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 вантажного ближньомагістрального літака вантажопідйомністю до 5,5 тон»

Виконала: ____________ **В.Д. Кравченко**

Керівник: д-р техн. наук., проф. ____________ **М.В. Карускевич**

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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 y.

DIPLOMA WORK

(EXPLANATORY NOTE)
OF ACADEMIC DEGREE
«BACHELOR»

Theme: «Preliminary design of cargo short-range aircraft with cargo capacity 5,5 tons»

Performed by: __________ **V.D. Kravchenko**

Supervisor: Doctor of science, Professor __________ **M. V. Karuskevich**

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

Kyiv 2020

NATIONAL AVIATION UNIVERSITY

Aerospace Faculty

Department of Aircraft Design

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

KRAVCHENKO VALERIIA

1. Theme: «Preliminary design of of cargo short range aircraft with cargo capacity 5,5 tons»

   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}=440 km/h, flight range L=1100 km, operating altitude H_{op}=6 km, max payload is 5500 kg.

4. Explanation note argument (list of topics to be developed): choice and substantiations of the airplane scheme, choice of initial data; engine selection, aircraft layout, center of gravity position calculation, designing of new winch component.

5. List of the graphical materials: general view of the airplane (A1×1); layout of the airplane (A1×1); overhead cargo equipment drawing (A1×1); hook suspension drawing (A1×1).

Graphical materials are performed in AutoCad, SolidWorks.

6. Calendar Plan

Task	Execution period	Signature
Task receiving, processing of statistical data	25.05.2020–28.05.2020	
Aircraft geometry calculation	29.05.2020–30.05.2020	
Aircraft layout	29.05.2020–31.05.2020	
Aircraft centering	31.05.2020–04.06.2020	
Graphical design of the parts	02.06.2020–10.06.2020	
Preliminary defence	10.06.2020–10.06.2020	
Completion of the explanation note	11.06.2020–15.06.2020	

7. Task issuance date: 25.05.2020 year

Supervisor of diploma work ____________ M.V. Karuskevich

Task for execution is given for ____________ V.D. Kravchenko

ABSTRACT

Explanatory note to the diploma work «Preliminary design of cargo short-range aircraft with cargo capacity 5,5 tons» contains:

sheets, figures, tables, references, poster and 4 drawings

Object of the design is development of cargo short-range aircraft with cargo capacity 5,5 tons.

Subject of the design – the conceptual design of the hook suspension improvement with stress-strain analysis of the hook.

The aim of the diploma work is the preliminary design of the aircraft and its design characteristic estimation.

The method of design is analysis of the prototypes and selections of the most advanced technical decisions, the geometrical characteristics estimation, centre of gravity calculations of the designing aircraft, stress-strain analysis of the hook and its modification.

The diploma work contains drawings of the short-range aircraft with a carrying capacity of 5,5 tons, calculations and drawings of the aircraft layout and hook suspension design.

The materials of the diploma could be recommended for the students of aviation specialties, for the aircraft operational companies, also it can be used for the design bureaus.

AIRCRAFT, PRELIMININARY DESIGN, LAYOUT, CENTER OF GRAVITY POSITION, OVERHEAD CARGO EQUIPMENT, HOOK SUSPENSION

CONTENT

				NAU 20 08K 00 00 00 56 EN			
Performed	Kravchenko			CONTENT	Letter	Sheet	Sheets
Supervisor	Karuskevich						
Adviser					402 AF 134		
Stand.contr.	Khizhnyak S.						
Head of dep.	Ignatovych S.						

				NAU 20 08K 00 00 00 56 EN		
Performed	Kravchenko			CONTENT	Letter	Sheet	Sheets
Supervisor	Karuskevich						
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Stand.contr.	Khizhnyak S.						
Head of dep.	Ignatovych S.						

INTRODUCTION

Air transportation plays a major role in the world economy. According to Boeing forecasts, over the next 20 years, the annual growth of air cargo was 4.7%, which means that air cargo in 2033 will increase almost 2 times.

In 2018, the global volume of regular freight transport, measured in tonne-kilometre freight, grew by 4.5%. In the segment of international air cargo transportation, which accounts for almost 87% of all air cargo transportation, the growth was about 4.6%, and the load factor on international scheduled cargo flights reached approximately the level, which amounted to about 55%. We can observe an increase in demand for freight transportation every year. But first of all, it is worth considering which planes it is worth using, what will be more economically profitable and what range to count on when choosing an airplane.

Small aircraft are not so picky about the infrastructure of the airport, moreover, small planes can get along the runway simpler and shorter.

The diploma paper presented below comprises the decision of two problems: preliminary design of short-range plane and development of the winch component.

The aircraft is a high-wing construction with a typical tail unit with two AI-24VT turboprop engines with a take-off power of 2820 hp each, mounted on the wing. The engine is equipped with a four-blade vane AB-72T propellers. Auxiliary power unit (APU) - RU-19A-300 turbojet engine. Rudder and elevators are equipped with aerodynamic balance.

In the rear of the fuselage is a cargo hatch for loading and unloading cargo. A special ramp closes the cargo hatch and can be used as a gangway for loading and unloading equipment and goods. Sliding the ramp under the fuselage ensures that vehicles approach the threshold of the cargo compartment and load the aircraft directly from the side of the vehicle.

The project reflects demand on regional commercial transportation and the inability to deliver cargo to local places due to poor airfield conditions. There is a need for cargo aircraft with a relatively short range, competing payload and, most importantly, unpretentious landing conditions, which is a remarkably difficult task.

The requirements of national and international (FAR-25, CS-25) airworthiness lows, rules, recommendations, as well as current trends have been taken into account at the process of the aircraft design. These are high level of flight safety; low fuel consumption; unpretentious landing and take-off conditions; short take-off and landing distances; high payload efficiency, ease of operation.

A special part of the work deals with the development of the hook suspension. The key idea of this design is to install the scale and safe lock to the hook suspension.

1. PRELIMINARY DESIGN OF THE PLANE

1.1. Analysis of similar planes

Among the numerous procedures of the preliminary design stage, the accumulation and analysis of data related to the previous successful machines play an important role. The analysis of the aircraft can be divided into these types of parameters, which in the future will represent the achievement of high economic efficiency of the plane: aerodynamics, bulk cargo, structural strength and fuel consumption.

This procedure can be called as an “Analysis and Synthesis”. For the plane designed in the frame of presented work, the prototypes have been selected on the base of similar take-off weight, cruising speed, cruising altitude, cargo capacity, fuel consumption, etc. The primary role in this selection plays, of course, the question “How much the former planes are successful?”

When choosing the design parameters of the aircraft, it is necessary to be guided by the already achieved level of technical excellence of flying machines, it is meant to use the main characteristics of the plane (prototype aircraft), the purpose and parameters of which are most similar to those most like incorporated in the project. It should be stated here that experience of the Antonov Design Bureau reveals a set of machines for the “analysis and synthesis”. It is first of all because this design bureau during the many years has been focused on the design of cargo airplanes.

The experience of Avions de transport regional (ATR) company has been taken into account as well. For the preliminary design, presented below, the planes Antonov 32, Antonov 24, Antonov 26 and ATR 42 have been selected.

Statistic data of prototypes are presented in table 1.1.

Table 1.1 – Statistic data of prototypes

PARAMETER	PLANES			
	AN 32	AN 24	AN 26	ATR 42
The purpose of airplane	Cargo	Cargo	Cargo	Cargo
Crew/flight attend (Persons)	4/4	4/4	4/4	4/4
Maximum take-off weight, m_{tow}, kg	27000	21000	24000	18600
Passenger's seat	4	4	4	4
The height of the flight $V_{w.\ ек.}$, m	7500	6000	7500	7600
Most pay-load, $m_{c.max}$, kg	6700	6500	5500	5450
Range $m_{к.max}$, km	780	1000	1000	1130
Take off distance $L_{зл.д.}$, m	620	870	870	1040
Landing distance, m	420	650	650	1030
Landing speed, km / h	190	165	190	190
Cruising speed, Vkm/h	470	500	440	540
Number and type of engines	2×AI20-20D-5M	2×AI24	2×AI24VT	2×P&WC PW121
Cruising power of engines, kW	2×5180	2×3470	2×3835	2×2925
Pressure ratio	9,45	6,4	7,65	7,5
Fuel consumption of cruising, kg/hour	1270	800	1000	650
Mass of fuel, kg	5500	4850	5500	4500
Landing gear scheme	TLG	TLG	TLG	TLG
The form of the cross-section fuselage	Circular	Circular	Circular	Circular
Length of aircraft, m	23,78	23,53	23,87	22,7
Height of aircraft, m	8.75	8,32	8,575	7,59
Diameter of fuselage, m	2,9	2,9	2,9	2,87

The end of the table 1.1.

Extension of the fuselage	8,2	8,1	8,2	7,83
Wingspan, m	29,2	29,2	29.2	24,57
Aspect ratio	11,37	11.76	11,37	11,07
Taper ratio	2,92	3,2	2,92	2,5
Sweepback on 1/4 chord, °	17°	6°50'	6°50'	5°25'

The scheme of the aircraft is determined by the relative position of the units, their number and shape. Its aerodynamic and technical and operational properties depend on the scheme and aerodynamic layout of the plane. The successfully chosen scheme allows increasing the safety and regularity of flights and economic efficiency of the aircraft. In case of our task: preliminary design of short-range cargo plane, it is possible to improve some units that have been successfully applied on other planes. Therefore, we will make inconspicuous changes in the design of main units (wing, chassis, plumbing, fuselage) in accordance with given technical, economic and other requirements.

The choice of the designed aircraft scheme is preceded by the study and analysis of the schemes of the aircraft adopted as prototypes. The data presented in the table above has provided the possibility to select the optimum characteristic for new aircraft.

The following are subject to substantiation:

— location of the wing and plumage relative to the fuselage, as well as the choice of their shape;

— location of engines, their number and type, if it is not specified in the design assignment;

— type and location of chassis supports.

1.1. Selection of the new plane parameters

In accordance with our task, a transport aircraft was designed, which is designed to carry load up to 5.5 tons (packaged on standard pallets or without pallets, as well as in containers) and wheeled vehicles on short and medium-haul overhead lines.

As a prototype for construction, we choose An 26 because of relatively good characteristics, small fuel consumption and parameters, which meet our requirements to carry cargo for domestic transportation. The new plane can operate with the help of the largeness of the cargo door with a width of 2.4 m and a special ramp-door. This huge opening width of the cargo hatch and improved ramp can make possible to perform convenient loading from both the ground and a truck bed thus facilitating aircraft loading and unloading.

The following methods are possible for loading:

— Travelling on their own (vehicles);

— By using an airborne crane (lifting capacity is up to 2 tons);

— By using a forklift loader;

— By using a manual loader.

Tie-down straps and nets are used for cargo lashing.

The designed aircraft is designed to carry cargo at a distance of 1100 km with a cruising speed of 440 km / h, at an altitude of 7500 m.

The prototype is made as a twin-engine monoplane with a highly located wing, at the ends of which specially profiled surfaces to reduce inductive resistance, are installed. Also, it has a tail vertical unit with an aerodynamic tail fence (ridge) and two dorsal ridges. The wing is straight in the root, trapezius closer to the tips has sweepback angle on ¼ chord 6°50'. The wing is a caisson structure, which consists of upper and lower panels, a transverse set - ribs and a longitudinal set - spars and stringers. Under it, two turboprop engines with a power ratio of 0.2800 (kW / kg) are installed.

Our aircraft is constructed as an all-metal freestanding pressurized fuselage, semi-monocoque type, and circular section. The fuselage is divided into four compartments: forward, middle, hatch and tail. Docking of the compartments among the fuselage is carried out by skin and stringers using docking tapes, fittings and overlays. In the area between frames 1–40, the fuselage is sealed. The fuselage has the largest cross section in the area between frames 9–28.

Most of the structural elements of the fuselage are made of sheet and profiled duralumin, as well as aluminum alloys.

In the manufacture of the airframe, adhesive and adhesive joints, chemical processes, monolithic large-sized panels, plastics, oriented organic glass, high-strength aluminum alloys and steel are widely used.

Glue-welded joints are used on aircraft for the manufacture of fuselage panels, plumage and engine nacelles. The material for the casing and stringer set of glue-welded panels is D16T duralumin. In the process of making glue-welded joints of stringers with skin, structural members are first spot-welded without adhesive, and then glue is applied to the edges of the stringers, which penetrates into the gaps between the stringers and the skin under the influence of capillary pressure forces. After heat treatment of the glue, X-ray inspection and application of anti-corrosion coatings, the panels provided to the assembly.

To facilitate the weight of the fuselage and tail panels, chemical milling was used. Non-metallic materials are widely used on the aircraft - fiber, press powder, oriented organic glass, polystyrene, ethrol, paronite, laminated plastics and fiberglass, polyamide resins, polyethylene, fluoroplastic, foam rubber.

In the fuselage of the aircraft there is a crew cabin and a cargo compartment. The crew cabin is located between frames 1–7 and is separated from the rest of the fuselage by a partition along frame 7. There is a door in the partition that opens toward the cargo compartment.

The aft part of the fuselage to the frame 1 is not sealed and is covered with a special fairing, under which a radar antenna is installed. The frame 1, the area of the floor of the crew cabin between frames 1–4, and the lower part of the frame 4 – are pressurized. The crew cabin floor area between frames 1–4 is located above the floor of the cargo compartment, under the floor is the compartment for the front landing gear. Between the frames 2–5 is a lantern of the cockpit. On the right side between frames 5–6 there is a radio operator window, and on the left between frames 5–7 there is a navigator blister. In the upper part of the cab between the frames 5–7 is located the upper emergency hatch.

