МІНІСТЕРСТВО ОСВІТИ ТА НАУКИ УКРАЇНИ

НАЦІОНАЛЬНИЙ АВІАЦІЙНИЙ УНІВЕРСИТЕТ

КАФЕДРА КОНСТРУКЦІЇ ЛІТАЛЬНИХ АПАРАТІВ

ДОПУСТИТИ ДО ЗАХИСТУ
Завідувач кафедри
д.т.н., професор.
___________ С.Р. Ігнатович
«___» __________ 2021 р.

ДИПЛОМНА РОБОТА

ВИПУСКНИКА ОСВІТНЬОГО СТУПЕНЯ БАКАЛАВРА

ЗІ СПЕЦІАЛЬНОСТІ
«АВІАЦІЙНА ТА РАКЕТНО-КОСМІЧНА ТЕХНІКА»

Тема: «Аванпроект ближньомагістрального літака вантажопідйомністю 16 тонн»

Виконавець: ___________ **С.В. Бевз**

Керівник: к.т.н., доцент ___________ **Т.П. Маслак**

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Київ 2021

MINISTRY OF EDUCATION AND SCIENCE OF UKRAINE
NATIONAL AVIATION UNIVERSITY
DEPARTMENT OF AIRCRAFT DESIGN

APPROVED BY
Head of department
D.Sc., professor
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«___» _________ 2021

BACHELOR THESIS

ON SPECIALITY
"AVIATION AND SPACE ROCKET TECHNOLOGY"

Topic: «Preliminary design of a cargo regional jet with a payload capacity of 16 tons»

Prepared by: __________ **S.V. Bevz**

Supervisor: PhD, associate professor __________ **T.P. Maslak**

Standard controller: PhD, associate professor __________ **S.V. Khyzhnyak**

Kyiv 2021

НАЦІОНАЛЬНИЙ АВІАЦІЙНИЙ УНІВЕРСИТЕТ

Факультет аерокосмічний

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Освітній ступінь «Бакалавр»

Спеціальність 134 «Авіаційна та ракетно-космічна техніка»

Освітня програма «Обладнання повітряних суден»

ДОПУСТИТИ ДО ЗАХИСТУ

Завідувач кафедри

д.т.н., професор.

_______________ С.Р. Ігнатович

«___» ___________ 2021 р.

ЗАВДАННЯ

на виконання дипломної роботи студента

1. Тема роботи: «Аванпроект ближньомагістрального літака вантажопідйомністю 16 тонн», затверджена наказом ректора від 21 травня 2021 року №815/ст.

2. Термін виконання проекту: з 24 травня 2021 р. по 20 червня 2021 р.

3. Вихідні дані до проекту: крейсерська швидкість V_{cr}=800 км/год, дальність польоту L =1300 км, крейсерська висота польоту H_{op}=11 км, 16 тонн.

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 вантажної кабіни, розрахунок центрування літака, спеціальна частина, яка містить проектування конструкції закрилка.

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

6. Календарний план-графік

№ пор.	Завдання	Термін виконання	Відмітка про виконання
1	Отримання завдання, обробка статистичних даних.	24.05.2021-30.05.2021	
2	Розрахунок мас літака та його основних льотно-технічних характеристик.	31.05.2021-02.06.2021	
3	Розрахунок центрування літака.	03.06.2021-04.06.2021	
4	Розробка креслень по основній частині.	05.06.2021-06.06.2021	
5	Проектування конструкції закрилка.	07.06.2021-11.06.2021	
6	Оформлення пояснювальної записки.	12.06.2021-13.06.2021	
7	Захист дипломної роботи.	14.06.2021-20.06.2021	

7. Дата видачі завдання: «24» травня 2021 року.

Керівник дипломної роботи _____________ Т.П. Маслак

Завдання прийняв до виконання _____________ С.В. Бевз

NATIONAL AVIATION UNIVERSITY

Faculty Aerospace
Department of Aircraft Design
Educational degree «Bachelor»
Specialty 134 «Aviation and space rocket technology»
Educational program «Aircraft Equipment»

APPROVED BY
Head of department
D.Sc., professor
___________ S.R. Ignatovych
«___» __________ 2021

TASK
for the bachelor thesis
BEVZ SERHII

1. Topic: «Preliminary design of a cargo regional jet with a payload capacity of 16 tons» confirmed by Rector's order № 815/ст from 21.05.21
2. Thesis term: from 24.05.2021 to 20.06.2021
3. Initial data: maximum payload 16 tonns; flight range with maximum payload 1300 km; cruise speed 800 km/h at operating altitude 11 km.
4. Content (list of topics to be developed): selection of design parameters; choice and substantiations of the airplane scheme; calculation of aircraft masses; determination of basic geometrical parameters; aircraft layout; center of gravity position calculation; determination of basic flight performance; description of the aircraft design; engine selection; conceptual design of the flaps.
5. Required materials:
   – general view of the airplane (A1×1);
   – layout of the airplane (A1×1);
   – assembly drawing of the flaps (A1×1).

   Graphical materials are performed in AutoCad.

6. Thesis schedule

№	Task	Time limits	Done
1	Task receiving, processing of statistical data.	24.05.2021-30.05.2021	
2	Aircraft take-off mass determination and flight performances calculation.	31.05.2021-02.06.2021	
3	Aircraft centering determination.	03.06.2021-04.06.2021	
4	Graphical design of the aircraft and its layout.	05.06.2021-06.06.2021	
5	Design flaps. Drawings of the special part.	07.06.2021-11.06.2021	
6	Completion of the explanation note.	12.06.2021-13.06.2021	
7	Preliminary examination and defense of the diploma work.	14.06.2021-20.06.2021	

Date: 24 May 2021 year.

Supervisor ____________ T.P. Maslak

Student ____________ S.V. Bevz

РЕФЕРАТ

Дипломна робота **«Аванпроект ближньомагістрального літака вантажопідйомністю 16 тонн»** містить:

сторінок, рисунків, таблиць, літературних посилань, креслень

Об'єкт проектування: ближньомагістральний вантажний літак з максимальним комерційним навантаженням 16 тонн.

Предмет проектування: концепція конструкції внутрішніх закрилків, розрахунок на міцність вузлів навіки закрилків крила.

Мета роботи: аванпроект ближньомагістрального вантажного літака відповідно до прототипів, планування вантажної кабіни та проектування механізації заднької кромки крила.

Методи дослідження: аналіз прототипів і вибір найбільш досконалих технічних рішень, оцінка геометричних характеристик, розрахунок центру мас літака, розрахунок на міцність конструкції закрилків, проектне креслення закрилків.

Наукова новизна результатів полягає у визначенні геометрії конструктивних елементів закрилків для забезпечення їх міцності під дією експлуатаційних навантажень.

Практична цінність роботи: визначається розширенням лінійки ближньомагістральних літаків підвищеної вантажопідйомності. Результати роботи можуть бути використані в авіаційній галузі та в навчальному процесі авіаційних спеціальностей.

АВАНПРОЕКТ ЛІТАКА, КОМПОНУВАННЯ ВАНТАЖНОЇ КАБІНИ, ЦЕНТРУВАННЯ ЛІТАКА, МЕХАНІЗАЦІЯ КРИЛА, ЗАКРИЛОК

ABSTRACT

Bachelor thesis «**Preliminary design of a cargo regional jet with a payload capacity of 16 tons**»

sheets, figures, tables, references and drawings

Object of study – design is a short-range aircraft with 16 tons of payload.

Subject of study – the conceptual design of the flaps.

Aim of bachelor thesis – preliminary design of a short-range cargo aircraft, according to the prototypes, the layout of the cargo compartment, and the conceptual design of the flaps.

Research and development methods – the design method is the analysis of prototypes and selection of the most advanced technical solutions, evaluation of geometric characteristics, the calculation of the center of gravity of the aircraft, strength analysis of the structure elements of the wing flaps.

Novelty of the results – practical implementation is determined by the expansion of the line of short-range cargo aircraft, conceptual design of the flaps structure.

Practical value – the results of the work can be implemented in the aviation industry and in the educational process of aviation specialties.

PRELIMINARY DESIGN, CARGO CABIN LAYOUT, CENTER OF GRAVITY POSITION, HIGH LIFT DEVICES, FLAPS

Format	№	Designation	Name	Quantity	Notes
			<u>General documents</u>		
A4	1	NAU 21 02 B 00 00 00 22 TW	Task for work	1	
	2	NAU 21 02 B 00 00 00 22	Short-range cargo aircraft	2	
A1		Sheet 1	General view		
A1		Sheet 2	Fuselage layout		
A4	3	NAU 21 02 B 00 00 00 22 EN	Short-range cargo aircraft	66	
			Explanatory note		
			<u>Documentation for assembly units</u>		
A3	4	NAU 21 02 B 00 00 00 22	Assembly drawing of flaps	1	

NAU 21 02 00 00 00 22 EN

Done by	Bevz S.V.	Preliminary design of a cargo regional jet with a payload capacity of 16 tons (List of diploma work)	list	sheet	sheets
Checked by	Maslak T.P				
St.control.	Khyzhniak S.V.		402 AF 134		
Head of dep.	Ignatovich S.R.				

INTRODUCTION

A transport cargo aircraft is an aircraft designed to carry cargo of various types, dimensions and purposes. The major feature of a cargo aircraft is the availability of a massive cargo compartment, special cargo doors and hatches, a hardened floor, and cargo equipment that allows cargo to be loaded on the aircraft. Nowadays, delivering certain types of cargo via cargo aircraft is by far the fastest way to transport it. When it comes to delivery speed, cargo planes are many times faster than trains and trucks. It is necessary to design a type of cargo aircraft that will have a high level of reliability and flight regularity, which will be in high demand in the global market.

One of the major tasks of aircraft design at the preliminary stage is to consider all modern general requirements, like flight safety; low fuel consumption; low noise level; low exhaust emissions; and reliability of operation. There is also a list of specific requirements for cargo aircraft: maximum payload, ease of loading and unloading, rational cargo cabin area, possibility of takeoff and landing on different runways, use of aircraft in different weather situations, ease of maintenance, the most advantageous ratio between price and effectiveness.

