МІНІСТЕРСТВО ОСВІТИ І НАУКИ УКРАЇНИ

НАЦІОНАЛЬНИЙ АВІАЦІЙНИЙ УНІВЕРСИТЕТ

Кафедра конструкції літальних апаратів

«ДОПУСТИТИ ДО ЗАХИСТУ»
Завідувач кафедри
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(підпис)
« _____»____________________ 2021 р.

ДИПЛОМНИЙ ПРОЕКТ

(ПОЯСНЮВАЛЬНА ЗАПИСКА)

ВИПУСКНИКА ОСВІТНЬО-КВАЛІФІКАЦІЙНОГО РІВНЯ

«БАКАЛАВР»

Тема: «Аванпроект ближньо-магістрального літака вантажепідйомністю до 5,5 тон»

Виконав: ____________ Мі Лен

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Київ 2021

MINISRY OF EDUCATION AND SCIENCE OF UKRAINE

NATIONAL AVIATION UNIVERSITY

Aircraft Design Department

AGREED
Professor, Dr. of Sc.
_________S.R. Ignatovych
«___» _______ 2021

DIPLOMA WORK

(EXPLANATORY NOTE)

OF EDUCATIONAL DEGREE

«BACHELOR»

Theme: «Preliminary design of short-range aircraft with cargo capacity up to 5.5 tons »

Performed by: __________ Leng Mi

Supervisor: Dr. of Science, Professor __________ M. V. Karuskevich

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

Kyiv 2021

NATIOONAL AVIATION UNIVERSITY

Aerospace Faculty

Aircraft Design Department

Educational degree «Bachelor»

Speciality 134 "Aviation and Space Rocket Technology"

APPROVED
Professor, Dr. of Sc.
_________S.R. Ignatovych
«____ » ___________ 2021

TASK
for bachelor diploma work
Leng Mi

1. Theme: «**Preliminary design of short-range cargo aircraft**»

Confirmed by Rector's order from …….. year № …..

2. Period of work execution __________________ to __________________

3. Work initial data:

– Maximum payload – m = 5.5 tons;

– Flight range with maximum payload – $L_{пол}$= 1100 km;

– Cruise speed – V_{cr} = 440 km/hour at operating altitude H_{op}= 5800 m;

– Landing speed – V_{land} =170 km/hour.

4. Explanation notes (list of topics to be developed):

– selection of design parameters;

– choice and substantiations of the airplane scheme;

– calculation of aircraft masses;

– determination of basic geometrical parameters;

– aircraft layout;

– center of gravity position calculation;

– determination of basic flight performance;

– description of the aircraft design;

– engine selection;

– special part.

5. List of the graphical materials:

– general view of the airplane (A1×1);

– layout of the airplane (A1×1);

– assembly drawing of the winch (A1×2).

6. Calendar Plan

№ п/п	Task	Execution period	Signature
1	Task receiving, processing of statistical data	15.05.21	
2	Aircraft take-off mass determination	19.05.21	
3	Aircraft layout	19.05.21	
4	Aircraft centering determination	25.05.21	
5	Graphical design of the parts	25.05.21	
6	Preliminary defence		
7	Completion of the explanation note		

7. Task date: «___» ____________ 2021

Supervisor of diploma work: ________________ M.V. Karuskevich
(signature)

Task is given for: _________________ Leng Mi
(signature)

ABSTRACT

Explanatory note to the diploma work « **Preliminary design of short-range cargo aircraft** » contains:

63 sheets, 14 figures, 11 tables, 20 references and 4 drawings.

Object of the design is development of cargo short-range aircraft with cargo capacity 5.5 tons.

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

Methods of the work – analysis and synthesis of technical, geometrical and mass characteristics.

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, cargo hold roller system concept, calculations and drawing.

AIRCRAFT, PRELIMINININARY DESIGN, LAYOUT, RANGE OF CENTER OF GRAVITY POSITIONS, ROLLER SYSTEM.

CONTENT

LIST OF ABBREVIATIONS

LG	Landing gear
TLG	Tricycle landing gear with front single-strut and two main gears
APU	Auxiliary power unit
SRA	Short-range airplane
CM	Center of mass
CCAR-25	China Civil Aviation Regulations 25
AC	Aircraft
AE	Aircraft equipment
MAC	Mean aerodynamic chord
CLS	Cargo loading system
PDU	Power drive unit
CAGR	Compound annual growth rate
STOL	Short taking-off and landing

LIST OF DRAWINGS

№ п/п	Name of drawings	Format	Number of sheets
1	Cargo SRA (general view)	A1	1
2	Cargo SRA (layout)	A1	1
3	Roller system (assembly drawing)	A1	1
4	Detailed view of the cargo hold roller system	A1	1

INTRODUCTION

In 2020, air transportation is playing an important role in international cooperation that fight against COVID-19. According to the Air Transport Global Market Report 2021, with recovering from the COVID-19 impact, the global air transport market is expected to grow from $571.52 billion in 2020 to $648.87 billion in 2021 at a compound annual growth rate (CAGR) of 13.5%. And the market is expected to reach $832.79 billion in 2025 at a CAGR of 6% [1].

Asia Pacific was the largest region in the global air transport market, accounting for 32% of the market in 2020. North America was the second large region accounting for 30% of the global air transport market. Africa was the smallest region in the global air transport market. The Asia, especially China, and North America will be the hot region of the air transportation. Therefore, it is worth designing a new airplane that is suitable for the feature of Asia air transport market [1].

The development of air transportation needs support from airport, however, the airport of many medium or mountain cities are small and the runway is shorter and simple. According to the condition of airports for most cities in China, the regional aircraft will be widely used in future.

The aim of the work is to design cargo short-range aircraft with payload 5.5 tons.