In the fuselage from the frame 10 to the frame 33 in cargo compartment integrated conveyor is installed. The front door is located between the frames 7–9 in the starboard side, and the lower emergency hatch is located in the lower part of the fuselage between the frames 7–10. Four round windows are installed on the cargo compartment in both sides of the fuselage. The window in the starboard side between frames 23–24 and the window in the starboard side between frames 14–15 are combined with emergency hatches.

Between the frames 33–40 is a cargo hatch.

The opening of the cargo hatch is closed by a cargo ramp ending in a tapered collision. A monorail is installed on the fuselage ceiling in the plane of symmetry between the frames 29–39, along which the hoist moves, designed for loading and unloading. Between the frames 11–33 two conveyor rails are built into the floor.

The wing center section is mounted in the upper part of the fuselage to the frames 17 and 20. The tail compartment of the fuselage carries the empennage of the aircraft and is not sealed. The compartment contains units for navigation and radio equipment of the aircraft. Access to the tail section realized through a hatch in the bottom panel of the compartment between frames 41–42. The fuselage structure consists of a transverse and longitudinal power sets, floor, casing, cockpit lantern, windows, doors and hatches.

The transverse power set of the fuselage consists of 51 frames, which by design can be divided into normal and reinforced. The lower parts of the formers together with the longitudinal profiles form the frame of the fuselage floor.

The longitudinal power set of the fuselage consists of stringers and a number of longitudinal beams in the aft and middle compartments.

1.1.1. Wing

The wing of the aircraft is highly located, rectangular with a trapezoidal center section (in the area between ribs № 7 and tips).

Structurally, the wing consists of the center, caisson and tail parts. It also has mechanization (flaps, ailerons, interceptors), which significantly improves the take-off and landing characteristics and maneuverability of the aircraft.

The wing has connections on ribs № 7 and 12 and is divided into a center section, two middle and two separate parts. The center part of the wing carries two deflected single-slot flaps, the middle parts of the wing - one double-slot retractable flap, detachable parts of the wing - two sections of ailerons. Docking of the wing parts to each other is carried out using the connector profiles, fittings and butt squares. Dihedral angle of the wing in the area between ribs № 12 is 0 °, and in the area of the ribs it is −2°. The wing installation angle is + 3°.

The wing structure is a caisson type, consisting of 2 spars, 23 ribs, skin and stringers. The wing skin has a different thickness in different areas. The leading edge of the wing is air-heated to prevent icing. In the trailing edge of the wing, there are flap control shafts and ailerons control rods.

On the center section there are installed nodes for docking with the fuselage, profiles of the connector for docking with the middle parts of the wing, attachment points for engines, main landing gear legs and single-slotted flap linkage assemblies.

Center section spars - beam type, one-piece, reinforced with uprights from extruded profiles. Each spar has two brackets for docking with the fuselage.

Center wing ribs - reinforced, beam type. Each of them consists of a blank duralumin web, reinforced by uprights from extruded corners, as well as upper and lower belts from extruded profiles of T-section.

The caisson of the middle part of the wing is a fuel tank compartment. The caisson is sealed with sealant. The middle part of the wing has connector profiles for docking with the center section and the removable part of the wing.

Spars of the middle part of the wing - beam type, consist of the upper and lower flanges of the T-section and the webs supported by pressed duralumin struts. Flanges are made of extruded profiles, machined to obtain a variable cross-sectional length. At the ends of the spars there are racks for joining the webs of the side members of the middle part of the wing with the webs of the spars of the center section and the removable part of the wing.

The detachable part of the wing is structurally similar to the center section and the middle part of the wing. The detachable part joins the middle part with the help of profiles, and end connector strut spars.

Spars have webs and flanges of variable cross-section. The web of the spars are supported by struts of extruded profiles. There are openings in the web of the front side member for the exit of warm air from the nose of the removable part to the end fairing.

The ribs of the detachable part are beam type, in design they are similar to the ribs of the middle part of the wing.

On each wing there is a single-slot center wing flap located between the fuselage and the engine nacelle, and a double-slot flap in the middle part of the wing.

A profiled deflector is riveted on the brackets to the double-slotted flap. When the flap deviates between the tail of the wing, the deflector and the flap, a double shaped gap is formed.

Each flap consists of a skin, a set of ribs, a spar and two carriages. Flaps are fastened using brackets.

The flaps are released and cleaned hydraulically by means of a transmission shaft and six screw lifts. The system is controlled by a push switch installed on the central control panel of the pilots. There is also the possibility of emergency release of the flaps using the rocker switch installed in the same place.

The flap deflection angle is controlled by the position indicator installed on the center console. Flap deflection angle at take-off - $15^{\circ\pm2^\circ}$, at landing - $38^{\circ-1^\circ}$.

Ailerons are attached to the rear side members of the removable wing parts. Each of them consists of a root and end sections. The design of the aileron includes a spar, ribs and skin. On the root section of the left aileron, a trim tab and servo tab are installed, on the right - only the servo compensator.

Ailerons are controlled by rotating the helms. When the helms rotate to the maximum angle of rotation from the neutral position - 90 degrees - the right aileron deviates upward by 24°, and the left aileron - down 16° (when rotating clockwise) and vice versa (when rotating counterclockwise).

1.1.2. Tail Unit

The empennage of the aircraft - T-type, mounted on the fuselage, consists of horizontal and vertical stabilizers.

The horizontal tail includes a stabilizer consisting of two consoles. Elevator consoles are balanced by horn and axial compensations. On each half of the elevator there are servo tabs - internal and external.

Vertical empennage includes: fin, fence and rudder. The rudder has horn and axial compensation. Servo tab is mounted on the rudder.

Dihedral angle of the horizontal stabilizer is 9°.

The stabilizer consists of two symmetrical consoles. Each console consists of an upper and lower panel, a forward, a tail, and an end fairing. The stabilizer panel consists of two half-spars, a set of half-ribs, a stringer and a skin. The stringers are fastened to the casing with the help of spot welding and glue, and the casing is glued to the ribs and side members. Docking of the stabilizer with the fuselage is carried out on the spars, by means of bolts and fittings.

Each half of the elevator consists of two glue-welded panels connected in the plane of the chords, the ending profile and the beam for attaching the trim tab. Trim tab are installed on each half of the elevator.

The elevator is controlled by moving the rudder: “from yourself” - the steering wheel deviates downward by 20°, towards itself - the steering wheel deviates upward by 25°. Steering wheels - duplicated, carried out from both places of pilots. An autopilot steering machine is included in the elevator control wiring.

1.1.3. Power Plan

The power plant consists of two turboprop engines (AI-24VT with AB-72T propellers) with take-off power of 2820 hp each and RU-19A-300 APU. The engines are installed in nacelles located on the center section of the wing. Each engine is attached to the truss, which is mounted on the aft spar of the wing center section using a frame through the reinforced bulkhead.

Besides the propeller, the following are mounted on the engine: gear fairing, cowl, anti-icing system, external oil system, generators and engine ventilation system, fuel system and fire extinguishing system. The hot part of the engine and the exhaust pipe are separated from the wing structure by special firewalls and screens.

APU RU19A-300 is installed in the rear of the right nacelle. It provides:

— additional thrust when climbing;

— necessary thrust in case of AI-24 engine failure;

— start of AI-24 engines;

— power supply of the aircraft onboard network in the parking with engines in idle ;

— power supply of the aircraft’s onboard network in case of starter-generator failure.

For the needs of life support on each engine air extraction is provided, and also the generator of alternating current which is the main source of the electric power is established. The power plant is controlled and monitored from the cockpit, where all the necessary units and devices are installed. In the control of the power plant, mechanical, electrical and automatic systems are used.

1.1.4. Landing Gear

The aircraft has a retractable landing gear, made according to tricycle landing gear type with front single-strut and two strut main gears. The main gears are installed in the engine nacelles and are retracted forward into special compartments under the engines in flight. On each main gear, on a common fixed axis, two wheels with pneumatics and disc brakes are installed. The wheels are equipped with inertial sensors.

The front gear is installed in the aft part of the fuselage and in flight retracts forward into the compartment under the cockpit. On the front strut, on a common rotating axis, two non-brake wheels with pneumatics are installed.

In the extracted and retracted positions, the legs are fixed with mechanical locks that open with the help of hydraulic cylinders. With the extracted position of the landing gear, only small sash located directly at the amortization struts remain unclosed; thus, the units and components located in the compartments of the landing gear are protected from dirt during taxiing. The landing gear sashes open and close using mechanisms kinematically connected with amortization struts. The wheels of the front landing gears are orientable. The wheels can be turned by the steering mechanism to the right and left by an angle of 45° from the neutral position to increase the maneuverability of the aircraft during taxiing. To maintain the straight-line direction of movement during take-off and run on an airplane, there is a takeoff and landing control of the front leg associated with steering the rudder. At the same time, the maximum steering angle of the front legs 10° at each side.

Extracting and retracting of the landing gear, opening locks, braking the wheels of the main legs and turning the wheels of the front leg are carried out by the power cylinders of the aircraft’s hydraulic system. In the event of a failure of the hydraulic system, the locks of the retracted position of the landing gear can be manually opened using a mechanical system. In this case, the gears are extracted and installed on the locks of the released position under the influence of its own weight and the air flow.

1.1.5. Flight Control System

Control of the aircraft - booster-free, carried out through mechanical wiring.

The control system provides control of:

- ailerons
- rudder
- elevator
- flaps
- interceptors in the roll automatic control mode

The aircraft is controlled by aerodynamic control surfaces – the elevator, rudder and ailerons. Trim tabs are installed on each half of the elevator and on the left aileron, and a servo tab is installed on the rudder. In addition, servo tabs are installed on each aileron. The aircraft flight control system provides control of rudders, ailerons, flaps and servo tabs, locking of rudders and ailerons while parking, control of the aft landing gear rotation and pressure reducing valves for braking the main landing gear wheels. Steering wheels and ailerons - double, that is, can be carried out from the workplaces of both pilots. To ensure synchronized control, the helm and pedals of the left pilot are kinematically connected with the helm and pedals of the right pilot.

Steering wheels and pedals are mounted on a common control panel mounted on the floor of the cockpit under the dashboard. Reduction brake valves are located on the same panel and the mechanisms of their drive from the pedals are mounted, as well as the parking brake mechanism.

Steering wheels for elevator trim tabs, flap control switches, rudder and aileron trim tabs, as well as a rudder and aileron locking knob are located on the pilots' central console. The handle for steering the wheels of the aft landing gear is located on the left pilots' control console. The autopilot control panel is located on the central panel.

1.1.6. Crew cabin

The cockpit must be as small as possible, but at the same time provide normal conditions for work and rest of the flight crew.

The most stringent requirements are imposed on the jobs of pilots. In addition to convenience, they should provide a good overview.

The crew consists of the ship's commander (first pilot), co-pilot, on-board technician (between the rear pilots), in the cargo compartment, a technician's place is equipped.The pilots are seated in seats next to each other.

The flight crew compartment is separated from other rooms by a rigid partition.

The crew cabin is designed like a prototype.

1.3. Calculations of the new aircraft parameters

1.3.1. Geometry calculations for the main parts of the aircraft

Geometrical characteristics of the wing are determined from the take off weight m_0 and specific wing load P_0.

Wing loading value has been accepted on the base of similar planes analysis.

For the designed plane it was chosen to be equal to 2573 kg/ m^2.

From this:

Full wing area with extensions is:

$$S_W = \frac{G_0}{P_0} = \frac{m_0 \cdot g}{P_0} = \frac{23479 \cdot 9{,}8}{2573} \approx 89.43 \ [m^2];$$

Relative wing extensions area is 0.1.

Wing area is:

$$S_w = 89.43 \cdot (1 - 0.1) \approx 80.5 \ [m^2]$$

An important characteristic of the wing is an aspect ratio. The greater the aspect ratio of the wing, the more efficient is the wing aerodynamically.

Some examples of the aspect ratio values are presented in table 1.2.

Table 1.2 – Aspect ratio examples

№	**Aircraft type**	**Aspect ratio**
1	Hang glider	4-8
2	Glider (sailplane)	20-40
3	Homebuilt	4-7
4	General Aviation	5-9
5	Jet trainer	4-8
6	Low subsonic transport	6-9
7	High subsonic transport	8-12
8	Supersonic fighter	2-4
9	Tactical missile	0.3-1
10	Hypersonic aircraft	1-3

For proposed plane the value of aspect ratio 11.37 has been selected.

Wing span is:

$l_w = \sqrt{S_w \cdot \lambda} = \sqrt{80.5 \cdot 11.37} \approx 30.25$ [m], where

λ - aspect ratio;

Root chord is:

$b_o = \frac{2S_w \cdot \eta_w}{(1+\eta_w) \cdot l_w} = \frac{2 \cdot 80.5 \cdot 2.92}{(1+2.92) \cdot 30.25} = 3.96$ [m], where

η_w – taper ratio

Tip chord is:

$b_t = \frac{b_o}{\eta_w} = \frac{3.96}{2.92} = 1.35$ [m];

Board chord is:

$b_{ob} = b_0 \cdot \left(1 - \frac{(\eta_w - 1) \cdot D_f}{\eta_w \cdot l_w}\right) = 3.96 \cdot \left(1 - \frac{(2.92-1) \cdot 2.9}{2.92 \cdot 30.25}\right) = 3.71$ [m], where

D_f - diameter of fuselage

Among the main wing geometrical parameters, the following values have been selected.

