МІНІСТЕРСТВО ОСВІТИ ТА НАУКИ УКРАЇНИ

НАЦІОНАЛЬНИЙ АВІАЦІЙНИЙ УНІВЕРСИТЕТ

Кафедра конструкції літальних апаратів

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

ДИПЛОМНА РОБОТА

(ПОЯСНЮВАЛЬНА ЗАПИСКА)
ЗДОБУВАЧА ОСВІТНЬОГО СТУПЕНЯ
"БАКАЛАВР"

Тема: «Аванпроект ближньомагістрального літака пасажиромісткістю до 50 осіб»

Виконав: ___________ **А.І. Зозуля**

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

Нормоконтролер: ___________ **С.В. Хижняк**

Київ 2020

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

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

DIPLOMA WORK

(EXPLANATORY NOTE)
OF ACADEMIC DEGREE
«BACHELOR»

Theme: «Preliminary design of short range aircraft with up to 50 passenger»

Performed by: ___________ **A.I. Zozulia**

Supervisor: ___________ **V.S. Krasnopolskii**

Standard controller: ___________ **S.V. Khizhnyak**

Kyiv 2020

NATIONAL AVIATION UNIVERSITY

Aerospace Faculty
Aircraft Design Department
Academic degree «Bachelor»
Speciality: 134 «Aviation and Rocket-Space Engineering»

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

TASK
for bachelor diploma work
ZOZULIA ANTON

1. Theme: «**Preliminary design of short range passenger aircraft with up to 50 seats capacity**».
Confirmed by Rector's order from 05.06.2020 year №801/ст
2. Period of work execution: from 25.05.2020 year to 21.06.2020 year.
3. Work initial data: cruise speed V_{cr} =500 km/h, flight range L=1400 km, operating altitude H_{op} =6.5 km, 50 passenger capacity.
4. Explanatory note argument (list of topics to be developed): choice and substantiations of the airplane scheme, reliability analysis, choice of initial data; engine selection, aircraft layout, center of gravity position calculation, design of control system elements.
5. List of the graphical materials: general view of the airplane (A1×1); layout of the airplane (A1×1); design of control system elements (A1×2).
Graphical (illustration) material is performed in AutoCAD, Microsoft Office PowerPoint, and given in form of posters and drawings of appropriate format.
6. Calendar Plan

Task	Execution period	Signature
Task receiving, processing of statistical data	25.05.2020-29.05.2020	
Aircraft take-off mass determination	30.05.2020-03.06.2020	
Aircraft layout	04.06.2020-08.06.2020	
Aircraft centering determination	09.06.2020-13.06.2020	

Graphical design of the parts	14.06.2020-16.06.2020	
Completion of the explanation note	17.06.2020-19.06.2020	
Preliminary examination and defence of the diploma work	20.06.2020-21.06.2020	

7. Task issuance date: 25.05.2020 year.

Supervisor of diploma work:_________ V.S. Krasnopolskii

Task for execution is given for:__________A.I. Zozulia

ABSTRACT

Explanatory note to the diploma work «Preliminary design of short range passenger aircraft with up to 50 seats capacity» contains:

60 sheets, 9 figures, 6 tables, 13 references

Object of the design is development of passenger aircraft with the possibility to carry 50 passengers.

Aim of the diploma work is the development of the aircraft preliminary design and its characteristic estimation.

The method of design is analysis of the prototypes and selections of the most advanced technical decisions.

The diploma work contains drawings of the short range aircraft with 50 passengers, calculations and drawings of the aircraft layout and design of control system elements.

The result of diploma work can be implemented in working process, aircraft documentation formation, and it can be used in design bureaus.

AIRCRAFT, PRELIMININARY DESIGN, LAYOUT, CENTER OF GRAVITY POSITION, DESIGN OF CONTROL SYSTEM ELEMENTS.

List of diploma work

Format	№	Designation	Name	Quantity	Notes
			General documents		
A4	1	NAU 20 05Z 00 00 00 99 TW	Task for work	1	
	2	NAU 20 05Z 00 00 00 99	Short range passenger aircraft	2	
A1		NAU 20 05Z 00 00 00 99 GV	General view		
A1		NAU 20 05Z 00 00 00 99 FL	Fuselage layout		
A4	3	NAU 20 05Z 00 00 00 99 EN	Short range passenger aircraft	60	
			Explanatory note		
			Documentation for assembly units		
A1	4	NAU 20 05Z 00 00 00 99 AD	Design of control system elements	4	

NAU 20 05Z 00 00 00 99 EN

Done by	Zozulia A.I.	Preliminary design of short range passenger aircraft with up to 50 seats capacity (List of diploma work)	list	sheet	sheets
Checked by	Krasnopolskii V.S.			6	60
			AF 402 134		
St.control.	Khyzhniak S.V.				
Head of dep.	Ignatovich S.R.				

Content

INTRODUCTION

Nowadays, the aviation is developing firstly throughout the world. There is need of different types of aircraft which can carry out the various functions. For this diploma work the creation of the preliminary outline of the short range airplane was design. The main performances are taken: cruise speed V_{cr} =500 km/h, flight range L = 1400 km, operating altitude H_{op} = 6.5 km, 50 passenger capacity.

It is substantiated by the necessity of the development the competitively cheap, and effective air transportation service inside Ukraine and the nearest abroad countries.

Internal flights will be popular because of the short duration and planned low cost. That is why designed aircraft is actual to be created.

Besides, the new fully digitalized aircraft maintenance manual format is planned to use for designed aircraft operation. The main aim is to shorten the procedure performance sequence and shorten the quantity of mistakes by attaching the new information modules joining all system in convenient form.

In general, aim of this diploma work is to create the aircraft which will meet the following requirements:

- High safety level
- Efficiency of operation
- Light maintenance support.

1. PROJECT PART. PRELIMINARY DESIGN OF SHORT RANGE AIRCRAFT

1.1 Analysis of prototypes

1.1.1 Overview general performances

The selection of optimal parameters is straight dependent of planned aim and area of aircraft usage, economical requirements based on complicity of construction, ability to degrees the usage of some of expensive construction materials and methods of assembly, planned passenger and cargo capacity and complexity in maintenance.

The creation of basic aircraft outline includes aerodynamic calculation, geometrical parameters calculation and centering of equipped fuselage. These steps form the final exterior and interior appearances of a designed aircraft.

For designed aircraft there were chosen the prototypes in range of 50-100 passengers and middle-range of usage. Such aircraft like Xian Ma6000, Ил-114 and F-28-0100 will compete with designed aircraft in chosen market segment. Performances of prototypes are presented in table 1.1.

Table 1.1

Performances of prototypes

Parameter	Airplanes		
	Xian Ma600	Ил-114	F-28-0100
1	2	3	4
The purpose of airplane	Passenger	Passenger	Passenger
Crew/flight attend. persons	2/2	2/2	2/2
Maximum take-off weight, m_{tow}, kg	21800	23500	44920
Maximum payload, $m_{к.max}$, kg	5500	6500	11563

Ending of the table 1.1

1	2	3	4
Passengers	60	60	100
The flight altitude $V_{w.\ ек}$, m	6500	7600	11300
Flight range $m_{к.\ max}$, km	1400	2000	3111
Take off distance $L_{TO.\ d.}$, m	1900	1700	2010
Number and type of engines	2xPW-127J	2xTB7-117CT	2xRR Tay-620
The shape of the fuselage cross-section	circular	circular	circular
Fineness ratio	8	9	10.38
Fineness ratio the nose and rear part	4	4.85	4,2
Sweepback angle at 1/4 chord line, ^{0}	6	6	20

The typical location of constructional elements, principal aerodynamic schemes of listed prototypes became the baseline for designed aircraft outline. For layout formation the mix of the most effective characteristics from all three prototypes are used. Besides the Xian Ma600 is chosen as a main prototype because it meets almost all requirements for middle-range economy class passenger airplane.

1.1.2 Brief description of the main parts of the aircraft

The plane is a cantilever high-wing monoplane with turboprop engines placed on the wing and twin-cycle landing gear with a front single-strut landing gear and two main gears.

Fuselage has circular cross section. Empennage has a conventional design. Rudder and elevators are equipped with aerodynamic balance.

1.1.2.1 Fuselage

The fuselage has semimonocoque design. It is pressurize between the first and the fourteenth formers.

The fuselage framework consists of 49 formers, longitudinal beams and stringers manufactured from extruded profiles, and working skin.

The cockpit, passenger cabin and all auxiliary units are located in the fuselage. There is a cargo bay behind the cockpit on the lower part of the fuselage, in front of which located a large cargo hatch. In the non-pressurize nose compartment (up to the first frame) the units of radio equipment are located. The passenger cabin is separated of the rest compartments by the bulkhead. In the tail section there is an entrance hall, a sideboard with a flight attendant’s seat, a toilet and wardrobe. At the rear of the compartment is the trunk. On the left side is the passenger front door with a sidewalk.

1.1.2.2 Wing

The wing of the aircraft has high taper ratio and trapezoidal planform. There is a set of structural elements of different thickness in vertical planform of cross-section, providing good loaded drag during insignificant parasitic, good lateral stability and controllability during significant angles of attack.

The wing is torsion box type. It is divided into a center section, two middle and two detachable parts, joined along ribs with the help of fitting connections.

The wing consists of a central (made by spars, upper and lower panels and ribs), nose and tail parts, end fairings, ailerons and slotted flaps. The wing center section consists of solid-pressed large-sized panels and spars that reduces its weight and greatly simplifies the process of assembly, and also increases the reliability of the design. There are four soft fuel tanks in the torsion box of the center section of the wing. The middle parts of the wing are the sealed fuel tank.

1.1.2.3 Tail unit

The empennage consists of vertical and horizontal part. Vertical tail unit includes fin and rudder, horizontal stabilizer and elevator. In front of the fin dorsal fin is mounted on the fuselage.