The airplane is cantilever high wing with conventional empennage and 2 AI-24VT turboprop engines under the wing. Tricycle landing gear with a nose single-strut landing gear and two main landing gears under the fuselage.

There are two doors in the one side of the fuselage for the crews and cargos getting in to the airplane, and in the rear of the fuselage, there is a catch for loading and unloading cargo. A ramp can be used to load and unload cargo; an electrical hoist is on the floor of the cabin to lift the cargo. On the floor there is a roller system to transport cargo into the cargo cabin.

This airplane is designed as a regional commercial transporter and has ability to take off and land on poor runway. To meet this requirement, the transporter should be short range, economic payload and short taking-off and landing. Last but not least, this aircraft should satisfy the safe, reliable, economic requirement.

For the airworthiness requirement of this aircraft, it need to satisfy the regulations of the national and international administration, such as all rules and regulations about CCAR-25, FAR-25 and CS-25 [2]. In a word, this aircraft is high level safety; low fuel consumption; short taking-off and landing; high payload, easily loading and unloading.

The special part of this project is aim to design the new cargo hold roller system. the key point is designing the roller system that meets the strength requirement and quickly load and unload cargo containers.

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PART 1. PRELIMINARY DESIGN OF THE PLANE

1.1. Analysis of similar planes

Three phases are usually used in aircraft design; conceptual, preliminary, and detail phases. In the conceptual design phase, the decision process and selection techniques will decide the all parameters of aircraft. In contrast, the preliminary design phase will employ the outcomes of calculation procedure. In this stage, three fundamental parameters are determined: (1) aircraft maximum take-off weight; (2) wing reference area; and (3) engine thrust. These main parameters determine the aircraft size, manufacturing cost, and the complexity of calculation [3].

There are two techniques are usually adopted in the preliminary design stage. First, a technique considered the statistics is used for determination of the three fundamental parameters. Second is based on the airplane performance requirements.

“Analysis and Synthesis [4]” is the widely used method in the designing procedure. At the beginning of the preliminary design, we usually choose the similar prototypes that have base of similar cruising speed, cruising altitude, payload, take-off weight, etc.

Therefore, when we choose the prototypes, it is important to select the successful and advanced airplane that was already achieved. Therefore, we can consider the airplane that was designed by the Ukraine Antonov company. Secondly Avions de transport regional (ATR) company is also great example in reginal airplane. And Xi'an Aircraft Company Limited of China is rising star in transport aircraft with relying on the huge Chinese air transport market. These three companies have long and successful history in designing cargo airplane.

The aircraft Yun-7, Antonov 26 and ATR 42 have been chosen for the preliminary design stage.

Statistic data of aircraft are presented in table 1.1.

Table 1.1. – Statistic data of prototypes

PARAMETER	AIRCRAFT		
	Yun-7	AN 26	ATR 42
The purpose of airplane	Cargo	Cargo	Cargo
Crew/flight attend (Persons)	3/3	4/4	4/4
Maximum take-off weight, m_{tow}, kg	21800	24000	18600
Passenger's seat	3	4	4
The height of the flight $V_{w.\ ек.}$, m	8553	7500	7600
Most pay-load, $m_{c.max}$, kg	5500	5500	5450
Range $m_{к.max}$, km	910	1000	1130
Take off distance $L_{зл.д.}$, m	640	870	1040
Landing distance, m	645	650	1030
Landing speed, km / h	165	190	190
Cruising speed, Vkm/h	450	440	540
Number and type of engines	2×Dongan WJ-5A	2×AI24VT	2×P&WC PW121
Cruising power of engines, kW	2×2162	2×3835	2×2925
Pressure ratio	6.4	7,65	7,5
Fuel consumption of cruising, kg/hour	800	1000	650
Mass of fuel, kg	2145	5500	4500
Landing gear scheme	TLG	TLG	TLG
The form of the cross-section fuselage	Circular	Circular	Circular
Length of aircraft, m	24.22	23,87	22,7
Height of aircraft, m	8.55	8,575	7,59
Diameter of fuselage, m	2,9	2,9	2,87
Extension of the fuselage	8,2	8,2	7,83
Wingspan, m	29,6	29.2	24,57
Aspect ratio	11,69	11,37	11,07
Taper ratio	3.2	2,92	2,5
Sweepback on 1/4 chord, °	6°50'	6°50'	5°25'

The relative position and shape of units of airplane determine the aerodynamic scheme of the aircraft. The aircraft layout decides the aerodynamic properties of the aircraft. Therefore, successful prototypes can bring great benefit on the safety and economy. In this work, we will improve cargo hold equipment that have been used on modern aircraft. Hence, we will make some unobtrusive changes in the design of main units (wing, landing gear, fuselage) to satisfy the new requirement of safety, technology, and economic.

At the starting process, we need to study and analysis the data of the prototypes to find the possibility that selects the optimum characteristic for new aircraft.

1.2. New plane description

The transport airplane is designed to carry load 5.5 tons of the cargo.

I have chosen An-26 [5] as prototype because of relatively similar characteristics, economic fuel consumption and parameter that is convenient to carry cargo for Asian domestic transportation. The width of cargo door is 2.4 m and a ramp-door of rear fuselage to conveniently load and unload cargo.

The aircraft will be used for the Chinese cargo company or express company in mid-short range, it will have good taking-off and landing performance for simple airport.

This aircraft is designed to carry payload with the speed of 440 km/h, the range is 1100 km, and the altitude is 7.5 km.

It has sweep wing. The wing structure includes the skin, ribs, two spars and stringers. And there are a vertical tail unit and horizonal tail wings on the rear of fuselage.

The fuselage of this cargo aircraft is all-metal pressurized and semi-monocoque fuselage. The fuselage consists of two cabins, they are cockpit and cargo cabin. There are 56 frames, and the largest crosses section of the fuselage is in the area between 9-28.

