aileron} = \frac{0.35 \cdot l_{wing}}{2} = \frac{0.35 \cdot 43.3}{2} = 7.58\,(\text{m}).$$

Aileron area:

$$S_{aileron} = \frac{0.06 \cdot S_{wing}}{2} = \frac{0.06 \cdot 178.5}{2} = 5.35\,(\text{m}^2).$$

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

$$S_{trim\,tabs} = 0.06 \cdot S_{aileron} = 0.06 \cdot 5.35 = 0.32\,(\text{m}^2).$$

High lift device of a wing. Double slotted Fowler flaps together with slats: Fowler flaps are a type of high-lift device designed to increase wing lift. When equipped with two slots, they create two separate airflow channels over the wing, allowing for increased airflow and reduced air pressure on the upper surface of the wing. This, in turn, generates more lift, allowing the aircraft to operate effectively at lower speeds.

When combined with slats, which are movable leading-edge devices positioned at the front of the wing, the aircraft can further maximize lift and control. Slats serve to delay the onset of stall, making it possible for the aircraft to maintain stable flight even at slow speeds, such as during take-off and landing.

The interaction between double-slotted Fowler flaps and slats results in improved lift, maneuverability, and the ability to operate safely in challenging conditions, like short runways or heavy payloads.

The relative coordination of high-lift devices on the wing chord for single-slotted and double-slotted flaps:

$$C_f = 0.28 \cdot C_i = 0.28 \cdot 3.53 = 0.99\,(\text{m}).$$

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

$$C_s = 0.12 \cdot C_i = 0.12 \cdot 3.53 = 0.42\,(\text{m}).$$

2.2. Fuselage layout

For this airplane, the fuselage layout is designed to provide comfortable seating for passengers in the cabin.

In the preliminary design of the fuselage structure, we rely on one of the main types of construction. For this aircraft, we have chosen a typical semi-monocoque structure, the main advantage of which is that its strength and stiffness depends on many structural elements, not just the skin, as in a monocoque structure. Thus, the fuselage structure consists of formers (frames and bulkhead), stringers, spars, and skin. The formers shape the fuselage, support the stringers with the skin and prevent the latter from buckling. Bulkheads take the main loads, concentrating all loads from other parts. At the aft and at the forward parts of the fuselage there are special pressure bulkheads. They both perform the function of sealing the cabin.

Technologically, the fuselage is divided into three parts. The first part is the cockpit, which, in addition to the crew, houses many instruments and devices.

The next part of the fuselage is the passenger compartment and the luggage compartment under the floor. The wing center section is also located there.

The last tail section of the fuselage consists of a compartment for system equipment, smaller forms, spars and stringers. Since the formers are smaller, they must be more rigid. Because of this design solution, there are no structural problems with the support of both horizontal and vertical stabilizers.

When selecting the fuselage parameters, aerodynamic requirements for streamlining and cross-section must be taken into account. The round fuselage cross-section is the most efficient as it provides minimum weight and maximum strength, and meeting the strength and weight reduction requirements are important for aircraft design.

We also focus on geometric parameters such as fuselage diameter, fuselage length, fuselage thinness ratio, nose and tail unit geometry. We calculate the length of an aircraft fuselage based on the purpose of the aircraft, the number of passengers, the cabin layout, and the characteristics of the aircraft's center of gravity and angle of attack.

First of all, it is necessary to find the length of the entire fuselage of the airplane:

$$L_{fus} = FR_f \cdot D_{fus} = 11 \cdot 3.55 = 39.1\,(\text{m}),$$

where FR_f – fineness ratio of the fuselage.

Length of aircraft fuselage forward part. To calculate the length of the nose section of the airplane fuselage, it is necessary to find the fineness ratio of the nose section of the prototype fuselage. To do this, measure the length of the nose section from the beginning to the end of the cone-shaped part and divide it by the fuselage diameter. The resulting number will be the coefficient required for the calculation. Accordingly, the length of the nose section of the airplane fuselage will be equal to:

$$L_{fwd} = FR_{np} \cdot D_{fus} = 1.15 \cdot 3.55 = 4.08 \text{ (m)},$$

where FR_{np} – fineness ratio of the nose part.

Length of the fuselage tail part. Similarly, to the previous calculation, it is necessary to do the same in this part, but in this case, instead of the length of the nose of the aircraft fuselage, the length of the tail of the fuselage will be taken and divided by the diameter:

$$L_{tail\ unit} = FR_{tu} \cdot D_{fus} = 2.27 \cdot 3.55 = 8.06 \text{ (m)},$$

where FR_{tu} – fineness ratio of the tail unit.

Subtracting the length of the entire fuselage from the length of the fuselage forward and tail part, we get the approximate length of the passenger cabin at 26.92 meters.

Cabin width. To find the width of the passenger compartment of a passenger airplane at the point where the passenger seats are located, use the formula:

$$B_{cab} = n_2 \cdot b_2 + n_3 \cdot b_3 + n_{aisle} \cdot b_{aisle} + 2 \cdot \delta + 2 \cdot \delta_{wall},$$

where n_2, n_3 – number of blocks of seats with 2 or 3 seats in a cross section; b_2, b_3 – width of block of 2 seats or 3 seats, mm; n_{aisle} – number of aisles; b_{aisle} – aisle width, mm; δ – distance between external armrests to the decorative panels, mm; δ_{wall} – width of the wall (fuselage structure, insolation, decorative panels), mm.

For the economy class cabin, it was decided to design the passenger seats as 3+3 in each row. This is the corresponding width of the economy class cabin:

$$B_{cab\ econom} = 0 \cdot 0 + 2 \cdot 1450 + 1 \cdot 410 + 2 \cdot 40 + 2 \cdot 80 = 3550 \text{ (mm)}.$$

The result perfectly matches the diameter of the aircraft fuselage. For the business class, where the comfort conditions should be higher, it was decided to design passenger seats as 2+2 in each row. Accordingly, the width of the business class cabin:

$$B_{cab\ business} = 1 \cdot 1340 + 0 \cdot 0 + 1 \cdot 490 + 2 \cdot 90 + 2 \cdot 100 = 3550 \text{ (mm)}.$$

This also perfectly matches the diameter of the fuselage.

Cabin height. The height of the passenger compartment plays an important role in the size of the middle part of the passenger compartment, which value is:

$$H_{cab} = 1.48 + 0.17 \cdot B_{cab} = 1.48 + 0.17 \cdot 3.55 = 2.08 \text{ (m)}.$$

The windows, which are rectangular in shape with rounded corners, are placed in a row on each side of the fuselage. This choice of shape is due to the fact that the windows will be stress concentrators and rounding in the corners and reinforcement with structure elements will help to avoid this. The distance between the two windows will be about 550 mm, as they will be located between two bulkheads in this aircraft.

Length of the cabin. Using the formula below, it is necessary the length of the economy class cabin. Accordingly, for this aircraft it is:

$$L_{econom} = L_1 + (N-1) \cdot L_{seatpitch} + L_2,$$

$$L_{econom} = 1200 + (29-1) \cdot 760 + 250 = 22730 \text{ (mm)}.$$

The same calculation must be made to get the length of the business class cabin:

$$L_{business} = L_1 + (N-1) \cdot L_{seatpitch} + L_2,$$

$$L_{business} = 1200 + (4-1) \cdot 810 + 250 = 2760 \text{ (mm)}.$$

As a result, the length of the entire passenger cabin (economy and business class) is 26.61 meters, which is approximately 70% of the length of the entire fuselage and is a good result.

Baggage compartment. The aircraft's baggage compartments are located under the passenger cabin floor. This decision was made taking into account the fact that baggage affects the center of gravity of the aircraft. After all, incorrect placement of cargo and passengers can lead to in-flight accidents, so it is extremely important not only to accurately calculate the placement of cargo, but also to limit its weight.

Therefore, it is necessary to take into account the fact that the unit load on the floor K = 400…600 kg/m^2.

The area of cargo compartment is defined:

$$S_{cargo} = \frac{M_{bag}}{0.4 \cdot K} + \frac{M_{cargo\&mail}}{0.4 \cdot K} = \frac{16 \cdot 190}{0.4 \cdot 500} + \frac{15 \cdot 190}{0.4 \cdot 500} = 24.7\ (\mathrm{m}^2),$$

where M_{bag} – mass of baggage of all passengers, $M_{bag} = m \cdot n_{pass}$, kg; 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 16 kilograms for each passenger.

Cargo compartment volume is equal to:

$$V_{cargo} = \nu \cdot n_{passenger} = 0.2 \cdot 190 = 38\ (\mathrm{m}^3).$$

The design of the airplane's luggage compartment is similar to that of the prototype.

Galleys and buffets. According to international standards, if an airplane has a mixed layout, two meals must be prepared. If the flight duration is less than 3 hours, the passenger is not provided with food, only drinks, including water and tea. If the flight duration is even shorter, namely 1 hour, then buffets and toilets may not be issued.

As for the location of kitchen cupboards and refreshment, they should be located near the door, preferably between the cockpit and passengers or cargo, and have separate doors.

In turn, it is forbidden to place them next to toilet facilities or connected to a wardrobe. Also, according to international standards, the galley volume should be about 0.1 cubic meters per passenger, so the galley volume of this aircraft should be:

$$V_{galley} = 0.1 \cdot n_{passenger} = 0.1 \cdot 190 = 19\ (\mathrm{m}^3).$$

The total area of galley floor:

$$S_{galley} = \frac{V_{galley}}{H_{cab}} = \frac{19}{2.08} = 9.12\ (\mathrm{m}^2).$$

Passengers should be served meals every 3.5 to 4 hours of flight. The number of dishes per passenger, namely breakfast, lunch and dinner, is 0.8, kg; tea and water – 0.4, kg; and the total weight of food for passengers and crew is about 235, kg. The design of the airplane's buffet is similar to that of the prototype.

Lavatories. The number of toilet facilities is determined by the number of passengers and the duration of the flight: with

t > 4:00 one toilet for 40 passengers,

t = 2...4 hours and 50 passengers

t < 2 hours to 60 passengers.

$$t = \frac{Range_{flight}}{V_{cruise}} + 0.5 = \frac{6400}{825} + 0.5 = 8.26\,(\text{h}),$$

$$N_{lavatory} = \frac{N_{passenger}}{40} = \frac{190}{40} < 5.$$

As a result, 5 toilets of the same design as the prototype were chosen for this aircraft, given their 1-meter width and area: $S_{lavatory} = 1.5\,(\text{m}^2)$.

In addition to the lavatories, 2 galley areas are also designed for the airplane, with a similar design to the prototype. The location can be seen in the design layout of the aircraft.

2.3. Layout and calculation of basic parameters of tail unit

The choice of tail unit (TU) position plays an important role in this design. It can be considered one of the most important tasks in aerodynamics.

To ensure the longitudinal stability of the aircraft during maneuvering flight, its center of gravity should be in front of the aerodynamic center of the aircraft (also called the "focus"). And the longitudinal stability indicator itself can be determined by finding the arm of the aerodynamic moment of lift, referred to the average value of the aerodynamic chord of the wing.

$$m_x^{Cy} = \overline{x}_{cg} - \overline{x}_F < 0,$$

where $m_x^{C_y}$ – is the moment coefficient; $\bar{x}_{cg}$, $\bar{x}_F$ – center of gravity and focus coordinates.

Determination of the TU geometrical parameters. Usually, the areas of vertical S_{VTU} and horizontal S_{HTU} of TU is:

$$S_{HTU} = (0.18...0.25)\cdot S = (32.13...44.625),$$

$$S_{VTU} = (0.12..0.20)\cdot S = (21.42..35.7).$$

For more exact:

$$S_{HTU} = \frac{C_{MAC}\cdot S_{wing}}{L_{HTU}}\cdot A_{HTU} = \frac{4.53\cdot 178.5}{19.75}\cdot 0.8 = 32.8\ (\text{m}^2),$$

$$S_{VTU} = \frac{l_{wing}\cdot S_{wing}}{L_{VTU}}\cdot A_{VTU} = \frac{43.3\cdot 178.5}{18.37}\cdot 0.08 = 33.6\ (\text{m}^2),$$

where L_{HTU} and L_{VTU} – arms of horizontal TU and vertical TU; l_{wing}, S_{wing} – wing span and wing area; A_{HTU}, A_{VTU} – coefficients of static moments, values of which may be taken from the prototype.

The values of L_{HTU} and L_{VTU} are influenced not only by the length of the nose and tail sections of the fuselage, sweepback and location, but also by the stability and controllability of the aircraft. In the first approach it can be assumed that $L_{HTU} \approx L_{VTU}$ and we may find it from the dependences:

Trapezoidal scheme, normal scheme:

$$L_{VTU} = 2.1\cdot C_{MAC} = 2.1\cdot 4.35 = 9.52\ (\text{m}).$$

Light airplane:

$$L_{VTU} = 2.2\cdot C_{MAC} = 2.2\cdot 4.35 = 9.97\ (\text{m}).$$

Determination of the elevator area and direction:

$$S_{elevator} = 0.35\cdot S_{HTU} = 0.35\cdot 32.8 = 11.8\ (\text{m}^2).$$

Rudder area:

$$S_{rudder} = 0.21\cdot S_{VTU} = 0.21\cdot 33.6 = 6.88\ (\text{m}^2).$$

Choose the area of aerodynamic balance. If the speed of the flight M ≥ 0.75, than:

$$S_{able} \approx S_{rudder} = 6.88\,(\text{m}^2).$$

The area of trim tab:

$$S_{tabs} = 0.12 \cdot S_{rudder} = 0.12 \cdot 6.88 = 0.83\,(\text{m}^2).$$

Determination of the TU span. TU span is related to the following dependence:

$$l_{HTU} = 0.4 \cdot l_{wing} = 0.4 \cdot 43.3 = 17.3\,(\text{m}).$$

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 accordingly to the location of the engines. Taking it into account we assume:

$$h_{VTU} = 0.18 \cdot l_{wing} = 0.18 \cdot 43.3 = 7.79\,(\text{m}).$$

For high wing airplanes we need to set the upper limit. Tapper ratio of horizontal and vertical TU:

$$\eta_{HTU} = 2.3 \qquad \eta_{VTU} = 1.2$$

TU aspect ratio for transonic planes:

$$\lambda_{HTU} = 1.15 \qquad \lambda_{VTU} = 4$$

Determination of TU chords C_{tip}, C_{MAC}, C_{root} for HTU:

$$C_{tip\,HTU} = \frac{2 \cdot S_{HTU}}{(\eta_{HTU} + 1) \cdot l_{HTU}} = \frac{2 \cdot 32.8}{(2.3 + 1) \cdot 17.3} = 1.08\,(\text{m}),$$

$$C_{MAC\,HTU} = \frac{2}{3} \cdot \frac{\eta_{HTU}{}^2 + \eta_{HTU} + 1}{\eta_{HTU} + 1} \cdot C_{tip\,HTU} = \frac{2}{3} \cdot \frac{2.3^2 + 2.3 + 1}{2.3 + 1} \cdot 1.08 = 2.05\,(\text{m}),$$

$$C_{root\,HTU} = C_{tip\;\;HTU} \cdot \eta_{HTU} = 1.08 \cdot 2.3 = 2.81\,(\text{m}).$$

Determination of TU chords C_{tip}, C_{MAC}, C_{root} for VTU:

$$C_{tip\,VTU} = \frac{2 \cdot S_{VTU}}{(\eta_{VTU} + 1) \cdot l_{VTU}} = \frac{2 \cdot 33.6}{(1.2 + 1) \cdot 7.79} = 3.82\,(\text{m}),$$

$$C_{MAC\,HTU} = \frac{2}{3} \cdot \frac{\eta_{VTU}{}^2 + \eta_{VTU} + 1}{\eta_{VTU} + 1} \cdot C_{tip\,VTU} = \frac{2}{3} \cdot \frac{1.2^2 + 1.2 + 1}{1.2 + 1} \cdot 3.82 = 4.17\,(\text{m}),$$

$$C_{root\,HTU} = C_{tip\,VTU} \cdot \eta_{VTU} = b_{tipVTU} \cdot \eta_{VTU} = 4.59\,(\text{m}).$$

Width/chord ratio of the airfoil. For horizontal and vertical TU in the first approach, $\bar{C}_{TU} = 0.8 \cdot \bar{C}_{wing}$.