The taper ratio of the wing has been selected equal to 2.92 by the following considerations:

a) The planform shape should not give rise to an additional lift distribution that is so far from elliptical that the required twist for low cruise drag results in large off design penalties;

b) The chord distribution should be such that with the cruise lift distribution, the distribution of lift coefficient is compatible with the section performance. Avoid high C_L (lift coefficient) which may lead to buffet or drag rise or separation;

c) The chord distribution should produce an additional load distribution which is compatible with the high lift system and desired stalling characteristics;

d) Lower taper ratios lead to lower wing weight;

e) Lower taper ratios result in increased fuel volume;

f) The tip chord should not be too small as Reynolds number effects cause reduced C_L capability;

g) Larger root chords more easily accommodate landing gear.

Here, again, a diverse set of considerations are important. The major design goal is to keep the taper ratio as small as possible (to keep the wing weight down) without excessive C_L variation or unacceptable stalling characteristics. Since the lift distribution is nearly elliptical, the chord distribution should be nearly elliptical for uniform C_L. Reduced lift or t/c (thickness to chord) outboard would permit lower taper ratios. The thickness to chord value has been selected equal to 0.120 because:

a) we would like to make the t/c as large as possible to reduce wing weight (thereby permitting larger span, for example);

b) Greater t/c tends to increase $C_L$max up to a point, depending on the high lift system, but gains above about 12% are small if there at all;

c) Greater t/c increases fuel volume and wing stiffness;

d) Increasing t/c increases drag slightly by increasing the velocities and the adversity of the pressure gradients;

e) The main trouble with thick airfoils at high speeds is the transonic drag rise which limits the speed and CL at which the airplane may fly efficiently.

The sweepback angle on 1/4 chord has been selected equal to 6.5°. The main advantage of a direct (or with a small sweepback angle) wing is its high coefficient of lift. The purpose of the swept back design is to prevent wave drag, and therefore the drag of the wing as a whole. But the disadvantage that determines the unsuitability of sweepback angle a wing at subsonic and supersonic flight speeds is a sharp increase in drag coefficient when the critical Mach number value is exceeded.

In our case, this drawback does not matter because of given flight speed of 400 - 500 km/h.

At a choice of structural scheme of the wing we determine quantity of spars and their position, and the places of wing portioning.

On the modern transport planes two spars or three spars designs are most conventional. The two spars design have been selected as appropriate (fig 1.1.).

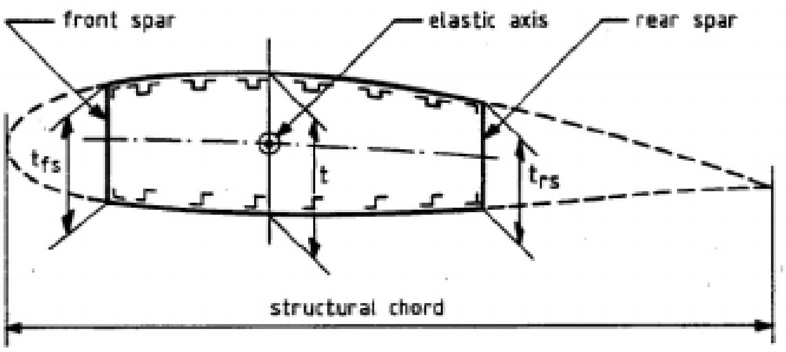

Figure 1.1 – Two spars wing design

I use the geometrical method of mean aerodynamic chord determination (fig. 1.2.).

Mean aerodynamic chord is equal: b_{MAC}=2.8688 m

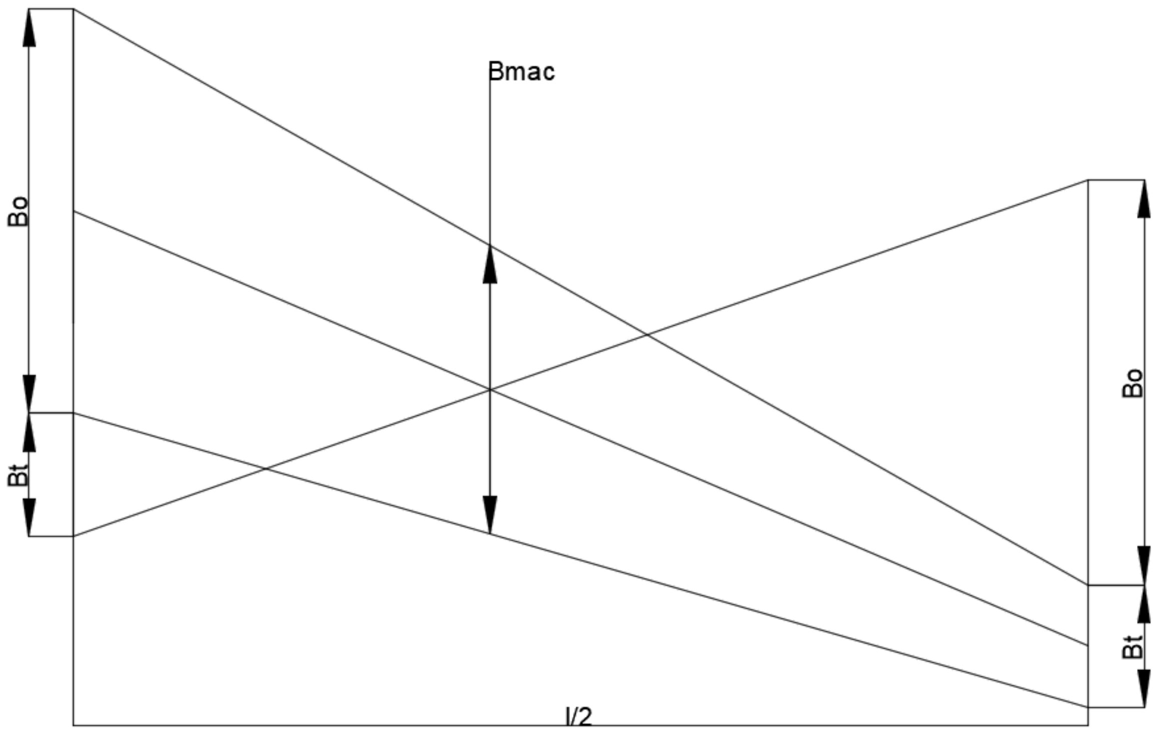

Figure 1.2 – Geometrical method of wing mean aerodynamic chord determination

After determination of the geometrical characteristics of the wing we come to the estimation of the ailerons geometrics and high-lift devices.

Ailerons geometrical parameters are determined in next consequence:

Ailerons span:

$l_{ail} = (0.3…0.4)·l_w /2 = 0.35·15.125 = 5.3$ [m];

Ailerons chord:

$b_{ail} = (0.22…0.26)·b_t = 0.24·1.35 = 0.324$ [m];

Aileron area:

$S_{ail} = (0,05…0,08)·S_w/2 = 0.065·40.25 = 2.62$ [m^2];

Increasing of l_{ail} and b_{ail} more than recommended values is not necessary and convenient. With the increase of l_{ail} more than given value the increase of the ailerons coefficient falls, and the high-lift devices span decreases. With b_{ail} increase, the width of the xenon decreases.

In the airplanes of the third generation there is a tendency to decrease relative wing span and ailerons area. In this case for the transversal control of the airplane we use spoilers together with the ailerons. Due to this the span and the area of high-lift devices may be increased, which improves take off and landing characteristics of the aircraft.

Aerodynamic compensation of the aileron.

Axial $S_{axinail} \leq (0.25\ldots0.28)\cdot S_{ail} = 0.265\cdot2.62 = 0.6943$ [m^2];

Inner axial compensation:

$S_{inaxinail} = (0.3..0.31)\cdot S_{ail} = 0.305\cdot2.62 = 0.8$ [m^2];

Area of ailerons trim tab.

For two engine airplane:

$S_{tail} = (0.04\ldots0.06)\cdot S_{ail} = 0.05\cdot2.62 = 0.131$ [m];

Range of aileron deflection:

Upward $\delta'_{ail} \geq 20^\circ$;

Downward $\delta''_{ail} \geq 10^\circ$

The aim of determination of wing high-lift devices geometrical parameters is the providing of take of and landing coefficients of wing lifting force, assumed in the previous calculations with the chosen rate of high-lift devices and the type of the airfoil profile.

Eventually, it can be stated that effective high lift systems have a more complicated design than simple and less effective systems, and this leads to higher purchase and maintenance costs. On the other hand, the more effective system saves fuel through the possibility of building a lighter wing with better lift-to-drag ratio than would be possible in the case of an overall design with a simple high lift system. An optimum compromise can – as is so often the case – only be found through detailed studies. In the early stages of design, it is, therefore, advisable to follow the designs of successful aircraft models.

Taking into the account the design of planes-prototypes and current trends in the design of high lift devices the two-slotted flap design has been proposed.

$b_{fl} = (0.28\ldots0.3)\cdot b_t = 0.29\cdot1.35 = 0.3915$ [m];

Before doing following calculations it is necessary to choose the type of airfoil due to the airfoil catalog, specify the value of lift coefficient $C_{y\max bw}$ and determine necessary increase for this coefficient $C_{y\max}$ for the high-lift devices outlet by the formula:

$$\Delta C_{y\max} = (\frac{C_{y\max l}}{C_{y\max bw}}) \quad (1.1.)$$

where $C_{y\max l}$ – necessary coefficient of the lifting force in the landing configuration of the wing by the aircraft landing insuring (it is determined during the choice is the aircraft parameters).

In the modern design the rate of the relative chords of wing high-lift devices is:

b_{sf} = 0.25..0.3 – for the split edge flaps;

b_f = 0.28..0.3 – one slotted and two slotted flaps;

b_f = 0.3..0.4 – for three slotted flaps and Faylers flaps;

b_s = 0.1..0.15 – slats.

Effectiveness of high-lift devices (C^*_{ymaxl}) rises proportionally to the wing span increase, serviced by high-lift devices, so we need to obtain the biggest span of high lift devices due to use of flight spoiler and maximum diminishing of the are of engine and landing gear nacelles.

During the choice of structurally-power schemes, hinge-fitting schemes and kinematics of the high-lift devices we need to come from the statistics and experience of domestic and foreign aircraft construction. We need to mention that in the majority of existing constructions elements of high-lift devices are done by spars construction schemes.

1.3.2. Fuselage Layout

During the choice of the shape and the size of fuselage cross section we need to come from the aerodynamic demands (streamlining and cross section).

Applicable to the subsonic cargo aircrafts (V < 800 km/h) wave resistance doesn’t affect it. Based on this, we need to choose friction resistance C_{xf} and profile resistance C_{xp} from the conditions of the list values.

During the subsonic flights, shape of fuselage nose part affects the value of wave resistance C_{xw}. Using the circular shape of fuselage nose part significantly diminishing its wave resistance.

For susonic airpcraft fuselage nose part must be:

$l_{nfp} = 2.1 \cdot D_f = 2.1 \cdot 2.9 = 6.09$ [m];

Based on the fact that the cause of the one AN 26 crash (October 4, 2007 Africa An-26, 9Q-COS) was the design mistakes caused by the wear fatigue of the skin of the aircraft, which has long been operated in tropical climates, we need to consider the strength and layout requirements during the choice of cross section shape. For ensuring of the minimal weight, the most convenient fuselage cross section shape is circular cross section. In this case, we have the minimal fuselage skin width. As the partial case we may use the combination of two or more vertical or horizontal series of circles.

Geometrical parameters that we concern:

- fuselage diameter D_f;
- fuselage length l_f ;
- fuselage aspect ratio λ_f ;
- fuselage nose part aspect ratio λ_{np} ;
- tail unit aspect ratio λ_{TU} .

Fuselage length is equal:

$l_f = D_f \cdot \lambda_f = 2.9 \cdot 8.2 = 23.78$ [m];

Fuselage nose part aspect ratio is equal:

$\lambda_{fnp} = \frac{l_{fnp}}{D_f} = 2.1$;

Sum of nose and rear parts must be equal 4.8.

Length of the fuselage rear part is equal:

$\lambda_{frp} = 4.8 - \lambda_{fnp} = 2.7$;

Length of the fuselage rear part is equal:

$l_{frp} = \lambda_{frp} \cdot D_f = 2.7 \cdot 2.9 = 7.83\ [m]$;

During the determination of fuselage length we seek for approaching minimum mid-section S_{ms} from one side and layout demands from the other.

For our prototype cargo airplane fuselage mid-section first of all comes from the size of cargo cabin.

For short range airplanes we may take the height as: h_1=1.75m; the distance from the window to the flour h_2=1m.

Step of formers (bulkheads) in the fuselage construction is in the range of 350...500mm.

1.3.3. Layout and calculation of basic parameters of tail unit

One of the most important tasks of the aerodynamic layout is the choice of tail unit placing. For ensuring longitudinal stability during overloading its center of gravity should be placed in front of the aircraft focus and the distance between these points, related to the mean value of wing aerodynamic chord, determines the rate of longitudinal stability.

$$m^{C_y}_x = \bar{x}_T - \bar{x}_F < 0 \qquad (1.2.)$$

Where $m^{C_y}_x$ –is the moment coefficient; $\bar{x}_T, \bar{x}_F$ - center of gravity and focus coordinates. If $m^{C_y}_x=0$, then the plane has the neutral longitudinal static stability, if $m^{C_y}_x > 0$, then the plane is statically instable. In the normal aircraft scheme (tail unit is behind the wing), focus of the combination wing – fuselage during the install of the tail unit of moved back.