The single piece or a built-up construction of the fuselage structure element are made of sheet and profiled duralumin D16AT, V95.

In the past manufacture of airframe, riveting, bolting, adhesive is widely use. With the development of the new technology, monolithic large-sized panels, plastics, oriented organic glass, high-strength aluminum alloys and steel are also widely used for the manufacture of airframe.

On the front of the fuselage, it is cockpit which is between frame 1 to frame 7. There is a door that is installed at frame 7; it can open be for crew into cargo hold. there is a radar antenna installed on the frame. The cockpit and its lower part are pressurized and sealed [6]. The compartment for nose landing gear is located the area where it is under the floor of frames 1-4. The later of the crew cabin is between the frames 2-5. the radio operator window is installed on the right side between frames 6-7. And on the left side of frames 5-6, it is the navigator blister. The instrumentation and control are located on the pilot's dashboard which consist of three panes (left, middle and right). In addition, for the emergency conditions, an emergency hatch is installed on the upper part of the cockpit, which is between the frame 5 to frame 7. Besides, a lower emergency hatch is designed too. So, there is two emergency hatches separately installed on the fuselage.

The cargo compartment starts from frame 10 to frame 38, which provides spaces for the cargo, sanitary equipment, blocks and components of aircraft system avionics and radio electronics. There are two doors used for the crew and cargo. Four round windows that avoid stress concentration are installed on the cargo cabin in both of sides of the fuselage. Between the frames 14-15, there is a window in the starboard side and it combined with emergency hatch. Otherwise, there are 8 windows in the starboard side.

There is wing center section is installed on the top of fuselage between the frames 20-23.

The empennage is carried by the rear fuselage. The compartment provides the space for the navigation and radio equipment of the cargo airplane. The structure of the fuselage includes the crew cabin and cargo hold deck floor, fuselage frames, windows, crew and cargo doors and emergency hatches. The fuselage is consisted by the 54 frames. Some of frames are normal circular section, some of frames are intended to carry concentrated loads, these are reinforced.

1.2.1. Wing

The wing of the cargo plane is sweep wing because the plane is subsonic. The wing is a composition of three parts, nose, caisson (torsion box) and rear parts. There are mechanical control surface and high lift device installed on the wing to provide short taking-off and landing (STOL) and maneuverability of the aircraft.

The structure of the wing is caisson type, included 2 spars, 25 ribs, skin and stringers. For the skin, different areas of the wing have different thickness. The thickness of the skin panels is bigger near the root, where the bending moment is bigger. To prevent the icing, the leading edge has air heat device. The hot air is taken from the engine compressor.

The spar is I-beam type with extruded aluminum profiles and there are two brackets of the spars to dock with fuselage.

The center ribs are reinforced beam with a duralumin web. Rib reinforced upright from the corner of the extrusion, as well as the upper and lower belts from the T-section of the extruded profile. The sealed fuel tank is in the wing middle part (torsion box).

For the detachable outer part of the wing, it is structurally similar to the central and intermediate part of the wing. It can be attached to the center section with the help of curves and end connector support bars.

The plane has simple High Lift Devices. A single-slotted center wing flap located between the fuselage and the engine nacelle on each wing, and a double-slotted flap is in the middle part of the wing.

The structure of flap is skin, a set of ribs, two carriages and a spar. The brackets are used to fasten flaps.

For the flap retraction and extending, the pilot can press a switch mounted on the pilot's central control panel to control the drive shaft and the six screw jacks. For emergency flap release, a dual rocker switch mounted in the same position can be used.

The flaps can be deflected at an angle of 15° for takeoff and 38° for landing, with a position indicator mounted on the center console to control the flap deflection angle.

For the ailerons, they are installed to the rear part of the removable wing. They are consisting of the roots and end sections. The aileron consists of the spar, skin and ribs. On the rear left aileron, a trim and serve tab are installed, on the right rear aileron, here is only servo tab.

The differential design of the ailerons prevents the problem of adverse yaw.

1.2.2. Tail Unit

The empennage is swept and conventional layout where the vertical tail is located on the top fuselage, and the horizontal tail is located the sides of the fuselage.

The horizontal tail includes a fixed surface, stabilizer, and movable surface, elevator. The elevator is balanced with horn and axial compensation, where there are two servo sheets internal and external. For the vertical empennage, it is consisted of fin, fence and rudder. The rudder has a servo tab and horn balance and axial compensation.

The horizontal stabilizer's dihedral angle is 9°.

There are two symmetrical consoles for stabilizer. There are two panels in respective upper and lower direction, and a tail, a forward and an end fairing. 2 half-spars, ribs, skin and stringers are included in the stabilizer. The stringers are fixed to sleeves with spot welds and glues, and ribs and side members are glued together. The bolts and fittings are used to hold in on the spars to dock the stabilizer with fuselage.

Each half of the elevator consists of two bonded panels with the panels attached to the plane of the stringers, the end profile and the crossbeam for attaching the decorative labels. Decorative labels are installed on each half of the elevator.

The sweep angle of vertical and horizontal tail is more than the sweep angle of wing, therefore the aerodynamic characteristics of the tail with an increase Mach number doesn't deteriorate faster than the wing.

The horizontal tail has symmetrical airfoil. The symmetric airfoil will keep the same aerodynamic load and have smaller drag.

1.2.3. Power Plan

This aircraft installs two AI-24VT engines that is turboprop engine with power 2820 hp. the nacelles at the wing will used for installing the engines.

In addition, the following equipment is installed on the engine:

- gear fairing;
-engine fairing;
- anti-icing system;
- generator;
-external oil system;
-engine ventilation system;
-fire suppression system.
- fuel system.

In the rear of the right nacelle, there is an APU RU19A-300 to offer additional thrust when aircraft is climbing and emergency thrust when AI-24 engine fail. Additionally, when the cargo aircraft's engines are in idle or starter-generator failure, it can supply the power to the aircraft onboard network.