For more accuracy: Subsonic $\bar{C}_{TU} = 0.09$.

TU sweptback. TU sweptback is taken 5° (not more than wing sweptback). This was done in order to provide the control of the airplane in shock stall on the wing.

2.4. Calculation of basic parameters and layout of landing gear

At this design stage, only part of the parameters can be determined. The reason for this is not only the lack of a drawing of the general view of the aircraft, but also data on the position of its center of gravity.

The distance from the center of gravity to the main LG:

$$B_m = 0.3 \cdot C_{MAC} = 0.3 \cdot 4.53 = 1.36 \text{ (m)}.$$

If the distance is too large, there will be a problem with lift of the nose gear during take-off. In the opposite case, if the distance is too small, there is a possibility of striking of the tail of the airplane.

In addition, the load on the nose landing gear will be too small and the airplane will be unstable during run-up on a slippery runway and in the presence of wind.

Landing gear wheel base comes from the expression:

$$B = 0.35 \cdot L_{fus} = 0.4 \cdot 39.1 = 13.67 \text{ (m)}.$$

An airplane with an engine on the wing is of the greatest importance.

The latter equation means that the nose support accounts for 6...10% of the weight of the aircraft.

The distance from the center of gravity to the nose LG:

$$B_n = B - B_m = 13.67 - 1.36 = 12.3 \text{ (m)}.$$

Wheel track is:

$$T = (0.7...1.2) \cdot B \le 12 \text{ (m)},$$

$$T = 0.8 \cdot 15.62 = 10.93 \text{ (m)}.$$

To prevent nose roll, the value of T should be > $2H$, where H is the distance from the runway to the center of gravity.

In order to select wheels for the landing gear, several factors need to be taken into account, such as their size, the take-off weight load and the dynamic load for the front support.

The type of tires and their pressure will be determined by the runway surface to be used for the aircraft. Brakes, in turn, will be located on the main wheel and sometimes on the front wheel. The load on the wheel is determined:

$$F_{main} = \frac{(B - B_m) \cdot m_0 \cdot g}{B \cdot n \cdot z} = \frac{(13.67 - 1.36) \cdot 94203 \cdot 9.81}{13.67 \cdot 2 \cdot 2} = 208047 \text{ (N)},$$

$$F_{nose} = \frac{B_m \cdot m_0 \cdot g \cdot K_g}{B \cdot z} = \frac{1.36 \cdot 94203 \cdot 9.81 \cdot 1.5}{13.67 \cdot 2} = 78148.94 \text{ (N)},$$

where n, and z – is the quantity of the supports and wheels on the one leg; K_g = 1.5…2.0 – dynamics coefficient.

Based on the calculated F_{main} and F_{nose} and the value of $V_{take\text{-}off}$ and $V_{landing}$,, the pneumatics from the catalog are selected, which must meet the following ratios.

$$P^K_{slmain} \geq P_{main}; P^K_{s\ln ose} \geq P_{nose}; V^K_{landing} \geq V_{landing}; V^K_{takeoff} \geq V_{takeoff},$$

where K is an index indicating the value of the parameter allowed in the catalog.

To ensure the possibility of aircraft operating on unpaved runways, the pressure in the wheel pneumatics should be within $P = (3...5) \cdot 10^5$ (Pa).

The last step in the design of the landing gear of this aircraft is the selection of tires. The GOODYEAR data book was used for this purpose. Since the units of measurement in the data book are different from those used in the calculation, the first task is to convert the units of measurement obtained to the required ones:

$$F_{main} = 208047 \text{ (N)} = 46771.03 \text{ (lbs)},$$

$$F_{nose} = 78148.94 \text{ (N)} = 17568.58 \text{ (lbs)}.$$

Also, Rated Speed (Airplane Lift-off Speed) from Initial Data will be used:

$$V_{rated} = 271.06 \text{ (km/h)} = 168.4 \text{ (mph)}.$$

The most durable wheels were Type IV Size 40×14 (Table 2.1)

Table 2.1

Wheels characteristics

Size		40×14
CONSTRUCTION	ly Rating	28
	T or TL	TL
	Rated Speed (MPH)	174K
SERVICE RATING	Rated Load (lbs)	33500
	Rated Inflation (psi)	200
	Max. Braking Load (lbs)	50250
	Max. Bottoming Load (lbs)	100500
Tread Design / Trademark		Aircraft Rib
Part Number		461B-3208-TL

2.5. Determination of designed aircraft center of gravity

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 2.3 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

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)	0.1336	12583.6	2.04	25651.7
2	Fuel system	0.0081	763.0	1.63	1244.4
3	Flight control system, 30%	0.0018	166.7	2.72	453.2
4	Electrical equipment, 20%	0.0063	597.2	0.45	270.6
5	Anti-icing system, 70%	0.0047	447.1	0.45	202.5
6	Hydraulic system, 30%	0.0050	466.3	2.72	1267.4
7	Power plant	0.0803	7563.6	-3.73	-28212.1
	Equipped wing without landing gear and fuel	**0.2398**	**22587.6**	**0.04**	**877.7**

Ending of the table 2.2

1	2	3	4	5	6
8	Nose landing gear	0.0079	742.9	-11.18	-8305.5
9	Main landing gear	0.0315	2971.5	2.49	7399.1
10	Fuel for flight	0.2735	25762.6	1.36	35011.4
11	Reserve fuel	0.0274	2579.3	1.59	4089.4
	Totally equipped wing	**0.5801**	**54644.0**	**0.72**	**39072.3**

Table 2.3

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	0.0810	7626.7	19.53	148910.8
2	Horizontal tail unit	0.0102	957.1	37.76	36143.1
3	Vertical tail unit	0.0099	935.4	36.54	34181.5
4	Radiolocation equipment	0.0030	282.6	0.91	257.7
5	Dashboard and instrument equipment	0.0053	499.3	1.50	748.9
6	Aero navigation equipment	0.0045	423.9	1.50	635.9
7	Radio equipment	0.0023	216.7	3.15	682.5
8	Flight control system, 70%	0.0041	389.1	21.48	8356.0
9	Electrical equipment, 80%	0.0254	2389.0	19.53	46645.0
10	Hydraulic system, 70%	0.0116	1088.0	18.80	20455.2
11	Anti icing system, 30%	0.0020	191.6	36.97	7083.4
12	Air-conditioning system	0.0158	1490.3	17.57	26188.1
13	Emergency equipment	0.0110	1040.2	18.74	19492.4
14	Tools	0.0002	20.0	3.50	70.0
15	Water and liquid	0.0053	500.0	10.92	5460.0
16	Lavatory 1	0.0011	100.0	5.70	569.5
17	Lavatory 2	0.0042	400.0	10.92	4368.0
18	Lavatory 3	0.0000	0.0	0.00	0.0
19	Wardrobe 1	0.0004	35.0	5.70	199.3
20	Wardrobe 2	0.0004	35.0	31.33	1096.4
21	Wardrobe 3	0.0000	0.0	0.00	0.0
22	Galley 1	0.0021	200.0	5.70	1139.0
23	Galley 2	0.0042	400.0	31.33	12530.0
24	Baggage equipment	0.0061	570.0	10.00	5700.0
25	Interior panels, lining and insulation	0.0059	555.8	17.57	9766.8
26	Passengers' seats 1 (business)	0.0008	72.0	8.53	613.8

Ending of the table 2.3.

1	2	3	4	5	6
27	Passengers' seats 2 (economic)	0.0046	435.0	20.76	9029.3
28	Pilots' seats	0.0004	40.0	2.29	91.6
29	Flight attendants' seats	0.0006	60.0	6.49	389.1
30	Non-typical equipment	0.0050	471.0	20.76	9776.9
	Equipped fuselage without commercial load	**0.2274**	**21423.6**	**19.16**	**410580.2**
31	Passengers 1(business class)	0.0127	1200.0	8.53	10230.0
32	Passengers 2(economic class)	0.1385	13050.0	20.76	270878.9
33	Passengers' baggage	0.0323	3040.0	17.57	53421.9
34	Cargo. mail	0.0016	150.0	28.00	4200.0
35	On board meal	0.0025	235.4	5.70	1340.6
36	Flight attendants	0.0032	300.0	6.49	1945.5
37	Crew	0.0017	160.0	2.29	366.4
	Totally equipped fuselage	**0.4199**	**39559.0**	**19.03**	**752963.5**

Table 2.4

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	22587.6	17.58	397076.0
2	Nose landing gear (extended)	742.9	6.36	4725.1
3	Main landing gear (extended)	2971.5	20.03	59521.4
4	Fuel for flight	25762.6	18.90	486901.1
5	Reserve fuel	2579.3	19.13	49331.3
6	Equipped fuselage without commercial load	21423.6	19.03	407775.7
7	Passengers 1 (business class)	1200.0	8.53	10230.0
8	Passengers 2 (economic class)	13050.0	20.76	270878.9
9	Baggage of passengers	3040.0	17.57	53421.9
10	Cargo. mail	150.0	28.00	4200.0
11	On board meal	235.4	5.70	1340.6
12	Flight attendants	300.0	6.49	1945.5
14	Crew	160.0	2.29	366.4
15	Nose landing gear (retracted)	742.9	5.33	3960.0
16	Main landing gear (retracted)	2971.5	20.03	59521.4

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

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

$$X_{MAC} = \frac{m_f x_f + m_w \cdot x_w' - m_0 C}{m_0 - m_w},$$

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.

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:

$$\bar{X}_T = \bar{X}_C = \frac{X_{C.G.} - X_{MAC}}{b_{MAC}} \cdot 100\% = \frac{C}{b_{MAC}} \cdot 100\%.$$

In table 2.5 is shown position of gravity center in different variants.

Table 2.5

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)	94203.0	1747714.0	18.55	22.34
2	Take-off mass (landing gear retracted)	94203.0	1746948.9	18.54	22.16
3	Landing variant (landing gear extended)	68205.0	1259472.3	18.47	20.43
4	Transportation variant (without payload)	76227.6	1404932.0	18.43	19.65
5	Parking variant (without fuel and payload)	47725.7	869098.3	18.21	14.79

Conclusion for project part

In this part, the preliminary design was performed, which is an important part of the aircraft development, as it helped to determine such characteristics as wing shape and configuration, fuselage diameter and length, tail unit geometry, landing gear configuration, and tire selection. The center of mass was also calculated, which affects the stability and controllability of the aircraft. These characteristics also greatly affect further calculations, so great attention must be paid to this part to avoid mistakes.

3. OPTIMIZATION OF THE PASSENGER CABIN FLOOR BEAM

3.1. Introduction

The floor beam is one of the key structural elements that provides a solid foundation for the aircraft floor. It performs an important function of supporting passengers, cargo, control unit and other equipment. Floor beams are supported at both ends by frames, which provide additional rigidity and strength to the structure. The frames, in turn, are vertical elements that connect the floor beams to other parts of the aircraft airframe, thereby ensuring the overall strength and stability of the entire structure.

Seat tracks are attached to the upper chords of the floor beams to provide comfortable and safe seating. These tracks are firmly attached to the floor beams to efficiently transfer the load from the seats to the main beam structure. This ensures that all floor elements work together to carry the applied loads that occur during flight, passenger boarding and disembarking, and cargo transportation.

This prototype uses a fully extruded I-shaped beam made of high-strength 7075-T6 aluminum alloy (Fig. 3.1 and Table 3.1). This alloy is known for its excellent strength and corrosion resistance, making it an ideal material for aircraft structures. However, it was decided to optimize the design of this beam to improve its efficiency and functional characteristics.

Table 3.1

Geometric properties section of a solid beam

Notation		Value	Measurement units
1	2	3	4
h	Section height	155.0	mm
b	Chord width	55.0	mm
d	Wall thickness	2.00	mm
t_1	Upper chord thickness	5.00	mm

Ending of the table 3.1

1	2	3	4
t_2	Lower chord thickness	5.00	mm
w_u	Stiffener chord width	10.0	mm
t_u	Stiffener thickness	3.00	mm

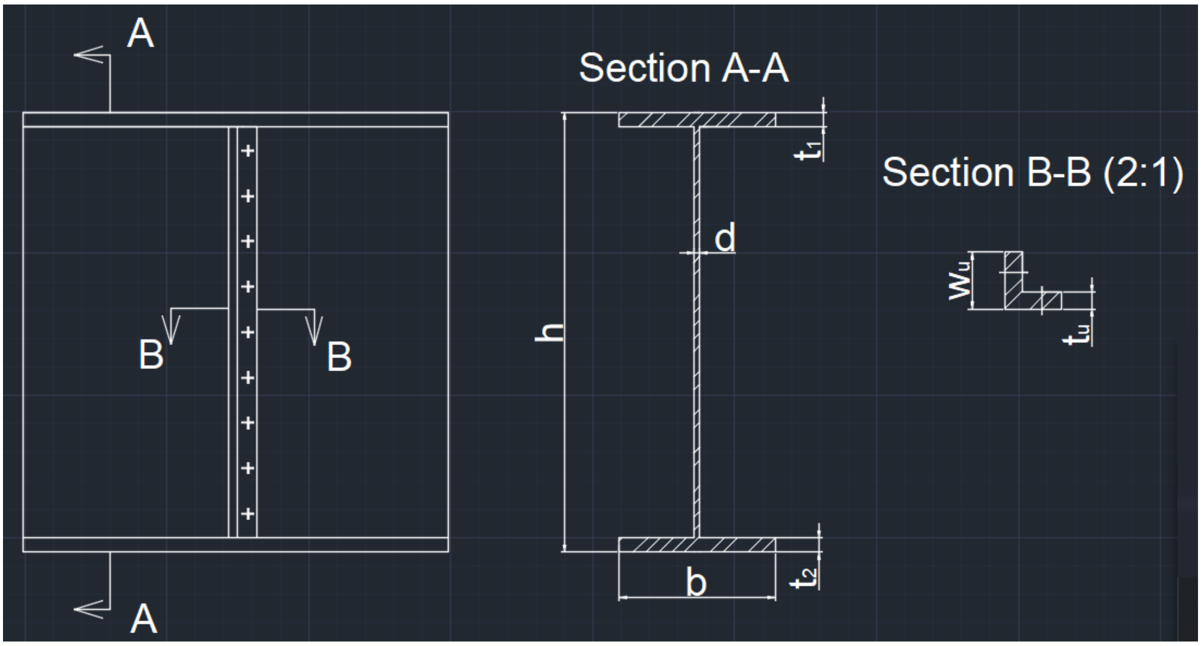


Fig. 3.1. Scheme of the geometric section of a solid beam.