The aim of the thesis is to make a preliminary design of a short-haul cargo aircraft with a maximum payload of 16 tons. The aircraft to be designed is also based on these requirements in order to meet the requirements for a cargo aircraft. The basic prototypes for the projected aircraft are An-148, An-178, CRJ-900. The main flight and technical characteristics of the aircraft are cruising speed of 800 km/h, range of 1300 km, operating altitude of 11 km.

CONTENT

1. PRELIMINARY DESIGN OF A SHORT RANGE CARGO AIRCRAFT

1.1 The analysis of prototypes and choice of design parameters

The aim of the thesis is to complete a preliminary design, following by the initial data of prototypes and market demands. The first stage of the work is the choice of data from the analysis of the prototypes shown in table 1.1. The prototypes for the aircraft to be designed are short-haul aircraft: An-148-100A, An-178 and CRJ-900. In accordance with the thesis task, the designed aircraft must be a short-haul, transport cargo aircraft with a cargo capacity of up to 16 tons.

Flight performance, statistics and geometric parameters of prototypes are given in tables 1.1 and 1.2.

Table 1.1 - Performance and technical data of the prototypes

Parameter	Aircraft		
	AN 148-100A	CRJ-900	AN-178
Crew/flight attend. persons	2/2	2/2	2/2
Maximum payload, kg	9000	10319	18000
Cruise speed, km/h	820	848	830
Flight altitude, m	11000	12500	12200
Maximum range, km	1920	2955	5500
Thrust to weight ratio, N/kg	3,25	3,19	3,66
Take-off distance, m	1580	1779	2500
Landing distance, m	1600	1596	2300
Take-off weight, kg	38550	36514	51000
Type of engine	2ТРДД	2ТРДД	2ТРДД
Take of thrust кN, кVт	62,7	58,4	77,8
Pressure ratio	21	23,09	21
Bypass ratio	4,8	5,13	5,6

Table 1.2 - Main geometrical parameters of prototypes

Parameter	Aircraft		
	AN 148-100A	CRJ-900	AN-178
Fuselage shape	Circular	Circular	Circular
Fuselage diameter	3,35	2,69	3,55
Fineness ratio of fuselage	7,82	11,8	8,38
Sweep back angle 1/4 chord	25	24	25
Wing aspect ratio	9,5	9	10

The following design step is the first estimation of weight, optimisation of wing loading and selection of engines. Having completed the first iteration, it is possible to start preliminary design of the wing, fuselage, tail and landing gear layout, determine the required wing area to support the payload, tail area to ensure aircraft stability, and calculate the position of the aircraft's centre of gravity.

Based on the prototype configuration, the presented aircraft is a high-plane with a T-shaped tail section, two turbojet engines mounted on pylons under the wings, and tricycle landing gear retractable into the fuselage. Such a design ensures high efficiency of lift devices; easy trim of the aircraft in flight; tail section location, reduces destabilizing moment; areas of vertical and horizontal stabilizers are reduced. Initial data of the designed aircraft are given in Appendix A, obtained by a special computer program at the Department of Aircraft Design

1.2. Aircraft geometry calculation

1.2.1. Wing geometry

The geometrical characteristics of the wing are determined based on the take-off weight m_0 and specific load on the wing P_o. These data were received by the computer program, special designed at the aircraft design department.

First, the wing area can be estimated:

$$S_w = \frac{m_0 \times g}{P_0} \frac{53122 \cdot 9.8}{4223} = 87.32 m^2 \ .$$

Taking the aspect ratio and wing area, a wing span is calculated:

$$l = \sqrt{S_w \cdot \lambda_w} = 28{,}92 \text{ m},$$

where λ_w – aspect ratio of the wing.

Root chord of the wing can be calculated by the following formula:

$$b_0 = \frac{2S_w \times \eta_w}{(1 + \eta_w) \times l} = 4{,}84m,$$

where η – taper ratio.

Tip chord:

$$b_t = \frac{b_0}{\eta_w} = 1{,}19m$$

The relative position of the spars along the chord of a wing with two spars are $\overline{X}_1 = 0{,}2$; $\overline{X}_2 = 0{,}6$.

The position of the spars in the root of a wing:

$$X_1 = \overline{X_1} \times b_0 = 0{,}2 \times 4{,}84 = 0{,}968\, m$$
$$X_2 = \overline{X_2} \times b_0 = 0{,}6 \times 4{,}777 = 2{,}904\, m.$$

The position of the spars in the tip of a wing:

$$X_1 = \overline{X_1} \times b_K = 0{,}2 \times 1{,}19 = 0{,}238\, m$$
$$X_2 = \overline{X_2} \times b_K = 0{,}6 \times 1{,}18 = 0{,}714\, m$$

The mean aerodynamic chord of the wing (MAC) is determined by the geometrical method (Fig. 1.1.).

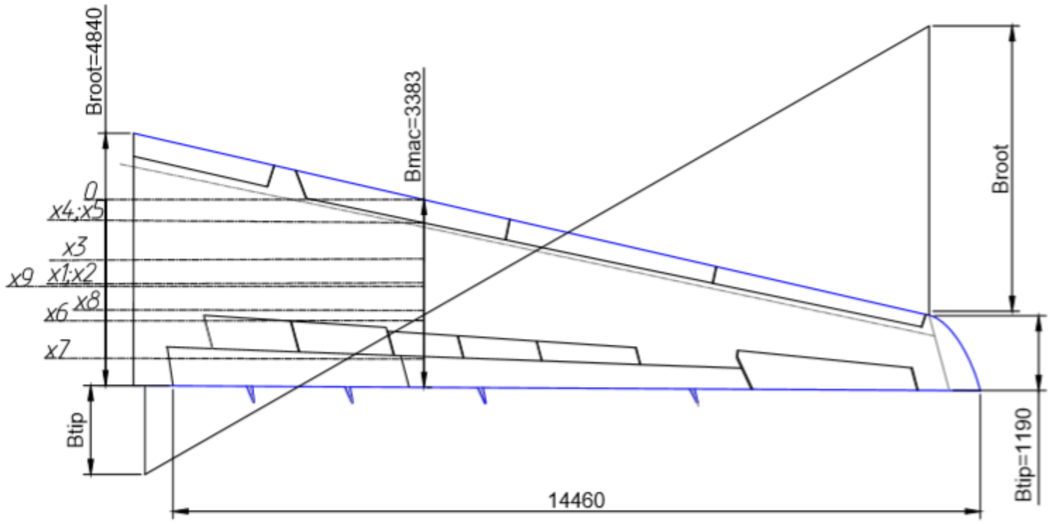

Fig. 1.1 - Mean aerodynamic chord of the wing

The received value of mean aerodynamic chord is b_{mac} = 3,341 m.

The thickness of airfoil is also taken from the experience of the companies, which develop the wing for the prototypes, $\bar{c}_i$=0.118.

As a result, the airfoil thickness in the root and in the tip parts is:

$$c_{root} = \bar{c}_{root} \times b_0 = 0{,}118 \times 4{,}84 = 0{,}571 \text{ m.}$$

$$c_{tip} = \bar{c}_{tip} \times b_0 = 0{,}118 \times 1{,}19 = 0{,}14 \text{ m.}$$

Double slotted flaps, spoilers and ailerons without aerodynamic balance are used as extendable devices on the wing.

The principal task of high-lift devices is to provide extra lift during takeoff and landing. The area of the slats can be calculated by the next equations:

$$S_{slat} = 0{,}1 \times S_{wing}$$

$$S_{slat} = 0{,}1 \times 87{,}42 = 8{,}742m$$

The area of the flaps can be determined by the next formula:

$$S_{flaps} = 0{,}17 \times S_{wing}$$

$$S_{flaps} = 0{,}17 \times 87{,}42 = 14{,}86m$$

The trailing edge of the wing tip is filled with ailerons.

The aileron's span is:

$$l_{ail} = 0{,}375 \times \frac{l_w}{2} = 0{,}375 \times \frac{28{,}92}{2} = 5{,}72m;$$

The aileron area is:

$$S_{ail} = 0{,}065 \times \frac{S_w}{2} = 0{,}065 \times \frac{87{,}32}{2} = 2{,}83m^2$$

The range of the aileron deflection is equal to upward deflection $\delta_{ail} \geq 20^\circ$ and downwards $\delta_{ail} \geq 10^\circ$.

1.2.2 Fuselage layout

Choosing the shape and dimensions of the fuselage cross-section, it is necessary to proceed from the requirements of aerodynamics. For transport aircraft, the speed of which is less than the speed of sound (V <800 km/h), the shock wave drag has almost no effect. Therefore, the shape should be selected to provide the lowest values of the corresponding friction resistance C_{xf} and form drag C_{xp} are ensured.

For transonic aircraft, the nose part of the fuselage should be found:

$$L_{ns}=(2\ldots3)\times D_f,$$

where D_f - fuselage diameter.

In addition to the requirements of aerodynamics during the choosing of a cross-section of the fuselage, it is necessary to take into account the layout conditions (especially for cargo aircraft) and strength requirements.

To provide the minimum weight, the most appropriate shape for the cross-section of the fuselage must be recognized as a circular cross-section. In this case, the thickness of the fuselage skin will be minimum.

The geometric parameters of the fuselage include:

- the diameter of the fuselage D_f;
- the length of the fuselage $L_{f;}$
- fuselage fineness ratio:

$$\lambda_f = \frac{D_f}{L_f}$$

- fineness ratio of the nose of the fuselage:

$$\lambda_{ns} = \frac{l_{ns}}{D_f};$$

- fineness ratio of the tail part of the fuselage:

$$\lambda_{tp} = \frac{l_{tp}}{D_f}$$

where l_{ns} and l_{tp} - respectively, the lengthes of the nose section and tail part of the fuselage.

The length of the fuselage is determined by the aircraft layout, features layout and centering, as well as the landing angle of attack $\alpha_{lan.}$

Determine the following fuselage parameters:

$$l_f = \lambda_f \times D_f = 8 \times 3{,}35 = 26{,}8m;$$

$$l_{ns} = \lambda_{ns} \times D_f = 3{,}1 \times 3{,}35 = 10{,}3m;$$

$$l_{tp} = \lambda_{tp} \times D_f = 2{,}1 \times 3{,}35 = 7{,}035m;$$

It is also necessary to make the minimum mid-section of the fuselage. For cargo aircraft, the mid-section of the fuselage is determined by the dimensions of the cargo cabin.