The sweep of the vertical and horizontal tail unit is greater than the sweep of the wing, so that the aerodynamic characteristics of the tail unit with an increase in the Mach number do not deteriorate faster than the characteristics of the wing. The grater sweep of the fin is also suitable, because at the same time the horizontal stabilizer efficiency is increased due to the increase of its moment arm.

The airfoil of vertical and horizontal stabilizer is symmetrical. Symmetric airfoil allows to maintain the same character of aerodynamic loads during deflection of rudders in different directions and, in addition, has a smaller drag.

Vertical tail unit in comparison with horizontal one has an increased relative thickness of the airfoil in order to reduce the mass of fin loaded by forces, from both vertical and horizontal parts.

The design feature of the aircraft empennage is the attachment of the assembled panels along the spars web that provides high manufacturability of the assembly.

High performance control column provides aircraft controllability over the entire range of flight speeds, at all altitudes in a wide range centering's.

1.1.2.4 Landing gear

The landing gear consists of three struts. All undercarriage struts are retractable. The direction of retraction is counter the flight.

The nose landing gear strut is located under the cockpit canopy. The main landing gear struts are installed under the engine nacelles and retract in flight forward into special compartments under the engines. On a fixed axis of each main strut two wheels with disc brakes are installed. Wheels are equipped with inertial sensors.

In extended and retracted positions landing gear struts are locked with the mechanical locks actuated by the hydraulic cylinders.

Landing gear wheel well are closed by doors while landing gear struts are fully extended or retracted. The doors actuation is performed by mechanisms which kinematically joined with strut actuation system.

The nose landing gear is used for steering. The turn of the nose strut wheel is performed by the actuators powered by aircraft hydraulic system. Besides the extension and retraction, braking, locks opening, doors actuations are performed by hydraulic system too. In case of hydraulic system failure the retraction and extension of a landing gear can be performed with use of mechanical system. In this case the extension of nose or main landing gear is performed partially due to their own weight.

The struts location was chosen to reach the optimum balance between aircraft stability and controllability. That is why during the calculation of wheel base and the wheel track the center of gravity of an aircraft should be considered.

1.1.2.5 Power plant

The power plant: The Pratt & Whitney engine is a three shaft, turboprop engine. The centrifugal low-pressure impeller is driven by a single stage low pressure turbine and the high-pressure compressor by a two-stage high pressure turbine. Power is delivered to the offset propeller reduction gearbox via a third shaft, connected to a two-stage power turbine.

The control system operates all engine functions, including power regulation. Engines PW 127H, PW 127J, PW 127B – are three-shaft, turboprop engines, in various modifications installed on passenger aircrafts Fokker 60 (PW 127B), Ил-114 (PW 127B) and transport Xian Ma600 (PW 127J).

1.2 Aircraft layout and center of gravity calculation

1.2.1 Geometry calculations for the aircraft principles structural units

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

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

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

1.2.1.1 Wing geometry calculation

Full wing area is:

$$S_w=\frac{m_0 \cdot g}{P_0}\frac{20800 \cdot 9.8}{3114}=65.46\ (m^2);$$

where m_0 – take-off weight;

g – gravity acceleration;

P_0 – specific wing load.

Wing span is:

$$I_w=\sqrt{S_w \cdot \lambda_w}=\sqrt{65.46 \cdot 11.38}=27.3\ (m);$$

where λ_w – wing aspect ratio.

Root chord is:

$$b_0=\frac{2S_w \cdot \eta_w}{(1+\eta_w) \cdot I_w}=\frac{2 \cdot 65.46 \cdot 3}{(1+3) \cdot 27.3}=3.6\ (m);$$

where η_w – wing taper ratio.

Tip chord is:

$$b_t=\frac{b_0}{\eta_w}=\frac{3.6}{3}=1.2\ (m);$$

Maximum wing thickness is:

$$C_{max}=C_w \cdot b_t=0.12 \cdot 1.2=0.144\ (m);$$

where c_w – medium wing relative thickness.

On board chord is:

$$b_{ob} = b_0 \cdot (1 - \frac{(\eta_w - 1) \cdot D_f}{\eta_w \cdot l_w}) = 3.6 \cdot \left(1 - \frac{(3-1) \cdot 2.9}{3 \cdot 27.3}\right) = 3.345\ (m);$$

where D_f – fuselage diameter.

For mean aerodynamic chord determination the geometrical method was used (fig. 1.1).

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

Thus, the mean aerodynamic chord is equal 2.6 m.

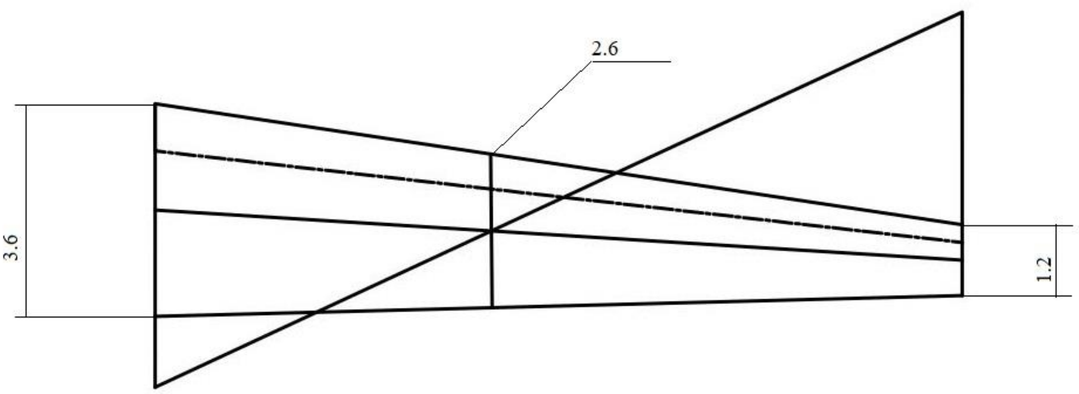

Fig. 1.1 Geometrical method of determination of mean aerodynamic chord

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

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

Ailerons geometrical parameters are determined in the next order:

Ailerons span:

$$I_{ail} = 0.4 \cdot \frac{I_w}{2} = 0.4 \cdot \frac{27.3}{2} = 5.250\ (m);$$

Aileron chord:

$$b_{ail} = 0.44 \cdot b_t = 0.44 \cdot 1.2 = 0.528\ (m);$$

Aileron area:

$$S_{ail} = 0.06 \cdot \frac{S_w}{2} = 0.06 \cdot \frac{65.46}{2} = 2(m^2);$$

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

Aerodynamic compensation of the aileron:

$$\text{Axial } S_{ax.ail} \leq (0.25 \dots 0.28) \cdot S_{ail}$$

$$S_{ail} = 0.26 \cdot 2 = 0.52\ (m^2)$$

Area of ailerons trim tab. For two engine airplane:

$$S_{tail} = 0.04 \dots 0.06 \cdot S_{ail} = 0.05 \cdot 2 = 0.1\ (m^2).$$

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

1.2.1.2 Fuselage layout

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

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

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

The next steps are necessary to calculate the main geometrical characteristics of the fuselage and consequently to obtain its outline.

Nose part length is:

$$I_{nfp} = 2 \cdot D_f = 2 \cdot 2.9 = 5.8\ (m);$$

Fuselage length is:

$$I_f = \lambda_f \cdot D_f = 8 \cdot 2.9 = 23.2\ (m);$$

where: λ_f – fuselage fineness ratio.

Fuselage nose part fineness ratio is:

$$\lambda_{fnp} = \frac{I_{fnp}}{D_f} = \frac{5.8}{2.9} = 2;$$

Length of the fuselage rear part is:

$$I_{frp} = \lambda_{frp} \cdot D_f = 2.4 \cdot 2.9 = 6.96\ (m);$$

where: λ_{frp} – fuselage fineness ratio.

Cabin height is:

$$H_{cab} = 1.48 + 0.17 B_{cab} = 1.48 + 0.17 \cdot 2.560 = 1.9152\ (m);$$

where B_{cab} – width of the cabin.

For economic class passenger cabin the location of seats in the one row (2 + 2) determine the next parameter:

$$B_{cab} = n_{2chblock} \cdot b_{2chblock} + b_{aisle} + 2\delta = 2 \cdot 1000 + 500 + 2 \cdot 30 = 2.560\ (m);$$

where $n_{3chblock}$ – width of 2 chairs;

$b_{3chblock}$ – number of 2 chair block;

b_{aisle} – width of aisle.

The length of passenger cabin is:

$$L_{cab} = L_1 + (n_{raws} - 1) \cdot L_{seatpitch} + L_2 = 1200 + (13 - 1) \cdot 750 + 300 = 10.5\ (m);$$

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where L_1 – distance between the wall and the back of first seat;

n_{rows} – number of rows;

$L_{seat\ pitch}$ – seat pitch;

L_1 – distance between the back of last seat and the wall.

1.2.1.3 Luggage compartment

Cargo compartment volume is:

$$V_{cargo} = v \cdot n_{pass} = 0.23 \cdot 52 = 11.96\ (m^3);$$

where ν – relative mass of baggage;

n_{pass} – number of passengers.

Luggage compartment design is similar to the prototype.

1.2.1.4 Galleys and buffets

Volume of buffets (galleys) is:

$$V_{galley} = (0.1 \ldots 0.12) \cdot n_{pass} = 0.1 \cdot 52 = 5.2\ (m^3);$$

where ν – volume of buffets;

n_{pass} – number of passengers.

Area of buffets (galleys) is:

$$S_{galley} = \frac{V_{galley}}{H_{cab}} = \frac{5.2}{1.9152} = 2.715\ (m^2);$$

Number of meals per passenger breakfast, lunch and dinner – 0.7 kg, tea and water – 0.4 kg. Buffet design similar to prototype.