After a comprehensive study of the scientific and technical literature, several approaches to selecting the appropriate material and cross-sectional shape for the floor beam were identified. This study included the analysis of the mechanical properties of materials, their behavior under load, and methods of connecting structural elements. Despite the possibility of changing the material, it was decided to retain the 7075-T6 aluminum alloy for the new beam design, given its excellent mechanical properties and compliance with aviation standards. However, the beam design will be changed. It will now have a comparable I-shaped cross-section, but will consist of four separate corners and a center wall, which will be riveted together (Fig. 3.2 and Table 3.2). A solid beam may have greater mass efficiency, but a prefabricated beam may be more efficient in terms of maintainability, as it will be easier to replace the top shelf in case of corrosion, and a prefabricated beam may be effective in stopping crack growth.

Table 3.2

Geometric properties section of a build-up beam

Notation		Value	Measurement units
h	Section height	125.0	mm
b	Chord width	51.5	mm
b_2	Corner width	25.0	mm
d	Wall thickness	1.50	mm
t_1	Doubler thickness	1.00	mm
t_2	Corner thickness	3.00	mm
w_u	Stiffener chord width	10.0	mm
t_u	Stiffener thickness	1.00	mm

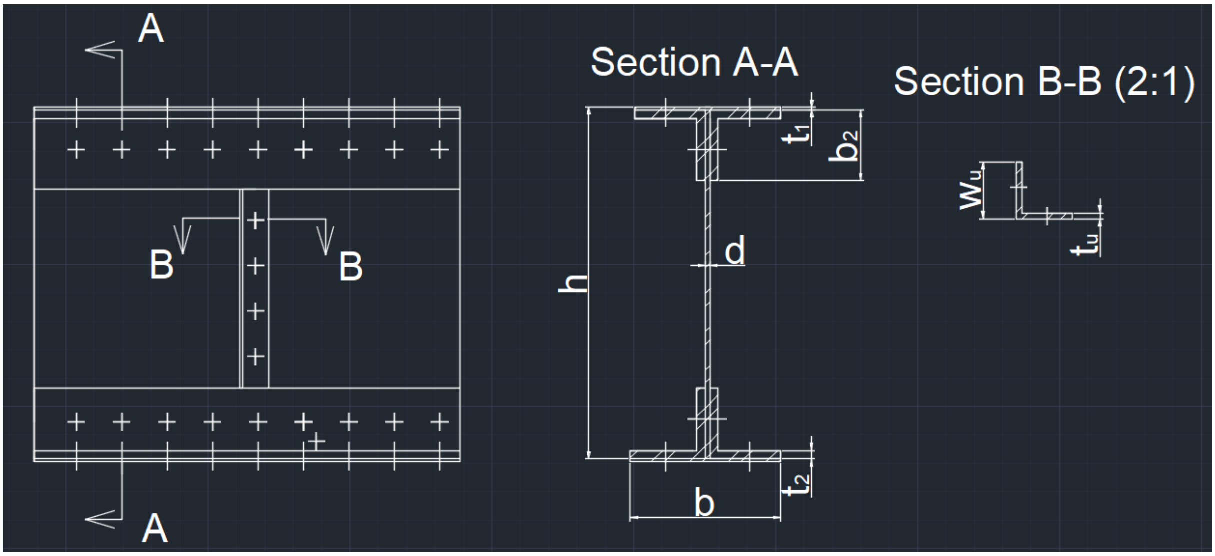


Fig. 3.2. Scheme of the geometric section of a build-up beam.

3.2. Selection of calculation methods and scheme

Since the design of a floor beam is concerned with its stiffness, it is most appropriate to consider the SR (shear resistant) beam model, i.e., a beam whose wall will not lose stability until the beam as a whole collapse. It should be noted that the model of such a beam does NOT provide for the presence of holes, except for those intended for fasteners. Any other holes will reduce the bearing capacity of the beam. The strength assessment will be performed analytically, and the stiffness assessment will be performed numerically using the ANSYS software.

Considering that the airplane is in flight most of the time, this is the state that will be considered in this calculation. The design scheme of the beam will be an articulated beam with five concentrated forces. Four of them are located at the connection of the seat track with the floor and are calculated as the load coming from the seats multiplied by the load factor value of 2.5 and a safety factor of 1.5. The last concentrated force is applied in the middle of the span, which is calculated as the weight of a person multiplied by the load factor value of 2.5 and a safety factor of 1.5.

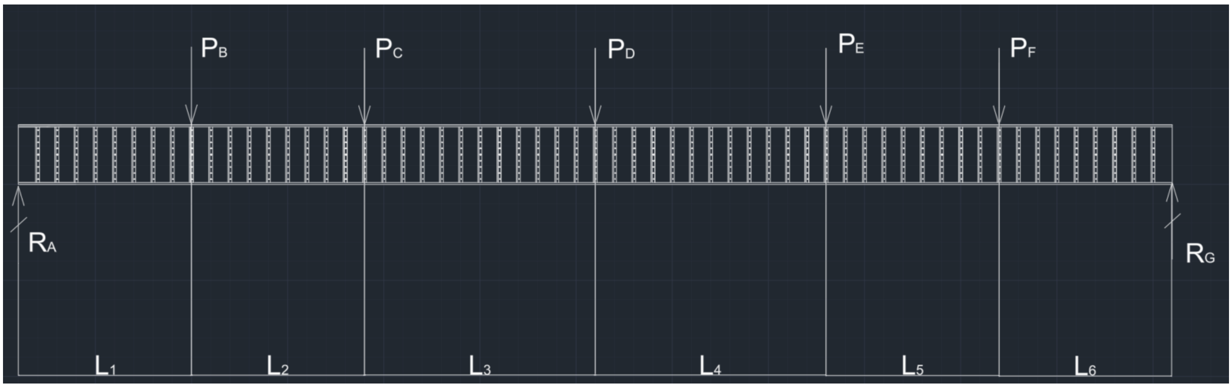


Fig. 3.3. The main calculation diagram of a beam.

It is necessary to determine the reactions in the supports using equilibrium equations:

$$R_G = \frac{-P_B \cdot L_1 - P_C \cdot (L_1 + L_2) - P_D \cdot (L_1 + L_2 + L_3) - P_E \cdot (L_1 + L_2 + L_3 + L_4)}{(L_1 + L_2 + L_3 + L_4 + L_5 + L_6)} =$$

$$\frac{-P_F \cdot (L_1 + L_2 + L_3 + L_4 + L_5)}{(L_1 + L_2 + L_3 + L_4 + L_5 + L_6)} = 11576\,(\text{N}),$$

$$R_A = \frac{-P_G \cdot L_6 - P_E \cdot (L_5 + L_6) - P_D \cdot (L_4 + L_5 + L_6) - P_C \cdot (L_3 + L_4 + L_5 + L_6)}{(L_1 + L_2 + L_3 + L_4 + L_5 + L_6)} =$$

$$\frac{-P_B \cdot (L_2 + L_3 + L_4 + L_5 + L_6)}{(L_1 + L_2 + L_3 + L_4 + L_5 + L_6)} = 11576\,(\text{N}),$$

The next step is to determine the values of the shear force and bending moment in different sections, which are shown in the table 3.3.

Table 3.3

Value of the shear force and bending moment on the section

Point		A	B	C	D	E	F	G
Location of the point	mm	0	450	900	1500	2100	2550	3000
Magnitude of the shear force Q	kg	0	11576.3	6615.0	1653.8	-1653.8	-6615.0	-11576.3
Magnitude of the bending moment M	N·mm $\cdot 10^6$	0	5.21	8.19	9.18	8.19	5.21	0

The diagrams for the shear force and bending moment were also plotted are shown in the fig. 3.4 and fig. 3.5.

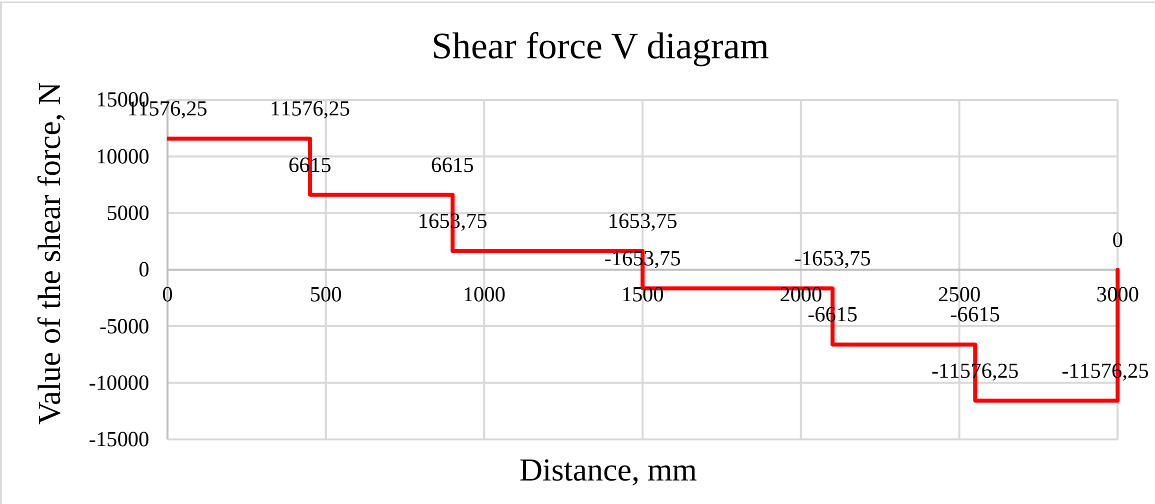


Fig. 3.4. Shear forces diagram.

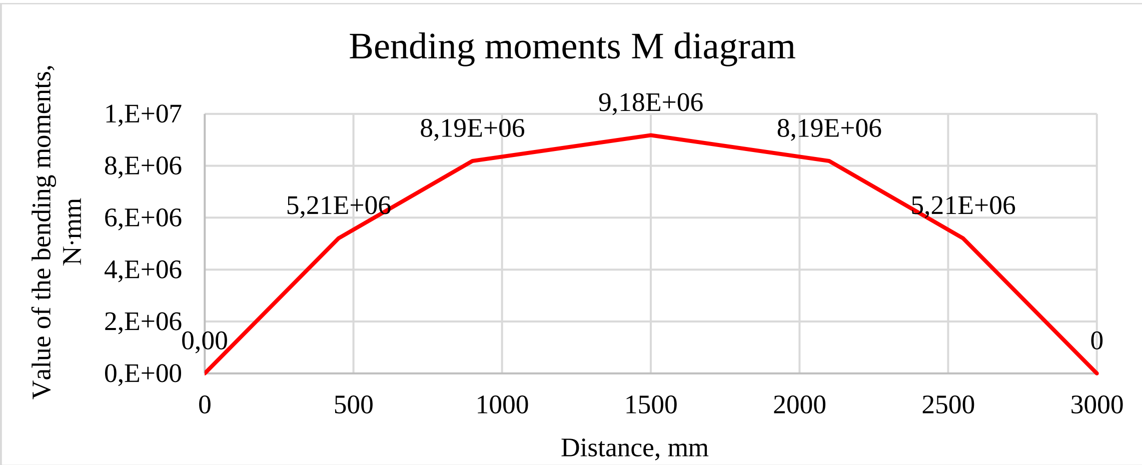


Fig. 3.5. Bending moments diagram.

For verification, the calculation was also performed in the Ansys software as show in fig. 3.6.

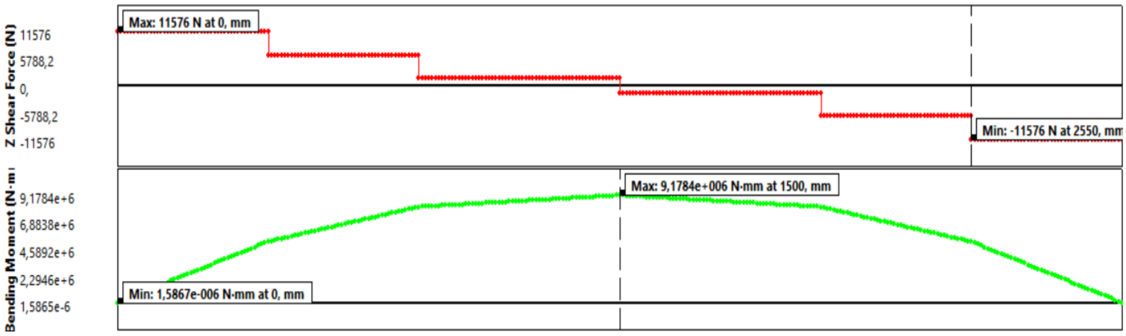


Fig. 3.6. Shear forces and bending moments diagram.

3.3. Calculation of the initial (solid) section

For evaluation of the web strength under combination of bending and shear loading, it is necessary to determine the geometric characteristics of the section. This will be done using the table method (Table 3.4). When calculating the initial cross-section (solid section), it is divided into 3 simple rectangles. The cross-sectional characteristics were taken as close as possible to those of the prototype in order to complete all the checks and achieve maximum weight efficiency. I-beams from the product range were not considered because they were too thick.

Table 3.4

Geometrical properties of cross-section

№	b_i	h_i	A_i	y_i	$A_i \cdot y_i$	$b_i \cdot h_i^3/12$	$A_i \cdot y_i^2$
1	50.0	5.0	250.0	152.5	38 125.0	520.8	$5.81 \cdot 10^6$
2	145.0	2.0	290.0	77.5	22 475.0	96.7	$1.74 \cdot 10^6$
3	50.0	5.0	250.0	2.5	625.0	520.8	$1.56 \cdot 10^3$
	mm	mm^2	mm	mm^3	mm^4	mm^4	

The value of the axial moment of inertia for the entire section is calculated using the following formulas:

$$J = \sum \frac{b_i \cdot h_i^3}{12} + \sum A_i \cdot y_i^2 - A \cdot y^2 = 3.09 \cdot 10^6 \text{ (mm}^4\text{)},$$

where b_i and h_i – element width and height, mm; A_i and A – area of element/section, mm^2 which calculeted by formula:

$$A = \sum A_i = 840\,(\text{mm}^2),$$

y_i and y – distance from the center of mass of the element/section to the axis, mm, which calculeted by formula:

$$y = \frac{\sum A_i \cdot y_i}{\sum A_i} = 77.5\,(\text{mm}).$$

After calculating the axial moment of inertia, it becomes possible to determine the active maximum bending and shear stresses in the critical section. Section D was chosen for determination maximum active bending stresses because the maximum bending moment is applied at this point. Section A was chosen for determination maximum active shear stresses because the maximum shear force is applied at this point.