Static range of static moment coefficient: horizontal Ahtu, vertical Avtu given in the table with typical arm Htu and Vtu correlations. Using table we may find the first approach of geometrical parameters determination.

Determination of the tail unit geometrical parameters

Area of vertical tail unit is equal:

$S_{vtu} = (0{,}12\ldots0{,}2) \cdot S_w$;

Area o horizontal tail unit is equal:

$S_{htu} = (0{,}18\ldots0{,}25) \cdot S_w$;

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

Trapezoidal scheme, normal scheme $L_{htu} = (0.2...3.5) \cdot b_{mac}$;

Light airplane $L_{htu} = (2.0...2.3) \cdot b_{mah}$;

Heavy airplane $L_{htu} = (3.2...3.3) \cdot b_{mah}$;

In the first approach we may count that $L_{htu} \approx L_{vtu} = 3 \cdot 2.86 \approx 8.55$ m;

Much better could be calculated like:

$$S_{htu} = \frac{b_{mac} \cdot S_w}{L_{htu}} \cdot A_{htu} = \frac{2.8688 \cdot 80.5}{8.55} \cdot 0.5 = 13.505\ [m^2];$$

$$S_{vtu} = \frac{l_w \cdot S_w}{L_{vtu}} \cdot A_{vtu} = \frac{30.25 \cdot 80.5}{8.55} \cdot 0.08 = 22.78\ [m^2];$$

where L_{htu} and L_{vtu} - lenght of horizontal and vertical empennage, l_w and S_w - wingspan and area of the wing, A_{htu} and A_{vtu} - coefficients of static moments.

Determination of the elevator area and rudder area:

Elevator area:

$S_{el} = 0.345 \cdot S_{htu} = 0.345 \cdot 13.505 = 4.66\ [m^2]$;

Rudder area:

$S_{rd} = 0{,}4 \cdot S_{vtu} = 0{,}4 \cdot 22.78 = 9.112\ [m^2]$;

Choose the area of aerodynamic balance: $0.3 \leq M \leq 0.6$

Elevator balance area is equal:

$S_{eb} = (0{,}22\ldots0{,}25) \cdot S_{el} = 0.235 \cdot 4.66 = 1.095\ [m^2]$;

Rudder balance area is equal:

$S_{rb} = (0{,}2\ldots0{,}22) \cdot S_{rd} = 0.21 \cdot 9.112 = 1.914\ [m^2]$;

The area of elevator trim tab:

$S_{etr} = 0.08 \cdot S_{el} = 0.08 \cdot 4.66 = 0.373\ [m^2]$;

The area of rudder trim tab is equal:

$S_{rtr} = 0{,}06 \cdot S_{rd} = 0{,}06 \cdot 9.112 = 0{,}54672\ [m^2]$;

Tapper ratio of horizontal and vertical tail unit we need to choose:

For planes M<1 corresponds to η_{htu} = 2...3; η_{vtu} = 1…3.3;

Aspect ratio of horizontal and vertical tail unit we may recommend:

For subsonic planes λ_{vtu} = 0.8...1.5 ; λ_{htu} = 3.5...4.5;

Determination of horizontal and vertical tail unit chords b_{tip}, b_{MAC}, b_{root}:

Tip chord of horizontal stabilizer is:

$$b_{HTUtch} = \frac{2 \cdot S_{HTU}}{(\eta_{htu}+1) \cdot l_{htu}} = \frac{2 \cdot 13.505}{(2.2+1) \cdot 8.55} = 0.987 \text{ [m]};$$

Root chord of horizontal stabilizer is:

$$b_{HTUrch} = b_{HTUtch} \cdot \eta_{htu} = 0.987 \cdot 2.2 = 2.172 \text{ [m]};$$

Tip chord of vertical stabilizer is:

$$b_{VTUtch} = \frac{2 \cdot S_{VTU}}{(\eta_{vtu}+1) \cdot l_{vtu}} = \frac{2 \cdot 22.78}{(2+1) \cdot 8.55} = 1.776 \text{ [m]};$$

Root chord of vertical stabilizer is:

$$b_{VTUrch} = b_{VTUtch} \cdot \eta_{vtu} = 1.776 \cdot 2 = 3.552 \text{ [m]};$$

Horizontal tail unit mean aerodynamic chord:

$$b_{MACHTU} = 0{,}66 \cdot \frac{\eta_{htu}^2 + \eta_{htu} + 1}{\eta_{htu} + 1} \cdot b_{HTUtch} = 1.6367 \text{ [m]};$$

Vertical tail unit mean aerodynamic chord:

$$b_{MACVTU} = 0{,}66 \cdot \frac{\eta_{vtu}^2 + \eta_{vtu} + 1}{\eta_{vtu} + 1} \cdot b_{VTUtch} = 2.735 \text{ [m]};$$

The sweepback angle of the empennage is taken as 3...50° more than the sweepback of the wing.

So: χ_{HTU} = 9 °; χ_{VTU} = 22 °.

1.3.4. Landing gear design

In the primary stage of design, when the airplane center-of-gravity position is defined and there is no drawing of airplane general view, only the part of landing gear parameters may be determined.

Main wheel axel offset is:

$$e_g = 0{,}15 \cdot b_{MAC} = 0.15 \cdot 3.86 = 0.579 \text{ [m]};$$

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

Landing gear wheel base comes from the expression:

$B_g = (0{,}3\ldots0{,}4) \cdot l_{\mathrm{f}} = 0.32 \cdot 23.78 = 7{,}61$ [m];

The last equation means that the nose support carries 6...10% of aircraft weight.

Front wheel axial offset will be equal:

$d_{ng} = B_g - e_g = 7.61 - 0{,}579 = 7.031$ [m];

Wheel track is:

$\mathrm{T} = (0{,}7 \ldots 1{,}2) \cdot B_g = 1{,}04 \cdot 7.61 = 7.9$ [m];

On a condition of the prevention of the side nose-over the value T should be > 2H, where H – is the distance from runway to the center of gravity.

Wheels for the landing gear is chosen by the size and run loading on it from the take off weight; for the front support we consider dynamic loading also.

Type of the pneumatics (balloon, half balloon, arched) and the pressure in it is determined by the runway surface, which should be used. We install breaks on the main wheel.

The position of c.m. can be taken by height:

- for high planes (with the location of the engines on the wing) c.m. is above the construction horizontal of the fuselage at a distance. The load on the wheel is determined:

$K_g = 1.5\ldots2.0$ – dynamics coefficient.

Nose wheel load is equal:

$$P_{nlg} = \frac{e_g \cdot m_0 \cdot g \cdot K_g}{B_g \cdot z_{nlg}} = \frac{0.579 \cdot 23479 \cdot 9.81 \cdot 1.75}{7.61 \cdot 2} = 15333.83 \text{ [N]};$$

Main wheel load is equal:

$$P_{mlg} = \frac{(B_g - e_g) \cdot m_0 \cdot g}{B_g \cdot z_{mlg} \cdot n_{mlg}} = \frac{(7.61 - 0.579) \cdot 23479 \cdot 9.81}{7.61 \cdot 2 \cdot 2} = 53201.15 \text{ [N]};$$

Where n and z – number of the number of supports and wheels on one support, respectively;

From the table of aircraft wheels with high pressure tires catalog by calculated P_{main} and P_{nose} and the value of $V_{take\ off}$ and $V_{landing,}$ pneumatics is chosen from the catalog, the following correlations should correspond:

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

Table 1.3 – Avition tires for prototype

Main gear		Nose gear	
Tire size	Ply rating	Tire size	Ply rating
1050x400mm	18	720x2506A mm	14

1.3.5. Power plant

AI-24 is a single-shaft turboprop engine with a 10-stage axial compressor, an annular combustion chamber and a three-stage turbine. We choose the best one of AI-24 - AI-24VT - an updated variant of AI24, which produced 2,100 kW (2,820 hp) differentiated because it was coupled with a RU19A-300 APU/booster instead of the regular TG-16M auxiliary power unit (APU) turbine.

Table 1.4 – Examples of modifications AI 24

Model	Power	Pressure ratio	Dry weight
AI-24	1,790 kW (2,400 hp)	6.4	600 kg
AI-24A	1,875 kW (2,515 hp)	6.4	600 kg
AI-24T	2,100 kW (2,820 hp)	7.05	600 kg
AI-24VT	2,100 kW (2,820 hp)	7.65	600 kg

1.4. Determination of the aircraft centre of gravity position

The layout and centering is the whole inseparable process. To ensure the desired degree of static stability and controllability of the aircraft, its center of gravity must be in a certain range along the length of the MAC wing.

During the operation of the aircraft, the position of the center of gravity may change as fuel is produced, as well as by loading the aircraft. The rear centering must be as close as possible to ensure the minimum required margin of static stability of the aircraft, which is determined by its purpose.

The maximum allowable front centering of the aircraft is determined by the efficiency of its longitudinal controls (balancing). The greater the efficiency of the longitudinal controls, the more acceptable the front centering of the aircraft, therefore, the range of its operational centering will be acceptable.

The main requirements for the layout of the aircraft are:

- The layout must best ensure compliance with the operational and technical requirements for the aircraft;
- Each unit (cargo) of the aircraft must be located so, that it most successfully performs its functions;
- The layout of the aircraft must provide ease of control and maintenance of these basic systems and units, as well as the convenience of removal and installation of removable parts and units;
- Technological division of the structure should provide a wide front of work in production, as well as the convenience of the overall assembly of the aircraft;
- The reinforced scheme should provide (with possible full compliance with the previous requirements) less weight of the structure with sufficient strength and rigidity.

The basic principles of aircraft layout are: operational and technical requirements in the layout of the aircraft are reflected in the degree of their importance for the designed aircraft. First of all, the most important requirements are met, and others - as far as possible. Contradictions in the requirements are resolved by making compromise decisions.

The layout of the aircraft uses the principle of combining several functions performed by the same structural element or unit. For example, connectors and hatches are made so that they perform both technological and operational functions. The same reinforced bulkheads in the empennage are used for attachment to the fuselage of vertical tail unit, and for mounting horizontal nacelles of engines and others. The principle of combining functions in the elements of the reinforced scheme provides not only savings in weight, but also to obtain large volumes inside the aircraft, to accommodate cargo (units).

In addition, a number of ideas are laid down in this location principle:

- The transfer and balancing of all major power factors, according to the elements of the reinforced scheme should be carried out as soon as possible;

- It is better to transfer concentrated forces by stretching or compressing the force elements than by bending;

- It is advisable to transfer bending moments at the highest possible construction height (base), and torque - in a closed loop.

1.4.1. Determination of centering of the equipped wing

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

The example list of the mass objects for the aircraft, where the engines are located under the wing, included the names given in the table 1.5. The mass of aircraft is 23479 kg.

We assume that our projected plane is symmetrical on the Y axis, so we determine only the coordinate of the center of gravity X. Coordinates of the center of power for the equipped wing are defined by the formulas:

$$X_w^{/} = \frac{\Sigma m_i^{/} x_i^{/}}{\Sigma m_з^{/}} \quad (1.3.)$$

Table 1.5 - Trim sheet of equipped wing masses

№	Object name	Mass		C.G. coordinates X_i,м	Moment of mass
		Units	Total mass m_i		
1.	Wing (structure)	0,15432	3623,27928	1,335	4839,251806
2.	Fuel system,50	0,00124	29,11396	1,335	38,88460498
3.	Airplane control, 30%	0,00336	78,88944	1,908	150,5210515
4.	Electrical equipment, 30%	0,003	70,437	0,318	22,398966
5.	Anti-ice system, 50%	0,01379	323,77541	0,318	102,9605804
6.	Hydraulic systems, 50%	0,01918	450,32722	1,908	859,2243358
7.	Power Plant	0,10032	2355,41328	-1,2	-2826,495936
8.	Equipped wing without landing gear and fuel	0,29521	6931,23559	0,4597	3186,745409
9.	Nose landing gear	0,0039552	92,8641408	-6,81	-632,4047988
10.	Main landing gear	0,0454848	1067,937619	1,749	1867,822896
11.	Fuel	0,10749	2523,75771	1,367	3450,986293
	Total	0,45214	10615,79506	0,741	7873,149799

$X_w = \sum m_i \cdot X_i / \sum m_i = 0.742$ m

1.4.2. Determination of centering of the equipped fuselage

Origin of the coordinates is chosen in the projection of the nose of the fuselage on the horizontal axis. For the axis X the construction part of the fuselage is given. The example list of the objects for the aircrft, which engines are mounted in the rear part of fuselage, is given in table 1.6.

The central gravity coordinates of the equiped fuselage are determined by formulas:

$$X_f = \frac{\Sigma m_i^{/} X_i^{/}}{\Sigma m_i^{/}}; \qquad (1.4.)$$

where X_i – fuselage center of gravity coordinate;

$\sum m_i$ – sum of total mass of fuselage.