1.2.1.5 Layout and calculation of basic parameters of tail unit

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

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

Area of vertical tail unit is:

$$S_{VTU} = \frac{l_{wx} S_w}{L_{VTU}} \cdot A_{VTU} = \frac{27.3 \cdot 65.46}{5.205} \cdot 0.0402 = 13.793\ (m^2);$$

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

Area o horizontal tail unit is:

$$S_{HTU} = \frac{b_{MAC} \cdot S_w}{L_{HTU}} \cdot A_{HTU} = \frac{2.6 \cdot 65.46}{9.996} \cdot 1.067 = 18.175\ (m^2);$$

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

Determination of the elevator area and direction:

Altitude elevator area is:

$$S_{el} = 0.2765 \cdot S_{HTU} = 0.2765 \cdot 18.175 = 4.662\ (m^2);$$

where k_{el} – relative elevator area coefficient.

Rudder area is:

$$S_{rud} = 0.2337 \cdot S_{VTU} = 0.2337 \cdot 13.793 = 3.223\ (m^2);$$

where k_r – relative rudder area coefficient.

Choose the area of aerodynamic balance:

$$0.3 \le M \le 0.6;$$

$$S_{eb} = (0.22..0.25)\ S_{el};$$

$$S_{rb} = (0.2..0.22)\ S_{rod};$$

Elevator balance area is:

$$S_{eb} = 0.22 \cdot S_{el} = 0.22 \cdot 4.662 = 1.026\ (m^2);$$

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

Rudder balance area is:

$$S_{rb} = 0.2 \cdot S_{rud} = 0.2 \cdot 3.223 = 0.6446\ (m^2);$$

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

The area of altitude elevator trim tab is:

$$S_{te} = 0.08 \cdot S_{el} = 0.08 \cdot 4.662 = 0.37296\ (m^2);$$

where k_{te} – relative elevator trim tab area coefficient.

Area of rudder trim tab is:

$$S_{tr} = 0.06 \cdot S_{rud} = 0.06 \cdot 3.223 = 0.19338\ (m^2);$$

where S_{tr} – relative trim tab area coefficient.

Root chord of horizontal stabilizer is:

$$b_{OHTU} = \frac{2S_{HTU} \cdot \eta_{HTU}}{(1+\eta_{HTU}) \cdot l_{HTU}} = \frac{2 \cdot 18.175 \cdot 2.2}{(1+2.2) \cdot 9.996} = 2.5\ (m);$$

where η_{HTU} – horizontal tail unit taper ratio;

l_{HTU} – horizontal tail unit span.

Tip chord of horizontal stabilizer is:

$$b_{OHTU} = \frac{b_{OHTU}}{\eta_{HTU}} = \frac{2.5}{2.2} = 1.14\ (m);$$

Root chord of vertical stabilizer is:

$$b_{OVTU} = \frac{2S_{VTU} \cdot \eta_{VTU}}{(1+\eta_{VTU}) \cdot l_{VTU}} = \frac{2 \cdot 13.793 \cdot 2.786}{(1+2.786) \cdot 5.205} = 3.9\ (m);$$

where η_{VTU} – vertical tail unit taper ratio;

l_{VTU} – vertical tail unit span.

Tip chord of vertical stabilizer is:

$$b_{OVTU} = \frac{b_{OVTU}}{\eta_{VTU}} = \frac{3.9}{2.786} = 1.4;$$

1.2.1.6 Landing gear design

To estimate the landing gear outline in this project it is necessary to calculate the location of every strut in relatively to each other, to determine the loads on landing gear system, and its location considering center of gravity of an airplane.

In this layout the principal scheme of landing gear is fully based on the prototype data.

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

Main wheel axes offset is:

$$e = 0.2673 \cdot b_{MAC} = 0.3 \cdot 2.6 = 0.78\ (m);$$

where k_e – coefficient of axes offset;

b_{MAC} – mean aerodynamic chord.

Landing gear wheel base is:

$$B = k_b \cdot L_f = 0.3836 \cdot 23.2 = 8.9\ (m);$$

where k_b – wheel base calculation coefficient.

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

Front wheel axial offset is:

$$d_{ng} = B - e = 10.5 - 0.78 = 9.72;$$

Wheel track is:

$$T = k_T \cdot B = 0.8315 \cdot 8.9 = 7.4\ (m);$$

where k_b – wheel track calculation coefficient.

Nose wheel load is:

$$P_{NLG}=\frac{(9.81\cdot e\cdot k_g\cdot m_0)}{(B\cdot z)}=\frac{(9.81\cdot 0.78\cdot 1.75\cdot 20800)}{(8.9\cdot 2)}=13263.12\ (N);$$

where K_g – dynamics coefficient;

z – number of wheels.

Main wheel load is equal:

$$P_{MLG}=\frac{(9.81\cdot(B-e)\cdot m_0}{(B\cdot n\cdot z)}=\frac{(9.81\cdot(8.9-0.78)\cdot 20800)}{(8.9\cdot 2\cdot 4)}=23611.3\ (N);$$

where n – number of main landing gear struts.

1.2.1.7 Choice and description of power plant

In accordance with the performance of aerodynamic calculations for the design of the aircraft, the required maximum thrust at take-off mode is 165. The PW127 is a free turbine propulsion engine consisting of turbomachine and reduction gearbox modules connected by a drive shaft and integrated strucural intake case, given in the table. 1.2.

The turbomachine is a three concentric shaft design incorporating two centrifugal compressors each driven separately by single-stage turbines, and a two-stage power turbine.

The reduction gearbox features a twin layshaft design with antifriction bearings and an offset propeller shaft. The combustion system is comprised of an annular reverse flow combustor, 14 piloted air blast fuel nozzles, and two ignitors.

Table 1.2

Examples of application PW 127

Engine Model	Overall Length (mm)	Overall Width (mm)	Dry Spec. Weight (kg)	Maximum Take-off Power – 5 min.		Normal Take-off Power – 5 min.		Maximum Continuous Power	
				Shaft Power (kW)	Maximum Air Temp for Rated Power (°C)	Shaft Power (kW)	Maximum Air Temp for Rated Power (°C)	Shaft Power (kW)	Maximum Air Temp for Rated Power (°C)
PW127	2130	679	480.8	2051	32	1846	32	1864	41

1.3 Determination of the aircraft center of gravity position

1.3.1 Determination of centering of the equipped wing

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

The centering is important aircraft characteristic as it affects on the balancing, stability and controllability of the aircraft. That's why it is necessary to keep it inside strict limits.

To calculate the centering it is necessary to determine the mass of main structural units and devices. The list of the units masses for the aircraft given in the table 1.3. The mass of aircraft is 20800 kg.

The longitudinal static stability of the aircraft is determined by the location of its center of mass relatively to the focuses. The closer the center of mass is to the nose part of the aircraft, the more longitudinally stability the aircraft have. Coordinates of the center of gravity for the equipped wing are:

$$X_w' = \frac{\Sigma m_i' x_i}{\Sigma m_i'}.$$

Table 1.3

Trim sheet of equipped wing masses

Name	Mass		Center of gravity coordinates	Moment (kgm)
	Units	Total (kg)		
1	2	3	4	5
Wing (structure)	0.16	3328	1.17	3893.76
Fuel system, 40%	0.0038	79.04	1.17	92.4768
Control system, 30%	0.00336	69.888	1.56	109.02528
Electrical equip. 30%	0.009	187.2	0.26	48.672
Anti-icing system 70%	0.01785	371.28	0.26	96.5328
Hydraulic system, 70%	0.0189	393.12	1.56	613.2672
Power plant	0.12042	2504.736	1.2	3005.6832
Equipped wing without landing gear and fuel	0.33333	6933.264	1.133581	7859.41634
Nose landing gear	0.016057	333.978736	-5.73	-1913.69816
Main landing gear	0.032113	667.957264	1.3	868.3444432
Fuel	0.13388	2784.704	1.092	3040.896768
Equipped wing with landing gear and fuel	0.51538	10700	0.919314	9836.6598

1.3.2 Determination of the centering of the equipped fuselage*:*

The list of the unit for the aircraft is given in table 1.4.

The center gravity coordinates of the equipped fuselage are:

$$X'_f = \frac{\Sigma m'_i X_i}{\Sigma m'_i};$$

Table 1.4

Trim sheet of equipped fuselage masses

Objects	Units	Total (kg)	Coordinates of center of gravity. (m)	Momentum (kgm)
1	2	3	4	5
Fuselage	0.11355	2361.84	11.6	27397.344
Horizontal stabilizer	0.01567	325.936	20	6518.72
Vertical stabilizer	0.01553	323.024	18.2	5879.0368
Radiolocation equipment	0.0046	95.68	11	1052.48
Dashboard with equipment	0.008	166.4	2.2	366.08
Aero navigation	0.0069	143.52	3.5	365
Radio equipment	0.0034	70.72	3.5	502.32
Lavatory	0.000967	113.68386	13.4	247.52
Galley	0.005466	113.68386	12.8	1523.36367
Baggage equipment	0.000967	20.10676	5.1	1455.153357
Control system	0.00784	163.072	9.8	102.5443536
Electrical system	0.021	436.8	12.8	1598.1056
Hydraulic system	0.0081	168.48	10.1	5591.04
Anti-ice system 30%	0.00765	159.12	9	1701.648
Onboard equipment	0.008	166.4	3.5	1432.08
Passengers seats	0.014645	240	9	582.4
Emergency	0.004838	100.64	5.1	2160
Cockpit seats	0.000962	20	2.5	513.264
Attendant seat	0.000192	8	12.9	51
Non-typical equipment	0.0024	49.92	0	51.6
Equipped fuselage without commercial load	0.250869	5247.0264 48	11.20183	58776.29978
Passengers	0.173077	3600	8.7	31320
Passenger baggage	0.023077	480	5.1	2448
Cargo	0.026058	542	5.1	2764.2
Crew	0.006731	140	2.5	350
Flight attendant	0.002404	50	12.9	645
Equipped fuselage with commercial	0.484619	10100	9.59029	96948.49978

1.3.2.1 Calculation of center of gravity positioning variants

The list of mass objects for center of gravity variants calculation given in Table 1.4 and Center of gravity calculation options given in table 1.5 completes on the base of both previous tables.