Formula for calculating maximum active bending stresses:

$$f_{b\,\max} = \frac{M_{\max} \cdot (y - t_2)}{J} = 229.83\left(\frac{\text{N}}{\text{mm}^2}\right),$$

where $M_{\max}$ – maximum bending moment at beam, $\text{N} \cdot \text{mm}$.

Formula for calculating maximum active shear stresses:

$$f_{s\,\max} = \frac{Q_{\max} \cdot S^*}{J \cdot t_2} = 33.3\left(\frac{\text{N}}{\text{mm}^2}\right),$$

where $Q_{\max}$ – maximum shear force, kg; S^* – the static moment of inertia of half the section, mm^3, which is calculated by the formula:

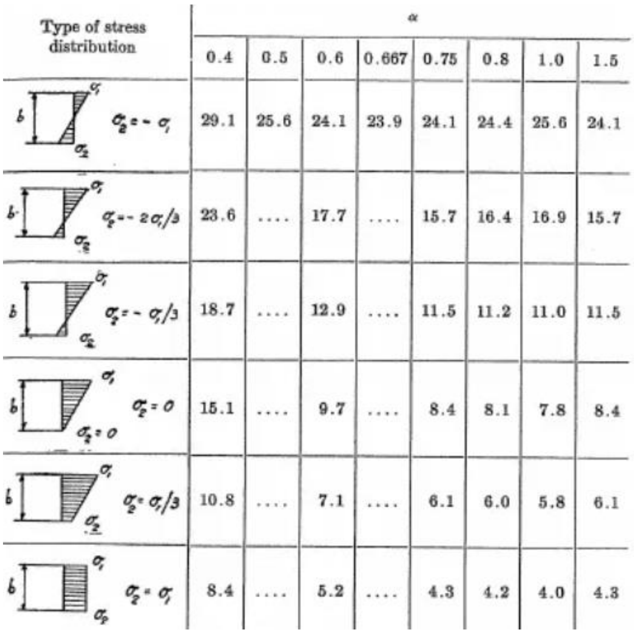
$$S^* = \sum A_i \cdot y_i = 17815.6\,\text{mm}^3.$$

It is also necessary to determine the allowable bending and shear stresses. Thin-walled structure is subjected to the bending and shear may buckle. So buckling strength must be checked. At first iteration of calculation is assumed that stresses don`t excess proportional limit.

Formula for calculating allowable bending stresses:

$$F_{cr} = \frac{\pi^2 \cdot E}{12 \cdot (1-\mu^2)} \cdot k \cdot \left(\frac{t_2}{h}\right)^2 = 317.35\left(\frac{\text{N}}{\text{mm}^2}\right),$$

where E – modulus of elasticity for a given material, N/mm^2; μ – Poisson's ratio; k – end fixity factor, which chosen for ratio between compressed and tension bending as show in Fig. 3.7 [2, p. 401].



Type of stress distribution	α							
	0.4	0.5	0.6	0.667	0.75	0.8	1.0	1.5
$\sigma_2 = -\sigma_1$	29.1	25.6	24.1	23.9	24.1	24.4	25.6	24.1
$\sigma_2 = -2\sigma_1/3$	23.6	....	17.7	....	15.7	16.4	16.9	15.7
$\sigma_2 = -\sigma_1/3$	18.7	....	12.9	....	11.5	11.2	11.0	11.5
$\sigma_2 = 0$	15.1	....	9.7	....	8.4	8.1	7.8	8.4
$\sigma_2 = \sigma_1/3$	10.8	....	7.1	....	6.1	6.0	5.8	6.1
$\sigma_2 = \sigma_1$	8.4	....	5.2	....	4.3	4.2	4.0	4.3

Fig. 3.7. Plate factor for simply supported plates in nonuniform longitudinal compression.

End fixity factor are chosen for ratio of the long side and short side of the bending panel 0.4 when real ratio is 0.35. Raising of the ratio makes calculation close to the reality.

Comparison of the critical bending stresses with proportional limits must be provided to check necessity of recalculation of critical bending stresses. Empirical value of proportional limit is 2/3 of yield limit.

$$F_{prop} = 330.72\left(\frac{\text{N}}{\text{mm}^2}\right),$$

$$F_{cr} < F_{prop}.$$

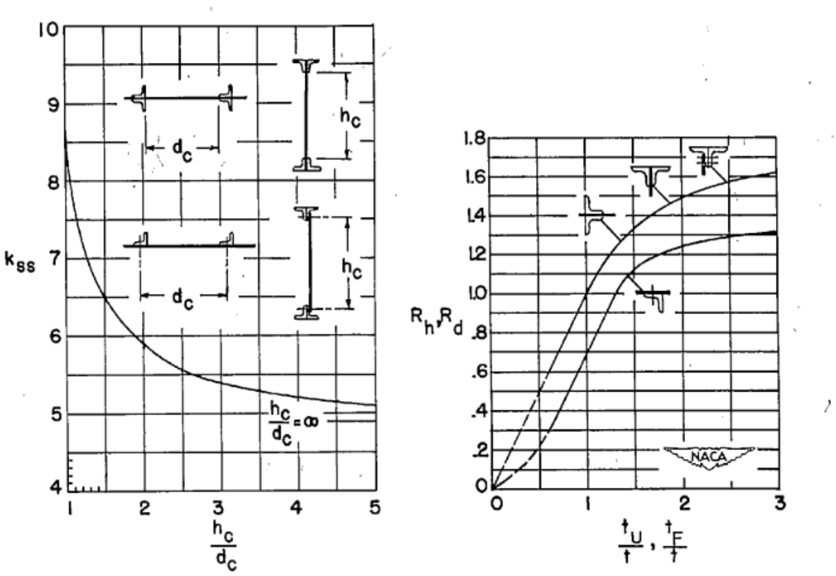
Formula for calculating allowable shear stresses:

$$F_{s\,cr}=k_{ss}\cdot E\cdot\left(\frac{t_2}{d_c}\right)^2\cdot\left(R_h+\frac{1}{2}\cdot(R_d-R_h)\cdot\left(\frac{d_c}{h_c}\right)^3\right)=161.7\left(\frac{\text{N}}{\text{mm}^2}\right),$$

where h_c – height of the web which strive to buckle, mm, which calculate by formula:

$$h_c=h-\frac{1}{6}\cdot h-\frac{1}{6}\cdot h=120\,(\text{mm}),$$

d_c – distance between stiffeners, mm; k_{ss}, R_d and R_h – empirical restrain coefficient for plate shear, which determined from the fig. 3.8 [3, p. 479].



(a) Theoretical coefficients for plates with simply supported edges. (b) Empirical restraint coefficients.

Fig. 3.8. Graphs for calculating buckling stress of webs.

At this stage, it is possible to check the safety factor of the web under combined load. This is done using a formula:

$$MS=\frac{1}{\sqrt{\left(\frac{f_{b\max}}{F_{b\,cr}}\right)^2+\left(\frac{f_{s\max}}{F_{s\,cr}}\right)^2}}-1=0.328.$$

The next step is evaluation of the compressed and tensile chords strength under bending moment, it is necessary to determine the values of the maximum effective stresses of the compressed (upper) chord, which calculeted by formula:

$$f_{ch} = \frac{M_{\max}}{J} \cdot (h - y) = 229.8 \left(\frac{\text{N}}{\text{mm}^2} \right).$$

Compressed chord of the beam may buckle as compered column. So it is necessary to determine the column buckling strength must be provided. To provide acceptable accuracy and safety compressed chord may be considered as column. Cross-section of this column is loaded by compression area of the beam: upper chord and half of the web.

Scheme of loading is a simply supported beam subjected by the concentrated force. The length of this beam is the longest distance, namely from the inner seat track to the intercostal, which is located in the middle of the beam span.

$$L_c = 600 \text{ (mm)}.$$

At first step, checking of elastic buckling strength must be provided. Critical elastic bending stresses are found using the Euler formula, for which it is necessary to find the value of the axial moment of inertia relative to the axis about which it will buckling. In this structure stiffness of web and stretched portion won`t let column buckles about X axis. Moment of inertia of the column about Y axis is found by table method (Table 3.5).

Table 3.5

Determination of the moment of inertia of upright about y-y axis

№	b_i	h_i	A_i	y_i	$A_i \cdot y_i$	$b_i \cdot h_i^3/12$	$A_i \cdot y_i^2$
1	55.0	5.00	275.0	2.50	687.50	572.9	$1.72 \cdot 10^2$
2	72.5	2.00	145.0	36.3	5 256.3	48.33	$1.91 \cdot 10^5$
	mm		mm^2	mm	mm^3	mm^4	

The value of the axial moment of inertia:

$$J = \sum \frac{b_i \cdot h_i^3}{12} + \sum A_i \cdot y_i^2 - A \cdot y^2 = 6.94 \cdot 10^4 \text{ (mm}^4\text{)}.$$

The value of the area of the section:

$$A = \sum A_i = 420 \text{ (mm}^2\text{)}.$$

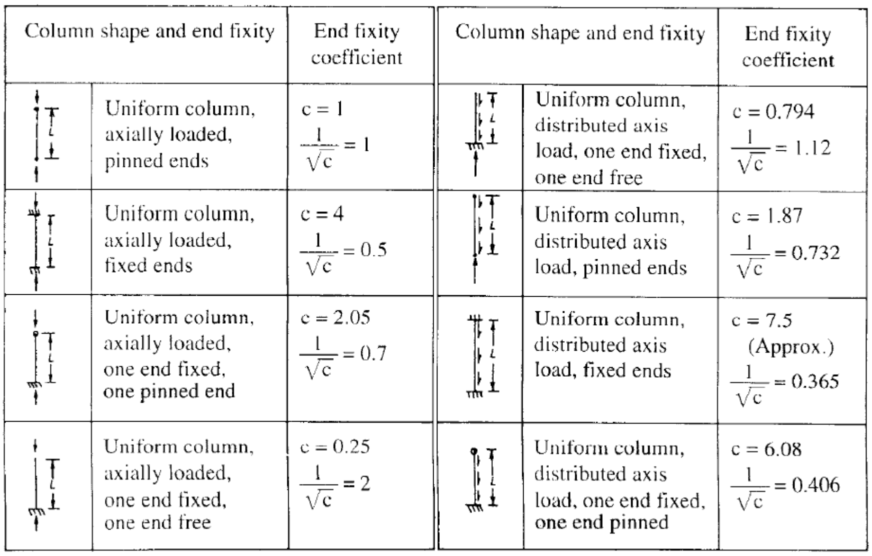
The value of the distance from the center of mass of the section:

$$x = \frac{\sum A_i \cdot x_i}{\sum A_i} = 27.5 \text{ (mm)}.$$

Then, it is possible to calculate the allowable critical stresses for the upper chord:

$$F_{ch\,cr} = \frac{\pi^2 \cdot E \cdot J}{\left(\frac{L}{\sqrt{c}}\right)^2 \cdot A} = 321.36 \left(\frac{\text{N}}{\text{mm}^2}\right),$$

where c is end fixity factor for column buckling, which determined from the Fig. 3.9 [4, p. 106].



Column shape and end fixity	End fixity coefficient	Column shape and end fixity	End fixity coefficient
Uniform column, axially loaded, pinned ends	$c = 1$, $\frac{1}{\sqrt{c}} = 1$	Uniform column, distributed axis load, one end fixed, one end free	$c = 0.794$, $\frac{1}{\sqrt{c}} = 1.12$
Uniform column, axially loaded, fixed ends	$c = 4$, $\frac{1}{\sqrt{c}} = 0.5$	Uniform column, distributed axis load, pinned ends	$c = 1.87$, $\frac{1}{\sqrt{c}} = 0.732$
Uniform column, axially loaded, one end fixed, one pinned end	$c = 2.05$, $\frac{1}{\sqrt{c}} = 0.7$	Uniform column, distributed axis load, fixed ends	$c = 7.5$ (Approx.), $\frac{1}{\sqrt{c}} = 0.365$
Uniform column, axially loaded, one end fixed, one end free	$c = 0.25$, $\frac{1}{\sqrt{c}} = 2$	Uniform column, distributed axis load, one end fixed, one end pinned	$c = 6.08$, $\frac{1}{\sqrt{c}} = 0.406$

Fig. 3.9. Column End-fixity coefficient.

It is also necessary to find the critical stresses at the buckling of flange:

$$F_{cr\,f} = K_c \cdot E \cdot \left(\frac{t}{b}\right)^2 = 79.0 \left(\frac{\text{N}}{\text{mm}^2}\right),$$

where K_c is end fixity factor for compressed plate, which determined from the fig. 3.10. It is determined for simply supported with 3 edges plate with 1 free unloaded edge [4, p. 458].

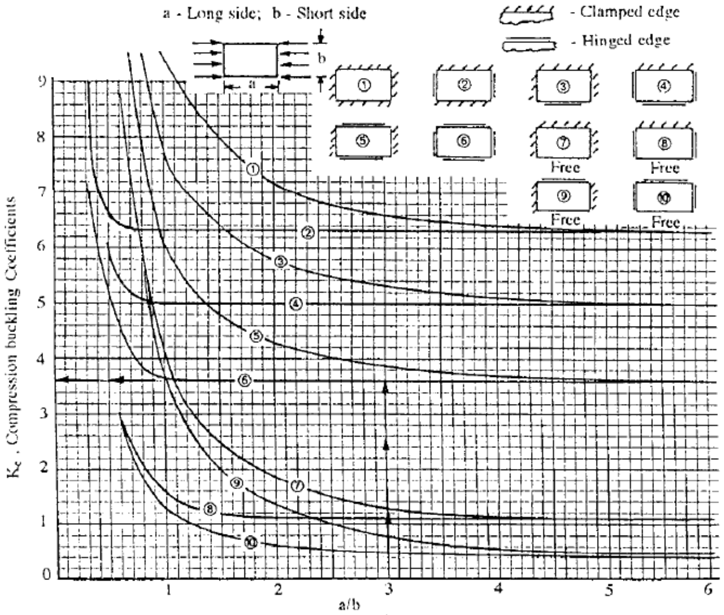


Fig. 11.3.1 K_c Coefficients (Compression)

Fig. 3.10. End fixity factor for compressed plate.

$$F_{ch\,cr} < f_{cr\,f}$$

Elastic column buckling occurs much more later than plate buckling. So, there is necessarily to take into account plate buckling during allowable stresses calculation. For this purpose, a formula of Johnson Euler can be used, which looks like this:

$$F_{jcr} = F_{cc} - \frac{F_{cc}^2}{4 \cdot \pi^2 \cdot E} \cdot \left(\frac{L^`}{\rho} \right)^2 = 249.35 \left(\frac{\text{N}}{\text{mm}^2} \right),$$

where L – the effective length of the column, which is calculated by the formula:

$$L^` = \frac{L}{\sqrt{c}} = 600\,(\text{mm}),$$

ρ – column flexibility, which is calculated by the formula:

$$\rho = \sqrt{\frac{J}{A}} = 12.85\,(\text{mm}),$$

F_{cc} – crippling stress, which calculated for T-section by formula:

$$F_{cc} = 0.67 \cdot \left(\left(\frac{g \cdot t^2}{A} \right) \cdot \left(\frac{E}{F_{cy}} \right)^2 \right)^{0.4} \cdot F_{cy} = 338.48 \left(\frac{\text{N}}{\text{mm}^2} \right),$$

g – coefficient for T-section, which equal 3.