Air extraction is provided to meet the life support needs of each engine, while the alternator is set as the main power source. In the cockpit, the pilots can control and monitor the power plant. And there are mechanical, automatic and electrical system are used to control the power plant.

1.2.4. Landing Gear

The landing gear is retractable and is of tricycle type. It includes nose and two main landing gear. There are fuselage nacelles are used to retract the main landing gear.

There is a retractable landing gear which is tricycle landing great type with nose single-strut landing gear and two strut main landing.

The main landing gears are installed in the fuselage nacelles and are retracted forward into the fuselage nacelles. Especially, inertial sensors are equipped on the wheels braking system.

The nose landing gear is retractable to the lower compartment of the cockpit. Two low pressure pneumatic tires are installed on the strut of the nose landing gear, Shimmy damper and steering mechanism.

When the aircraft is in the extracted and retracted position, the struts are held in place with a mechanical lock with an auxiliary opening for the hydraulic cylinder. When the landing gear is removed, only the small sash located directly on the amortization strut remains open; therefore, the units and components located in the landing gear compartment are vulnerable to dirt during the taxing period. A mechanism kinematically connected to the amortization strut is used to open and close the landing gear doors.

The tires of the nose landing gear are steerable. The wheels can be rotated from the neutral position of the steering mechanism to the right or left side. The angle is approximately 45°. Wheel rotation increases maneuverability. The takeoff and landing control of the front strut associated with the steering rudder is used to maintain a straight direction of motion during takeoff and to operate the aircraft.

The aircraft's hydraulic system powers the extension and retraction of the landing gear, opens the locks, breaks the wheels of the main outriggers, and turns the wheels of the front outriggers.

If the hydraulic system fails, the pilot can manually operate the lock in the retracted position of the landing gear. In this case, the weight of the gear and the airflow can affect the gear to be extracted and installed in the release position of the lock.

1.2.5. Flight Control System

The aircraft control surface includes: elevator control system, stabilizers, rudder, ailerons, air brakes, and flap.

Flight control system is subdivided on:

- primary flight control, it includes ailerons, elevators, and the rudder. The ailerons provide rolling control, rudder provides yaw control, and elevator provided pitch control.

- secondary control surface, there are flaps in our aircraft. The single slotted flaps are used to change camber of the wing to provide high lift and lower landing speed.

Trim tabs are mounted on the elevators, rudders, and ailerons. These are used for trim control.

There is a servo tab installed on the rudder and each of the ailerons.

The aircraft flight control system controls the servo tabs, flaps, ailerons, and rudder, it can lock of ailerons and rudder when aircraft parks on the ground. The main landing gear can be rotated and pressure decrease to break the main landing gear tires.

There are double steering wheels and ailerons for the pilots operating. On the cockpit floor, the pedals and steering wheels are installed on the control panel. And also, there is a reduction brake valve on the common control panel. Finally, their drive mechanism is installed from the pedal, and the brake mechanism for parking.

Steering wheels, which is for elevator trim tabs, flap control switches, rudder and aileron trim tab, as well as a rudder and aileron locking knob, are installed on the pilot’s central console. The autopilot control panel are on the central panel, and the left pilot’s control

Handle for turning the wheels of the main landing gear on the console of the left pilot. The autopilot control panel is on the central panel.

1.2.6. Crew cabin

The cockpit is small, and provides normally work and rest of the flight pilots

For the pilots, there is most stringent requirement are imposed for him. As well as it should provide a good overview for them.

There are first pilots, co-pilots, on-board technician in the crew cabin. And in the cargo cabin, a technician's place is offered. There are two pilot seats which are next to each other.

The flight crew cabin is separated from cargo hold by a rigid bulkhead.

1.3. Substantiation of the new aircraft parameters

1.3.1. Geometry calculations for the main units of the airplane

The taking-off weigh m_0 and wing loading P_0 (aircraft weight per unit of the wing area) are used for determination of the geometrical characteristics.

Determination of the wing loading is based on the similar aircraft analysis.

From this:

Wing area is:

$$S_w = \frac{G_0}{P_0} = \frac{m_0 \cdot g}{P_0} = \frac{27975 \times 9.8}{5.596 \times 10^3} \approx 105.61 \ (m^2).$$

Determination of the wing aspect ratio. If we choose greater value of the aspect ratio, the aerodynamic of wing is more efficient.

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

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

Wing span is:

$$l_w = \sqrt{S_w \cdot \lambda} = \sqrt{105.61 \times 11.30} \approx 34.55 \ (m),$$

where:

λ - aspect ratio; S_w is the area of the wing.

Root chord is:

$$b_o = \frac{2 S_w \cdot \eta_w}{(1+\eta_w) \cdot l_w} = \frac{2 \times 105.61 \times 2.80}{34.55 \times (1+2.80)} = 4.50 \ (m),$$

where:

η_w – taper ratio

Tip chord is:

$b_t = \frac{b_o}{\eta_w} = \frac{4.50}{2.8} = 1.61$ (m).

Table 1.2. – Aspect ratio examples

№	**Aircraft type**	**Aspect ratio**
1	Hypersonic aircraft	1-3
2	Tactical missile	0.3-1
3	Supersonic fighter	2-4
4	High subsonic transporter	8-12
5	Low subsonic transporter	6-9
6	Jet trainer	4-8
7	Glider	4-8
8	Sailplane	20-40
9	Homemade	4-7
10	General aircraft	5-9

Board chord can get from:

$b_{ob} = b_0 \cdot \left(1 - \frac{(\eta_w - 1)\cdot D_f}{\eta_w \cdot l_w}\right) = 4.50 \times \left(1 - \frac{(2.8-1)\times 3}{2.8\times 34.55}\right) = 4.23$ (m).

where:

D_f - diameter of fuselage.