The mean aerodynamic chord center of gravity is:

$$X_{MAC} = \frac{m_f \cdot x_f + m_w \cdot x_w - m_0 \cdot c_n}{m_0 - m_0};$$

where m_0 – aircraft takeoff mass, kg;

m_f – mass of equipped fuselage, kg;

m_w – mass of equipped wing, kg.

Table 1.5

Calculation of center of gravity positioning variants

Name	Mass, kg	Coordinates	Moment
1	2	3	4
Object	m_i	m	kgm
Equipped wing without fuel	6933.264	10.11955311	70161.53
Nose landing gear (retracted)	333.978	8.9	1087.423
Main landing gear (retracted)	667.9	10.60447197	7082.727
Fuel	2784	10.40997197	28981.36
Equipped fuselage	5247.026	11.20183029	58776.3
Passengers	3600	8.7	31320
Baggage of passengers	480	5.1	2448
Cargo	542	5.1	2764.2
Crew	140	2.5	350
Attendants	65	12.9	838.5
Nose landing gear (extended)	333.978	2.25597197	753.445
Main landing gear (extended)	667.9	10.60447197	7082.727

Table 1.6

Aircrafts center of gravity position variants

Variants of the loading	Mass, kg	Momentum of the mass, кg·m	Center of mass, m	Centering
1	2	3	4	5
Take-off mass (nose landing gear extended)	20800	203648.5448	9.811	25.50
Take-off mass (nose landing gear retracted)	20800	203314.5668	9.795	25.00
Landing variant (main landing gear extended)	17258	167170.876	9.686	21.64
Transportation variant (without payload)	16236.17	167782.3668	10.333	18.87
Parking variant (without fuel and payload)	15182.17	137107.9829	10.401	15.35

Conclusion to the project part

In this part the main geometric dimensions and centering of designed aircraft were determined. That made the creation of the drawing outline possible.

During the calculation the main geometrical parameters caused by operational purpose, planned quantities of passengers and cargo, speed and altitude of flight, conditions of landing and take-off, were considered. All obtained values meet requirements for the short range passenger aircrafts.

The centering of the designed aircraft was performed. The most forward center of gravity position of equipped aircraft is 17.96 from the origin of the leading edge of main aerodynamic chord. The most aft center of gravity position of equipped aircraft is 25.5 from the origin of the leading edge of main aerodynamic chord. Between these values centering of the aircraft should be performed.

Geometrical parameters almost math with chosen prototypes. That fact allows to make a conclusion that designed aircraft will successfully concur with another models on the chosen market segment.

Furthermore, the engine PW-127J that meets the requirements considering efficiency for designed aircraft was approximated. Main peculiarities of basic section of an aircraft and their influence on outline creation were figured out. Reliability analysis was performed.

As was shown in results of the reliability analysis we should pay attention to the design of control system elements as it is important part of the aircraft and it is directly effects on its operational characteristics.

2. SPECIAL PART. DESIGN OF CONTROL SYSTEM ELEMENTS

2.1 Output loads for calculation

The forces in the control parts must be determined in the neutral position of the steering surfaces. In the presence in the control system of various automatic devices such as autopilot, etc. the effort in the control parts must be determined taking into account the action of these devices.

Most parts and components of the aircraft control system are based on the strength of the design (destructive) load of the *PP*. Calculations for contact voltages are performed on operational loads *P*.

The value of the safety factor $f = PP / PE$ is normalized (within 1.5 ... 2).

In addition to the basic safety factor of NLPL, additional safety factors are provided:

- $f_{add} = 1.25$ – additional safety factor provided for the main butt and split joints;

- $f_{add} = 1.5$ – additional safety factor provided for casting parts.

The source material for determining the design loads are:

- kinematic diagrams with information about the location of the axes of rotation of all nodes, the dimensions of the kinematic elements;
- operational aerodynamic loads;
- normalized operational efforts of pilots applied to the rudder;
- operating capacity (load) from autopilot steering machines;
- normalized or accepted safety factors for different calculation cases.

When determining the operating aerodynamic loads, the deviation of the steering surfaces in manual control is taken instantaneously.

To move to the calculated aerodynamic loads is accepted $f = 2$

Hinge moments are determined taking into account the action of servo surfaces. The normalized operating forces on the command levers for non-maneuverable aircraft are determined depending on the weight of the aircraft. The maximum value of effort at single management makes:

$$P_1^{el} = 80kgf = 785N;$$
$$P_1^{PB} = 120kgf = 1177N;$$
$$P_1^{PH} = 120kgf = 1177N.$$

The force is applied "on itself" or "from itself" to the left or right horn of the rudder, the force is applied down the tangent to the rim of the rudder on one side.

In the case of dual control, in addition to the load by the force of one pilot, the simultaneous application by each pilot to his rudder of the load, which is 75% of the above, is considered. These loads can be directed both in one and in opposite directions.

After determining the operational values of the aerodynamic hinge moments of the steering surfaces, the normalized forces on the control levers is balanced.

Balancing-bringing the hinge moments of the steering surfaces in accordance with the forces applied to the steering wheel.

$$M_1^{el} = P_1^{el} \cdot h;$$

where h (m) is the gear ratio of the control wiring.

In most cases, when balancing, it is necessary to take into account the uneven distribution of aerodynamic load between the left and right halves of the steering surface.

In this case, the initial hinge moments of the left and right ailerons are divided into symmetrical and antisymmetric components. Antisymmetric components of the hinge moments of both ailerons that are balanced on the command lever.

Antisymmetric components of hinge moments are determined by the formulas:

$$M_{lev}^{as} = P_{kr} \cdot h_{lev}/2;$$

$$M_{pr}^{as} = P_{kr} \cdot h_{kr}/2.$$

The given hinge moments for the left and right ailerons are obtained as the sum of the corresponding values of symmetrical components and corrected antisymmetric components.

At manual control of rudders (ailerons) the hinge moments received as a result of balancing (f = 2) are calculated. In addition, the wiring of the manual aileron control in each wing console must be tested for strength from the design force on the steering wheel

$$P_1^{el} = 90kgf = 883N;$$

(at double control – 135 kgf – 1324 N), if at this position of the center of pressure nowhere on the scope of the aileron will not be more less than 50% of the local chord.

Units and parts that belong to two or more control channels (steering wheel) must be designed in case of simultaneous control of two channels; it is assumed that in each channel there is 75% of the design load.

2.1.1. Calculation of the strength of the elements of the steering column.

Case 1. Single control , loaded right-hand steering wheel.

Calculated loads:

$$P^{c}_{1\ n.\ e.r.} = P^{c}_{1\ l.\ e.r} = 78 \cdot 2 = 156\ kgf = 1530\ N;$$

Case 2. Single aileron control, loaded right-hand steering wheel**.**

Calculated loads:

$$P^c_{1p.el} = P^c_{1l.el}=171.4\cdot 2\cdot 1.25=428.5\ kgf=4202\ N;$$

$$M^c_{cr.el}=4500\ kgf\cdot cm=441\ N\cdot m.$$

Case 3. Single control of ailerons, loaded at the helm. Load one horn

$$P^c_{1p.el\cdot} = P^c_{1l.el}=171.4\ kgf=1681\ N;$$

$$M^c_{cr\ el}=7200\ kgf\cdot cm\ =706\ N\cdot m.$$

Case 4. Single compatible control P.B. and ailerons, loaded right horn.

Calculated loads

$$P^c_{el}=156\cdot kgf=1530\ N;$$

$$P^c_{el}=428.5\ kgf=4202\ N;$$

Case 5. Single control RV, loaded at the helm horns.

Calculated loads:

$$P^c_{t.r.b.} = P^c_{e.}=272\ kgf=2667\ N;$$

Calculation of the steering wheel

The worst case is the work of one pilot for one rudder, the combined deviation of the ailerons and the rudder

$$P^c{}_{el}=428.5\ kgf=4202\ N;$$

$$M^c{}_{cr\ el}=4500\ kgf{\cdot}cm\ =441\ N{\cdot}m;$$

$$P^c{}_e=78{\cdot}2=156\ kgf=1530\ N.$$

Since the steering wheel is made by casting of alloy ML5P4 having σv = 152 MPa, the coefficient.

Margin of safety is taken $f = 1.5$

$$P^c{}_e=156{\cdot}1.5=234\ kgf=2295\ N;$$

$$P^c{}_{el}=428.5{\cdot}1.5=642.75\ kgf=6303\ N.$$

According to the sketch design, the cross section is at a distance of 7.8 cm from the point of application of effort, and the section has the following dimensions (mm); Fig. (2.1)