At this stage, it is possible to check the coefficient of safety of the calculation.

$$MS = \frac{F_{jcr}}{f_{ch}} - 1 = 0.084.$$

It is also necessary to determine the value of the maximum effective stresses of the tensile (lower) chord.

$$f_t = \frac{M_{\max}}{J} \cdot y = 229.8\left(\frac{\mathrm{N}}{\mathrm{mm}^2}\right).$$

At this stage, it is possible to check the margin of safety of the tensile chord:

$$MS = \frac{F_{tu}^*}{f_t} - 1 = 0.956,$$

where F_{tu}^* – an underestimated F_{tu} value due to the presence of holes in the area and, accordingly, lower strength. It is calculated by the formula:

$$F_{tu}^* = F_{tu} \cdot 0.9 \cdot \frac{A_{net}}{A_{gross}} = 449.6\left(\frac{\mathrm{N}}{\mathrm{mm}^2}\right).$$

Also, it is necessary to check the safety factor of the shear web strength.

$$MS = \frac{F_{su}}{f_{s\max}} - 1 = 8.719.$$

For determination of the coefficient of safety of tensile chord, the unloaded stiffeners must be checked. To prevent bending of the web, the racks must have sufficient bending stiffness. Therefore, it is necessary to check it. Also, check the ratio between the thickness of the web and the attached flange of the upright.

Moment of inertia of the upright about x-x axis is found using table method (Table 3.6).

Table 3.6

Determination of the moment of inertia of upright about x-x axis

№	b_i	h_i	A_i	y_i	$A_i \cdot y_i$	$b_i \cdot h_i^3/12$	$A_i \cdot y_i^2$
1	7.00	3.00	21.0	1.5	31.50	15.75	47.25
2	3.00	10.0	30.0	5.0	150.0	250.0	750.0
	mm		mm^2	mm	mm^3	mm^4	

Moment of inertia of the upright:

$$J_u = \sum A_i \cdot y_i^2 + J_i = 1063.0\,(\text{mm}^4).$$

Required minimum stiffeners moment of inertia to dimension of web ratio is 0.65. Since minimum moment of inertia:

$$J_{u\,\min} = 0.65 \cdot h \cdot t_2^3 = 806.0\,(\text{mm}^4),$$

$$J_u > J_{u\,\min}.$$

Checking of thickness of the unloaded upright and strength of the loaded upright are not necessary for now as unloaded upright haven`t already answered to requirements of strength.

For checking of the local strength, needs to determinate the net area bending stresses in chord web attachment. To check strength of the web to frame attachment is necessary to determine moment of inertia of the cross-section of the net area.

To find net area moment inertia is necessary to subtract inertia moments of bolts about the X axis. Moments of inertia of bolts are determined using table method (Table 3.7).

Table 3.7

Geometrical properties of cross-section

№	b_i	h_i	A_i	y_i	$A_i \cdot y_i$	$b_i \cdot h_i^3/12$	$A_i \cdot y_i^2$
1	4.00	4.00	16.0	163.0	2 608.0	21.33	$4.25 \cdot 10^5$
	mm		mm^2	mm	mm^3	mm^4	

Assume that the number of bolts at the wall-frame connection in A section is 8 pieces. So, the value of the axial moment of inertia:

$$J_r = \sum \frac{b_i \cdot h_i^3}{12} + \sum A_i \cdot y_i^2 - A \cdot y^2 = 21.3\,(\text{mm}^4).$$

The value of the area of the section:

$$A = \sum A_i = 128.0\,(\text{mm}^2).$$

The value of the distance from the center of mass of the section:

$$y = \frac{\sum A_i \cdot y_i}{\sum A_i} = 163.0 \text{ (mm)}.$$

The value of net area moment of inertia of cross-section:

$$J_{net} = J - J_r \cdot 8 = 3.09 \cdot 10^6 \text{ (mm}^4\text{)}.$$

Net area bending stresses:

$$f_{net} = \frac{M \cdot (y - y_r)}{J_{net}} = 450.45 \left(\frac{\text{N}}{\text{mm}^2} \right),$$

where y_r is 7/16 inch or 11.1 mm.

Then, it is necessary to determine the shear stress over the grid area. The shear stresses of the web to the chordal fastener are determined in accordance with the structure idealization approach. This method allows us to obtain an increased value of the applied stresses.

$$f_s = \frac{Q}{(h - 2 \cdot y_r) \cdot t_2} = 43.59 \left(\frac{\text{N}}{\text{mm}^2} \right).$$

The value of net area shear stresses is calculated by formula:

$$f_{s\,net} = \left(\frac{p}{p-d} \right) \cdot \left(\frac{t}{t_p} \right) \cdot f_s = 58.12 \left(\frac{\text{N}}{\text{mm}^2} \right).$$

At this stage, it is possible to check the safety factor of the web to frame attachment:

$$MS = \frac{1}{\sqrt{\left(\frac{f_{b\,net}}{F_{tu}} \right)^2 + \left(\frac{f_{s\,net}}{F_{su}} \right)^2}} - 1 = 0.167.$$

Also, it is necessary to check the safety factor of the web to upright attachment

$$MS = \frac{F_{su}}{f_{s\,net}} - 1 = 4.571.$$

The local strength of the web to the stiffener must be checked in this calculation. For this purpose, it is necessary determinate to the distance between the inner stiffener attachment to the chord and the next attachment that attaches the web to the stiffener:

$$s = \frac{(h - 2 \cdot y_r - 5 \cdot p)}{2} = 26.39 \text{ (mm)}.$$

To determine the net area at attachment shear stresses, the following formula is used:

$$f_{s\,net} = \left(\frac{s}{s - d_c}\right) \cdot \left(\frac{t}{t_p}\right) \cdot f_s = 51.38 \left(\frac{\text{N}}{\text{mm}^2}\right).$$

At this stage, it is possible to check the safety factor of the web to stiffeners attachment strength:

$$MS = \frac{1}{\sqrt{\left(\frac{f_{b\,net}}{F_{tu}}\right)^2 + \left(\frac{f_{s\,net}}{F_{su}}\right)^2}} - 1 = 0.172.$$

It is additionally needed to determine the strength of the web fasteners to the chord. For this purpose, the effective shear flow must be determined, because the fasteners must be able to withstand the shear flow along the web. It is determined using the formula:

$$q = f_s \cdot t = 87 \left(\frac{\text{N}}{\text{mm}}\right).$$

The following is the determination of the allowable shear flow when the number of rows of fasteners is 3. The allowable shear flow is determined by the formula:

$$q_{all} = P_{all} \cdot \frac{N_f}{p} = 711.7 \left(\frac{\text{N}}{\text{mm}}\right),$$

where P_{all} – allowable load which rivet may withstand, which determined from the fig. 3.11 [4, p. 282].

Fastener Diameter		$\frac{3}{32}$	$\frac{1}{8}$	$\frac{5}{32}$	$\frac{3}{16}$	$\frac{1}{4}$	$\frac{5}{16}$	$\frac{3}{8}$	$\frac{7}{16}$	$\frac{1}{2}$	$\frac{9}{16}$
Fastener Material	F_{su} (Ksi)	Ultimate Single Shear Load (lbs./fastener)									
1100F(A)	9										
5056(B)	28	203	363	556	802	1450	2295	3275			
2117-T3(AD)	30	217	388	596	862	1555	2460	3510			
2017-T31(D)	34	247	442	675	977	1765	2785	3980			
2017-T3(D)	38	275	494	755	1090	1970	3115	4445			
2024-T31(DD)	41	296	531	814	1175	2125	3360	4800			
7075-T73(E)	41										
7075-T731 (E)	43	311	558	854	1230	2230	3525	5030			
A-286 CRES	90	—	—	1726	2485	4415	6906	9938			
Ti-6Al-4V & Alloy Steel	95	—	—	1822	2623	4660	7290	10490	14280	18650	23610
Alloy Steel	108	—	—	2071	2982	5300	8280	11930	16240	21210	26840
	125	—	—	2397	3452	6140	9590	13810	18790	24540	31060
	132	—	—	2531	3645	6480	10120	14580	19840	25920	32800

(Ref. 9.1)

Multiplied by the correction factors (see Fig. B.6.7) if t/D<0.33 (single shear) and t/D<0.67 (double shear)

Fig. 9.2.5 Rivet Shear-off Strength Allowables

Fig. 3.11. Rivet Shear-off Strength Allowables.

At this stage, it is possible to check the safety factor of the web to chord attachment fitting strength:

$$MS = \frac{q_{all}}{q} - 1 = 7.163.$$

The next step is determination of total mass of the structure. Total mass of the structure is determined using table method (Table 3.8) and for gross area.

Table 3.8

Mass of the gross area structure

Name of the element	Area of the element	Length	Volume	Density	Mass (m_i)
Web	290.0	3000	$8.70 \cdot 10^5$	$2.70 \cdot 10^{-5}$	23.49
Upper chord	275.0	3000	$8.25 \cdot 10^5$	$2.70 \cdot 10^{-5}$	22.28
Lower chord	275.0	3000	$8.25 \cdot 10^5$	$2.70 \cdot 10^{-5}$	22.28
Stiffener	51.00	2610.0	$1.33 \cdot 10^5$	$2.70 \cdot 10^{-5}$	3.59
	mm^2	mm	mm^3	kg/mm^3	kg

$$m_t = \sum m_i = 71.63\,(\text{kg}).$$

It was also decided to determine the largest displacement of the structure using Ansys software. For this purpose, were used "Static Structure" Analysis System where the geometry of the structure was set in "Space Claim" like a beam element (Fig. 3.12).

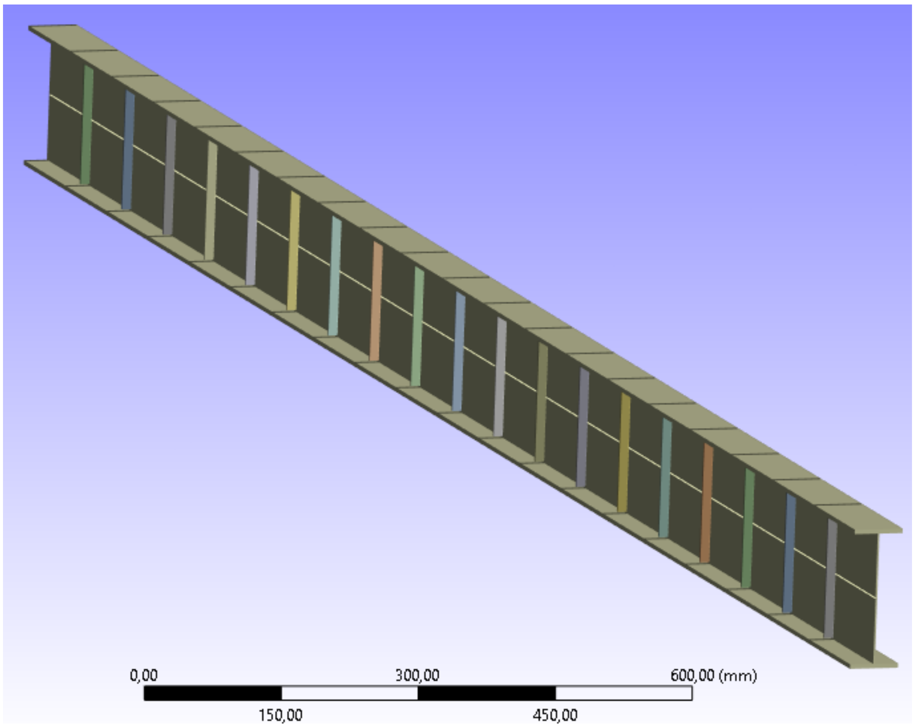


Fig. 3.12. Structure geometry.

Then, using the "Model", the mesh for the beam must be specified as shown in fig. 3.13.

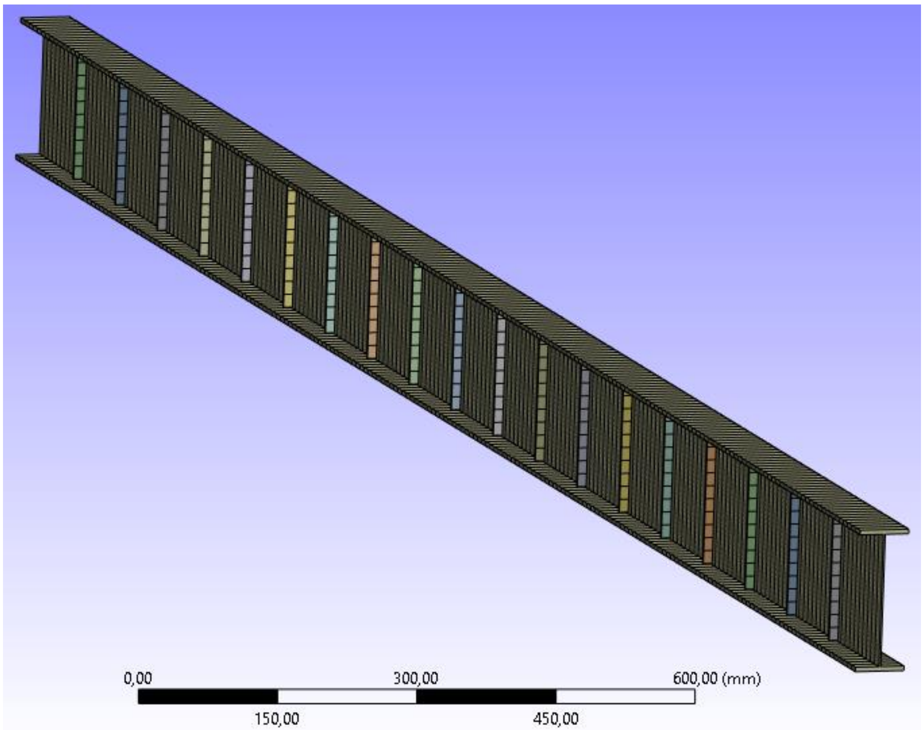


Fig. 3.13. Structure mesh.