Table 1.6 – Trim sheet of equipped fuselage masses

№	Objects names	Mass		Center of gravity coordinates Xi, м	Moment of mass
		Units	Total mass		
1.	Fuselage	0,15973	3750,30067	11,89	44591,07497
2.	Horizontal tail	0,01627	382,00333	1,11	423,7371938
3.	Vertical tail	0,01618	379,89022	1,488	565,0867023
4.	Radar	0,0034	79,8286	1	79,8286
5.	Radio equipment	0,0045	105,6555	1	105,6555
6.	Instrument panel	0,0079	185,4841	2,5	463,71025
7.	Fuel system,50	0,006	830,75	10	8307,58
8.	Power	0,07	10307,93	39	402010
9.	Aero navigation equipment	0,0068	159,6572	2	319,3144
11.	Aircraft control system 70%	0,0078	183,1362	11,89	2177,489418
12.	Hydro-pneumatic sys 30%	0,00822	192,99738	16,65	3212,634387
13.	Electrical equipment 70%	0,021	493,059	11,89	5862,47151
14.	Not typical equipment	0,0021	49,3059	11,89	586,247151
15.	Furnishing and thermal equipment	0,0107	251,2253	11,89	2987,068817
16.	Anti ice and airconditioning system	0,0197	462,5363	15,457	7149,423589
17.	Cargo equipment	0,00223	52,35817	11,5	602,118955
18.	Service load	0,02497	586,27063	3,5	2051,947205
19.	Additional equipment	0,00213	50,01027	11,89	594,6221103
20.	Equipped fuselage without payload	0,31363	7363,71877	9,75	71772,43076
21.	Cargo	0,2223	5219,3817	11,5	60022,88955
22.	Crew	0,01193	280	2,5	700
Total		0,54786	12863,10047	10,30	132495,3203
TOTAL fraction		1	23478,89553	-	

We can find fuselage center of gravity coordinate X_f:

$$X_f = \sum m_i \cdot X_i / \sum m_i = 10.3 \text{ [m]};$$

After we determined the center of gravity of fully equipped wing and fuselage, we construct the moment equilibrium equation relatively fuselage nose:

$$m_f \cdot x_f + m_w \cdot (x_{MAC} + x_w) = m_0 \cdot (x_{MAC} + C) \qquad (1.5.)$$

where m_0 – aircraft takeoff mass, kg;

m_f – mass of equipped fuselage, kg;

m_w – mass of equipped wing, kg;

C – distance from mean aerodynamic chord leading edge to the center of gravity point, determined by the designer.

$C = (0{,}23...0{,}32)\, B_{MAC}$ – high wing;

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

$$X_{MAC} = \frac{m_f x_f + m_w \cdot x_w^{/} - m_0 C}{m_0 - m_w} \qquad (1.6.)$$

$$X_{MAC} = \frac{132495.3203 + 7873.15 - 23479 \cdot 2.86 \cdot 0.23}{23479 - 10615.8} \approx 9{,}698 \text{ [m]};$$

$$X_T = \frac{X_{цт} - X_{сах}}{b_{сах}} \cdot 100\%$$

Table 1.7 - Calculation of center of gravity positioning variants

Name	Mass, Kg	Coordinate	Mass moment
Object	m_i	X, m	Kg· m
Equipped wing (without fuel and landing gear)	4575,82231	19,53	89369,013
Nose landing gear (extended)	479,47614	5,077	2434,3004
Main landing gear (extended)	1917,90456	18,04	34594,587
Fuel/fuel reserve	19842,29973	19,74	391641,36
Equipped fuselage (without payload)	11977,96261	18,58	222550,55
Passengers of business class	900	8,3	7470
Passengers of economy class	8100	20,2	163620
Baggage	2400	17	40800
Cargo	1677,53	17	28518,01
Crew	520	7,5	3900
Nose landing gear (retracted)	479,47614	4,077	1954,8242
Main landing gear (retracted)	1917,90456	18,04	34594,587
Reserve fuel	2570,98413	20	51419,683

Table 1.8 - Airplanes center of gravity position variants

Variants of the loading	Mass, kg	Moment of the mass, kg· m	Center of mass, m	Centering, X_c, %
Take off mass (L.G. extended)	63591	984897,8149	15,488	27,1
Take off mass (L.G. retracted)	63591	984418,3388	15,480	26,89
Landing weight (LG extended)	35119,67975	644676,1381	18,356	26,18
Ferry version	39313,46535	744010,3288	18,925	21
Parking version	18951,16562	348948,4455	18,413	21,5

Conclusion to the project part

Analyzing the results, that is the parameters and flight characteristics of the designed aircraft, we can conclude that the designed prototype, short-range cargo aircraft fully meets the requirements in the terms of reference for the project (even exceeds them in the class of base aerodrome), meets its purpose in terms of increased transport efficiency and operational capabilities from unprepared runways, and has a reserve to increase load capacity or range.

In the projected part of the diploma work we have considered and then selected the main geometrical parameters for the designing aircraft structure such as wing characteristics, fuselage construction, tail unit part and landing gear parameters. After designing of the wing and the fuselage we have made the calculations of the center of gravity of the equipped aircraft. Masses position of wing, fuselage, and other units in our airplane have been also taken into account.

2. IMPROVEMENT OF THE HOIST COMPONENT DESIGN

2.1. Analysis of overhead cargo equipment

The loading and unloading equipment is designed to lift cargo from the ground or cargo platform installed under the cargo hatch, move cargo along the cargo compartment and install it on a conveyor or cargo floor.

Loading and unloading equipment of our prototype consists of a loading and unloading device and equipment for loading non-self-propelled wheeled loads.

Table 2.1 - Basic characteristics of overhead cargo equipment

№	Parameter	Value
1.	Payload, kg	2000
2.	Carriage stroke between frames. № 28-40,mm	3560
3.	The speed of lifting and lowering the load with a BL-56 electric winch, m/min	at least 1
4.	The speed of lifting and lowering the load from the manual drive (with a crank speed of 35 rpm), m / min	0.25
5.	The force on the handle of the manual drive (with a force of 500 kg on the cable), kg	not more than 15
6.	The maximum height of the load (stroke of the cargo hook), mm	2330

The loading and unloading device is located on the ceiling of the cargo compartment between the frames 28-40 and consists of an electric winch with a cable, a monorail with a lock and a bypass roller, a carriage with a stopper, a movable block with a hook and a traverse.

An electric winch is used for lifting and lowering the load, as well as to keep it at the right level when moving along the cargo compartment. The carriage with the load moves manually on the rollers along the monorail located in the plane of symmetry of the fuselage on the ceiling of the cargo compartment. To limit the movement of the carriage on the monorail installed front and rear stops. An electric winch BL-56 is installed between the frames 28-29 and is attached to them by a overlay 5 (Fig. 2.1.) and a pin 4 mounted on the transverse profiles of the ceiling frame. The ledge of the base 11 of the winch is included in the pad, the second end of the base with its groove is put on the pin and pressed against the nut.

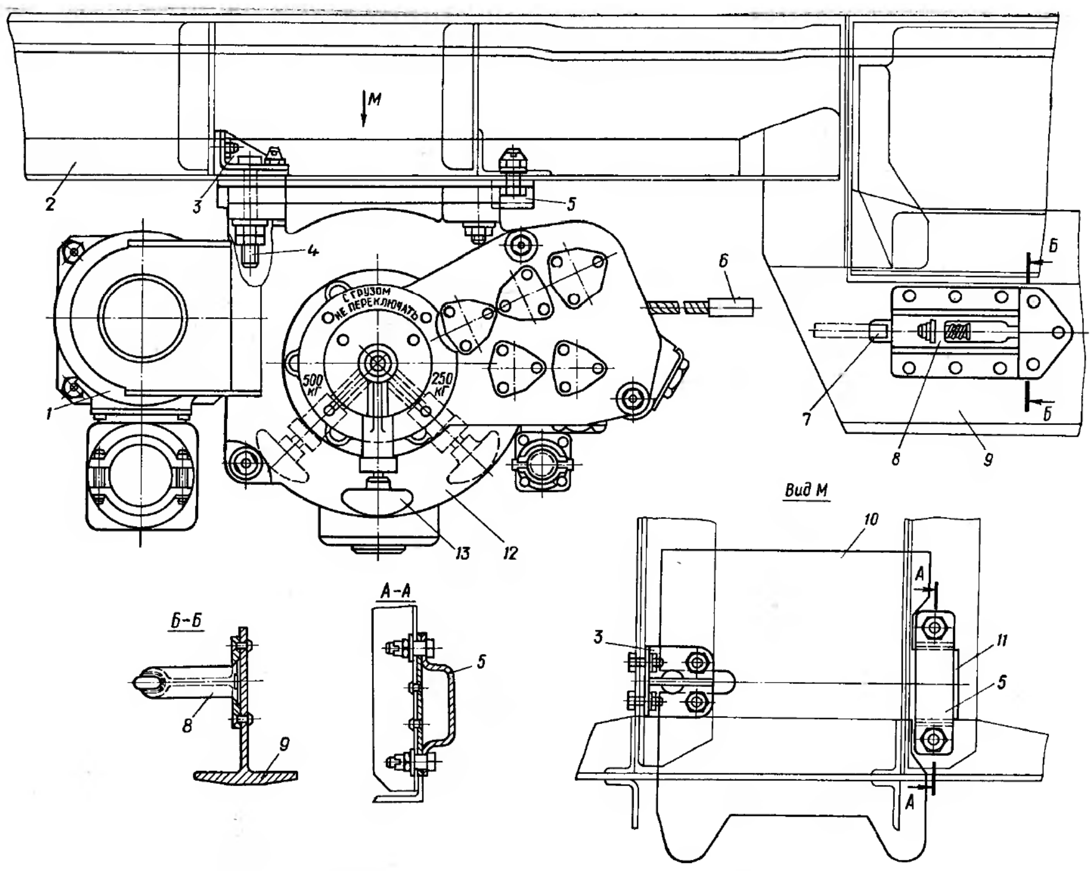

Figure 2.1 – Installation of winch BL-56: 1 – winch electromechanism; 2 – overhead longitudinal beam; 3 – bracket; 4 – pin; 5 – overlay; 6 – a tip of a cable; 7 – spring; 8 – stopper; 9 – monorail; 10 – the base of the winch; 11 – base ledge; 12 – winch body; 13 - winch shift knob for 500 and 250 kg

The clamp (Fig. 2.2.) is intended for fastening the free end of the winch cable. The latch is mounted on the monorail 3 in the dovetail socket and is locked by the spring 2. The axial load from the cable is perceived by the stopper 4. A stopper 8 is installed in the groove of the housing 5 of the clamp body, which is pressed against the cable end 7 by the spring 9. The tip is clamped between the stopper and the end of the latch housing.

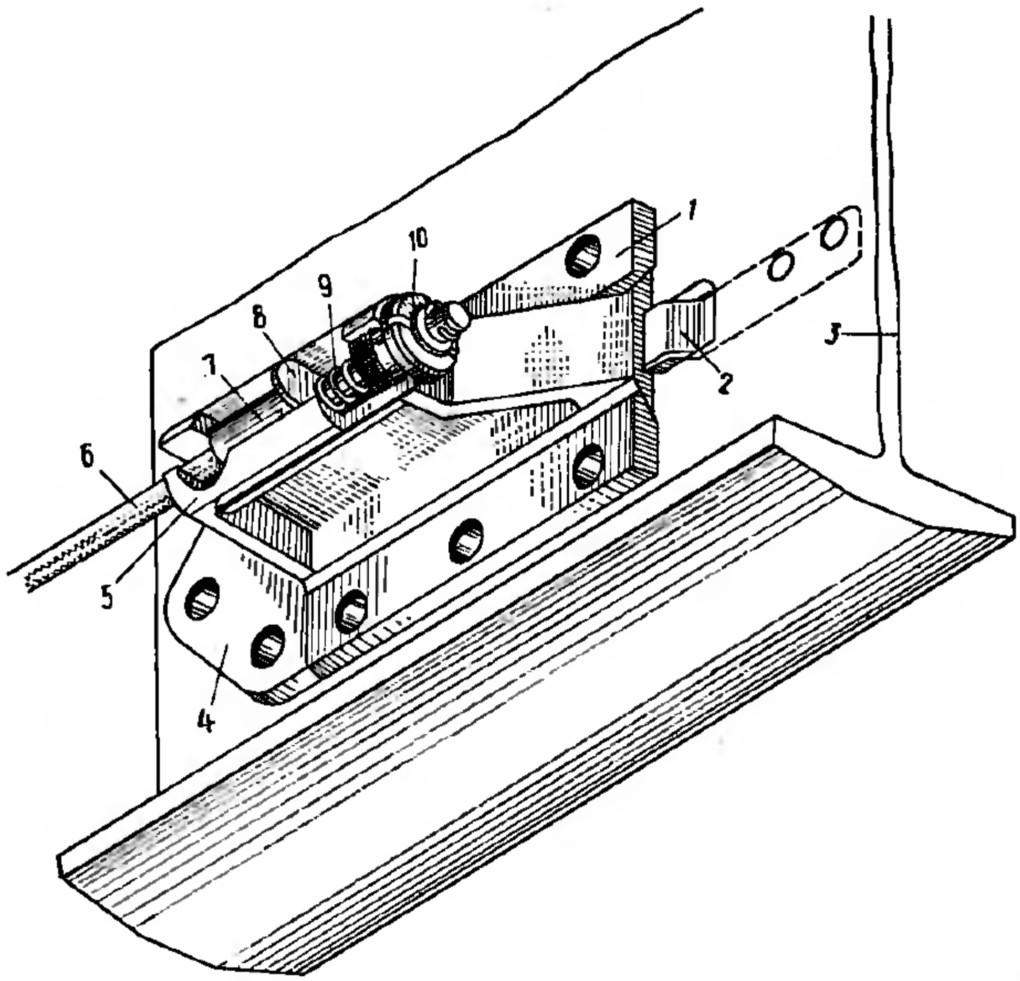

Figure 2.2 – The clamp of the cable end: 1 – overlay; 2, 9 – springs; 3 – monorail; 4 – stopper; 5 – clamp body; 6 – cable; 7 – the end of the cable; 8 – clamp stopper; 10 – screw

The carriage telpher with a loading capacity of 2000 kgf, (Fig. 2.3.) together with two cheeks forms a power frame, on which four roller slides 2, four rollers 4 and a carriage stopper 7 are installed. When moving the carriage along the cargo compartment, the roller slides roll along the flange of the monorail 1. The rollers are fixed blocks of the hoist. The carriage stopper provides its fixation on any part of the monorail during airplane movement. Locking is controlled by a strap 8.