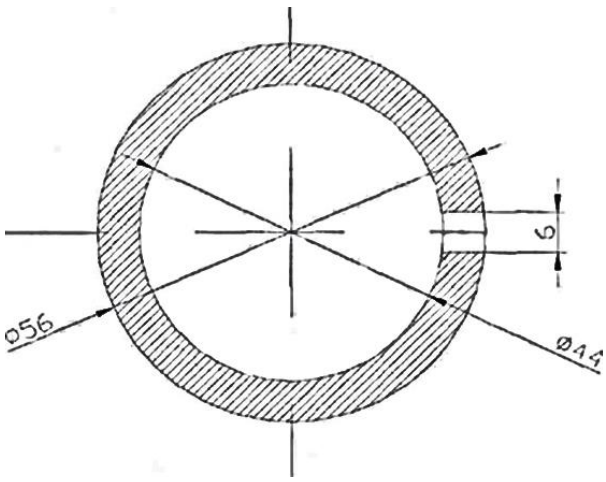

Fig. 2.1 Cross-Section 1-1

Find the cross-sectional area, the moment of bending resistance and the moment of resistance of torsion of the cross-section 1-1

$$F = S_3 - S_B - S_{OT} = 24.62 - 15.2 - 0.36 = 9.06 cm^2 = 9.06 \cdot 10^{-4} m^2;$$

$$W_b = \frac{\pi \cdot D^3}{32} \cdot \left[1 - \left(\frac{d}{D}\right)^4\right] = 10.66 cm^3 = 10.66 \cdot 10^{-6} m^3;$$

$$W_{cr} = \frac{\pi \cdot D^3}{16} \cdot \left[1 - \left(\frac{d}{D}\right)^4\right] = 21.32 cm^3 = 21.32 \cdot 10^{-6} m^3;$$

$$\sigma_p = \frac{P_{pb} \cdot 10.5}{W_b} - \frac{P_{pb}}{F} + \frac{P_{el} \cdot 7.8}{W_b} = 230.5 - 25.8 + 470.3 = 675 \cdot \frac{kgf}{cm^2} = 66 \cdot MPa;$$

$$T_{cr} = \frac{P_{el} \cdot 10.5}{W_{cr}} = \frac{642.75 \cdot 10.5}{21.32} = 316.5 \frac{kgf}{cm^2} = 31 \mathrm{MPa};$$

$$\sigma_c = \frac{P_{pb} \cdot 10.5}{W_b} + \frac{P_{pb}}{F} + \frac{P_{el} \cdot 7.8}{W_b} = 230.5 + 25.8 + 470.3 = 726.6 \frac{kgf}{cm^2} = 71 MPa;$$

$$\sigma = \sqrt{(\sigma_c)^2 + 4 \cdot (T_{cr})^2} = \sqrt{726.6^2 + 4 \cdot (316.5)^2} = 963.65 \frac{kgf}{cm^2} = 95 MPa;$$

The margin of safety is:

$$n = \frac{\sigma_{ns}}{\sigma} = \frac{152}{95} = 1.6;$$

Strength according to the third theory of strength is provided.

Calculation of crumpling of a wheel skeleton under a pin

Worst case: work with both hands on the roll channel

$$M^c_{cr.el} = 7200\ kgf \cdot cm = 706\ Nm;$$

$$D_{av} = 5\ cm = 0.05\ m;$$

n - is the number of cut planes

$$F = n \cdot h \cdot D_{sh} = 4 \cdot 0.6 \cdot 0.6 = 1.44cm^2 = 1.44 \cdot 10^{-4}m^2$$

$$\sigma_t = \frac{M_{cr\,\mathrm{el}}}{D_{av} \cdot F_t} = \frac{7200}{5 \cdot 1.44} = 1000\frac{kgf}{cm^2} = 98MPa$$

Steering

$$\sigma_b = 1550\frac{kgf}{cm^2} = 152MPa$$

Pin

$$\sigma_b = 11000\frac{kgf}{cm^2} = 1079MPa$$

The crumpling strength of the pin and the steering wheel is provided.

Calculation of pins on a cut

$$D_{av}=4.4cm=4.4\cdot 10^{-2}\ m;$$

$$D_{sh}=0.6\ cm=6\cdot 10^{-3}\ m;$$

$$n=4$$

$$M^{c}_{cr\ el}=7200\ кgf\cdot cm=706\ N\cdot m;$$

$$\sigma_b = 11000\frac{kgf}{cm^2} = 1079MPa;$$

$$F_{ct} = n \cdot \frac{\pi\cdot(D_{sh})^2}{4} = 4 \cdot \frac{3.14\cdot 0.6^2}{4} = 1.13cm^2 = 1.13 \cdot 10^{-4}m^2;$$

$$T_{av} = \frac{M_{crel}}{D_{av}\cdot F_{ct}} = \frac{7200}{4.4\cdot 1.13} = 1447.6\frac{kgf}{cm^2} = 142MPa;$$

$$T_b = 0.6 \cdot \sigma_b = 0.6 \cdot 11000 = 7150\frac{kgf}{cm^2} = 701MPa;$$

The margin of safety is

$$n = \frac{T_b}{T_{av}} = \frac{701}{142} = 4.94.$$

The strength under the condition of the cut of the pin is provided.

Calculation of the steering shaft in a critical place for torsion Fig. (2.1)

$$M^{c}_{cr\ el} = 7200\ кгf{\cdot}cm = 706\ N{\cdot}m;$$

$$D_{av} = 4\ cm = 4{\cdot}10^{-2}\ m;$$

$$h = 0.4\ cm = 4{\cdot}10^{-3} m \text{ – wall thickness;}$$

$$D_{z} = 4.4\ cm = 4.4{\cdot}10^{-2}\ m;$$

$$d_{b} = 3.6\ cm = 3.6{\cdot}10^{-2}\ m;$$

$$d_{wt} = 26\ cm = 26{\cdot}10^{-2}\ m$$

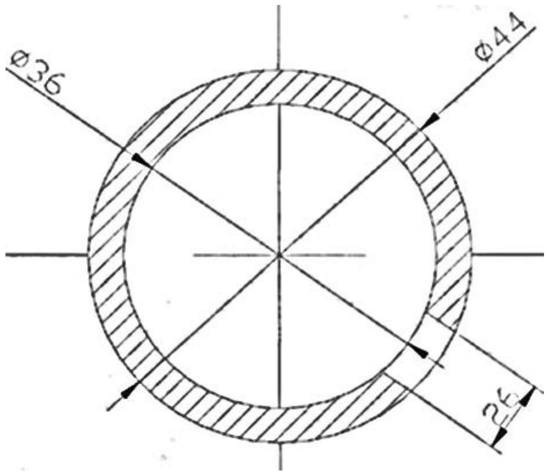

Fig. 2.2 Critical section of steering shaft.

$$\sigma_b = 4000\frac{kgf}{cm^2} = 392MPa;$$

$$W_{cr} = \frac{\pi{\cdot}D_{av}{\cdot}h^2}{3} = \frac{3.14{\cdot}4{\cdot}0.4^2}{3} = 0.67\ cm^3 = 0.67 \cdot 10^{-6} m^3;$$

$$T_{cr} = \frac{M_{crel}}{W_{cr}} = \frac{7200}{0.67} = 10746.3\,\frac{kgf}{cm^2} = 1054MPa;$$

$$T_b = 0.65 \cdot 4000 = 2600\,\frac{kgf}{cm^2} = 255MPa;$$

We select the sizes of section.

$$(T_{cr}) = \frac{3 \cdot M_{crel}}{\pi \cdot D_{av} \cdot h^2};$$

$$h = \sqrt{\frac{3 \cdot M_{crel}}{\pi \cdot D_{av} \cdot (T_{cr})}} = \sqrt{\frac{3 \cdot 706}{3.14 \cdot 0.04 \cdot 255 \cdot 10^6}} = 0.008m;$$

In this section, the outer diameter should be Ds = 4.8 cm

Calculation of the propeller fork

$$H=4.7cm=4.7 \cdot 10^{-2}\ m;$$

$$A=0.7cm=7 \cdot 10^{-3}\ m;$$

$$B=5\ cm=5 \cdot 10^{-2}\ m;$$

$$M^{c}_{cr\ el}=7200\ кgf \cdot cm=706\ N \cdot m;$$

$$L_{arm}=3\ cm=3 \cdot 10^{-2}\ m;$$

$$\sigma_b=40\ кgf/mm^2=392MPa.$$

Since it is a fork, the coefficient is introduced. margin of safety $f_{dop} = 1.25$

$$M^{P}_{cr\ el}=9000\ кgf \cdot cm=883\ N \cdot m.$$

Calculation of the critical cross section for bending.

$$P^p = \frac{M_{cr}}{H} = \frac{9000}{4.7} = 1914.9kgf = 18779N;$$

$$W_b = \frac{A \cdot B^2}{6} = \frac{0.7 \cdot 5^2}{6} = 2.92cm^3 = 2.92 \cdot 10^{-6}m^3;$$

$$\sigma_b = \frac{P^p \cdot L_{arm}}{W_b} = \frac{1914.93}{2.92} = 1967.4\frac{kgf}{cm^2} = 193MPa;$$

$$T_{av} = \frac{P^p}{F_{ct}} = \frac{1914.9}{5 \cdot 0.7} = 547.1\frac{kgf}{cm^2} = 54MPa;$$

$$\sigma = \sqrt{(\sigma_b)^2 + 4 \cdot (T_{av})^2} = \sqrt{1967.4^2 + 547.1^2} = 2246\frac{kgf}{cm^2} = 220MPa.$$

The margin of safety is

$$n = \frac{\sigma_b}{\sigma} = \frac{392}{220} = 1.78;$$

Strength according to the third theory of strength is provided.

Calculation of the square ring Fig. (2.3)

Material 07X16 H6, σ_b=110 kgs/mm^2 =1079 MPa

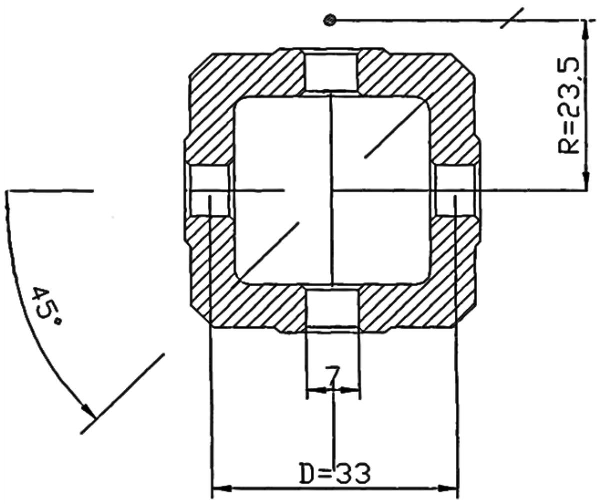

Fig. 2.3 Square ring

Since the most critical place is located at an angle of 45 degrees, this section is calculated for the gap and bend.