The sweepback angle on 1/4 chord has been selected equal to 8°. Small sweepback angle wing has a high coefficient of lift, therefore, it can prevent wave drag. On the contrary, the ineffective sweepback angle, the drag coefficient sharp increase if the critical Mach number is surpassed.

In this work, the drawback doesn't have influence because of given flight speed is 400-500 km/h.

In the previous paper, we have determined the numbers of spars and position, also included the positions and numbers of ribs.

For this aircraft, two spars structure are selected (fig 1.3.).

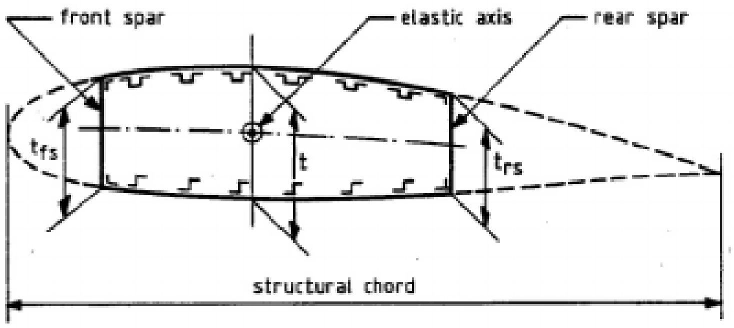

Fig. 1.3. - Two spars wing design

With the geometrical method of mean aerodynamic chord determination (figure 1.4.). I have got the Mean aerodynamic chord is equal: b_{MAC}=3.29 (m).

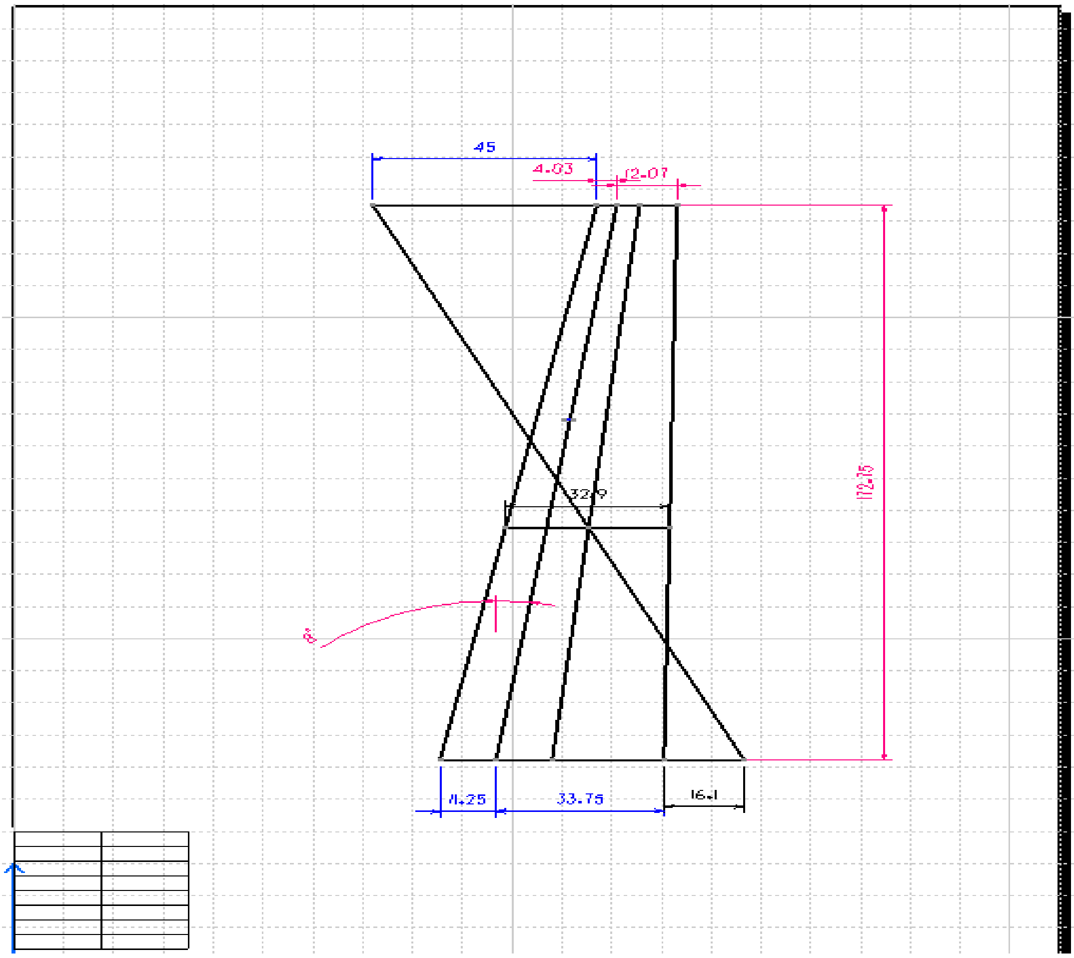

Figure 1.4.-Geometrical method of wing mean aerodynamic chord determination

Determination of the aileron's geometrics and high lift devices.

Ailerons geometrical parameters are determined in next consequence:

Aileron span:

$l_{ail} = (0.3\ldots0.4)\cdot l_w /2 = 0.375 \times \frac{34.55}{2} = 6.48$ (m);

Aileron chord:

$b_{ail} = (0.22\ldots0.26)\cdot b_t = 0.4 \times 1.61 = 0.644$ (m).

Aileron area:

$S_{ail} = (0{,}05\ldots0{,}08)\cdot S_w/2 = 0.065 \times \frac{105.61}{2} = 3.43$ (m^2).

For the third-generation airplanes, there is a new tendency to decrease relative ailerons area and wing span. In this work, spoilers and ailerons are applied to transversal control this aircraft. So, the span and area of high-lift devices are increased for improving taking-off and landing characteristics of this airplane.