1.2.3 Crew cabin

The length of the cockpit depends on the number and relative position of the crew's workplaces. The average cockpit length is 2300 ... 3300 mm. The cockpit is separated from other areas of the fuselage by the wall, equipped with a door with a lock.

When designing the cockpit, it should be noticed that pilot must be able to clearly see that part of the runways that allows him to control the flight path and prevent any collisions of any part of the aircraft with other aircraft or obstacles. In practice, this is ensured by the minimum angles of view during cruise flight, takeoff run, run and taxiing.

For designing aircraft, the forward view during cruise flight from the pilot's estimated eye position should be 17° down and 20° up. Therefore, to ensure the correct inspection, areas are determined in which objects should be absent, for example, the cockpit canopy racks, which limit the field of view, as well as areas in which the size of these items should be limited. For the same purpose, the windows of the cockpit canopy are made in a special form.

1.2.4 Landing gear design

The retractable tricycle landing gear is selected. This is the most frequently used landing gear type for modern aircraft. During takeoff, the wheels take all the weight of the airplane, and it also has better lateral stability on the ground.

At the initial design stage, when center of gravity has not been performed and there are no general arrangement drawings of the aircraft, only a part of the landing gear geometrical parameters can be determined.

The distance from the centre of gravity to the main landing gear strut:

$$e = 0{,}2673 \times b_{MAC} = 0{,}2673 \times 3{,}38 = 0{,}903m.$$

The landing gear base is found by the expression:

$$B = 0{,}4526 \times L_f = 0{,}4526 \times 26{,}8 = 12{,}129m.$$

The nose landing gear can carry 6-10% of the aircraft's total weight, so the nose landing can be estimated by the following formula:

$$d_{ng} = B - e = 11{,}226 \text{ m}.$$

Wheel track is one of the main paremeters of the designing aircraft, it is calculated using the formula:

$$T = 0{,}6072 \times B = 0{,}6072 \times 12{,}129 = 7{,}36m$$

The tires of the wheels of the landing gear are selected from take-off weight of the aircraft, the length of the runways needed for take off and landing, take off spped and landing speed. For the selecting of the nose landing gear tires, dynamic loads are taken into account also. The type of tires and the pressure in them are determined by the surface of the runway for which the aircraft is intended to operate. Brake wheels are installed on the main wheels, and shimmy depher is on the nose landing gear support. Wheel loads are determined:

For the main landing gear wheels load is equal:

$$P_{MLG} = \frac{(9{,}81 \times (B - e) \times m_0)}{B \times n \times z} = \frac{(9{,}81 \times (12{,}129 - 0{,}903) \times 53122)}{(12{,}129 \times 2 \times 2)} = 219240{,}5N$$

For the nose landing gear the wheel load is equal:

$$P_{NLG} = \frac{(9{,}81 \times e \times k_g \times m_0)}{B \times z} = \frac{(9{,}81 \times 0{,}903 \times 1{,}8 \times 53122)}{(12{,}129 \times 2)} = 71924{,}008N$$

K_g =1,5...2,0 - dynamic load factor.

The dimensions of tires for designing aircraft are shown in the table 1.3.

Table 1.3 - Tires characteristcs

Main gear		Nose gear	
Tire size	Ply rating	Tire size	Ply rating
1244x431	32	990x330	16

The airplane tires can withstand high pressure, near 13 bar (200 psi, 13 atm).

1.2.5 Tail unit parameters

One of the most important tasks of the aerodynamic layout is the choice of the location of the horizontal tail. To provide the longitudinal static stability of the aircraft in terms of overload, its centre of mass must be in front of the focus of the aircraft and the distance between these points, referred to the value of the mean aerodynamic chord (MAC) of the wing, and determines the moment of longitudinal stability:

$$m_x^{Cy} = \overline{X}_T - \overline{X}_F < 0$$

where m_x^{Cy} - moment coefficient, $\overline{X}_T$ and $\overline{X}_F$ respectively, the relative coordinate of the centre of gravity and focus. If $m_x^{Cy} = 0$, then the aircraft has neutral longitudinal static stability.

Based on the analysis of statistical data of prototype aircrafts, we have:

$$S_{HTU} = \frac{b_{MAC} \times S_w}{L_{HTU}} \times A_{HTU} = \frac{3{,}38 \times 87{,}32}{9{,}46} \times 0{,}55 = 18{,}87 m^2;$$

$$S_{VTU} = \frac{l_w \times S_w}{L_{VTU}} \times A_{VTU} = \frac{28{,}92 \times 87{,}32}{9{,}46} \times 0{,}09 = 18{,}87 m^2$$

where L_{HTU}, L_{VTU} – horizontal and vertical tail;

l and S – span and wing area;

A_{HTU}, A_{VTU} – coefficients of static moments.

Values L_{HTU}, L_{VTU} depend on a number of factors. First of all, their value is influenced by the length of the nose and tail parts of the fuselage; sweep angle and wing location, as well as conditions to provide stability and controllability of the aircraft. At the first iteration, we can assume that $L_{HTU} \approx L_{VTU}$.

The elevator area is usually taken:

$$S_{el} = 0{,}2765 \times S_{HTU} = 5{,}21 \text{ m}^2.$$

The rudder area is usually taken:

$$S_{rud} = 0{,}2337 \times S_{VTU} = 4{,}94 \text{ m}^2.$$

Determination of the span of the horizontal tail.

The wing span and span of tail unit are connected by statistic data

$$L_{HTU} = (0{,}32 \ldots 0{,}5) \times l_w = 0{,}323 \times 28{,}95 = 9.3 \text{ m}.$$

Vertical tail height h_{VTU} is determined by the placement of the wings due to the fuselage. Taper ratio of the horizontal and vertical tail should be selected from the range: η_{HTU} = 2…3 and η_{VTU} = 1…1,33. Based on the analysis of statistical data of prototype aircrafts we take η_{HTU} = 2,51 and η_{VTU} = 1,367.

The apect ratio for subsonic aircrafts: λ_{VTU} = 0,8…1,5 and λ_{HTU} = 3,5...4,5. Based on the analysis of statistical data of prototype aircrafts we take λ_{VTU} = 0,95 and λ_{HTU} = 4,5. Determination of the b_{tip}, b_{root} performs according to the formulas:

For horizontal tail unit:

$$b_{root} = \frac{b_{tip}}{\eta_{HTU}} = \frac{2,89}{2,51} = 1,15m$$

$$b_{tip} = \frac{2 \times S_{HTU} \times \eta_{HTU}}{(1 + \eta_{HTU}) \times l_{HTU}} = \frac{2 \times 18,87 \times 2,51}{(1 + 2,51) \times 9,32} = 2,89m$$

For vertical tail unit:

$$b_{root} = \frac{b_{tip}}{\eta_{VTU}} = \frac{5,32}{1,367} = 3,89m$$

$$b_{tip} = \frac{2 \times S_{VTU} \times \eta_{VTU}}{(1 + \eta_{VTU}) \times l_{VTU}} = \frac{2 \times 21,15 \times 1,367}{(1 + 1,367) \times 4,59} = 5,32m$$

Relative airfoil thickness: for horizontal or vertical tail as a first approximation $\overline{C}_t \approx 0,8 \cdot \overline{C}_w$. If we attach a horizontal stabilizer to the fin, we assume $\overline{C}_t$ = (0,08…0,1). For the designing aircraft we take $\overline{C}_t$ = 0,1.

The sweep back angle of the tail is taken by *3...5°* more than the wing sweep. This is done to provide aircraft controll when the wave drag appears on the wing.

1.2.6 Engine selection for the designing aircraft

Taking into account the ratio of thrust to weight of the prototypes, the value 3.2 is chosen for the designed aircraft. Based on the required thrust and the engine parameters selected earlier in the initial data, we select the bypass turbojet engine D436T1, the characteristics of which are given in Table 1.4.

Table 1.4 – Characteristics of D436T1 engine

№	Engine data	Units of measurement	Value
1	Type of engine	-	turbojet
2	Take-off power	kN	73,57

3	Power in cruise flight mode	kN	14,71
4	Specific fuel consumption	kg/N×hour	0,062
5	Engine pressure ratio	-	24
6	Dry engine mass	kg	1450
7	Bypass ratio	-	4,95

1.3 Aicraft centre of gravity calculation

At the pre-design stage, after the first calculation of geometric parameters, after choosing the schemes of aircraft components, the following task is to determine the weight of the main elements of the aircraft on the basis of statistical data. The weight of the airplane is divided into the weight of an empty airplane and the weight of a fully loaded airplane. It is common practice to mix aircraft equipment, different components and systems.

General aircraft layout requirements are as follows: each component (cargo) of the aircraft should be located so that it is most useful; the layout of the aircraft should allow control and maintenance of systems and units, as well as allow repair and mounting of parts and units; the layout should provide ease of overall assembly of the aircraft structure; the structural layout should provide minimum weight with maximum strength and durability.

1.3.1 Centre of gravity of equipped wing

The weight of the equipped wing includes the weight of its structure, weight of the equipment and the weight of the fuel. The beggining of the coordinates of the center of gravity is selected in the projection of the MAC to the plane.

Table 1.5 - Treem sheet of equipped wing

№	Name	Mass m_i		C.G. coordinates x_i, m	Moment m_i x_i, kgm
		Units	Total mass m_i, kg		
1.	Wing (structure)	0,11993	6370,92	1,4196	9044,16
2.	Fuel system 40%	0,00152	80,75	1,4196	114,62
3.	Control system (30%)	0,00219	116,33	1,014	117,96
4.	Electrical equipment (10%)	0,00208	110,49	0,338	37,34
5.	Anti-icing system (70%)	0,01197	635,87	0,338	214,92
6.	Hydraulic system (70%)	0,01358	721,39	2,028	1462,99

7.	Power units	0,08915	4735,82	2,65	12549,93
8.	Equipped wing without fuel and LG	0,24042	12771,59	1,8433	23541,95
9.	Fuel	0,13088	6952,60	1,4534	10104,91
	Equipped wing	0,3713	19724,19	1,7058	33646,87

The coordinate of the center of gravity of the equipped wing is determined by the formula:

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

1.3.2 Centre of gravity of equipped fuselage

All coordinates of the fuselage masses (table 1.6) are selected in the projection of the fuselage on the horizontal axis with the origin in the nose part of the fuselage. The horizontal line of the fuselage is taken as the X-axis.

$$x_f = \frac{\sum m_i^{/} \ x_i^{/}}{\sum m_i^{/}}$$

After the centre of gravity calculations of the fuselage masses, we need to find a point at wich we can attach the wing. It is necessary to calculate *Xmac:*

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

where x_{MAC} - position of the front part of the MAC relative to the nose of the fuselage.