The cross section has the following dimensions:

$$A = 1.6\ cm = 1.6 \cdot 10^{-2}\ m;$$

$$B = 0.8\ cm = 8 \cdot 10^{-3}\ m.$$

Calculated load

$$M^{c}_{cr.el} = 7200\ \kappa gf \cdot cm = 706\ N \cdot m;$$

$$F = 1.6 \cdot 0.8 = 1.28\ cm^2 = 1.28 \cdot 10^{-4} m^2;$$

$$W_b = \frac{A \cdot B^2}{6} = \frac{1.6 \cdot 0.8^2}{6} = 0.17 cm^3 = 0.17 \cdot 10^{-6} m^3;$$

$$\sigma_p = \frac{1.415 \cdot M_{crel}}{2 \cdot R \cdot 2 \cdot F \cdot cos(45)} = \frac{1.415 \cdot 7200}{2 \cdot 2.35 \cdot 2 \cdot 1.28 \cdot 0.707} = 1197.65 \frac{kgf}{cm^2} = 117 MPa;$$

$$\sigma_b = \frac{1.415 \cdot M_{crel} \cdot 3.3}{4 \cdot \pi \cdot 2 \cdot W_b \cdot cos(45)} = \frac{1.415 \cdot 7200 \cdot 3.3}{4 \cdot 3.14 \cdot 2 \cdot 0.17 \cdot 0.707} = 9477.13 \frac{kgf}{cm^2} = 929 MPa;$$

$$\sigma = \sigma_b + \sigma_p = 9477.13 + 1197.65 = 10674.78 \frac{kgf}{cm^2} = 1046 MPa;$$

The margin of safety is

$$n = \frac{\sigma_b}{\sigma} = \frac{1079}{1046} = 1.03;$$

Strength as a result of calculation is provided.

Calculation of a rocking chair

Calculation of the force transmitted to the thrust number 1

$$R=7\ cm=7\cdot 10^{-2}\ m;$$

$$M^{c}_{cr.el}=7200\ кgf\cdot см=706\ N\cdot m;$$

$$P_{thrust}=\frac{M_{cr}}{R}=\frac{7200}{7}=1028.6kgf=10087\cdot N.$$

Calculation of thrust at break

$$D_{thrust}=3.6\ cm=3.6\cdot 10^{-2}\ m;$$

$$h=0.2\ cm=2\cdot 10^{-3}\ m;$$

$$F=\pi\cdot D_{thrust}\cdot h=3.14\cdot 3.6\cdot 0.2=2.26\ cm^2=2.26\cdot 10^{-4}\ m^2;$$

$$\sigma_o=\frac{P_{thrust}}{F}=\frac{1028.6}{2.26}=455\frac{kgf}{cm^2}=45MPa.$$

The margin of safety is

$$n=\frac{\sigma_b}{\sigma_o}=\frac{392}{45}=8.71.$$

Strength is provided.

Calculation of crumpling of a framework of a rocking chair under the plug.

$$F_{dop}=1.25.$$

$$\sigma_o = \frac{f \cdot P_{thrust}}{n \cdot F_o} = \frac{1028.6 \cdot 1.25}{2 \cdot 0.5 \cdot 0.8} = 1607.2 \frac{kgf}{cm^2} = 158 MPa;$$

Strength is provided.

Calculation of a bolt of fastening on a cut.

Material 30ХГСА.

$$\sigma_b = 11000 \frac{kgf}{cm^2} = 1079 MPa;$$

$$D_b = 0.6\ cm = 6 \cdot 10^{-3}\ m;$$

The bolt has two cut planes.

$$T_{ct} = \frac{f \cdot P_{thrust}}{n \cdot F_{ct}} = \frac{1.25 \cdot 1028.6}{2 \cdot 0.2826} = 2274.9 \frac{kgf}{cm^2} 223 MPa;$$

$$T_b = 0.6 \cdot \sigma_b = 0.6 \cdot 11000 = 7150 \frac{kgf}{cm^2} = 701 MPa.$$

The margin of safety is

$$n = \frac{T_b}{T_{av}} = \frac{701}{223} = 3.14;$$

Strength is provided.

Calculation of the rocking chair in the neutral position of the steering wheel. Strength

$$P_{thrust1} = 128575\ кgf = 1261\ kN;$$

Traction force number 2

$$P_{thrust} = \frac{P_{thrust}\cdot 8.7}{8} 1118.6kgf = 10970N;$$

The force transmitted to the axis of rotation of the rocking chair.

$$P_{axes} = \sqrt{(P_{\text{т}1})^2 + (P_{\text{т}2})^2} = \sqrt{1028.6^2 + 1118.6^2} = 1519.6kgf = 14902N.$$

$$\sigma_b = 11000\frac{kgf}{cm^2} = 1079MPa;$$

$$D_b = 0.6\ cm = 6\cdot 10^{-3}\ m;$$

$$T_{ct} = \frac{f\cdot P_{axes}}{n\cdot F_{ct}} = \frac{1899.54}{2\cdot 0.2826} = 3360.1\frac{kgf}{cm^2} = 330MPa;$$

$$T_b = 0.6\cdot\sigma_b = 0.6\cdot 11000 = 7150\frac{kgf}{cm^2} = 701MPa;$$

The margin of safety is

$$n = \frac{T_b}{T_{ct}} = \frac{701}{330} = 2.12;$$

Strength is provided.

Calculation of the rocking chair in the most dangerous place in section 1-1.Fig (2.4)

$$P_{thrust2} = 1118.6\ \text{к}gf = 10970\ N;$$

$$W_b = \frac{2 \cdot 2.4 \cdot 0.5^2}{6} = 0.96cm^3 = 0.96 \cdot 10^{-6}m^3.$$

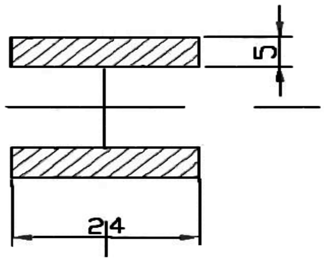

Fig. 2.4 Calculation of the rocking chair

$$\sigma_b = \frac{P^p \cdot L_{arm} \cdot f}{W_b} = \frac{1118.6 \cdot 2 \cdot 1.25}{0.96} = 2913 \frac{kgf}{cm^2} = 286MPa.$$

The margin of safety is

$$n = \frac{\sigma_{ns}}{\sigma_b} = \frac{412}{286} = 1.44;$$

Strength is provided.

So in the extreme position of the rocking chair the angle between the rod №1 and tangent to the rocking chair at this point is 7 degrees, then the increase in thrust will be negligible, and the koeff. of strength reserve in the rocking chair is big enough, then there is no point in holding calculations in a rocking chair in extreme positions.

Calculation of supports of fastening of a steering column (Fig. 2.5, 2.6)

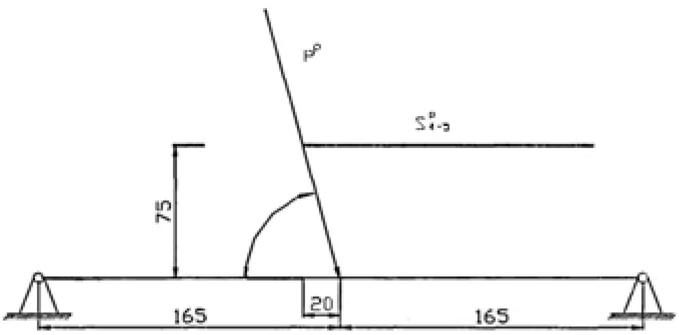

Fig. 2.5 Calculation scheme 1

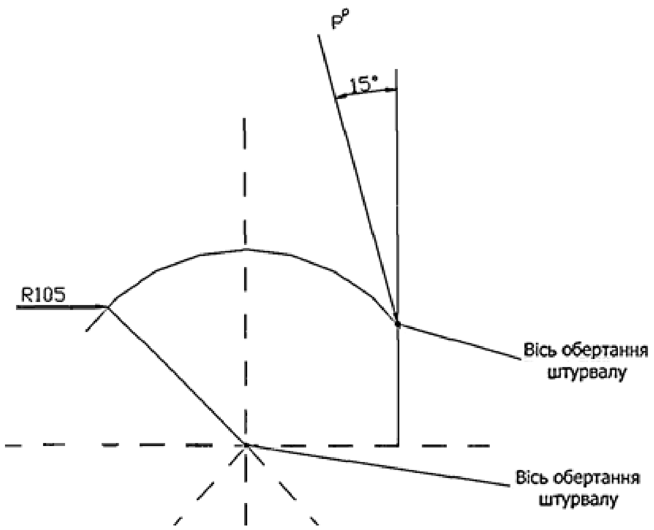

Fig. 2.6 Calculation scheme 2

The calculation is performed in the neutral position of the steering wheel the work of the pilot with both hands

Calculation of forces on supports in the O-Y plane

$$S_2=1118.6\ \text{кgf}=10970\ N;$$

$$P_{el}=342.8\ \text{кgf}=3362\ N.$$

The shoulders are indicated in the calculation diagram

$$P_{ya} = -P_{\mathrm{el}} \cdot \cos(15) + S_2 \cdot \frac{7.5}{2 \cdot 16.5} = -342.8 \cdot 0.965 + 1118.6 \cdot 0.227 = -76.88 kgf = -754N;$$

$$P_{y\text{в}} = -P_{\mathrm{el}} \cdot \cos(15) - S_2 \cdot \frac{7.5}{2 \cdot 16.5} = -342.8 \cdot 0.965 + 1118.6 \cdot 0.227 = -584.72 kgf = -5794N;$$

Sh. № doc. Sign Date

$$P_{za(\text{в})} = S_2 + P_{el} \cdot \sin(15) = 342.8 + 1118.6 \cdot 0.259 = 623.3kgf = 6200N.$$