Aerodynamic compensation of the aileron.

Axial $S_{axinail} \leq (0.25\ldots0.28)\cdot S_{ail} = 0.25 \times 3.43 = 0.8575$ (m^2).

Inner axial compensation:

$S_{inaxinail} = (0.3..0.31)\cdot S_{ail} = 0.305 \times 3.43 = 1.05$ (m^2).

Area of ailerons trim tab.

For two engine airplanes:

$S_{tail} = (0.04\ldots0.06)\cdot S_{ail} = 0.05 \times 3.43 = 0.171$ (m).

Range of the differential aileron deflection:

Upward $\delta'_{ail} \geq 18^\circ$;

Downward $\delta''_{ail} \geq 8^\circ$.

The geometrical parameters of wing high lift devices provide the taking-off and landing coefficients of the wing lift force [7].

Consider the prototypes and modern trends for the HLD, the two-slotted flap is a good choice.

$b_{fl} = (0.28\ldots0.3)\cdot b_t = 0.29\cdot1.61 = 0.4669$ (m).

We can determine the airfoil section by using the airfoil catalog.

By the following formula, we can raise the coefficient $C_{y\max}$ for the HLD with necessary value of lift coefficient $C_{y\max bw}$.

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

where:

$C_{y\max l}$ – necessary coefficient of the lifting force for the aircraft landing.

The rate of the relative chords of wing HLD for modern aircraft:

- b_{sf} = 0.25..0.3 – for the split edge flaps;
- b_f = 0.28..0.3 – two slotted flaps or one slotted;
- b_f = 0.3..0.4 – f Faylers flaps or three slotted flaps;
- b_s = 0.1..0.15 – slats.

The effectiveness of HLD increase proportionally with the wing span increasing, serviced but the HLD, therefore, it is necessary to use the biggest span of HLD because of using of flight spoilers and maximum reduction of the engine nacelle area.

For the choice of structurally-power schemes, high-fitting schemes and kinematics of the HLD, the statistics and experience of domestic and foreign aircraft construction are a good way to use. And considering the main of existing constructions of HLD, spars construction is widely used.

1.3.2. Fuselage Layout

For the fuselage layout calculation, it usually includes the calculation of the main geometrical dimensions and creation of the interior scheme creation.

In the geometry calculation, the special aerodynamic characteristics of the aircraft, the typical drag under normal and extreme flight conditions should be considered. The fuselage geometry of airplane should allow to avoid high values of drag, skin friction and wave drag, withstand the aerodynamic loads and provide the high safety factor. In order to decrease parasitic drag and provide strength characteristics avoiding the stress concentration in fuselage cross-section, the round shape is chosen.

The design of the interior scheme is based on the requirement of design competence. And the cargo compartment should be considered to load the cargo.

Subsonic aircraft fuselage nose part must be:

$l_{nfp} = 2.1 \cdot D_f = 2.1 \cdot 3 = 6.3$ (m);

Noting the fact that AN-26 crashed (African AN-26, 9Q-COS, October 4, 2007) [8], taking into account the errors caused by wear fatigue of the skin of the aircraft operating in a tropical climate for a long time. Therefore, the strength and layout requirements should be considered when designing the cross-section shape of the fuselage. In modern aircraft, the fuselage cross section shape is circular cross section. It provides the minimum value of the wetted surface.

The following geometrical parameters have been considered and accepted:

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

Fuselage length is equal:

$l_f = D_f \cdot \lambda_f = 3 \times 9.52 = 28.56$ (m);

Fuselage nose part aspect ratio (fineness ratio) is equal:

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

Sum of rear and nose parts must 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:

$$l_{frp} = \lambda_{frp} \cdot D_f = 2.7 \cdot 3 = 8.1\ (m) .$$

1.3.3. Layout and calculation of basic parameters of tail unit

In order to keep longitudinal stability for overloading aircraft, the aircraft lift force should be behind of the center of gravity. The distance of two points needs to be closed to the wing mean aerodynamic chord, which influence the 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;

$m^{C_y}{}_{x}$ is the moment coefficient.

If $m^{C_y}{}_{x}=0$, we can conclude the neutral longitudinal static stability, if $m^{C_y}{}_{x} > 0$, the aircraft is stability instable.

Static range of static moment coefficient.

Determination of the tail unit geometrical parameters:

Area of vertical tail unit is equal:

$$S_{htu} = \frac{b_{mac} \cdot S_W}{L_{htu}} \cdot A_{htu} = \frac{0.9 \times 105.61 \times 3.29}{9.87} = 31.7\ (m^2);$$

$$S_{vtu} = \frac{l_W \cdot S_W}{L_{vtu}} \cdot A_{vtu} = \frac{0.08 \times 105.61 \times 34.55}{9.87} = 29.57\ (m^2);$$

$L_{htu} \approx L_{vtu} = 3 \cdot 3.29 \approx 9.87$ m,

where:

L_{htu} and L_{vtu} are the length of vertical and horizontal tail wing;

l_w is the wing span;

S_w is the area of the wing;

A_{htu} is the coefficients of static moments of horizontal tail wing;

A_{vtu} is the coefficients of static moments of horizontal tail wing.

Determination of the rudder area and elevator area:

Elevator area:

$S_{el} = 0.35 S_{HTU} = 0.35 \times 31.7 = 11.1\ (m^2);$

Rudder area:

$S_{rd} = 0.4 S_{VTU} = 0.4 \times 29.57 = 11.8\ (m^2);$

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

Elevator balance area is equal:

S_{eb}= (0,22…0,25)· $S_{el} = 0.235 \cdot 11.1 = 2.6085$ (m^2).