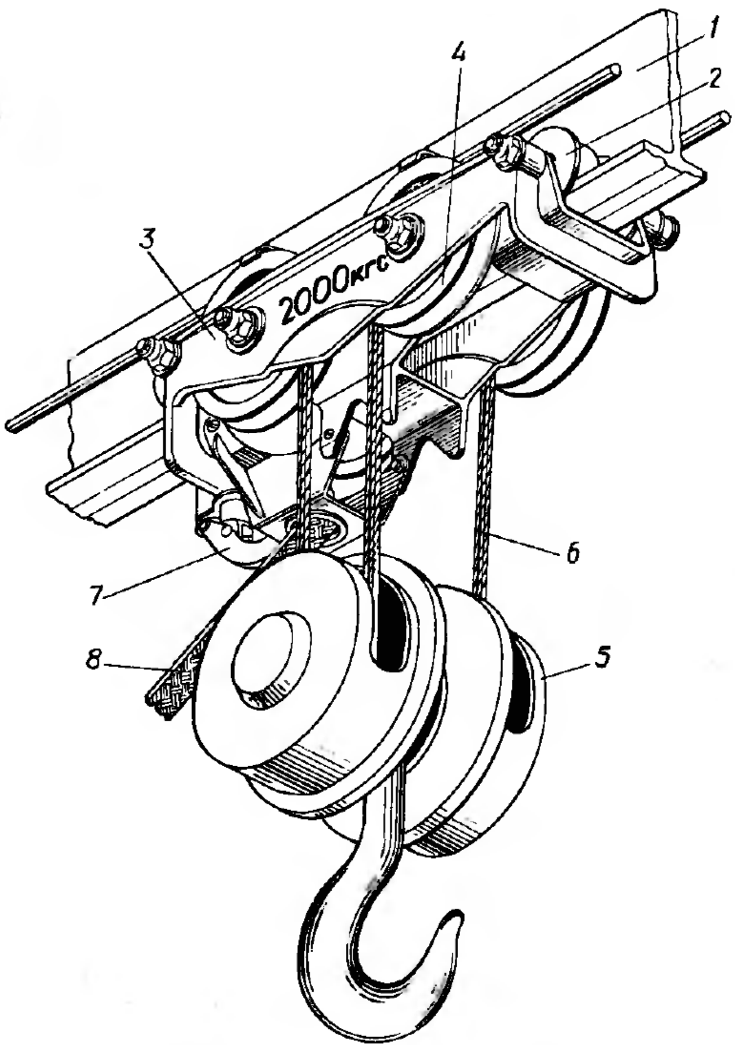

Figure 2.3 – Telpher carriage with a loading capacity of 2000 kg: 1 – monorail; 2 – roller slides; 3 – carriage body; 4 – rollers; 5 – movable block with cargo hook; 6 – cable; 7 – carriage stopper; 8 – strap

Let’s take a look at the hoist design at all. It is mounted to the trolley and performs the actual lifting function via a hook or lifting attachment. Based on this, we set the task to increase the efficiency of use of overhead cargo equipment.

2.2. Hook suspension modification

Hook suspension consists of the case with mechanism inside and hook, suspended to the case outside. Pulley consists of bearing, two inserts, protective washers and holder with groove for the rope, moving from the hoist drum. Shortened suspensions are used for cable single and double pulley blocks with even multiplicity. In a shorter suspension, the axis of the blocks and the crosshead of the hook are made as a whole, and the hook has an elongated shank. On each side of the hook is an equal number of blocks. Shortened pendants are smaller in height than normal pendants. This allows you to increase the height of the load, ceteris paribus. The sizes of hook hangers are normalized. The choice of a typical hook suspension is made from the catalog according to a given value of load capacity, type of chain hoist and its multiplicity, as well as the characteristic of the operating mode.

So, the first thing that I wanted to pay attention to was safe latches, which, in principle, should be installed on hooks in order to prevent the sudden sliding of the load from the hook.

The second problem is about cargo weight that hook can carry. Hook suspension is intended for cargo pick-up and measurement of the cargo weight. Even if we have weighed the load, the better option would be to place the weight indicator on the hook - this will be done primarily in order to estimate the loading weight, additional information about a specific container or item, as well as to avoid overloading of the winch, hook or beam structure as a whole. Each hook must withstand a static load exceeding its carrying capacity by 25%. Given the absence of the weight indicator, we will solve it this way - modify the hook by adding a spring. This spring will indicate the weight that is applied to this hook.

The design of the indicator will be very simple - a spring-type dial balance (Figure 2.4.).

Under the influence of gravity weight, the spring is stretched, respectively, moves on the arrow scale. Based on the position of the arrow, you can find out the weight of the load being weighed (Figure 2.5.). The extension of the spring is described by Hooke's law.

$$F = \Delta l \cdot k, \quad (2.1.),$$

where F is the force by which the spring is extend (compressed), Δl is the absolute elongation (compression) of the rod, and k is the constant factor characteristic of the spring (i.e., its stiffness).

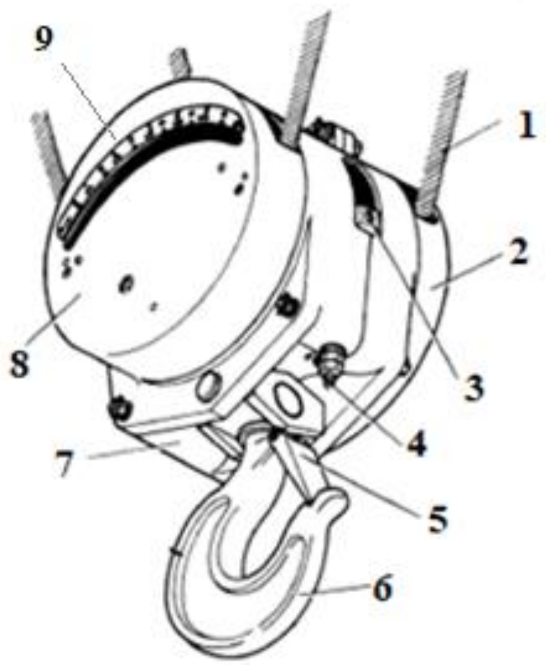

Figure 2.4 – Hook design: 1 – cable; 2 – caseback; 3 – lever; 4 – bolt; 5 – lock; 6 – hook; 7 – housing; 8 –front cover; 9 – scale

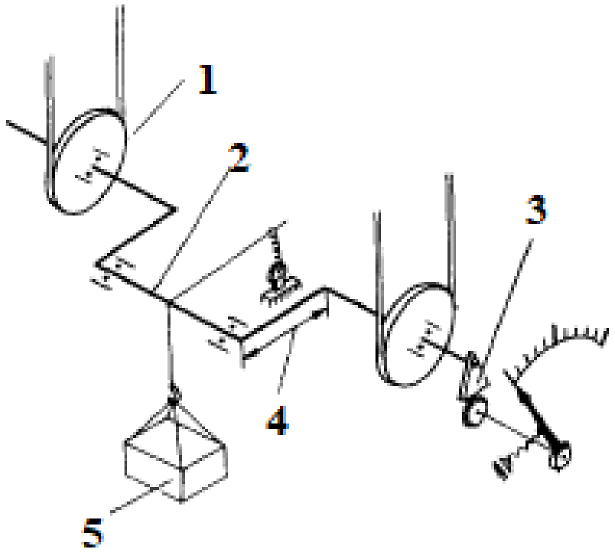

Figure 2.5 – Work concept scheme: 1 – block; 2 – axle; 3 – quadrant; 4 – eccentricity; 5 – cargo

2.2.1. Calculation of hook strength

When designing lifting mechanisms, based on a given load weight, type of drive and operating mode, the hook number is determined in accordance with GOST 6627–74 (Table 2.2.).

This standard applies to the preparation of single-handed hooks with a cylindrical tail, manufactured by hot stamping used in hoisting machines and mechanisms.

When choosing the size of the hook suspension, two conditions must be met:

a) The load capacity of the hook suspension must not be less than the specified load capacity: $Q_{hs} \geq Q_{cargo}$;

b) The mode of operation of the hook suspension must correspond to the mode of operation of the mechanism.

After choosing a hook suspension, we write out its conditional support and technical characteristics. A sketch of the suspension is shown in drawing 4. The sketch shows the overall and mounting dimensions.

After we clarified the size of the hook suspension, we must carry out the strength verification calculation of its individual elements.

Table 2.2 – One-horned hook for cargo weighing 20 kN (GOST 6627–74)

№	Weight, kN	Dimensions of the hook, mm																		
		D	S	b	h	d1	D	L	l1	l2	d2	r	r1	r2=r4	r3	r5	r6	r7	r8	r9
10	20	55	40	34	52	30	35	220	85	45	M30	10	60	8	28	70	38	55	20	2,5

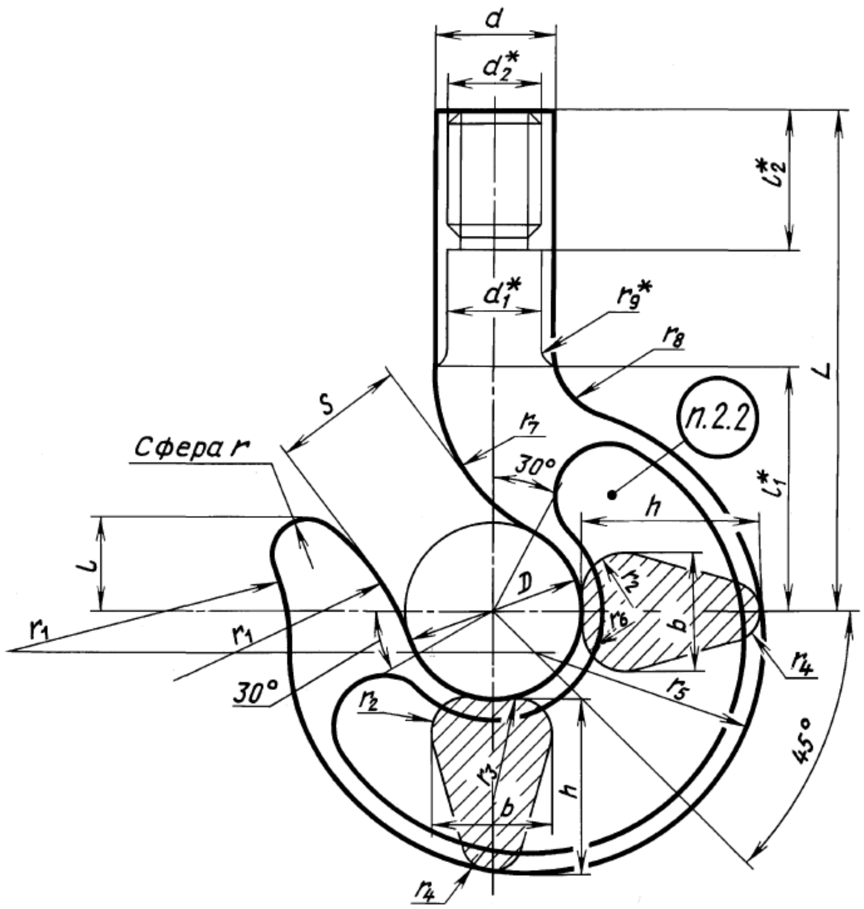

Figure 2.6 – Dimensions of one-horned hook for cargo weighing 20 kN (GOST 6627–74)

One-horned hook is a curved beam, stretched by an external load P. This load causes two stresses in different sections of the hook:

1. One is constant for all sections, depending only on tension and equal to the tensile force Pp divided by the area of the corresponding section F;

2. The second one is a variable from the bending caused in the section due to the eccentric action of the force Pp.

The bending moment acting in any section of the hook is equal to the product of the force Pp by the distance to the neutral axis of the section. The maximum of this moment will be in the horizontal section a-a, where y = ymax (Fig. 2.6., b).

This section, as the most dangerous, is subjected to verification calculation. The shank of the hook, in the sections of which only tensile stresses equal to:

$\sigma = Pp/F$

The remaining sections of the hook are strong if the transitions from the calculated section a-a to the spout and shank are smooth.