Calculation of forces on supports in the O-X plane

$$S_z\text{=802.7 кgf=7872 N;}$$

$$P_e\text{=272 кgf=2667 N.}$$

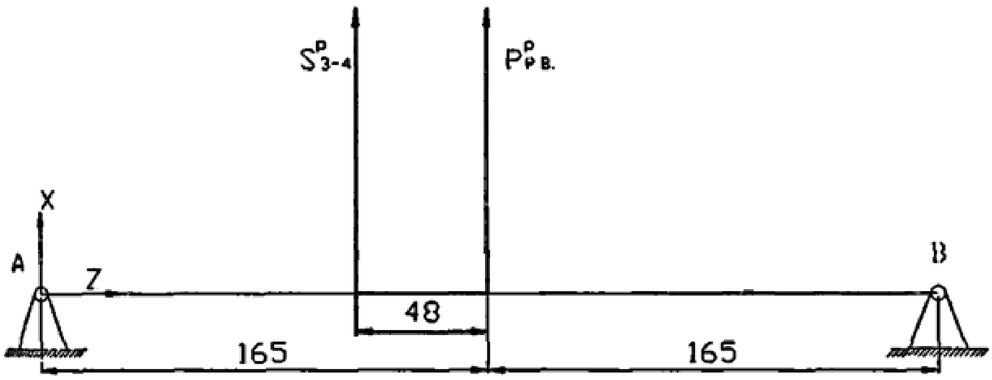

Fig. 2.7 Calculation scheme 3

$$P_{xa} = \frac{S_Z \cdot (16.5+4.8)+P_{\text{рв}}}{33} = \frac{802.7 \cdot 21.3+272}{33} = 526.3kgf = 5161N;$$

$$P_{xb} = \frac{S_Z \cdot (16.5+4.8)+P_{\text{рв}}}{33} = \frac{802.7 \cdot 11.7+272}{33} = 292.8kgf = 2871N;$$

Efforts in the direction O-X cause a cut of bolts of fastening of an arm.
Efforts in the O-Y direction cause the bracket mounting bolts to come off.
According to the preliminary design, the diameters of the bolts.
DB = 6 · 10-3 m made of material 30ХГСА

$$P_{AX}\text{=526.3 kgf=5161 N;}$$

$$P_{AY}\text{=78.88 kgf=754 N;}$$

$$P_{BX}=292.8\ kgf=2871\ N;$$

$$P_{BY}=584.72\ kgf=5734\ N;$$

$$F_{OT}=F_{av}=0.2826\ cm^2=0.2826\cdot 10^{-4}\ m^2;$$

n = 4 – number of bolts

$$\sigma_{\text{ОТ}}=\frac{P_{max}}{F_{\text{ОТ}\cdot p}}=\frac{584.72}{0.2826\cdot 4}=517.3\frac{kgf}{cm^2}=51MPa;$$

$$T_{ct}=\frac{P_{max}}{n\cdot F_{ct}}=\frac{526.3}{4.0.2826}=465.6\frac{kgf}{cm^2}=46MPa;$$

$$\sigma=\sqrt{(\sigma_{\text{ОТ}})^2+4\cdot(T_{ct})^2}=\sqrt{517.3^2+4\cdot(465.6^2)}=1065.2\frac{kgf}{cm^2}=104MPa;$$

$$T_b=0.6\cdot\sigma_b=0.6\cdot 11000=7150\frac{kgs}{cm^2}=701MPa;$$

The margin of safety is

$$n=\frac{T_b}{T_{ct}}=\frac{701}{104}=6.74;$$

Strength according to the third theory of strength is provided.

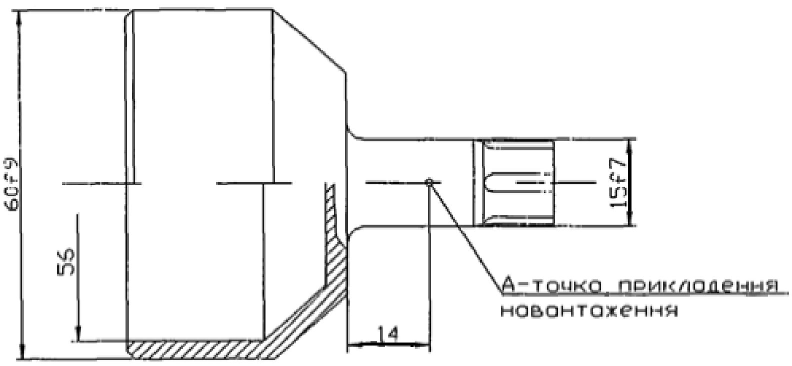

Fig. 2.8 Pin

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$$D = 15 \cdot 10^{-3}\ m;$$

$$F = 1{,}767 cm^2 = 1.767 \cdot 10^{-4}\ m^2;$$

$$W = 0.3313\ cm^2 = 0.3313 \cdot 10^{-3}\ m^2;$$

$$P_{AX} = 526.3\ кgf = 5161\ N.$$

Calculated case: Single control P.B.

$$\sigma_b = \frac{P_{ax} \cdot L_{arm}}{W_b} = \frac{526.3 \cdot 0.14}{0.3313} = 222.4 \frac{kgs}{cm^2} = 22 MPa;$$

$$\tau_{ct} = \frac{P_{ax}}{F_{ct}} = \frac{526.3}{1.767} = 297.8 \frac{kgs}{cm^2} = 29 MPa;$$

$$\sigma = \sqrt{(\sigma_b)^2 + 4 \cdot (\tau_{ct})^2} = \sqrt{(222.4)^2 + 4 \cdot (297.8)^2} = 635.8 \frac{kgs}{cm^2} = 62 MPa.$$

The margin of safety is

$$n = \frac{\sigma_b}{\sigma_{ct}} = \frac{1079}{62} = 17.4;$$

Strength according to the third theory of strength is provided.

2.1.2. Calculation of the loading spring PoSS-70

Case 1. Single control. Position of steering columns "on itself".

Load of the right steering column "from itself". Start of coupling disconnection.

Calculated loads:

$$P^c_{pr.} = 3 \cdot 75 = 225\ кgf = 2206\ N;$$

Case2. Single control. Position of steering columns "on itself". Load of the right steering column "from itself".

Final coupling disconnection.

Calculated loads:

$$P^{c}_{pr.} = 3 \cdot 90 = 270 кgf = 2648\ N;$$

Case 3. Dual control. Neutral position of the steering columns.

$$P^{c}_{pr.} = 3 \cdot 75 = 225\ кgf = 2206\ N.$$

Loading spring

Calculated pre-tightening effort:

$$P^{c}_{pret.} = P^{c}_{pret.} \cdot f = 70 \cdot 3 = 210\ кgf = 2059\ N.$$

The design force of the loading spring corresponds to the maximum stroke:

$$P^{c}_{max.} = P^{c}_{max.} \cdot f = 90 \cdot 3 = 270\ кgf = 2648\ N;$$

Ware 65С2ВА-ХН-9 ГОСТ 14963-78;

$$T_{dop} = 8400\ кgf/cm^2 = 824\ MPa;$$

$$G = 7.6 \cdot 10^5\ кgf/cm^2 = 74.531 \cdot 10^9\ MPa.$$

Spring parameters:

n = 8 – number of turns

d = 9 mm – wire diameter

$D = 53$ mm – the outer diameter of the spring

$D_0 = D - d = 53 - 9 = 44$ mm – the average diameter of the spring.

Coefficient taking into account the curvature of the turn

$$k = f \cdot \left(\frac{D_0}{d}\right) = 1.32;$$

$H_0 = 135$ mm – height of the spring in the free state;

Calculation of torsion springs.

Calculated case 2:

$$P^c_{pr} = 90\ \kappa gf = 883\ N;$$

Tangential torsion springs:

$$\tau = \frac{8 \cdot P \cdot D_0 \cdot k}{\pi \cdot d^3} = \frac{8 \cdot 90 \cdot 4.4 \cdot 1.32}{3.14 \cdot 0.9^3} = \frac{1827 kgs}{cm^2} = 179 MPa;$$

$$[\tau_{dop}] = 824 MPa;$$

$$\eta = \frac{824}{179} = 4.6 \geq 2.$$

Calculation of spring for stability

$$H / D_0 = 135/44 = 3.07;$$

Since the value of $N / D_0 < 5.1$, the loss of spring stability is impossible.

Calculation of the threaded connection of the glass

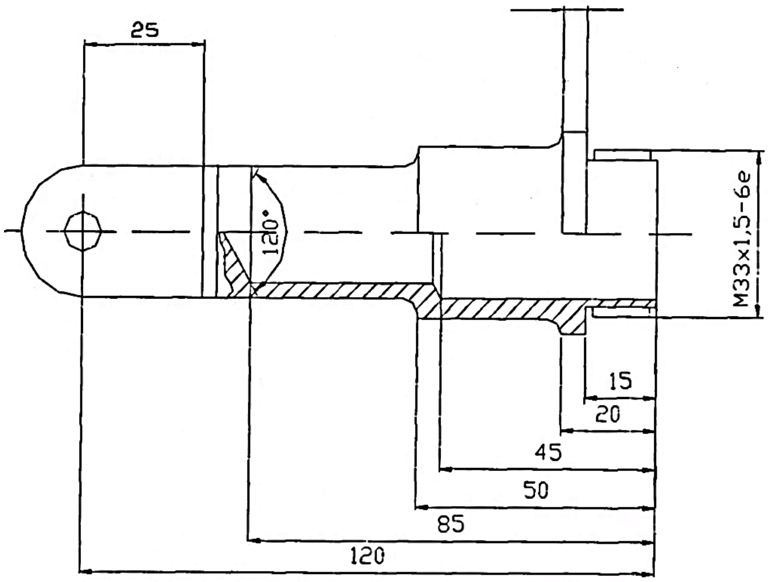

Fig. 2.9 Threaded connection of the glass

Case 2.