Rudder balance area is equal:

S_{rb}= (0,2…0,22) · $S_{rd} = 0.21 \cdot 11.8 = 2.478$ (m^2).

The area of elevator trim tab:

S_{etr}=$0.08 \cdot S_{el} = 0.08 \cdot 11.1 = 1.1088$ (m^2).

The area of rudder trim tab is equal:

S_{rtr}=$0,06 \cdot S_{rd} = 0,06 \cdot 11.8 = 0.708$ (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:

$$b_{HTUtch} = \frac{2 \cdot S_{HTU}}{(\eta_{\text{htu}}+1) \cdot l_{\text{htu}}} = \frac{2 \times 31.7}{9.87 \times (1+2.80)} = 1.69 \text{ (m)}.$$

Root chord of horizontal stabilizer is:

$$b_{HTUrch} = b_{\text{HTUtch}} \cdot \eta_{\text{htu}} = 1.69 \cdot 2.8 = 4.7 \text{ (m)}.$$

Tip chord of vertical stabilizer is:

$$b_{VTUtch} = \frac{2 \cdot S_{\text{VTU}}}{(\eta_{\text{vtu}}+1) \cdot l_{\text{vtu}}} = \frac{2 \cdot 29.57}{(2+1) \cdot 9.87} = 1.997 \text{ (m)}.$$

Root chord of vertical stabilizer is:

$$b_{VTUrch} = b_{\text{VTUtch}} \cdot \eta_{\text{vtu}} = 1.997 \cdot 2 = 3.994 \text{ (m)}.$$

Horizontal tail unit mean aerodynamic chord:

$$b_{MACHTU} = 0,66 \cdot \frac{\eta_{\text{htu}}{}^2 + \eta_{\text{htu}} + 1}{\eta_{\text{htu}} + 1} \cdot b_{\text{HTUtch}} = 3.417 \text{ (m)}.$$

Vertical tail unit mean aerodynamic chord:

$$b_{MACVTU} = 0,66 \cdot \frac{\eta_{\text{vtu}}{}^2 + \eta_{\text{vtu}} + 1}{\eta_{\text{vtu}} + 1} \cdot b_{\text{VTUtch}} = 2.636 \text{ (m)}.$$

The sweepback angle of the empennage is taken as 3.50˚ less than the sweepback angle of the wing.

So: $\chi_{HTU} = 9\,^{\circ}$; $\chi_{VTU} = 22\,^{\circ}$.

1.3.4. Landing gear design

Landing gear calculation is an important step when designing the aircraft. In this part, the calculation consists of two parts the geometry calculation of landing gear and load calculation of landing gear. There is two necessary parameter we should be considered, the wheel base, and wheel tack in geometry calculation. After finishing the calculation, we should choose the land gear wheel.

Main wheel axel offset is:

$e_g = 0{,}15 \cdot \mathrm{b_{MAC}} = 0.15 \cdot 3.29 = 0.4935$ (m).

Landing gear wheel base comes from the expression:

$B_g = (0{,}3\ldots0{,}4) \cdot l_\mathrm{f} = 0.35 \cdot 28.56 = 10$ (m).

Front wheel axial offset will be equal:

$d_{ng} = B_g - e_g = 8.56 - 0.4935 = 8.0665\ (m)$.

Wheel track is:

$\mathrm{T} = (0{,}7 \ldots 1{,}2) \cdot B_g = 0.85 \cdot 8.56 = 6.42\ (m)$.

To prevent the side nose-over, the value should be more than 2H. H is the distance from runway to the center of gravity.

$K_g = 1.5...2.0$ – dynamics coefficient.

Nose wheel load is equal:

$$P_{nlg} = \frac{e_g \cdot m_0 \cdot g \cdot K_g}{B_g \cdot z_{nlg}} = \frac{0.4935 \cdot 27975 \cdot 9.81 \cdot 1.75}{8.56 \cdot 2} = 13843.97\text{(N)}.$$

Main wheel load is equal:

$$P_{mlg} = \frac{(B_g - e_g) \cdot m_0 \cdot g}{B_g \cdot z_{mlg} \cdot n_{mlg}} = \frac{(8.56 - 0.4935) \cdot 27975 \cdot 9.81}{8.56 \cdot 2 \cdot 2} = 64653.27\ \text{(N)},$$

where:

n – number of the quantity of supports;

z – number of the wheels.

The last is the tires of the landing with the values of wheel loading and take-off speed.

For the nose landing gear, we choose the Flight Eagle DDT 18K88-5 [9] with parameters $P_{rated} = 3050\ lbf$, size 18×5.75-8.

For the main landing gear Aircraft Rib 277K08-1 with parameters $P_{rated} = 3050\ lbf$; size 22×7.75-10.

Table 1.3. – Aviation tires for nose landing gear

Size	Construction			Service Rating				Tread Design/ Trademark	Weight(Lbs)
	Ply Rating	TT or TL	Rated Speed (mph)	Rated load (Lbs)	Rated Inflation (Psi)	Max. Breaking Load (Lbs)	Max. Bottoming Load (Lbs)		
18×5.75-8	8	TL	190	3050	105	4570	9200	Rib	13.7
Inflated Dimensions (in)				Static Loaded Radius (in)	Aspect Ratio	Wheel (in)			
Outside DIA		Section Width				Width Between Flanges	Specified Rim Diameter	Flange Height	Min Ledge Width
Max	Min	Max	Min						
18	17.4	5.75	5.4	7.6	0.87	4.25	8	0.88	1.25

Table 1.4. – Aviation tires for main landing gear

Size	Construction			Service Rating				Tread Design/ Trademark	Weight(Lbs)
	Ply Rating	TT or TL	Rated Speed (mph)	Rated load (Lbs)	Rated Inflation (Psi)	Max. Breaking Load (Lbs)	Max. Bottoming Load (Lbs)		
size 22×7.75-10	10	TL	190	5500	110	7980	14900	Rib	22.11
Inflated Dimensions (in)				Static Loaded Radius (in)	Aspect Ratio	Wheel (in)			
Outside DIA		Section Width				Width Between Flanges	Specified Rim Diameter	Flange Height	Min Ledge Width
Max	Min	Max	Min						