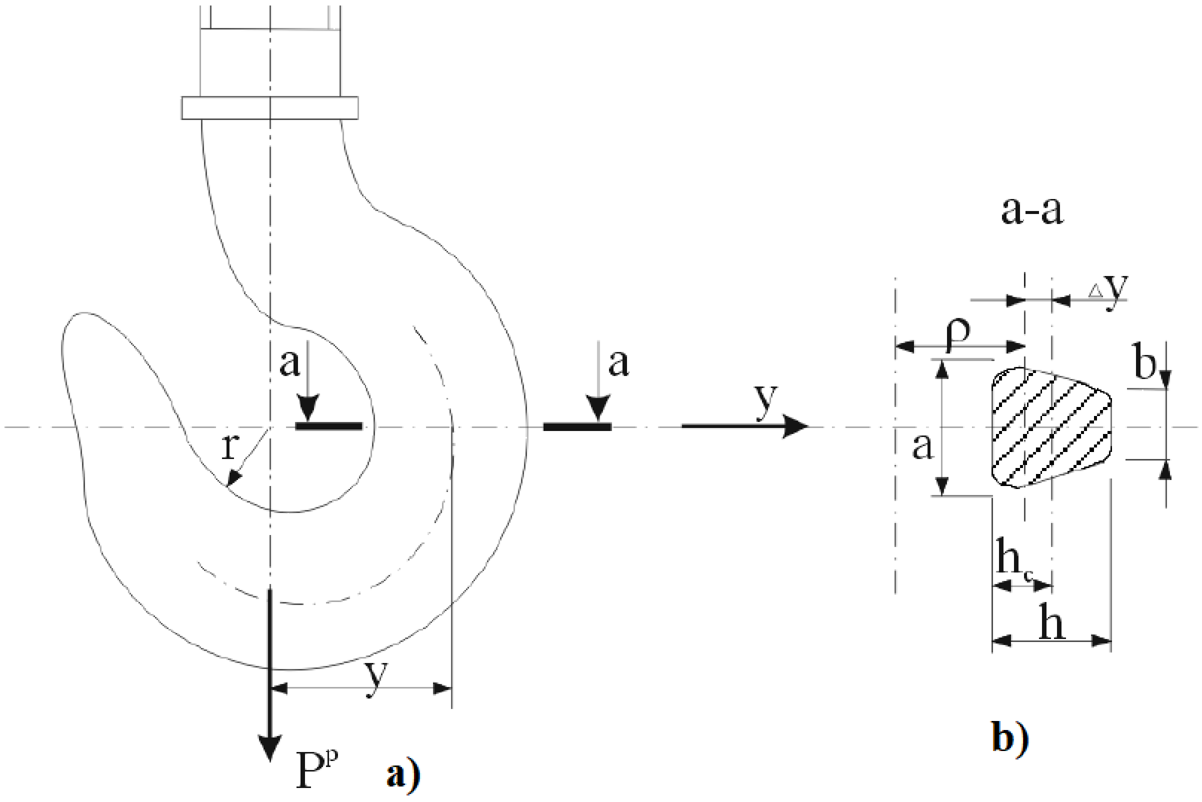

Figure 2.7 – One-horned hook for cargo weighing

In case of our hook suspension capacity (2000 kg) we take material 30ХГСА.

Given data:

- $Pp = 2000$ kg;
- $[\sigma] = 110$ kg/mm^2 = 11000 kg/mm^2;
- $a = 32$ mm;
- $b = 8$ mm;
- $R = 79{,}5$ mm;
- $r = 27{,}5$ mm;

In Appendix D you can see the cross-section of our hook a-a, and its calculation of area and center of mass position.

Area is equal: $F = 1207{,}11$ mm^2 = 12,0711 cm^2;

Center of mass: $h_c = 22{,}24$ mm = 2,224 cm;

The radius of curvature:

$$\rho = \frac{F}{\frac{a \cdot R - b \cdot r}{h} \cdot ln\frac{R}{r} - (a-b)} = \frac{1207{,}11}{23{,}44} = 51{,}48 \text{ [mm]} = 5{,}148 \text{ [cm]};$$

Stress appeared in cross-section inner surface:

$$\sigma_{in} = \frac{Pp}{F} + \frac{M \cdot h_1}{F \cdot |\Delta y| \cdot r};$$

Stress appeared in cross-section outer surface:

$$\sigma_{out} = \frac{Pp}{F} - \frac{M \cdot h_2}{F \cdot |\Delta y| \cdot (r+h)};$$

$M = Pp \cdot \rho = 2000 \cdot 51{,}48 = 102978$ [kg·mm] = 10297,8 [kg·cm]

$h_1 = \rho - r = 2{,}398$ cm;

$h_2 = h - h_1 = 2{,}801$ cm;

$\Delta y = h_c - h_1 = -\,0{,}174$ cm;

$\sigma_{in} = 593{,}21$ kg/cm^2;

$\sigma_{out} = -14{,}5$ kg/cm^2;

$$\eta = \frac{[\sigma]}{\sigma_{int}} = \frac{11000}{593{,}21} = 18{,}54;$$

$$\eta = \frac{[\sigma]}{|\sigma_{out|}} = \frac{11000}{14{,}5} = 758,\ 62;$$

Also we have provided the simulation of our hook loading in Solidworks, and obtained that hook suspension meet our requirements.

2.2.2. Hook suspension spring

The spring, which will be subjected to great stretching, should be selected based on GOST 18794-80 standards.

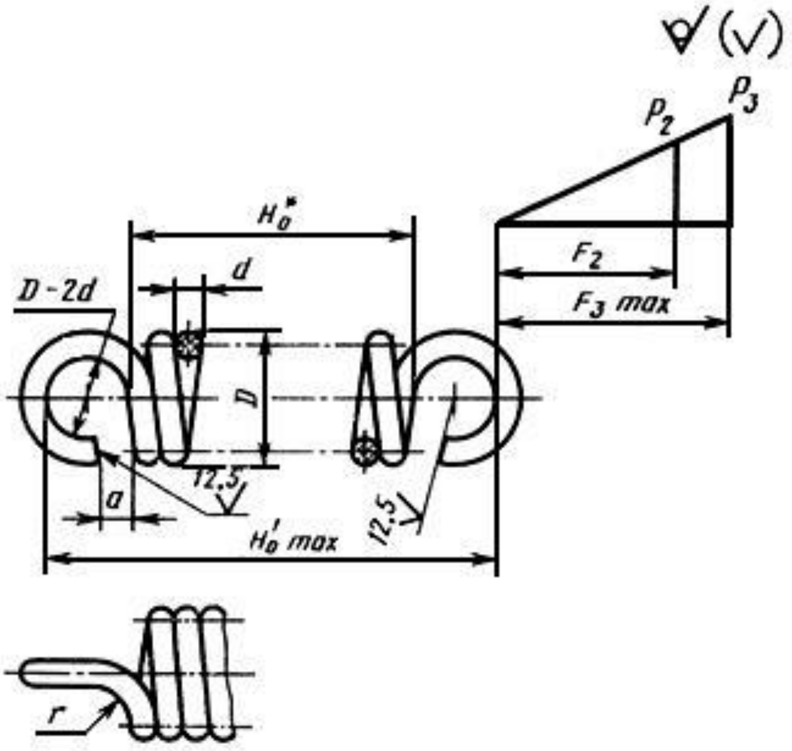

Figure 2.8 – Spring dimensions

Table 2.3 – GOST standarts on dimensions

Spring designation	Main parameters of coils		P_2, N (kgf)	P_3, N (kgf)	D,mm	d, mm	H_o,mm	H'$_o$ max,mm
	Standart designation	Number of spring						
1086-0304	GOST 13771-68*	208	23,60 (2,36)	25,00 (2,50)	8,0	0,8	8,8	21,6
1086-0305							13,6	26,4
1086-0306							20,8	33,6

General conclusions

In the process of working with this diploma, the following goals were set and achieved:

1. Preliminary design short-range cargo aircraft with payload 5,5 tons:

   a). The schematic design of the layout of the short-range cargo aircraft;

   b). The geometrical and mass parameters selection;

   c). The main aircraft components mass calculation (center of gravity position);

   d). The selection of the engine for the required power.

2. Improving of the hoist design:

   a). Calculation of the strength of the hook;

   b). Modification of the hook by adding a safer lock and weight scale.

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П Р О Е К Т
С А М О Л Е Т А С Т В Д
НАУ, кафедра К Л А

ПРОЕКТ дипломный Расчет выполнен 10.09.2019
Исполнитель Кравченко В.Д. Руководитель Карускевич М.В.

ИСХОДНЫЕ ДАННЫЕ И ВЫБРАННЫЕ ПАРАМЕТРЫ

Количество пассажиров	0.
Количество членов экипажа	2.
Количество бортпроводников или сопровождающих	2.
Масса снаряжения и служебного груза	674.64 кг.
Масса коммерческой нагрузки	5500.00 кг.
Крейсерская скорость полета	440. км/ч
Число "М" полета при крейсерской скорости	0.3855
Расчетная высота начала реализации полетов с крейсерской экономической скоростью	6.000 км
Дальность полета с максимальной коммерческой нагрузкой	1100. км.
Длина летной полосы аэродрома базирования	1.68 км.
Количество двигателей	2.
Оценка по статистике энерговооруженности в квт/кг	0.2800
Степень повышения давления	28.00
Относительная масса топлива по статистике	0.2200
Удлинение крыла	11.37
Сужение крыла	2.92
Средняя относительная толщина крыла	0.120
Стреловидность крыла по 0.25 хорд	6.5 град.
Степень механизированности крыла	0.800
Относительная площадь прикорневых наплывов	0.000

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

Диаметр фюзеляжа	2.90 м.
Удлинение фюзеляжа	8.20
Стреловидность горизонтального оперения	15.0 град.
Стреловидность вертикального оперения	21.0 град.

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

Значение оптимального коэффициента под'емной силы в расчетной точке крейсерского режима полета Cy 0.50122
Значение коэффициента Сх.инд. 0.00998

ОПРЕДЕЛЕНИЕ КОЭФФИЦИЕНТА Dm = Mкрит - Mкрейс

Число Маха крейсерское	Mкрейс	0.38545
Число Маха волнового кризиса	Mкрит	0.68130
Вычисленное значение	Dm	0.29585

Ch.	Shee	Doc. №	Sign.	Date	NAU 20 08K 00 00 00 56 EN		
Performed	KravchenkoV.				Appendix A	Letter	Sheet 1	Sheets 3
Checked	Karuskevich					402 AF 134	
Adviser							
Stand.	Khvzniak S.						
Head of	Ignatovych						

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

при взлете	2.573
в середине крейсерского участка	2.472
в начале крейсерского участка	2.520

Значение коэффициента сопротивления фюзеляжа и гондол	0.00796
Значение коэфф. профиль. сопротивления крыла и оперения	0.00998

Значение коэффициента сопротивления самолета:

в начале крейсерского режима	0.03164
в середине крейсерского режима	0.03144

Среднее значение Су при условном полете по потолкам	0.50122
Среднее крейсерское качество самолета	15.94081

Значение коэффициента Су.пос.	1.744
Значение коэффициента (при скорости сваливания) Су.пос.макс.	2.617
Значение коэффициента (при скорости сваливания) Су.взл.макс.	2.233
Значение коэффициента Су.отр.	1.608
Энерговооруженность в начале крейсерского режима	0.091
Стартовая энерговооруженн. по условиям крейс. режима No.кр.	0.132
Стартовая энерговоруж. по условиям безопасного взлета No.взл.	0.131

Расчетная энерговоруженность самолета No 0.135

Отношение Dn = No.кр / No.взл Dn 1.009

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

взлетный	0.2881
крейсерский (характеристика двигателя)	0.2468
средний крейсерский при заданной дальности полета	0.2476

ОТНОСИТЕЛЬНЫЕ МАССЫ ТОПЛИВА:

аэронавигационный запас	0.02356
расходуемая масса топлива	0.08393

ЗНАЧЕНИЯ ОТНОСИТЕЛЬНЫХ МАСС:

крыла	0.15432
горизонтального оперения	0.01627
вертикального оперения	0.01618
шасси	0.04944
силовой установки	0.10032
фюзеляжа	0.15973
оборудования и управления	0.1349
дополнительного оснащения	0.00213
служебной нагрузки	0.02497
топлива при Lрасч.	0.10749
коммерческой нагрузки	0.23425

Взлетная масса самолета "М.о" = 23479. кГ.

Потребная взлетная мощность двигателя 1585.6 kВт

Относительная масса высотного оборудования и противообледенительной системы самолета	0.0197
Относительная масса пассажирского оборудования (или оборудования кабин грузового самолета)	0.0014

Относительная масса декоративной обшивки и ТЗИ	0.0107
Относительная масса бытового (или грузового) оборудования	0.0067
Относительная масса управления	0.0112
Относительная масса гидросистем	0.0274
Относительная масса электрооборудования	0.0300
Относительная масса локационного оборудования	0.0045
Относительная масса навигационного оборудования	0.0068
Относительная масса радиосвязного оборудования	0.0034
Относительная масса приборного оборудования	0.0079
Относительная масса топливной системы (входит в массу "СУ")	0.0031
Дополнительное оснащение:	
Относительная масса контейнерного оборудования	0.0000
Относительная масса нетипичного оборудования	0.0021

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

ХАРАКТЕРИСТИКИ ВЗЛЕТНОЙ ДИСТАНЦИИ

Скорость отрыва самолета	180.82 км/ч
Ускорение при разбеге	1.48 м/с*с
Длина разбега самолета	851. м.
Дистанция набора безопасной высоты	578. м.
Взлетная дистанция	1430. м.