The value x_{MAC} can be calculated by next formula:

$$x_{MAC} = \frac{m_f x_f + m_w \times x_w^{/} - m_0 C}{m_0 - m_w} = 10{,}9.$$

where: m_f - mass of equipped fuselage,

m_w - mass of equipped wing.

c – distance from the front of MAC to the center of mass of the aircraft.

$$c = (0{,}28 \dots 0{,}32) \times b_{MAC}$$

The values of center of masses of equipped fuselage shown in table.

Table 1.6 - Centre of gravity of equipped fuselage

№	Object name	Weight m_i		Coordinate of C.G., m	Mass moment, kgm
		Units	Total, kg		
	Airframe				
1.	Fuselage	0,12476	6627,5	13,41	88881,41
2.	Horizontal Tail Unit	0,0174	928,04	0,91	844,51
3.	Vertical Tail Unit	0,02031	1078,9	2,07	2233,33
	Equipment				
4.	Anti-Ice System, (15%)	0,00256	135,99	21,45	2917,03
5.	Air-cond. system(15%)	0,00256	135,99	13,41	1823,79
6.	Thermal and sound isolation	0,0062	329,35	13,41	1823,79
7.	Control system, (70%)	0,00511	271,45	13,41	3640,46
8.	Hydraulic system, (30%)	0,0058	308,1	18,77	5783,17
9.	Electrical equip.(90%)	0,0187	993,38	13,41	13322,23
10.	Location equipment	0,0059	313,41	1,081	338,8
11.	Navigation equipment	0,0088	467,4736	1,88	878,85
12.	Radio communication equipment	0,0044	233,73	1,081	252,66
13.	Dashboard	0,0103	547,15	2,58	1411,66
14.	Seats of crew	0,00056	29,96	2,78	83,29
15.	Seats for ac.person	0,00037	19,97	3,6	71,9
16.	Cabin equipment	0,0006	31,87	6,26	199,52
17.	Cargo equipment	0,02845	1511,32	23,59	35655,08
18.	Atypical equipment	0,0037	196,55	896	1761,1
19.	Service equipment	0,01594	846,76	6,99	5918,88
20.	Nose landing gear	0,00297	157,77	2,78	438,6
21.	Main landing gear	0,04153	2206,15	12,4	27356,34
	Equipped fuselage with no payload	0,26286	13963,94	11,65	162754,77
22.	Cargo	0,295	15670,99	13,6	213125,46
23.	Crew	0,00564	299,6	2,78	832,91
	Total	0,999	53122	18,25	608589,72

1.3.3 Centre of gravity for different types of loading

At this step of the aircraft preliminary design we have already the location of the wing in relation to the fuselage of the aircraft on the drawing we have designed. We have already calculated the centers of gravity of the fuselage and the wing. Therefore, we need to calculate the centers of gravity for various types of aircraft loading and flight modes. The center of gravity of an airplane is the relative position of the CG in relation to the MAC, and this ratio is shown as a percentage. The center of gravity calculations of the masses are shown in Tables 1.7 and 1.8.

Table 1.7 - Summary list of centers of gravity

№	Object name	Weight m_i, kg	Coordinates, m	Static moment of mass, kgm
1.	Equipped wing (without fuel and LG)	12771,59	13,84	176867,3
2.	Removed nose LG	157,77	1,78	280,83
3.	Removed main LG	2206,15	12,4	27356,34
4.	Fuel	6952,6	12,37	86003,75
5.	Equipped fuselage	13963,64	11,65	162754,77
6.	Cargo	15670,99	13,6	213125,46
7.	Crew	299,6	2,78	832,91
8.	Opened nose LG	157,77	2,78	438,6
9.	Opened main LG	2206,15	12,4	27356,34

Table 1.8 Center of gravity for different aircraft modes

№	Loading options	Weight, kg	Static moment of mass, kgm	Center of the mass, m	Centering, %
1.	Takeoff weight (opened LG)	53122	664159,15	12,76	22,53
2.	Takeoff weight (removed LG)	53122	664121,38	12,76	22,51
3.	Landing option (opened LG)	48422,2	603382,24	12,46	15,48

4.	Transportation option (no payload)	36351,4	454095,91	12,49	16,39
5.	Parking option (no fuel and payload)	29099,16	367417,02	12,62	18,37

Conclusion to the part

In this part, a preliminary design of a short-haul cargo aircraft with a capacity of up to 16 tons was developed. The main parts and elements of the designed airplane were considered.

So, as a result of this part of the bachelor thesis we have a preliminary design of the aircraft with a wingspan of 28.92 m, the estimated fuselage length of 26.8 m with a fuselage diameter of 3.35 m. The D436T1 engine was chosen to provide thrust in all flight modes because of its low fuel consumption, low weight to thrust ratio and low emissions and noise. Center of gravity positions were also calculated. The most forward center of gravity position is at 15.48% of the leading edge of the mean aerodynamic chord, and the most rearward center of gravity position is at 22.53% for takeoff mode. Through these calculations, we can confirm that all the masses of the airplane are in equilibrium.

2. CONCEPTUAL DESIGN OF FLAPS

2.1 Brief description of the flaps design

The flaps are the main high lift devices of modern aircraft, which correspond to the structural, aerodynamic and strength requirements.

Experience of the high lift device operation indicates the necessity of double flaps implementation for this class of aircraft. The inner wing flap is double flap retractable with a fixed position of the deflector relative to the main link. The flap profile is formed by the tip part of the wing airfoil.

The flap is used to change the aerodynamic shape of the wing, to encrese the camber of airfoil at take off and landing flight modes. The increment of lift force at flap deflection is provided by changing not only the wing profile, it also increas the area, which ensures the stall safety to large angles of attack.

Structurally, the flap has the same structure as the wing. A new trands of the composite materials implementation for the structure of falps are also embedded for designing aircraft – in the leading edge and trailitng adge of flaps.

The part between spars (more loaded part) of the flaps are made from aluminum alloys.

The spars are longitudinal structural elements in the form of i-beams riveted by the caps to the skin structure. The transverse structure elements are made up of the flap ribs, which are spaced at 450 mm. The ribs are thin-walled panels, which are divided by the webs of the spars into three parts: the nose, tail and middle part.

At the lposition of the flap hinge assemblies, reinforced ribs are installed to take the loads and transmit them to the spars.

The nose part of flaps is made in the form of a separate unit with a composite materials. Attachment of the nose part to the middle part is carried out by screws.

The middle part of the main flap is made of D16AT alloy cladded by pure alluminum, it connected with the longitudinal and transverse sets by riveted joints.

The tail part of the flap is structurally similar to the nose and made of composite material. The retraction and extension of the high lift devices are hydraulically operated by the flap control box using two screw lifters.

2.2 Strength analysis of the flaps. Diagrams of Q, M_{bend}, M_{tors}

The internal flaps sections are located between the engine pylons and the fuselage, they are shown in fig. 2.1.

It is necessary to:

- determine the design operationg loads and its distribution over the flaps;
- find the total force of the control mechanisms, the total reaction in the hinges;
- determine the forces Q, M_{bend}, M_{tors} and build their diagrams.