The beam is loaded according to the design scheme shown in fig. 3.3. The load is specified as "force" and the supports are specified as "remote displacement" as shown in fig. 3.14:

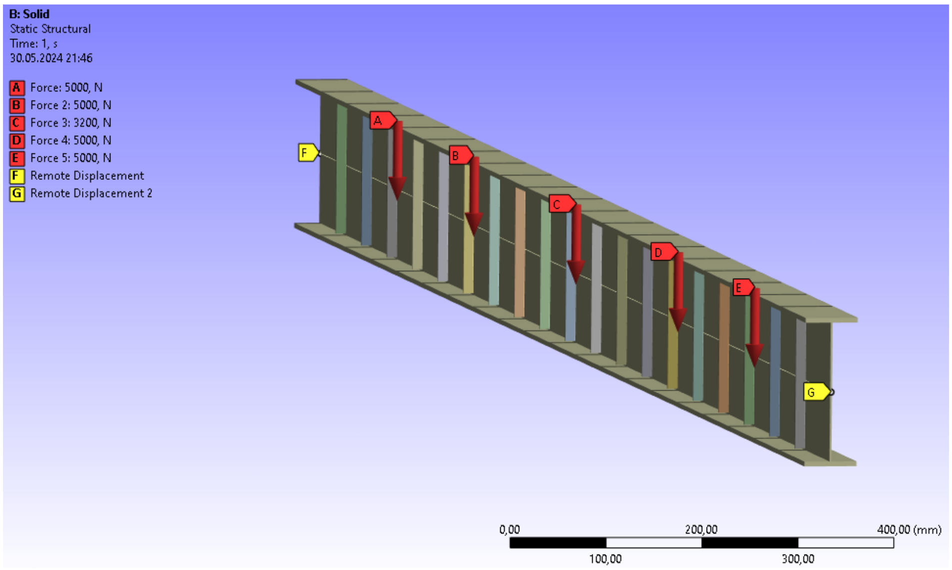


Fig. 3.14. Structural loading.

Then, by setting the Total deformations in the Solution, the result of the structure displacement was obtained as shown in fig. 3.15:

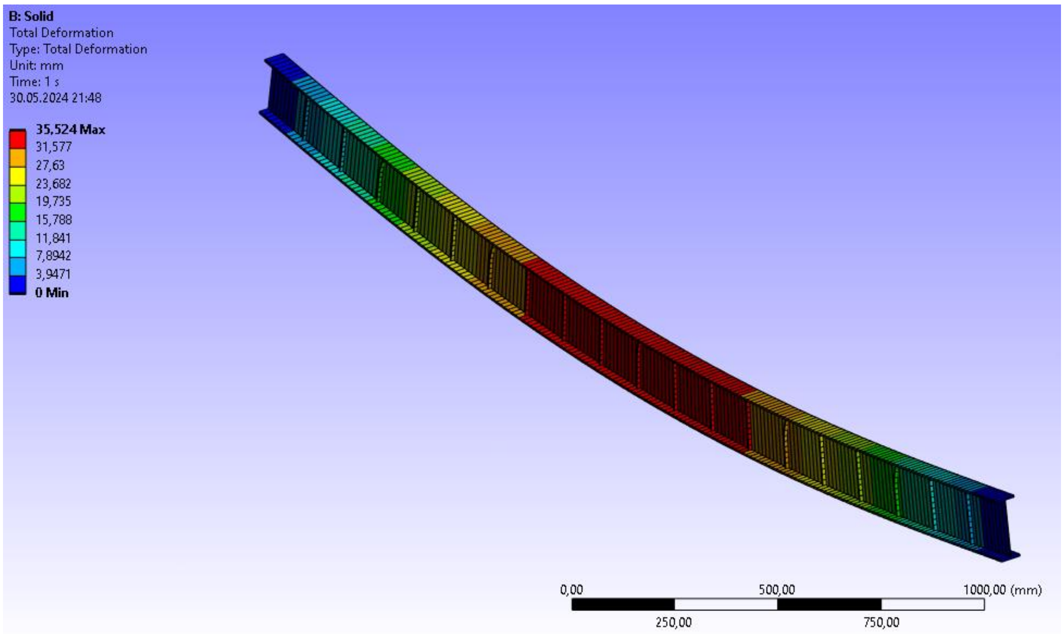


Fig. 3.15. The result of the calculation.

3.4 Calculation of the proposed (build-up) section

For evaluation of the web strength under combination of bending and shear loading, it is necessary to determine the geometric characteristics of the section. This will be done using the table method (Table 3.9). In calculating the new proposed cross-section (build-up cross-section), the values for the corners were taken from the assortment, and the other elements were calculated as simple rectangles.

Table 3.9

Geometrical properties of cross-section

№	b_i	h_i	A_i	y_i	$A_i \cdot y_i$	$b_i \cdot h_i^3/12$	$A_i \cdot y_i^2$
1	51.5	1.0	51.5	126.5	6 514.8	4.3	$8.24 \cdot 10^5$
2	–		143.0	118.0	16 874.0	8 100.0	$1.99 \cdot 10^6$
3	–		143.0	118.0	16 874.0	8 100.0	$1.99 \cdot 10^6$
4	1.5	125.0	187.5	63.5	11 906.3	244 140.6	$7.56 \cdot 10^5$
5	–	143.0	9.0	1 287.0	8 100.0	$1.16 \cdot 10^4$	5
6	–	143.0	9.0	1 287.0	8 100.0	$1.16 \cdot 10^4$	6
7	51.5	0.1	51.5	0.5	25.8	4.3	7
	mm	mm^2	mm	mm^3	mm^4		

The value of the axial moment of inertia:

$$J = \sum \frac{b_i \cdot h_i^3}{12} + \sum A_i \cdot y_i^2 - A \cdot y^2 = 2.38 \cdot 10^6 \ (\text{mm}^4).$$

The value of the area of the section:

$$A = \sum A_i = 862.5 \, (\text{mm}^2).$$

The value of the distance from the center of mass of the section:

$$y = \frac{\sum A_i \cdot y_i}{\sum A_i} = 63.5 \, (\text{mm}).$$

Similarly, to the previous calculation, further calculations were made. The value of the maximum active bending stresses:

$$f_{b\,\max} = \frac{M_{\max} \cdot (y - t_2)}{J} = 240.59 \left(\frac{\text{N}}{\text{mm}^2} \right).$$

The value of the maximum active shear stresses:

$$f_{s\,\max} = \frac{Q_{\max} \cdot S^*}{J \cdot t_2} = 70.4\left(\frac{\text{N}}{\text{mm}^2}\right).$$

The value of the static moment of inertia of half the section:

$$S^* = \sum A_i \cdot y_i = 21761.2\,(\text{mm}^3).$$

The value of the allowable bending stresses:

$$F_{cr} = \frac{\pi^2 \cdot E}{12 \cdot (1-\mu^2)} \cdot k \cdot \left(\frac{t_2}{h}\right)^2 = 274.15\left(\frac{\text{N}}{\text{mm}^2}\right).$$

Comparison of the critical bending stresses with proportional limits.

$$F_{prop} = 330.72\left(\frac{\text{N}}{\text{mm}^2}\right).$$

Comparing:

$$F_{cr} < F_{prop}.$$

The value of the allowable bending stresses:

$$F_{s\,cr} = k_{ss} \cdot E \cdot \left(\frac{t_2}{d_c}\right)^2 \cdot \left(R_h + \frac{1}{2} \cdot (R_d - R_h) \cdot \left(\frac{d_s}{h_s}\right)^3\right) = 179.9\left(\frac{\text{N}}{\text{mm}^2}\right).$$

The value of the height of the web which strive to buckle:

$$h_c = h - b_2 - b_2 = 75\,(\text{mm}).$$

The value of the safety factor of the web under combined load:

$$MS = \frac{1}{\sqrt{\left(\frac{f_{b\,\max}}{F_{b\,cr}}\right)^2 + \left(\frac{f_{s\,\max}}{F_{s\,cr}}\right)^2}} - 1 = 0.042.$$

The next step is evaluation of the compressed and tensile chords strength under bending moment. The value of the maximum effective stresses of the compressed (upper) chord:

$$f_{ch} = \frac{M_{\max}}{J} \cdot (h - y) = 240.6\left(\frac{\text{N}}{\text{mm}^2}\right).$$

Scheme of loading is a simply supported beam subjected by the concentrated force. The length of this beam is the longest distance, namely from the inner seat track to the intercostal, which is located in the middle of the beam span.

$$L_c = 600\,(\text{mm}).$$

Moment of inertia of the column about Y axis is found by table method (Table 3.10).

Table 3.10

Determination of the moment of inertia of upright about y-y axis

№	b_i	h_i	A_i	y_i	$A_i{\cdot}y_i$	$b_i{\cdot}h_i^3/12$	$A_i{\cdot}y_i^2$
1	51.5	1.00	51.50	0.50	25.72	4.92	12.88
2	–		143.0	9.00	1 287.0	8 100.0	$1.16{\cdot}10^4$
3	–		143.0	9.00	1 287.0	8 100.0	$1.16{\cdot}10^4$
4	1.50	25.0	37.50	13.50	506.3	1 953.2	$6.83{\cdot}10^3$
	mm		mm^2	mm	mm^3	mm^4	

The value of the axial moment of inertia:

$$J = \sum \frac{b_i \cdot h_i^3}{12} + \sum A_i \cdot y_i^2 - A \cdot y^2 = 4.95 \cdot 10^4\ (\text{mm}^4).$$

The value of the area of the section:

$$A = \sum A_i = 375.0\,(\text{mm}^2).$$

The value of the distance from the center of mass of the section:

$$x = \frac{\sum A_i \cdot x_i}{\sum A_i} = 25.8\,(\text{mm}).$$

The value of the allowable critical stresses for the upper chord:

$$F_{ch\,cr} = \frac{\pi^2 \cdot E \cdot J}{\left(\frac{L}{\sqrt{c}}\right)^2 \cdot A} = 256.75\left(\frac{\text{N}}{\text{mm}^2}\right).$$

It is also necessary to find the critical stresses at the buckling of flange:

$$F_{cr\ f} = K_c \cdot E \cdot \left(\frac{t}{b}\right)^2 = 578.3\left(\frac{\text{N}}{\text{mm}^2}\right).$$

Comparing:

$$F_{ch\,cr} > f_{cr\,f}.$$

Elastic column buckling occurs much earlier than plate buckling. So, there is no necessarily to take into account plate buckling during allowable stresses calculation. The value of the safety factor of the calculation:

$$MS = \frac{F_{ch\,cr}}{f_{ch}} - 1 = 0.067.$$

The value of the maximum effective stresses of the tensile (lower) chord.

$$f_t = \frac{M_{\max}}{J} \cdot y = 244.4\left(\frac{\text{N}}{\text{mm}^2}\right).$$

The value of the safety factor of the tensile chord:

$$MS = \frac{F_{tu}^*}{f_t} - 1 = 0.823.$$

The value of the underestimated F_{tu}:

$$F_{tu}^* = F_{tu} \cdot 0.9 \cdot \frac{A_{net}}{A_{gross}} = 445.49\left(\frac{\text{N}}{\text{mm}^2}\right).$$

The value of the safety factor of the shear web strength.

$$MS = \frac{F_{su}}{f_{s\,\max}} - 1 = 3.598.$$

Moment of inertia of the upright about x-x axis is found using table method (Table 3.11).

Table 3.11

Determination of the moment of inertia of upright about x-x axis

№	b_i	h_i	A_i	y_i	$A_i \cdot y_i$	$b_i \cdot h_i^3/12$	$A_i \cdot y_i^2$
1	9.00	1.00	9.00	0.5	4.50	0.75	2.250
2	1.00	10.0	10.0	5.0	50.0	83.3	250.0
	mm		mm^2	mm	mm^3	mm^4	

The value of the moment of inertia of the upright:

$$J_u = \sum A_i \cdot y_i^2 + J_i = 336.3\,(\text{mm}^4).$$

The value of the minimum moment of inertia:

$$J_{u\,\min} = 0.65 \cdot h \cdot t_2^3 = 274.2\ (\text{mm}^4),$$

$$J_u > J_{u\,\min}.$$

Checking of thickness of the unloaded upright and strength of the loaded upright are not necessary for now as unloaded upright haven`t already answered to requirements of strength.

For checking of the local strength, it is necessary find net area moment inertia for which the subtract inertia moments of bolts about the X axis is required. Moments of inertia of bolts are determined using table method (Table 3.12).

Table 3.12

Geometrical properties of cross-section

№	b_i	h_i	A_i	y_i	$A_i{\cdot}y_i$	$b_i{\cdot}h_i^3/12$	$A_i{\cdot}y_i^2$
1	4.00	4.00	16.0	125.0	2 000.0	21.33	$2.5{\cdot}10^5$
	mm		mm^2	mm	mm^3	mm^4	

Assume that the number of bolts at the wall-frame connection is 6 pieces. So, the value of the axial moment of inertia:

$$J_r = \sum \frac{b_i \cdot h_i^3}{12} + \sum A_i \cdot y_i^2 - A \cdot y^2 = 21.3\ (\text{mm}^4).$$

The value of the area of the section:

$$A = \sum A_i = 96.0\ (\text{mm}^2).$$

The value of the distance from the center of mass of the section:

$$y = \frac{\sum A_i \cdot y_i}{\sum A_i} = 125.0\ (\text{mm}).$$

The value of net area moment of inertia of cross-section:

$$J_{net} = J - J_r \cdot 6 = 2.38 \cdot 10^6\ (\text{mm}^4).$$

The value of the net area bending stresses

$$f_{net} = \frac{M \cdot (y - y_r)}{J_{net}} = 434.57 \left(\frac{\text{N}}{\text{mm}^2}\right).$$

The value of the shear stresses of the web to the chordal fastener:

$$f_s = \frac{Q}{(h - 2 \cdot y_r) \cdot t_2} = 75.09 \left(\frac{\text{N}}{\text{mm}^2} \right).$$

The value of the net area shear stresses is calculated by formula:

$$f_{s\,net} = \left(\frac{p}{p - d} \right) \cdot \left(\frac{t}{t_p} \right) \cdot f_s = 100.12 \left(\frac{\text{N}}{\text{mm}^2} \right).$$

The value of the safety factor of the web to frame attachment:

$$MS = \frac{1}{\sqrt{\left(\frac{f_{b\,net}}{F_{tu}} \right)^2 + \left(\frac{f_{s\,net}}{F_{su}} \right)^2}} - 1 = 0.155.$$

The value of the safety factor of the web to upright attachment:

$$MS = \frac{F_{su}}{f_{s\,net}} - 1 = 2.234.$$

The value of the distance between the inner stiffener attachment to the chord and the next attachment that attaches the web to the stiffener:

$$s = \frac{(h - 2 \cdot y_r - 5 \cdot p)}{2} = 11.39\ (\text{mm}).$$

The value of the net area at attachment shear stresses:

$$f_{s\,net} = \left(\frac{s}{s - d_c} \right) \cdot \left(\frac{t}{t_p} \right) \cdot f_s = 115.75 \left(\frac{\text{N}}{\text{mm}^2} \right).$$

The value of the safety factor of the web to stiffeners attachment strength:

$$MS = \frac{1}{\sqrt{\left(\frac{f_{b\,net}}{F_{tu}} \right)^2 + \left(\frac{f_{s\,net}}{F_{su}} \right)^2}} - 1 = 0.131.$$

The value of the effective shear flow:

$$q = f_s \cdot t = 113.0 \left(\frac{\text{N}}{\text{mm}} \right).$$

The following is the determination of the allowable shear flow when the number of rows of fasteners is 1. The allowable shear flow is determined by the formula:

$$q_{all} = P_{all} \cdot \frac{N_f}{p} = 237.2 \left(\frac{\text{N}}{\text{mm}} \right).$$

The value of the safety factor of the web to chord attachment fitting strength:

$$MS = \frac{q_{all}}{q} - 1 = 1.106.$$

The next step is determination of total mass of the structure. Total mass of the structure is determined using table method (Table 3.13) and for gross area.