ХАРАКТЕРИСТИКИ ВЗЛЕТНОЙ ДИСТАНЦИИ
ПРОДОЛЖЕННОГО ВЗЛЕТА

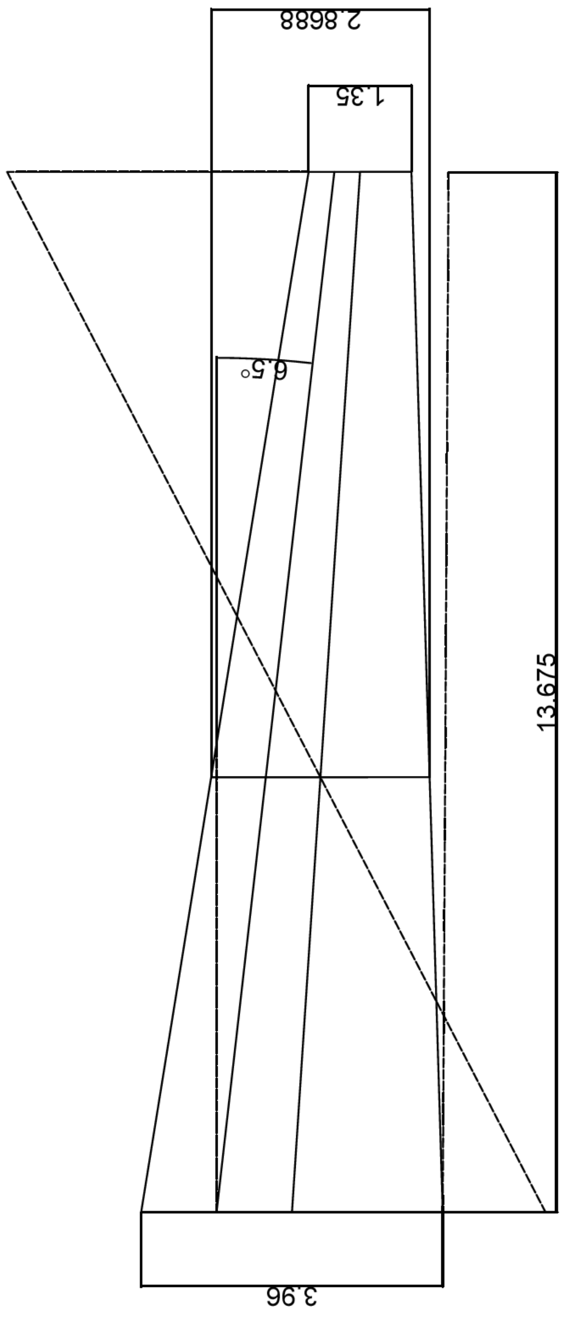
Скорость принятия решения	171.77 км/ч
Среднее ускорение при продолженном взлете на мокрой ВПП	0.06 м/с*с
Длина разбега при продолженном взлете на мокрой ВПП	2788.23 м.
Взлетная дистанция продолженного взлета	3336.68 м.
Потребная длина летной полосы по условиям прерванного взлета	3459.08 м.

ХАРАКТЕРИСТИКИ ПОСАДОЧНОЙ ДИСТАНЦИИ

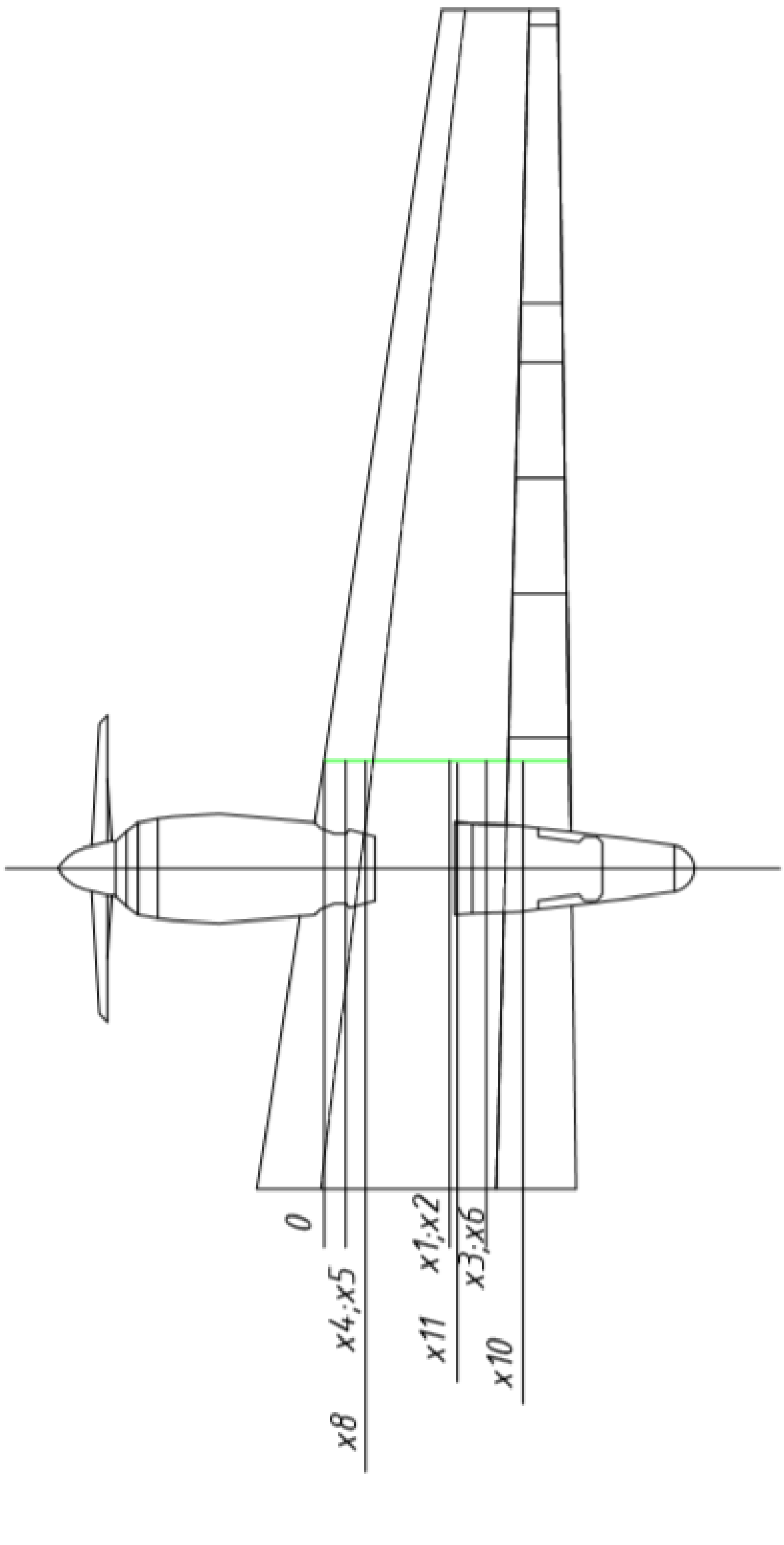
Максимальная посадочная масса самолета	22593.	кг.
Время снижения с высоты эшелона до высоты полета по кругу	12.2	мин.
Дистанция снижения	14.94	км.
Скорость захода на посадку	184.39	км/ч.
Средняя вертикальная скорость снижения	1.59	м/с
Дистанция воздушного участка	493.	м.
Посадочная скорость	169.09	км/ч.
Длина пробега	430.	м.
Посадочная дистанция	923.	м.
Потребная длина летной полосы (ВПП + КПБ) для основного аэродрома	1541.	м.
Потребная длина летной полосы для запасного аэродрома	1310.	м.

ПОКАЗАТЕЛИ ЭФФЕКТИВНОСТИ САМОЛЕТА

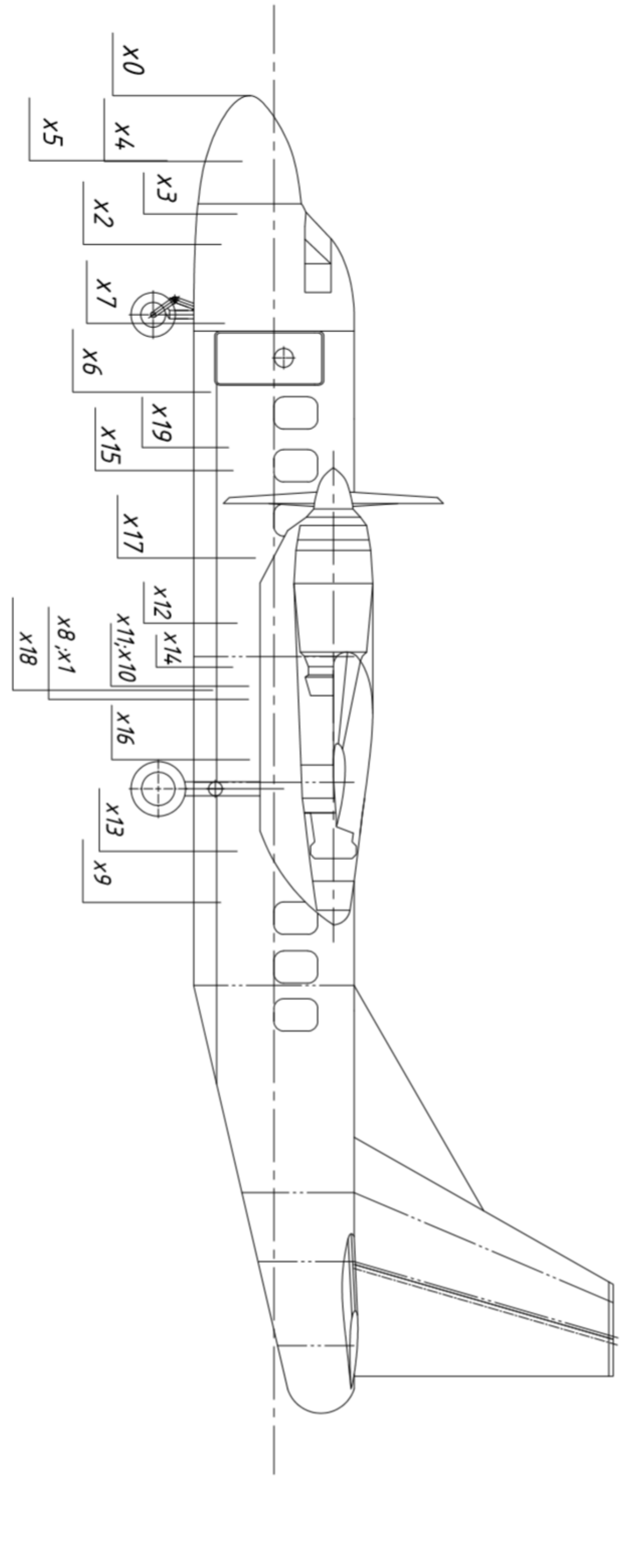
Отношение массы снаряженного самолета к массе коммерческой нагрузки	2.8008
Масса пустого снаряженного с-та приход. на 1 пассажира	0.00 кг/пас.
Относительная производительность по полной нагрузке	150.37 км/ч
Производительность с-та при макс. коммерч. нагрузке	2160.7 т*км/ч
Средний часовой расход топлива	703.750 кг/ч
Средний километровый расход топлива	1.79 кг/км
Средний расход топлива на тоннокилометр	325.703 г/(т*км)
Средний расход топлива на пассажирокилометр	0.0000 г/(пас.*км)
Ориентировочная оценка приведен. затрат на тоннокилометр	0.3297$/(т*км)



Ch.	Shee	Doc. №	Sign.	Date	NAU 20 08K 00 00 00 56 EN
Performed	KravchenkoV.				Appendix B
Checked	Karuskevich				Letter: / Sheet: 1 / Sheets: 2
Adviser					
Stand.	Khvzniak S.				402 AF 134
Head of	Ignatovych				

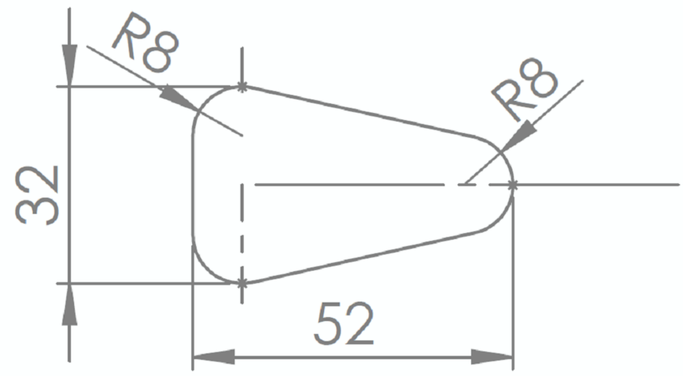


					NAU 20 08K 00 00 00 56 EN	Sh.
Ch.	Sheet	Doc. №	Sign.	Date		2



					NAU 20 08K 00 00 00 56 EN
Ch.	Shee	Doc. №	Sign.	Date	
Performed		Kravchenko			Appendix C
Checked		Karuskevich			
Adviser					
Stand.		Khvzniak S.			
Head of		Ignatovych			

Letter	Sheet	Sheets
402 AF 134		



Section properties of Sketch1 of Part1

Area = 1207.11 millimeters^2

Centroid relative to sketch origin: (millimeters)
X = 22.24
Y = 0.00

Centroid relative to part origin: (millimeters)
X = 22.24
Y = 0.00
Z = 0.00

Moments of inertia of the area, at the centroid: (millimeters ^ 4)
Lxx = 66010.51 Lxy = 0.00 Lxz = 0.00
Lyx = 0.00 Lyy = 233364.71 Lyz = 0.00
Lzx = 0.00 Lzy = 0.00 Lzz = 299375.21

Polar moment of inertia of the area, at the centroid = 299375.21 millimeters ^ 4

Angle between principal axes and sketch axes = -0.00 degrees

Principal moments of inertia of the area, at the centroid: (millimeters ^ 4)
Mx = 66010.51
My = 233364.71

UNLESS OTHERWISE SPECIFIED:
DIMENSIONS ARE IN MILLIMETERS
SURFACE FINISH:
TOLERANCES:
LINEAR:
ANGULAR:

Aircraft design department

DEBURR AND BREAK SHARP EDGES

NAU 20 08K 00 00 00 56EN

NAME	SIGNATURE	DATE
Performed by	Kravchenko V.D.	
Supervisor	Karuskevich M.V.	
Adviser		
Stand.contr.	Khizhnyak S.V.	
Head. of dep.	Ignatovich S.R.	

APPENDIX D

MATERIAL:

WEIGHT:

DWG NO.

Section a-a

A4

SCALE:1:2

SHEET 1 OF 1