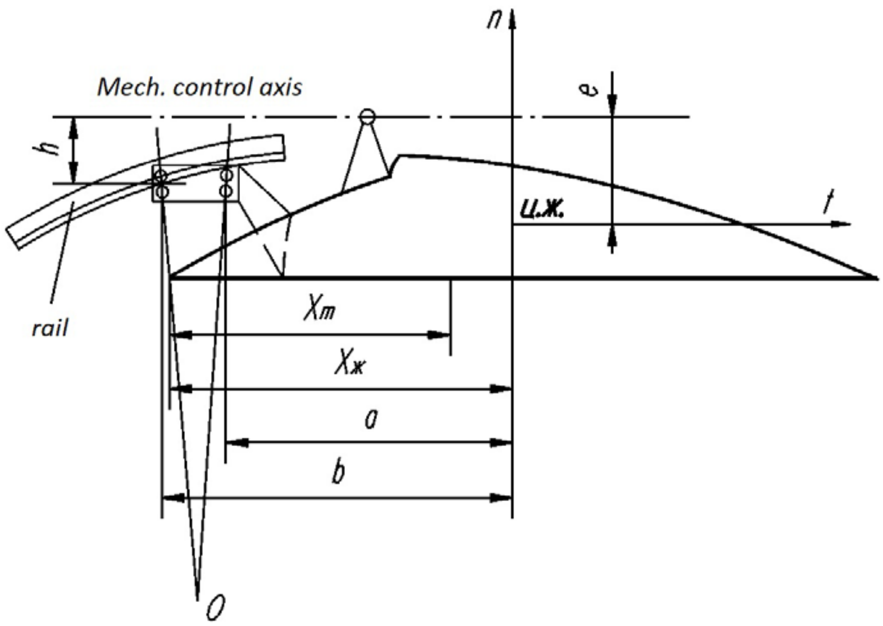

Fig.2.1- Flap scheme

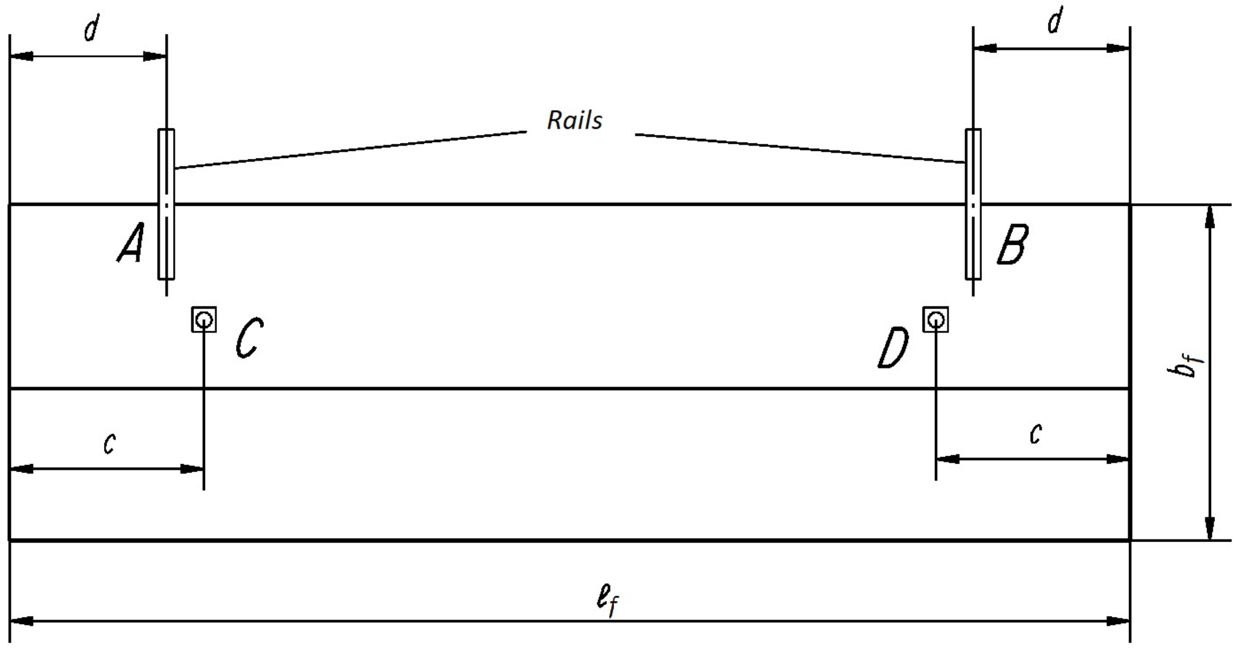

Fig.2.1 - Flap rail diagram

Initial data:

- minimal flight speed $V_{min} = 226\frac{km}{h}$;

- normal force coefficient $c_n = 4{,}5$;

- relative center of pressure $\overline{x_T} = 0{,}42$;

- safety factor $f = 2$;

- flap span $l_f = 5{,}72\ m$;

- flap chord $b_f = 1{,}2m$.

Dimensions: $d = 1{,}1m$; $c = 1{,}3m$;- $\overline{x_{ж}} = 0{,}6m$; $a = 0{,}62m$; $b = 0{,}47m$; $e = 0{,}3m$; $h = 0{,}12m$.

1. Determination of the P_f^p. For this, first it is necessary to find the value of q_f^E and S_f:

$$q_f^E = 3{,}56\frac{\rho_0 \times V_{min}^2}{2} = 3{,}56\frac{0{,}125}{2}\left(\frac{202}{3{,}6}\right)^2 = 700{,}5\frac{kg}{m^2}$$

$$S_f = b_f \times l_f = 1{,}2 \times 5{,}72 = 6{,}86m^2$$

Then

$$P_f^p = f \times c_n \times S_f \times q_f^E = 2 \times 4{,}5 \times 6{,}86 \times 700{,}5 = 43248\, kg$$

This force is perpendicular to the flap chord and is applied from its nose at a distance of x_p:

$$x_p = \overline{x_p} \times b_f = 0{,}42 \times 1{,}2 = 0{,}5m$$

Determination of the distributed load of the flap.

The linear air load on the flap is proportional to the chords:

$$q_f^p = \frac{P_f^p}{S_f} b_f$$

Since the chords are constant along the span, then

$$q_f^p = \frac{P_f^p}{l_f} = \frac{43248}{5{,}72} = 7561\, kg/m^2$$

The distribution of the specific air load along the chord could be found from the given equations:

- the chord load (Fig.2.2) is:

$$q_f^p = \frac{P_{1f} + P_{2f}}{2} \times b_f$$

- the coordinate of its attachment is:

$$x_p = \frac{b_f}{3} \times \frac{P_{1f} + 2P_{2f}}{P_{1f} + P_{2f}} \text{ , where } \overline{x_p} = \frac{b_f}{3} \times \frac{P_{1f} + 2P_{2f}}{P_{1f} + P_{2f}}$$

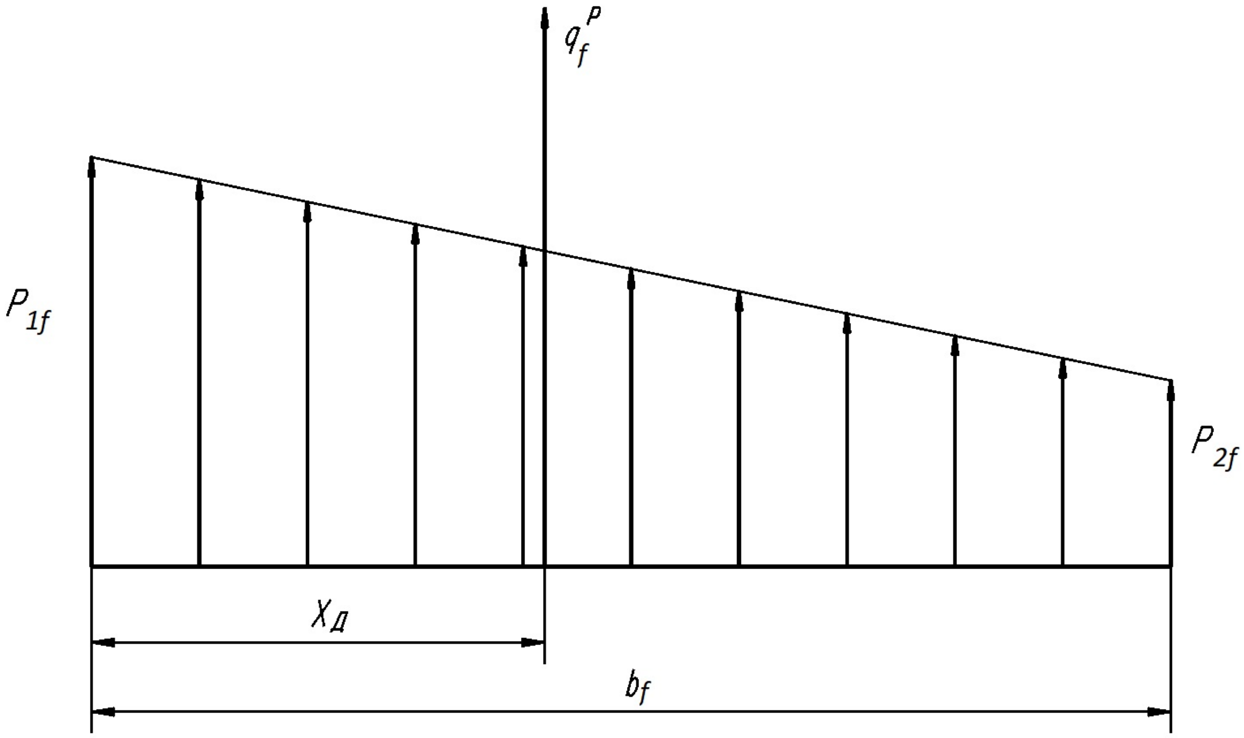

Fig.2.2 - Load distribution over the flap span

Solving these two equations, we determine the values of P:

$$p_{1f} = \frac{2q_f^p}{b_f} \times \left(2 - 3 \times \overline{x_Д}\right) = \frac{2 \times 7561}{1{,}2}(2 - 3 \times 0{,}42) = 9325\ kg/m^2$$

$$p_{2f} = \frac{2q_f^p}{b_f} \times \left(3 \times \overline{x_Д} - 1\right) = \frac{2 \times 7561}{1{,}2}(3 \times 0{,}42 - 1) = 3276\ kg/m^2$$

Correctness of the definition checked by the condition of equality of the two expressions of the average specific load

$$\frac{p_{1f} + p_{2f}}{2} = f \times c_n \times q_f^E$$

$$\frac{9325 + 3276}{2} = 2 \times 4{,}5 \times 700{,}5$$

The condition is satisfied: 6302 = 6302

2. Determination of the total force T in the control mechanisms and the total reaction $\sum R$ of the carriages.

On the flap, which is in equilibrium, there are three forces P_f^p, T and $\sum R$, which lie in the plane normal to the plane of the flap. The magnitude and direction of the load P_f^p is known. The direction of force T coincides with the axis direction of the control mechanism. The total reaction of the carriages $\sum R$ passes through point O, since the carriage rollers are arranged on an arc of a circle with center O.

To determine the magnitude of the unknown forces T, $\sum R$ and $\sum R$ direction, use the three-force theorem: If a solid body is in equilibrium under the action of three forces, the lines of action of these forces intersect at one point, and the triangle of forces is closed.

Draw the side projection of the flap (fig. 2.3) and draw the lines of action of forces P_f^p and T. The lines of action of these forces cross at point ε. The total reaction $\sum R$ of the carriages must also pass through this point.