Table 3.13

Mass of the gross area structure

Name of the element	Area of the element	Length	Volume	Density	Mass (m_i)
Web	187.5	3 000	$5.63 \cdot 10^5$	$2.70 \cdot 10^{-5}$	15.19
Upper chord	286.0	3 000	$8.58 \cdot 10^5$	$2.70 \cdot 10^{-5}$	23.17
Lower chord	286.0	3 000	$8.58 \cdot 10^5$	$2.70 \cdot 10^{-5}$	23.17
Upper doubler	51.50	3 000	$1.55 \cdot 10^5$	$2.70 \cdot 10^{-5}$	4.172
Lower doubler	51.50	3 000	$1.55 \cdot 10^5$	$2.70 \cdot 10^{-5}$	4.172
Stiffener	19.00	2142.0	$4.07 \cdot 10^4$	$2.70 \cdot 10^{-5}$	1.099
	mm^2	mm	mm^3	kg/mm^3	kg

$$m_t = \sum m_i = 70.96 \text{ (kg)}.$$

However, in the case of this proposed beam section, the weight of the rivets must be taken into account. For this purpose, their approximate number is calculated:

$$n = \frac{(L_1 + L_2 + L_3 + L_4 + L_5 + L_5)}{p} \cdot 6 = 1125.$$

Accordingly, the weight of the rivets:

$$m_r = m_r \cdot n = 1.32 \text{ (kg)}.$$

For this calculation, the value of the largest displacement was also determined using Ansys software. For this purpose, were used "Static Structure" Analysis System where the geometry of the structure was set in "Space Claim" like rods and shells elements (Fig. 3.16).

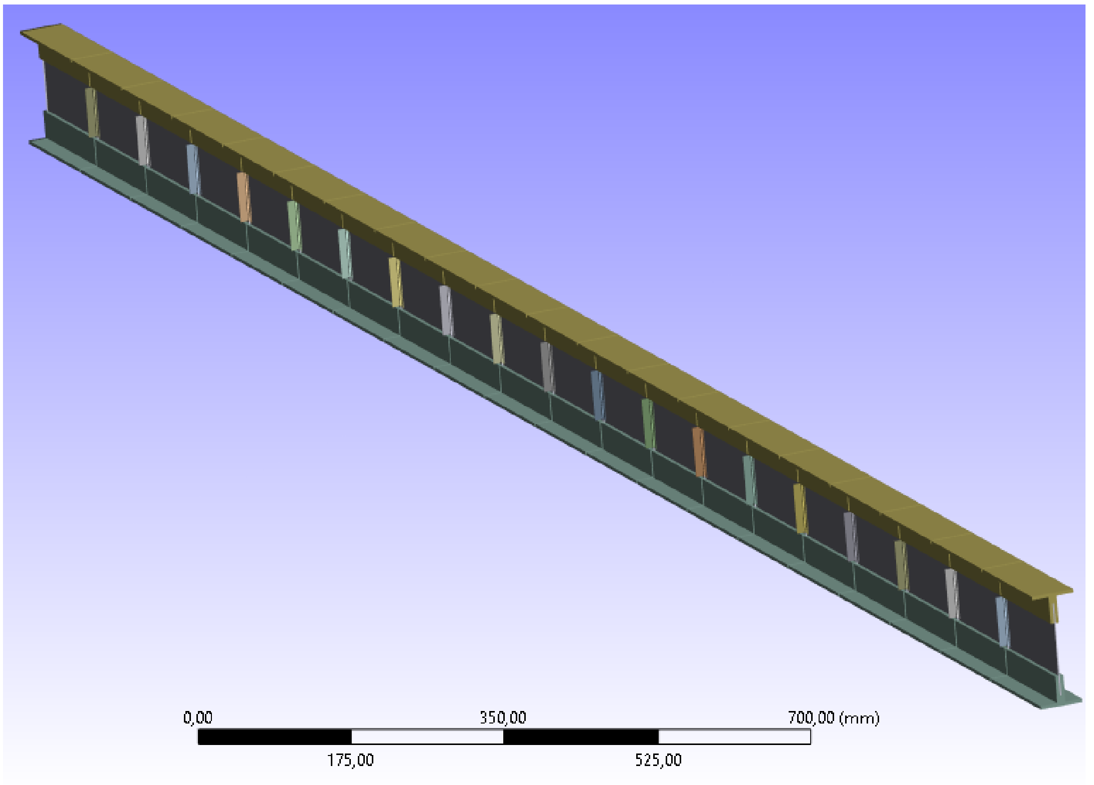


Fig. 3.16. Structure geometry.

Then, using the "Model", the mesh for the beam must be specified as shown in Fig. 3.17:

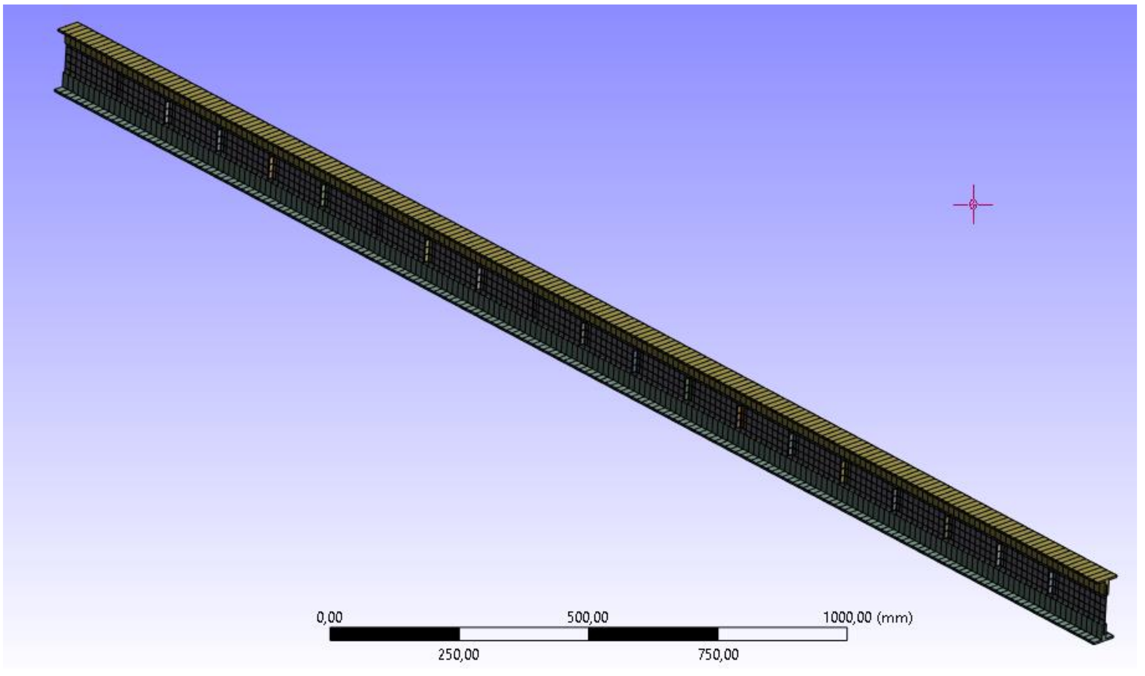


Fig. 3.17. Structure mesh.

The beam is loaded according to the design scheme shown in Fig. 3.3. The load is specified as "force" and the supports are specified as "remote displacement" as shown in Fig. 3. 18:

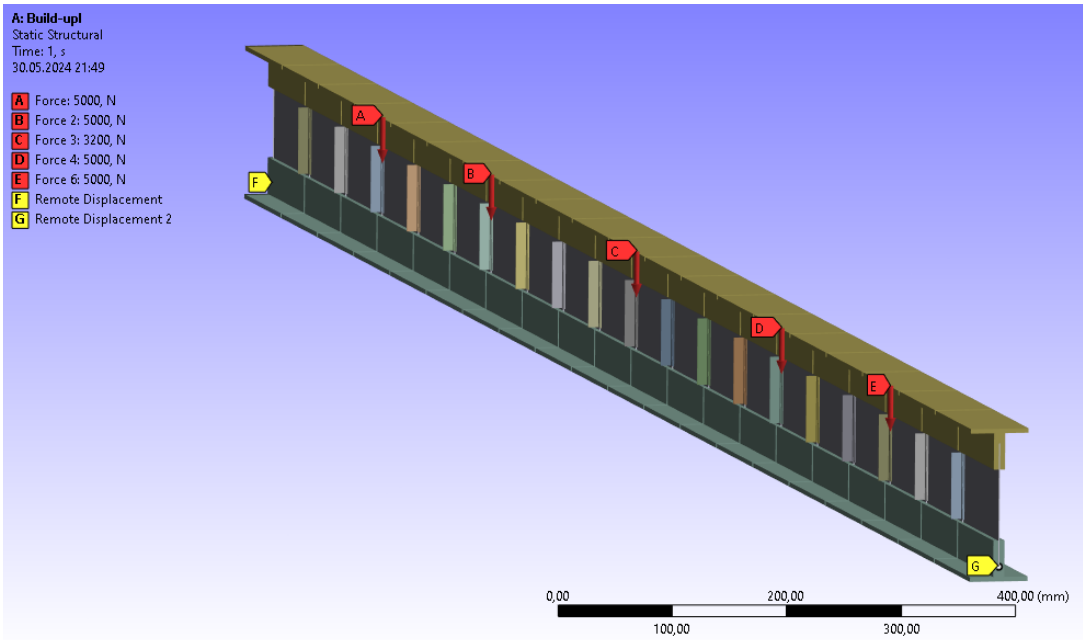


Fig. 3.18. Structural loading.

Then, by setting the Total deformations in the Solution, the result of the structure deflection was obtained as shown in fig. 3.19:

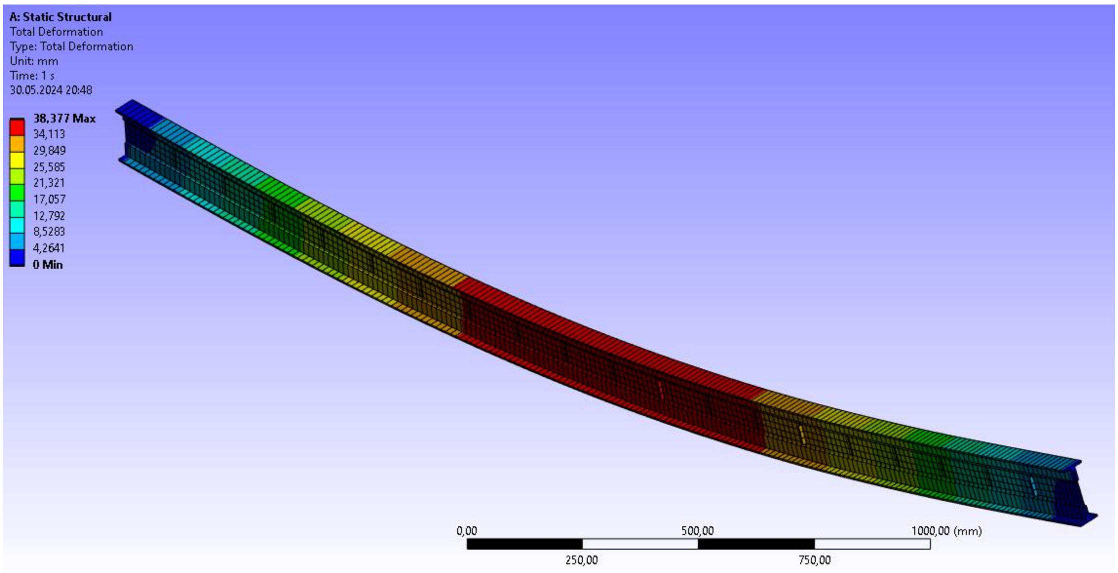


Fig. 3.19. The result of the calculation.

3.5. Comparison of the initial and proposed cross-section

All calculated safety factors are shown in Table 3.14 and Table 3.15.

Table 3.14

Safety factor of different elements of structure for different load cases

Load case	Initial cross-section	Proposed cross-section
Web		
Failure under combination of the bending and shear	0.328	0.042
Failure under ultimate shear stresses	8.719	3.598
Chords		
Rapture of the stretched chord	0.956	0.823
Column buckling of the compressed chord	0.085	0.067

Table 3.15

Safety factor of local strength

Failure	Initial cross-section	Proposed cross-section
Web to chord attachment		
Under combination of the bending and shear	0.167	0.155
Web to upright attachment		
Under ultimate shear stresses	4.571	2.234
Web to chord and upright attachment		
Under combination of the bending and shear	0.172	0.131
Fitting		
Under ultimate shear stresses of the fastener	7.163	1.106

Comparing the safety factor of the original and the proposed beam cross-section, it is clear that the composite cross-section has a smaller safety factor, which means that the structure makes better use of its strength reserves. It is also necessary to compare the values of the mass and the largest deflection of the structure as show in table 3.16:

Table 3.16

Cross-sectional efficiency

	Initial cross-section	Proposed cross-section
Total mass of the structure (kg)	71.63	72.28
Largest displacement of a structure (mm)	35.52	38.37

Conclusion to the special part

Comparing the repair capability of the aircraft, for example, in case of corrosion of the upper flanges in the lavatory and toilet areas, the proposed section(build-up) will be more effective, since in cases of severe corrosion it will be easier to replace the entire flange in this section rather than in the initial one (solid). Also, the proposed cross-section will have a better effect on the crack resistance of the beam, since when a crack occurs, it will not go to the other flange of the structure, as it would be in the case of the solid cross-section. In addition, with build-up cross-section, it is possible to manufacture elements from different materials to have better characteristics (an example is a spar in a wing, the flanges of which are made of different series of aluminum).

However, the beam with the proposed cross-section is 0.89% heavier. As a result, a variant was proposed that would be 0.65 kg heavier for one floor beam and approximately 50 kg heavier for the entire aircraft. This will make the entire aircraft heavier by less than 1%. Also, the beam with the proposed cross-section has a 7.4% greater deflection than the beam with the previous cross-section. However, even in this case, the deflection is only 1.3% of the length of the whole beam, so it is not very significant in this situation.

As a result, this cross-section can be chosen if the weight difference is less significant for the aircraft compared to the repair ability, crack resistance and etc.

GENERAL CONCLUTION

The purpose of this work was to optimize the passenger cabin floor beam. The main three stages of the work were:

1. Analyzing several prototypes to obtain data on the basis of which the design of the aircraft was started. As a result, the obtained values fully meet the requirements.

2. Calculating the geometric characteristics of all parts of the aircraft and the position of the center of mass. As a result, the obtained values fully meet the requirements.

3. Calculation and comparison of the initial and optimized floor beam structure. As a result, the obtained values do not fully meet the goal, since the improvement of some structural characteristics led to a slight decrease in other characteristics. However, in some cases, the improvements may play a greater role and the structure will be slightly more efficient.

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2. Bleich F. Buckling strength of metal structures. New York: McGraw-Hill, 1952.