Calculated loads:

$$P_{pr}^{c} = 3 \cdot 90 = 270kgf = 2648N;$$

Crushing of a carving of the case under a glass

$$P_{pct}^{c} = 3 \cdot 90 = 270kgf = 2648N;$$

$$d_{zn} = 30mm;$$

$$S^{I} = 1{,}5;$$

$$\sigma_{ct} = \frac{P_{prct}^{c}}{\pi \cdot d_{zn} \cdot S^{I}} = \frac{19.1kgf}{cm^2} = 1.87MPa;$$

$$\sigma = \frac{40kgf}{cm^2} = 3.92MPa;$$

$$\eta = \frac{3.92}{1.87} = 2.1 \geq 2$$

Crushing the thread of the glass under the body

$$P_{prct}^{c} = 3 \cdot 90 = 270 kgf = 2648 N;$$

$$d_{zn} = 30\ mm;$$

$$S^{I} = 1{,}5;$$

$$\sigma_{pr} = \frac{P_{prct}^{c}}{\pi \cdot d_{zn} \cdot S^{I}} = 19.1 kgf/cm^2 = 1.87 MPa;$$

$$\sigma = 38 kgf/cm^2 = 3.73 MPa;$$

$$\sigma = \frac{38 kgf}{cm^2} = 3.73 MPa$$

$$\eta = \frac{3.73}{1.87} \geq 2$$

Conclusion of special part

In this part the main strength calculations of designed control system elements were performed. That made the creation of the drawing outline possible.

During the calculation the main geometrical parameters caused by operational purpose, bending and torsion, strength of elements, were considered. All obtained values meet the requirements for the short range passenger aircrafts. Geometrical parameters almost match with chosen prototypes. That fact allows to make a conclusion that designed control system elements will successfully concur with another models on the chosen market segment.

General conclusions

During this designing work I’ve got the next results:

- preliminary design of the short range aircraft with 50 passengers;
- the schematic design of the layout of the short range aircraft with 50 passengers;
- the center of gravity of the airplane calculations;
- the calculation of the main geometrical parameters of the control system;
- the design of control system elements;

Designed aircraft satisfies the planned aim of usage, its geometrical characteristics will provide the necessary aerodynamic performance, which will lead to efficient usage.

In the second part the main strength calculations of designed control system elements were performed. That made the creation of the drawing outline possible.

During the calculation the main geometrical parameters caused by operational purpose, bending and torsion, strength of elements, were considered. All obtained values meet the requirements for the short range passenger aircrafts.

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Appendix

					NAU 20 05Z 00 00 00 99 EN
Done by	Zozulia A.I.			Appendix	list / sheet / sheets
Checked by	Krasnopolskii V.S.				— / 57 / 60
St.control.	Khyzhniak S.V.				AF 402 134
Head of dep.	Ignatovich S.R.				

ПРОЕКТ
САМОЛЕТА С ТВД
НАУ, кафедра КЛА

ПРОЕКТ diploma Расчет выполнен 23.09.2019
Исполнитель Zozulya A.I. Руководитель Krasnopolskii V.S.

ИСХОДНЫЕ ДАННЫЕ И ВЫБРАННЫЕ ПАРАМЕТРЫ

Количество пассажиров	50.
Количество членов экипажа	2.
Количество бортпроводников или сопровождающих	1.
Масса снаряжения и служебного груза	353.67 кг.
Масса коммерческой нагрузки	4377.50 кг.
Крейсерская скорость полета	450. км/ч
Число "М" полета при крейсерской скорости	0.3942
Расчетная высота начала реализации полетов с крейсерской экономической скоростью	6.000 км
Дальность полета с максимальной коммерческой нагрузкой	750. км.
Длина летной полосы аэродрома базирования	1.90 км.
Количество двигателей	2.
Оценка по статистике энерговооруженности в квт/кг	0.1800
Степень повышения давления	15.00
Относительная масса топлива по статистике	0.2000
Удлинение крыла	11.37
Сужение крыла	2.90
Средняя относительная толщина крыла	0.120
Стреловидность крыла по 0.25 хорд	6.0 град.
Степень механизированности крыла	0.580
Относительная площадь прикорневых наплывов	0.000

Профиль крыла - Ламинизированный типа NACA
Шайбы УИТКОМБА - не применяются
Спойлеры - установлены

Диаметр фюзеляжа	2.65 м.
Удлинение фюзеляжа	8.00
Стреловидность горизонтального оперения	15.0 град.
Стреловидность вертикального оперения	20.0 град.

РЕЗУЛЬТАТЫ РАСЧЕТА
НАУ, КАФЕДРА "КЛА"

Значение оптимального коэффициента под'емной силы в расчетной точке крейсерского режима полета Су 0.49680

Значение коэффициента Сх.инд. 0.00994

ОПРЕДЕЛЕНИЕ КОЭФФИЦИЕНТА Dм = Мкрит - Мкрейс

Число Маха крейсерское	Мкрейс	0.39421
Число Маха волнового кризиса	Мкрит	0.68116
Вычисленное значение	Dm	0.28694

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

при взлете	2.637
в середине крейсерского участка	2.562
в начале крейсерского участка	2.584

Значение коэффициента сопротивления фюзеляжа и гондол	0.00985
Значение коэфф. профиль. сопротивления крыла и оперения	0.00995
Значение коэффициента сопротивления самолета:	
в начале крейсерского режима	0.03328
в середине крейсерского режима	0.03319
Среднее значение Су при условном полете по потолкам	0.49680
Среднее крейсерское качество самолета	14.96679
Значение коэффициента Су.пос.	1.556
Значение коэффициента (при скорости сваливания) Су.пос.макс.	2.334
Значение коэффициента (при скорости сваливания) Су.взл.макс.	2.074
Значение коэффициента Су.отр.	1.493
Энерговооруженность в начале крейсерского режима	0.100
Стартовая энерговооруженн. по условиям крейс. режима No.кр.	0.145
Стартовая энерговоруж. по условиям безопасного взлета No.взл.	0.146

Расчетная энерговоруженность самолета No 0.150

Отношение Dn = No.кр / No.взл Dn 0.996

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

взлетный	0.3034
крейсерский (характеристика двигателя)	0.2603
средний крейсерский при заданной дальности полета	0.2611

ОТНОСИТЕЛЬНЫЕ МАССЫ ТОПЛИВА:

аэронавигационный запас	0.02030
расходуемая масса топлива	0.06738

ЗНАЧЕНИЯ ОТНОСИТЕЛЬНЫХ МАСС:

крыла	0.15047
горизонтального оперения	0.01799
вертикального оперения	0.01782
шасси	0.05217
силовой установки	0.11394
фюзеляжа	0.11076
оборудования и управления	0.16761
дополнительного оснащения	0.00294
служебной нагрузки	0.02082
топлива при Lрасч.	0.08769
коммерческой нагрузки	0.25771

Взлетная масса самолета "М.о" = 16986. кГ.
Потребная взлетная мощность двигателя 1276.8 kВт

Относительная масса высотного оборудования и противообледенительной системы самолета	0.0277
Относительная масса пассажирского оборудования (или оборудования кабин грузового самолета)	0.0206
Относительная масса декоративной обшивки и ТЗИ	0.0133
Относительная масса бытового (или грузового) оборудования	0.0082
Относительная масса управления	0.0129
Относительная масса гидросистем	0.0304

Относительная масса электрооборудования	0.0300
Относительная масса локационного оборудования	0.0047
Относительная масса навигационного оборудования	0.0070
Относительная масса радиосвязного оборудования	0.0035
Относительная масса приборного оборудования	0.0082
Относительная масса топливной системы (входит в массу "СУ")	0.0025
Дополнительное оснащение:	
Относительная масса контейнерного оборудования	0.0000
Относительная масса нетипичного оборудования	0.0029

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

ХАРАКТЕРИСТИКИ ВЗЛЕТНОЙ ДИСТАНЦИИ

Скорость отрыва самолета	189.97 км/ч
Ускорение при разбеге	1.72 м/с*с
Длина разбега самолета	806. м.
Дистанция набора безопасной высоты	409. м.
Взлетная дистанция	1215. м.

ХАРАКТЕРИСТИКИ ВЗЛЕТНОЙ ДИСТАНЦИИ ПРОДОЛЖЕННОГО ВЗЛЕТА

Скорость принятия решения	180.47 км/ч
Среднее ускорение при продолженном взлете на мокрой ВПП	0.17 м/с*с
Длина разбега при продолженном взлете на мокрой ВПП	1552.15 м.
Взлетная дистанция продолженного взлета	1921.90 м.
Потребная длина летной полосы по условиям прерванного взлета	2002.81 м.

ХАРАКТЕРИСТИКИ ПОСАДОЧНОЙ ДИСТАНЦИИ

Максимальная посадочная масса самолета	16511.	кг.
Время снижения с высоты эшелона до высоты полета по кругу	12.0	мин.
Дистанция снижения	15.05	км.
Скорость захода на посадку	198.65	км/ч.
Средняя вертикальная скорость снижения	1.68	м/с
Дистанция воздушного участка	378.	м.
Посадочная скорость	187.13	км/ч.
Длина пробега	573.	м.
Посадочная дистанция	950.	м.
Потребная длина летной полосы (ВПП + КПБ) для основного аэродрома	**1587.**	**м.**
Потребная длина летной полосы для запасного аэродрома	**1350.**	**м.**

ПОКАЗАТЕЛИ ЭФФЕКТИВНОСТИ САМОЛЕТА

Отношение массы снаряженного самолета к массе коммерческой нагрузки	2.5284
Масса пустого снаряженного с-та приход. на 1 пассажира	221.36 кг/пас.
Относительная производительность по полной нагрузке	155.43 км/ч
Производительность с-та при макс. коммерч. нагрузке	1669.4 т*км/ч
Средний часовой расход топлива	581.991 кг/ч
Средний километровый расход топлива	1.53 кг/км
Средний расход топлива на тоннокилометр	348.626 г/(т*км)
Средний расход топлива на пассажирокилометр	28.6146 г/(пас.*км)
Ориентировочная оценка приведен. затрат на тоннокилометр	1.3636 $/(т*км)