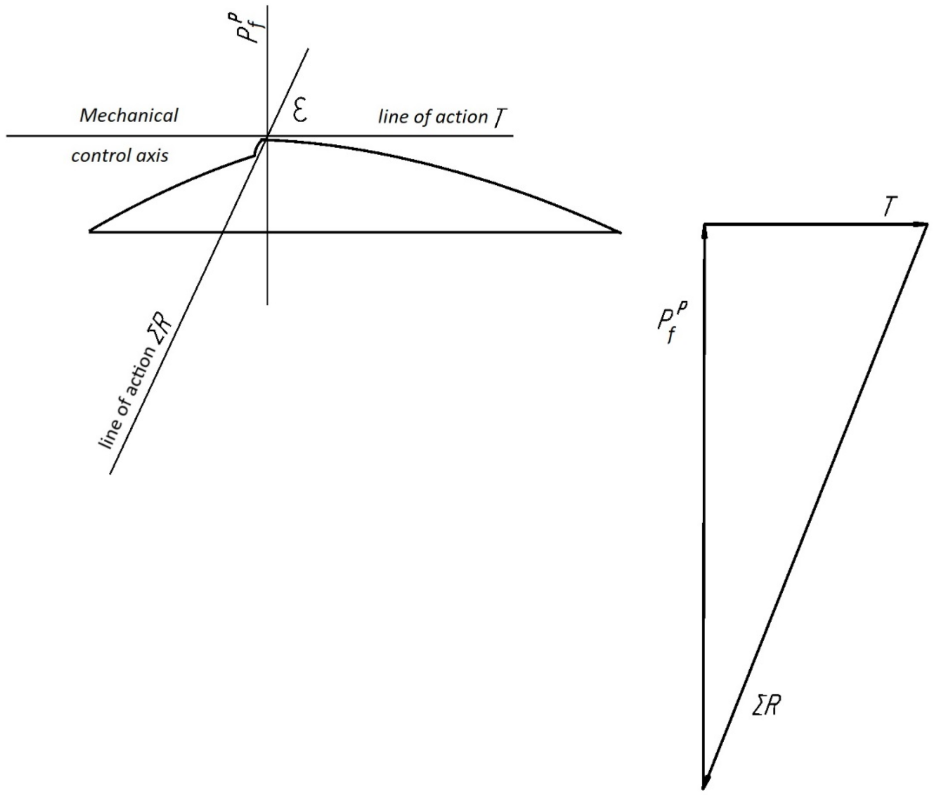

Fig.2.3 - Schematic diagram of the lines along which the loads act

Knowing the magnitude and direction of one force P_f^p and the lines of action of the other two forces, we construct a force triangle of equilibrium. As a result of the construction we obtain:

$$T = 17034\ kg;\ \sum R = 46480\ kg.$$

3. Determination of the rollers' reactions. The reaction of each carriage due to the symmetry of the structure and the load is $0{,}5 \times \sum R$.

The total reaction of each carriage is the resultant reaction of the rollers, whose lines of action pass through the axes of the rollers and point O, which is the center of the arc of the guideway. Decompose the value $0{,}5 \times \sum R$ by the indicated directions and obtain:

- total reaction of the front support rollers of each carriage $R_1 = 31446\ kg$
- total reaction of the rear support rollers of each carriage $R_2 = 52535\ kg$

Determination of forces Q, M_{bend}, M_{tors} and construction of epiures by the flap span.

Fig.2.1. shows axes t and n - the main central axes of the flap section. The t axis is parallel to the chord and lies in the plane of maximum rigidity, the n axis is parallel to this plane.

The forces acting on the flap are distributed along these axes.

The components of the reactions of the front rollers along the t and n axes are equal:

$$R_{1tA} = R_{1tB} = 6079\ kg$$
$$R_{1nA} = R_{1nB} = 30853\ kg$$

Taking into account the symmetry of the flap design and load, the calculation of forces Q, M_{bend}, M_{tors} will be given at the specific points only to the middle of the flap.

We limit ourselves to the consideration of Q, M_{bend} from the action of forces parallel to the axis n.

Force Q:

To the left of base A:

$$Q_A = q_f^p \times d = 7561 \times 1{,}1 = 8317\ kg$$

To the right of base A:

$$Q'_A = Q_A + R_{1nA} - R_{2nA} = 8317 + 30853 - 52535 = -13364\ kg$$

In the cross section in the middle of the flap:

$$Q = Q'_A + q_f^p \left(\frac{l}{2} - d\right) = -13364 + 7561 \left(\frac{5{,}72}{2} - 1{,}1\right) = 0$$

Force M_{bend}:

To the left of base A:

$$M_{bend} = q_f^p \times \frac{d^2}{2} = 7561 \times \frac{1{,}1^2}{2} = 4574\ kg \times m$$

To the right of base A:

$$M'_{bend} = I_{b\grave{e}nd} = 4574\ kg \times m$$

In the cross section in the middle part of the flap:

$$M_{bend} = q_f^p = \frac{l_f}{2}\frac{l_f}{4} - (R_{2nA} - R_{1nA}) \times \left(\frac{l_f}{2} - d\right) =$$

$$= 7561 \times \frac{5{,}72}{2} \times \frac{5{,}72}{4} - (52535 - 30853) \times \left(\frac{5{,}72}{2} - 1{,}1\right) =$$

$$= -7237\ kg \times m$$

Force M_{tors} :

To the left of base A:

$$M_{tors\,A} = q_f \times d \times (x_ж - x_д) = 7561 \times 1{,}1 \times 0{,}1 = 832\;kg \times m$$

To the right of base A:

$$M'_{tors\,A} = M_{tors\,A} + R_{1nA} \times b - R_{2nA} \times a - R_{1tA}(e - h) =$$
$$= 832 + 30853 \times 0{,}62 - 52535 \times 0{,}47 - 6079 \times 0{,}18$$
$$= -5825\;kg \times m$$

To the left of base C:

$$M_{tors\,C} = M'_{tors\,A} + q_f^p (c - d) \times (x_ж - x_д) = -5825 + 7561 \times 0{,}2 \times 0{,}1 =$$
$$= -5674 kg \times m$$

To the right of base A:

$$M'_{tors\,C} = M_{tors\,C} + \frac{T}{2} \times e = -5674 + \frac{17034}{2} \times 0{,}3 = -3118\;kg \times m$$

In the cross section in the middle of the flap:

$$M_{tors} = M'_{tors\,C} + q_f^p \times \left(\frac{l_f}{2} - c\right) \times (x_ж - x_д) =$$
$$= -3118 + 7561\left(\frac{5{,}72}{2} - 1{,}3\right) \times 0{,}1 = -1939\;kg \times m$$

Analyzing the character of changes in the forces Q, M_{bend}, M_{tors} on the flap span and taking into account the found values of the forces, we construct the diagrams of Q^p, $\grave{M}_{\text{bend}}$ and M_{tors}^p (Fig.2.4).

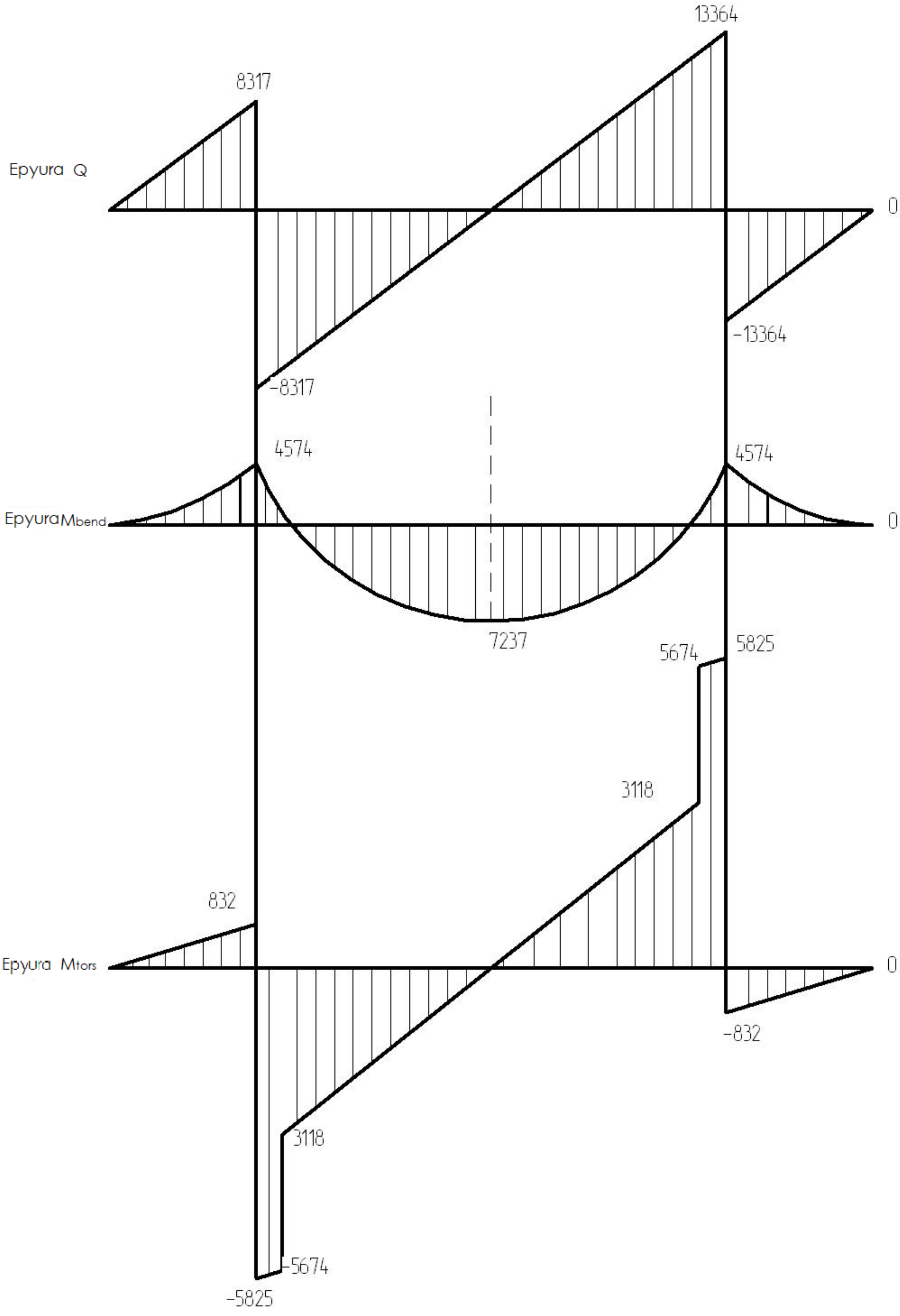

Fig. 2.4 – Diagrams of shear forces, bending and torsional moments along the flap span

2.3 Determination of the flaps structural element

Proceeding from the force diagrams acting in the section of the flap main link (fig.2.5), we have that the maximum M_{bend} acts in the middle of the spars length, and the maximum transverse force Q and M_{tors} act in the section along the bearings A and B. In the first approximation, we use the method of the combined section, that is, we will assume that all the maximum loads act in the section where M_{bend} is maximum.