3. National Advisor Committee for Aeronautics report. 2015.

4. Niu M. Airframe Stress Analysis & Sizing. Adaso Adastra Engineering Center, 1998.

Appendix

Appendix A

Performed by: Lanovenko Alina
Supervisor: Krasnopolskyi Volodymyr

PRELIMINARY DESIGN OF THE AIRCRAFT
INITIAL DATA AND SELECTED PARAMETERS

Passenger Number	190.
Flight Crew Number	2.
Flight Attendant or Load Master Number	5.
Mass of Operational Items	2 089.75 kg
Payload Mass	20 229.30 kg
Cruising Speed	825. km/h
Cruising Mach Number	0.7754
Design Altitude	11.50 km
Flight Range with Maximum Payload	6 400. km
Runway Length for the Base Aerodrome	3.30 km
Engine Number	2.
Thrust-to-weight Ratio in N/kg	3.5000
Pressure Ratio	35.00
Assumed Bypass Ratio	5.30
Optimal Bypass Ratio	5.50
Fuel-to-weight Ratio	0.2600
Aspect Ratio	10.50
Taper Ratio	3.40
Mean Thickness Ratio	0.120
Wing Sweepback at Quarter Chord	29.0 deg
High-lift Device Coefficient	1.050
Relative Area of Wing Extensions	0.050

Wing Airfoil Type - Supercritical
Winglets - Installed
Spoilers - Installed

Fuselage Diameter	3.55 m
Fineness Ratio	11.00
Horizontal Tail Sweep Angle	32.0 deg
Vertical Tail Sweep Angle	35.0 deg

CALCULATION RESULTS

Optimal Lift Coefficient in the Design Cruising Flight Point	0.49672
Induce Drag Coefficient	0.00910

ESTIMATION OF THE COEFFICIENT Dm = Mcritical - Mcruise

Cruising Mach Number	0.77540
Wave Drag Mach Number	0.79439
Calculated Parameter Dm	0.01898

Wing Loading in kPa (for Full Wing Area):

At Takeoff	5.173

At Middle of Cruising Flight	4.398
At the Beginning of Cruising Flight	4.973

Drag Coefficient of the Fuselage and Nacelles	0.00646
Drag Coefficient of the Wing and Tail Unit	0.00911

Drag Coefficient of the Airplane:

At the Beginning of Cruising Flight	0.02684
At Middle of Cruising Flight	0.02564
Mean Lift Coefficient for the Ceiling Flight	0.49672

Mean Lift-to-drag Ratio	19.36973
Landing Lift Coefficient	1.615
Landing Lift Coefficient (at Stall Speed)	2.422
Takeoff Lift Coefficient (at Stall Speed)	1.998
Lift-off Lift Coefficient	1.459
Thrust-to-weight Ratio at the Beginning of Cruising Flight	0.478
Start Thrust-to-weight Ratio for Cruising Flight	2.126
Start Thrust-to-weight Ratio for Safe Takeoff	2.425
Design Thrust-to-weight Ratio	2.522
Ratio Dr = Rcruise / Rtake-off	0.877

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

Takeoff	35.2544
Cruising Flight	56.2301
Mean cruising for Given Range	58.9319

FUEL WEIGHT FRACTIONS:

Fuel Reserve	0.02738
Block Fuel	0.24610

WEIGHT FRACTIONS FOR PRINCIPAL ITEMS:

Wing	0.13358
Horizontal Tail	0.01016
Vertical Tail	0.00993
Landing Gear	0.03943
Power Plant	0.08029
Fuselage	0.08096
Equipment and Flight Control	0.13034
Additional Equipment	0.00498
Operational Items	0.02218
Fuel	0.27348
Payload	0.21474

Airplane Takeoff Weight	94 203. kgf
Takeoff Thrust Required of the Engine	118.77 kN

Air Conditioning and Anti-icing Equipment Weight Fraction	0.0226
Passenger Equipment Weight Fraction (or Cargo Cabin Equipment)	0.0163
Interior Panels and Thermal/Acoustic Blanketing Weight Fraction	0.0059
Furnishing Equipment Weight Fraction	0.0148

Flight Control Weight Fraction	0.0059
Hydraulic System Weight Fraction	0.0165
Electrical Equipment Weight Fraction	0.0317
Radar Weight Fraction	0.0030
Navigation Equipment Weight Fraction	0.0045
Radio Communication Equipment Weight Fraction	0.0023
Instrument Equipment Weight Fraction	0.0053
Fuel System Weight Fraction	0.0081

Additional Equipment:

Equipment for Container Loading	0.0000
No typical Equipment Weight Fraction (Build-in Test Equipment for Fault Diagnosis, Additional Equipment of Passenger Cabin)	0.0050

TAKE-OFF DISTANCE PARAMETERS

Airplane Lift-off Speed	271.06	km/h
Acceleration during Takeoff Run	1.82	m/s*s
Airplane Take-off Run Distance	1 554.	m
Airborne Take-off Distance	578.	m
Take-off Distance	2 132.	m

CONTINUED TAKE-OFF DISTANCE PARAMETERS

Decision Speed	257.51	km/h
Mean Acceleration for Continued Take-off on Wet Runway	0.08	m/s*s
Take-off Run Distance for Continued Take-off on Wet Runway	4 759.44	m
Continued Take-off Distance	5 337.82	m
Runway Length Required for Rejected Take-off	5 537.12	m

LANDING DISTANCE PARAMETERS

Airplane Maximum Landing Weight	75 067.	kg
Time for Descent from Flight Level till Aerodrome Traffic Circuit Flight	22.7	min
Descent Distance	51.99	km
Approach Speed	247.30	km/h
Mean Vertical Speed	2.00	m/s
Airborne Landing Distance	516.	m
Landing Speed	232.30	km/h
Landing run distance	748.	m
Landing Distance	1 264.	m
Runway Length Required for Regular Aerodrome	2 111.	m
Runway Length Required for Alternate Aerodrome	1 795.	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

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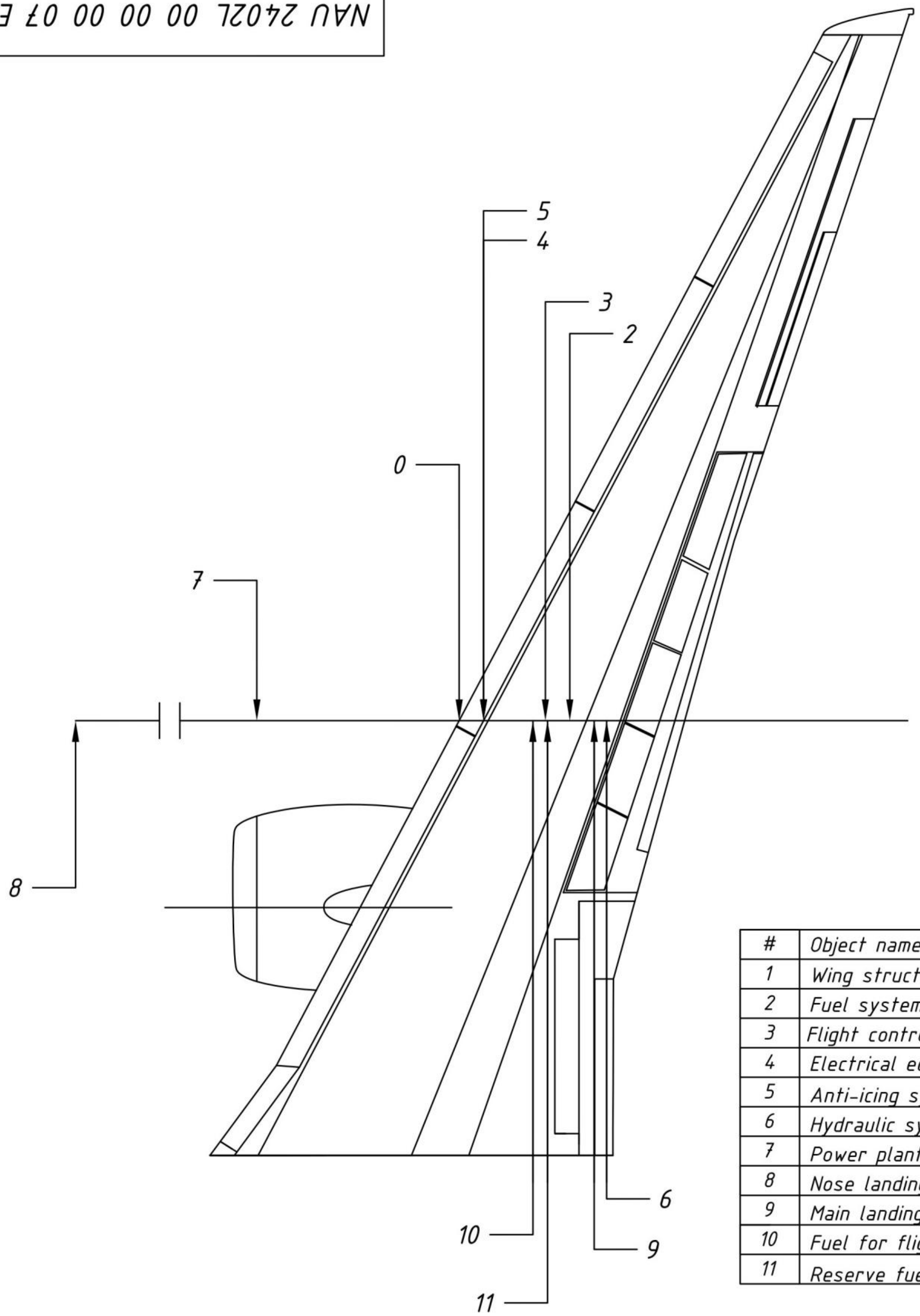

#	Object name
1	Wing structure
2	Fuel system
3	Flight control system, 30%
4	Electrical equipment, 20%
5	Anti-icing system, 70%
6	Hydraulic system, 30%
7	Power plant
8	Nose landing gear
9	Main landing gear
10	Fuel for flight
11	Reserve fuel

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Ch.	Sheet	№ docum.	Sign.	Ch.
Performed		Lanovenko Alina		
Supervisor		Krasnopolskyi V.		
Tech. control				
St. controler		Krasnopolskyi V.		
Approved		Yutskevich S.		

Center of gravity of wing

Letter	Mass	Scale
Q		1:100
Sheet 1	Sheets 1	

Appendix B

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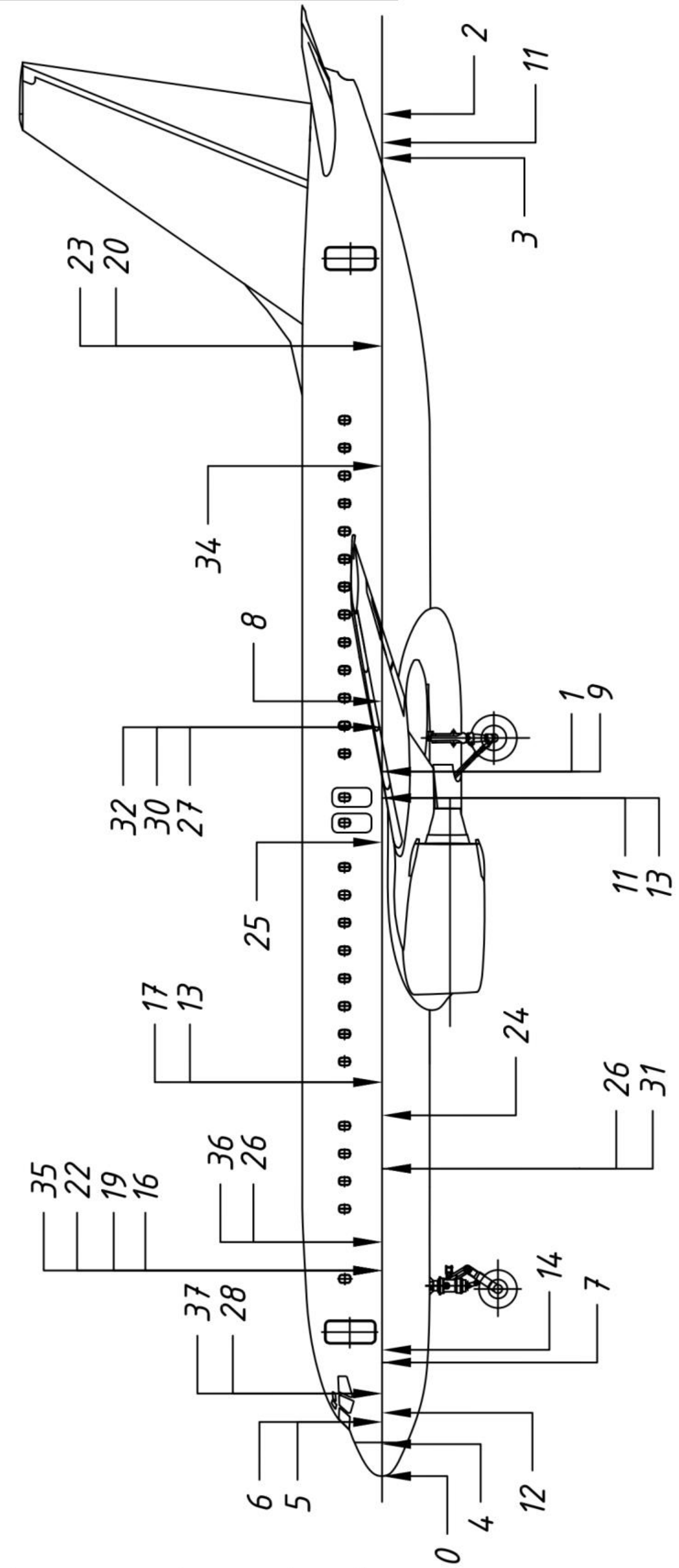

#	Object name
1	Fuselage
2	Horizontal tail unit
3	Vertical tail unit
4	Radiolocation equipment
5	Dashboard and instrument equipment
6	Aero navigation equipment
7	Radio equipment
8	Flight control system, 70%
9	Electrical equipment, 80%
10	Hydraulic system, 70%
11	Anti-icing system, 30%
12	Anti-conditioning system
13	Emergency equipment
14	Tools
15	Water and liquid
16	Lavatory 1
17	Lavatory 2
18	Lavatory 3
19	Wardrobe 1
20	Wardrobe 2
21	Wardrobe 3
22	Galley 1
23	Galley 2
24	Baggage equipment
25	Interior panels, lining and insulation
26	Passengers' seats 1 (business)
27	Passengers' seats 2 (economic)
28	Pilots' seats
29	Flight attendants' seats
30	Non-typical equipment
31	Passengers 1 (business class)
32	Passengers 2 (economic class)
33	Passengers' baggage
34	Cargo mail
35	On board meal
36	Flight attendants
37	Crew

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Ch.	Sheet	№ docum.	Sign.	Ch.
Performed		Lanovenko Alina		
Supervisor		Krasnopolskyi V.		
Tech. control				
St. controler		Krasnopolskyi V.		
Approved		Yutskevich S.		

Center of gravity of fuselage

Letter	Mass	Scale
Q		1:200
Sheet 1		Sheets 1

Appendix C

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