Figure 2.5 shows the cross section of the flap, which is formed by the nose, interspar and tail parts of the main part of the flap.

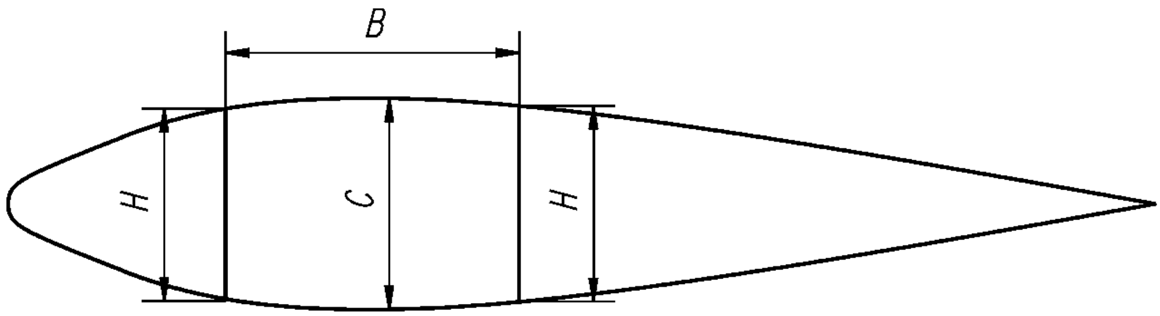

Fig.2.5 - Flap cross-section

Let's assume that all the loads acting on the main link will be taken by the inter-spar part, which is formed by the 1st and 2nd spars and the power cladding, upper and lower. Thus, the 1st and 2nd spars together with the upper and lower cladding form a beam of trapezoidal cross-section type, which works for transverse bending and torsion.

As a first approximation, we will assume that the transverse bending is taken by the spar chords together with the cladding attached to the chords. Normal tensile σ and compression stresses arise from the action of M_{bend} in the ribs and adjacent skin.

So we will distribute the M_{bend} between the spars and the cladding in the ratio:

- 0.7 M_{bend} for the spars
- on the cladding 0.3 M_{bend}

The calculated loading forces have the following values:

- $Q = 13364\ kg$;
- $M_{bend} = 7237\ kg \times m$;
- $M_{tors} = 5825\ kg \times m$.

The material of all power elements of the section is D16T.

$E = 7 \times 10^5\ kg/cm^2$, $\sigma = 4400\ kg/cm^2$. Spar height $H_1 = 8{,}4\ cm$ and $H_2 = 7{,}6\ cm$, spacing between spars $B = 18\ cm$.

1. Determination of the dimensions of the top panel elements.

a) Reduction area of the top panel

$$F_{red.up} = \frac{M\grave{b}end}{\acute{H} \times \sigma}$$

The working height H of the section of the monoblock flap with two spars is determined by the formula:

$$H = 0{,}95 \frac{\sum H_{\text{габ.i.}} + \text{c}}{m + 1} = 0{,}95 \frac{8{,}5 + 7{,}5 + 9{,}5}{2 + 1} = 8{,}1\ cm$$

Then

$$F_{red.up.} = \frac{7237 \times 100}{H \times 2400} = 37{,}43\ cm^2$$

b) Total area of the upper chords of the spars

$$F_{up.} = k \times F_{red.up.} = 0{,}7 \times 37{,}23 = 25{,}88\ cm^2$$

The area of the caps is distributed between the spars in proportion to their height:

$$F_{up.1} = \frac{H_1^2 \times F_{up.}}{H_1^2 + H_2^2} = \frac{8{,}5^2 \times 25{,}88}{8{,}5^2 + 7{,}5^2} = 14{,}55\ cm^2$$

$$F_{up.2} = \frac{H_2^2 \times F_{up.}}{H_1^2 + H_2^2} = \frac{7,5^2 \times 25,88}{8,5^2 + 7,5^2} = 11,33\ cm^2$$

c) Thickness of the compressed skin.

The reduced thickness of the upper skin

$$\delta_{up.} = \frac{f_{red.up.}}{\mathrm{B}} = \frac{11,55}{18} = 0,642\ cm$$

Where

$$f_{red.up.} = F_{red.up.} - F_{up.} = 37,05 - 25,88 = 11,55\ cm^2$$

Thickness of the top skin

$$\delta_{sk.up.} = 0,6 \times \delta_{up.} = 0,6 \times 0,642 = 0,39\ cm$$

2. Dimensioning the elements of the bottom panel

a) Reduction area of the bottom panel

$$F_{red.bot.} = \frac{M_{bend}}{H \times k \times \sigma} = \frac{7237 \times 100}{8,1 \times 0,9 \times 2400} = 41\ cm^2$$

The panel attenuation coefficient is taken $k = 0,9$.

b) The total area of the bottom chords of the spars and the area of each spar are equal:

$$F_{bot.} = k \times F_{red.bot.} = 0,7 \times 41 = 28,7\ cm^2$$

The area of the caps is distributed between the spars in proportion to their height:

$$F_{bot.1} = \frac{H_1^2 \times F_{bot.}}{H_1^2 + H_2^2} = \frac{8{,}5^2 \times 28{,}7}{8{,}5^2 + 7{,}5^2} = 16{,}14\ cm^2$$

$$F_{bot.2} = \frac{H_2^2 \times F_{bot.}}{H_1^2 + H_2^2} = \frac{7{,}5^2 \times 28{,}7}{8{,}5^2 + 7{,}5^2} = 12{,}56\ cm^2$$

c) Thickness of the bottom skin.

The reduced thickness of the bottom skin

$$\delta_{bot.} = \frac{f_{red.bot.}}{\text{B}} = \frac{12{,}3}{18} = 0{,}68\ cm$$

Where

$$f_{red.bot.} = F_{red.bot.} - F_{bot.} = 41 - 28{,}7 = 12{,}3\ cm^2$$

Thickness of the bottom skin

$$\delta_{sk.bot.} = 0{,}6 \times \delta_{bot.} = 0{,}6 \times 0{,}68 = 0{,}41\ cm$$

To take into account the work of the upper and lower skin on the shear stresses from M_{tors} and on the stress from local bending in the transfer of air load, its thickness, defined in the calculation of the normal stresses from M_{bend}, is necessary to increase by 20-30%, therefore:

$$\delta'_{sk.up.}\delta'_{sk.bot.} = 5\ mm$$

3. Determination of wall thickness and step of the stiffenners of the front spar, in which the directions of the tangential loads q_{Q1} and $q_{M_{tors}}$ are the same.

a) The reduced shear force Q_{np} in the calculated section is equal:

$$Q_{np} = Q_{en} - \frac{M_{bend}}{H}\alpha = 13364 - \frac{7237 \times 100}{8{,}1 \times 10} = 4429\,kg$$

b) Shear force, which act on the wall of the spar and is determined by the ratio:

$$Q_1 = Q_{np} \times \frac{H_1^2}{\sum H_1^2}$$

Then

$$Q_1 = Q_{np} \times \frac{H_1^2}{H_1^2 + H_2^2} = 4429 \times \frac{8{,}5^2}{8{,}5^2 + 7{,}5^2} = 2490\,kg$$

$$Q_2 = Q_{np} \times \frac{H_2^2}{H_1^2 + H_2^2} = 4429 \times \frac{7{,}5^2}{8{,}5^2 + 7{,}5^2} = 1939\,kg$$

c) The linear forces:

$$q_{Q1} = \frac{Q_1}{H_{1p}} = \frac{2490}{0{,}95 \times 8{,}5} = 308\,kg/cm$$

$$q_{Q2} = \frac{Q_2}{H_{2p}} = \frac{1939}{0{,}95 \times 7{,}5} = 272\,kg/cm$$

$$q_{M_{tors}} = \frac{M_{tors}}{2 \times F_{\text{конт}}} = \frac{5825 \times 100}{2 \times \frac{8{,}5 + 7{,}5 + 9{,}5}{3} \times 18} = 19\,kg/cm$$

The total shear force in the wall of the front spar is equal:

$$q_{w.1} = q_{Q1} + q_{M_{tors}} = 306 + 19 = 325\,kg/cm$$

d) The wall thickness of the front spar.

When designing the wall, we accept the condition that the wall must not lose stability to P^p. Therefore, the condition must be fulfilled $\tau^p \leq \tau_k$.

If we set $\tau^p = 0{,}3 \times \sigma = 0{,}3 \times 4400 = 1320\frac{kg}{cm^2}$, then the calculation of the wall can be done as follows:

$$\delta_{w.1} = \frac{q_{w.1}}{\tau_k} = \frac{325}{1320} = 0{,}24\ cm$$

The resulting thickness is rounded to the standard thickness of 2.5 mm.

e) Step of the spar stiffeners.

The step of the stiffeners on the wall is found by the formula:

$$t_{w.} = \delta_{w.1}\sqrt{\frac{5{,}04 \times E}{\tau^p - 3{,}4 \times E\left(\frac{\delta_{w.1}}{H_p}\right)^2}} = 0{,}25\sqrt{\frac{5{,}04 \times 7 \times 10^5}{1320 - 3{,}4 \times 7 \times 10^5\left(\frac{0{,}24}{8{,}1}\right)^2}} = 17{,}5\ cm$$

The required number of stiffeners between the two ribs:

$$n = \frac{t_{\text{нр}} - t_{w.}}{t_{w.}} = \frac{45 - 17{,}5}{17{,}5} = 1{,}57$$

Take a number of intermediate stiffeners n = 2.

Therefore, the step of the stiffeners is as follows:

$$t_{w.} = \frac{t_{\text{нр}}}{n+1} = \frac{45}{3} = 15\ sm$$

f) Calculation of the riveted attachment of the wall of the front spar to the cap.

The force acting on one rivet is equal:

$$T_r = \frac{q_{w.1} \times t_r}{n} = \frac{325 \times 3}{2} = 467\ kg$$

where n - is the number of rivets in the joint.

Rivet with diameter $d_r = 5\ mm$, the maximum shear force, $T_{br} = 480\ kg$, material D16T.

2.4 Flaps maintenance

During maintenance of flaps, special attention is paid to the most stressed places, namely:

- riveted joints;
- surfaces that are exposed to exhaust gases;
- seam joints.

These areas may include:

- cracks and chips;
- scratches and deformations;
- paintwork damage;
- loose rivets and bolts.

Scratches are measured with a spring indicator (up to 1 mm - cleaned and coated with paintwork, more than 1 mm - repaired). If the length of the scratch is up to 4 mm, then this place is drilled out and a rivet is installed, more than 4 mm - overlay is put on.

Signs of loose rivets:

- formation of dark marks around them;
- bending of the edges of the blind rivets;
- peeling of the paintwork.

Constant control of fatigue cracks is performed, which are recorded in the special form.

Main works during maintenance of flaps:

1. Check the condition of skins, fairings, hinges, carriage, gears, screw elevators, rails, bearings, electromechanisms (corrosion, cracks, dents, condition of paintwork, reliability of mounting).

2. Inspection of flap retracting and releasing systems from one position to another. This controls the release and retraction time, smoothness of movement, travel reserve to the stops, matching the deflection angles.

Preparatory work:

- connect a ground source of electricity;
- create pressure in the hydraulic system from the hydraulic unit;
- establish communication with the ground;
- make sure that there are no people under the means of mechanization;
- hang a sign "Aircraft under electric power".

During maintenance or after replacement of individual elements of the system, the flaps are checked for failure to reach the stops.

Methods of verification:

1. Checking with feeler gaps between the stop and the flap structure element.
2. With a special key release the hydraulic clutch. Another special key through the reducer bring the flap to the mechanical stop.
3. Checking the clearances between the rails and rollers in the support nodes with feelers.
4. Checking the flaps rocking in released position. To do this, the force is applied with a dynamometer and the backlash is measured.

Conclusion to the part

In a special part of the thesis the design of the wing flap is performed, as well as its strength calculation, the definition of the geometry of the structural parts of the flap are alculated. The construction materials for the manufacture of the flap are defined. The maintenance procedure of the flaps is described in the part of work.

GENERAL CONCLUSION

During the diploma thesis the following results were obtained:

- the preliminary design of the short-range cargo aircraft is performed;

- calculations of center of gravity and flight technical characteristics of the aircraft were performed;

- developed a structure of the flap, which ensures its strength and provide safe operation of the aircraft.

The chosen scheme of the aircraft - high-wing monoplane with two engines located under the wing, increases the protection of engines and wing structures from damage during take off and landing.

The installation of modern D436T1 engines on the designed aircraft, which meet all ICAO standards for noise and emissions into the atmosphere, allows customers to perform international flights.

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П Р О Е К Т
С А М О Л Е Т А   С   Т Р Д Д
Н А У, А К Ф,   кафедра   К Л А

ПРОЕКТ дипломный   Расчет выполнен 16.10.2020
Исполнитель Бевз С.   Руководитель Маслак Т.П.

ИСХОДНЫЕ ДАННЫЕ И ВЫБРАННЫЕ ПАРАМЕТРЫ

Количество пассажиров	0.
Количество членов экипажа	2.
Количество бортпроводников или сопровождающих	2.
Масса снаряжения и служебного груза	896.66 кг.
Масса коммерческой нагрузки	16000.00 кг.
Крейсерская скорость полета	800. км/ч
Число "М" полета при крейсерской скорости	0.7491
Расчетная высота начала реализации полетов с крейсерской экономической скоростью	11.000 км
Дальность полета с максимальной коммерческой нагрузкой	1300. км.
Длина летной полосы аэродрома базирования	2.55 км.
Количество двигателей	2.
Оценка по статистике энерговооруженности в квт/кг	3.2000
Степень повышения давления	5.50
Относительная масса топлива по статистике	0.3100
Удлинение крыла	9.58
Сужение крыла	4.05
Средняя относительная толщина крыла	0.118
Стреловидность крыла по 0.25 хорд	24.0 град.
Степень механизированности крыла	0.970
Относительная площадь прикорневых наплывов	0.000
Профиль крыла - Суперкритический	
Шайбы УИТКОМБА - не применяются	
Спойлеры - установлены	
Диаметр фюзеляжа	3.35 м.
Удлинение фюзеляжа	8.00
Стреловидность горизонтального оперения	32.0 град.
Стреловидность вертикального оперения	40.0 град.

Р Е З У Л Ь Т А Т Ы   Р А С Ч Е Т А
Н А У, А К Ф,   КАФЕДРА   "К Л А"

Значение оптимального коэффициента под'емной силы в расчетной точке крейсерского режима полета   Су   0.43976

Значение коэффициента   Сх.инд.   0.00916

ОПРЕДЕЛЕНИЕ КОЭФФИЦИЕНТА Dм = Мкрит - Мкрейс

Число Маха крейсерское	Мкрейс	0.74974
Число Маха волнового кризиса	Мкрит	0.76808
Вычисленное значение	Dm	0.01833

Значения удельных нагрузок на крыло в кПА(по полной площади):

при взлете	4.223
в середине крейсерского участка	3.937
в начале крейсерского участка	4.068

Значение коэффициента сопротивления фюзеляжа и гондол	0.00847
Значение коэфф. профиль. сопротивления крыла и оперения	0.000917
Значение коэффициента сопротивления самолета:	
в начале крейсерского режима	0.02820
в середине крейсерского режима	0.02789
Среднее значение Су при условном полете по потолкам	0.43976
Среднее крейсерское качество самолета	15.76555

Значение коэффициента Су.пос.	1.631
Значение коэффициента (при скорости сваливания) Су.пос.макс.	2.447
Значение коэффициента (при скорости сваливания) Су.взл.макс.	2.039
Значение коэффициента Су.отр.	1.488
Энерговооруженность в начале крейсерского режима	0.621
Стартовая тяговоруженность. по условиям крейс. режима Ro.кр.	2.608
Стартовая тяговоруж. по условиям безопасного взлета Ro.взл.	2.487

Расчетная тяговоруженность самолета No 2.713

Отношение Dn = Ro.кр / Ro.взл Dr 1.049

УДЕЛЬНЫЕ РАСХОДЫ ТОПЛИВА (в кг/кВт*ч):

взлетный	37.1755
крейсерский (характеристика двигателя)	58.1239
средний крейсерский при заданной дальности полета	58.6662

ОТНОСИТЕЛЬНЫЕ МАССЫ ТОПЛИВА:

аэронавигационный запас	0.03349
расходуемая масса топлива	0.09739

ЗНАЧЕНИЯ ОТНОСИТЕЛЬНЫХ МАСС:

крыла	0.11993
горизонтального оперения	0.01747
вертикального оперения	0.02031
шасси	0.04959
силовой установки	0.08915
фюзеляжа	0.12476
оборудования и управления	0.12984
служебной нагрузки	0.01688
топлива при Lрасч.	0.13088
коммерческой нагрузки	0.30120

Взлетная масса самолета "М.о" = 53122. кГ.

Потребная взлетная тяга двигателя 72.05 kH

Относительная масса высотного оборудования и противообледенительной системы самолета	0.0171
Относительная масса пассажирского оборудования (или оборудования кабин грузового самолета)	0.0006
Относительная масса декоративной обшивки и ТЗИ	0.0062
Относительная масса бытового (или грузового) оборудования	0.02845
Относительная масса управления	0.0073
Относительная масса гидросистем	0.0194

Относительная масса электрооборудования	0.0208
Относительная масса локационного оборудования	0.0059
Относительная масса навигационного оборудования	0.0088
Относительная масса радиосвязного оборудования	0.0044
Относительная масса приборного оборудования	0.0103
Относительная масса топливной системы (входит в массу "СУ")	0.0038
Дополнительное оснащение:	
Относительная масса контейнерного оборудования	0.0000
Относительная масса нетипичного оборудования	0.0037

[встроенные системы диагностики и контроля параметров, дополнительное оснащение салонов и пр.]

ХАРАКТЕРИСТИКИ ВЗЛЕТНОЙ ДИСТАНЦИИ

Скорость отрыва самолета	242.45	км/ч
Ускорение при разбеге	2.07	м/с*с
Длина разбега самолета	1094.	м.
Дистанция набора безопасной высоты	578.	м.
Взлетная дистанция	1678.	м.

ХАРАКТЕРИСТИКИ ВЗЛЕТНОЙ ДИСТАНЦИИ ПРОДОЛЖЕННОГО ВЗЛЕТА

Скорость принятия решения	230.32	км/ч
Среднее ускорение при продолженном взлете на мокрой ВПП	0.22	м/с*с
Длина разбега при продолженном взлете на мокрой ВПП	1973.98	м.
Взлетная дистанция продолженного взлета	2552.36	м.
Потребная длина летной полосы по условиям прерванного взлета	2646.29	м.

ХАРАКТЕРИСТИКИ ПОСАДОЧНОЙ ДИСТАНЦИИ

Максимальная посадочная масса самолета	50399.	кг.
Время снижения с высоты эшелона до высоты полета по кругу	21.5	мин.
Дистанция снижения	47.74	км.
Скорость захода на посадку	242.55	км/ч.
Средняя вертикальная скорость снижения	1.97	м/с
Дистанция воздушного участка	514.	м.
Посадочная скорость	227.55	км/ч.
Длина пробега	714.	м.
Посадочная дистанция	1228.	м.
Потребная длина летной полосы (ВПП + КПБ) для основного аэродрома	2051.	м.
Потребная длина летной полосы для запасного аэродрома	1744.	м.

ПОКАЗАТЕЛИ ЭФФЕКТИВНОСТИ САМОЛЕТА

Отношение массы снаряженного самолета к массе коммерческой нагрузки	1.8737	
Масса пустого снаряженного с-та приход. на 1 пассажира	0.00	кг/пас.
Относительная производительность по полной нагрузке	345.66	км/ч
Производительность с-та при макс. коммерч. нагрузке	10805.3	кг*км/ч
Средний часовой расход топлива	2687.411	кг/ч
Средний километровый расход топлива	3.98	кг/км
Средний расход топлива на тоннокилометр	248.715	г/(т*км)
Средний расход топлива на пассажирокилометр	0.0000	г/(пас.*км)
Ориентировочная оценка приведен. затрат на тоннокилометр	0.2038	$/(т*км)