1.3.5. Power plant

AI-24VT, CT7-9C3, PW150D - turboprop engine, in various modifications installed on passenger aircraft An-26 (AI-24VT), CN-235(CT7-9C3) and transport An-132 (PW150D). According to the performance of aerodynamic calculation for the designing aircraft we consider the AI-24VT engine is used for this aircraft. The AI-24 turboptop engines was produced in the 1950s by the Moter Sich. It is single-shaft turboprop with water injection. The length is 2.4 m, the dry weight 600kg. it consists of a compressor with 10-stag axial, a combustor with 8 flame tubes, and 3 stage axial turbines.

Table 1.5. – Examples of modifications AI 24

Name	AI-24VT	CT7-9C3	PW150D
Type	Single-shaft turboprop	Turboshaft engine	Turboprop
Compressor	10-stage axial	6-stage axial	2-spool, 3-stage axial, single centrifugal
Weight	1300 lb 600 kg	537 lb 244kg	1580 lb 716.9 kg
Power	2500–2800 shp 1900-2100 kW	1500-3000 shp 3415-3781 kW	2476–3415 lbf 1846–3415 kW
Length	95.9 in (243.6 cm)	47 in (120 cm)	95 in (242.2 cm)
Diameter	14 in (36 cm)	25 in (64 cm)	31 in (790 cm)
Overall pressure ratio	7.05:1	17:1	-

1.4. Determination of the aircraft center of gravity position

1.4.1. Determination of centering of the equipped wing

Mass of the equipped wing includes the mass of the wing structure, mass of the equipments installed in the wing and mass of the fuel. Regardless of the place of mounting, the main landing gear and the nose gear are contained in the mass register of the equipped wing. The mass table includes name of the objects, mass and the center of gravity coordinates relative with the leading edge of the mean aerodynamic chord (MAC) for the XOY.

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 1.6. The mass of aircraft is 27975 kg.

Assuming that the aircraft is symmetrical on the Y axis, then we only need to consider the coordinate of the center of gravity X. coordinates of the center of power for the equipped wing are defined by these formulas:

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

Table 1.6. - Trim sheet of equipped wing masses

№	Object name	Mass		CG coordinates	Moment of mass
		units	total mass		
1	wing	0.14459	4044.91	1.48	5988.48
2	fuel system	0.00310	86.72	1.48	128.39
3	Flight control system, 30%	0.00297	83.09	1.97	164.01
4	electrical equipment, 10%	0.00299	83.65	0.33	27.52
5	anti-ice system, 50%	0.0090	251.78	0.33	82.83
6	hydraulic systems, 70%	0.01722	481.73	1.97	950.93
7	Engines (-fuel system)	0.13083	3659.97	-1.36	-4977.56
8	Equipped Wing without landing gear and fuel	0.3107	8691.83	0.27	2364.62
9	nose landing gear	0.007416	207.46	-7.10	-1472.98
10	main landing gear	0.042024	1175.62	1.46	1716.41
11	fuel	0.10760	3010.11	1.35	4060.34
Total		0.46774	13085.03	0.69	9032.99

$X_w = \sum m_i \cdot X_i / \sum m_i = 0.69$ m

1.4.2 Determination of centering of the equipped fuselage

In this part, we choose the nose of the fuselage as origin of the coordinates on the horizontal axis. The list of the objects for the aircraft is given in table 1.7.

The center gravity coordinate of the equipped fuselage is 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 1.7 – Trim sheet of equipped fuselage masses

№	Objects names	Mass		C.G coordinates	Moment of mass
		units	total mass		
1	fuselage	0.17880	5001.93	14.28	71427.56
2	horizontal tail	0.01800	503.55	26.86	13525.35
3	vertical tail	0.01849	517.26	26.18	13541.80
4	navigation equipment	0.00670	187.43	1.20	224.91
5	radio equipment	0.00340	95.12	2.38	226.37
6	radar	0.00450	125.89	0.60	75.53
7	instrument panel	0.00780	218.21	1.21	264.03
8	Flight control system 70%	0.00693	193.87	14.28	2768.42
9	hydraulic system 30%	0.00738	206.46	19.99	4127.46
10	anti ice system, 25%	0.00450	125.89	22.85	2876.27
11	air-conditioning system, 25%	0.00450	125.89	14.28	1797.67
12	electrical equipment, 90%	0.02691	752.81	14.28	10750.09
13	lining and insulation	0.01120	313.32	14.28	4474.21
14	Load devices equipment	0.00380	106.31	9.75	1036.47
15	Not typical equipment	0.00100	27.98	2.00	55.95
16	Additional equipment (emergency equipment)	0.00080	22.38	2.00	44.76
17	Operational items	0.03055	854.64	3.00	2563.90
18	Cargo equipment	0.00170	47.56	10.33	491.17
19	equipped fuselage without payload	0.33696	9426.46	13.82	130271.96
20	Baggage, cargo, mail	0.19660	5499.89	10.33	56802.81
21	crew	0.01000	279.75	2.20	615.45
	TOTAL	0.54356	15206.09	12.34	187690.23
	TOTAL fraction	1	28291.12		

We can find fuselage center of gravity coordinate X_f:

$$X_f = \sum m_i \cdot X_i / \sum m_i = 10.3 \text{ (m)}$$

Then we consider the center of gravity of fully equipped wing and fuselage, we can use the moment equilibrium equation relatively fuselage nose:

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

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

$$X_{MAC} = \frac{15206.09 \times 12.34 + 13085.03 \times 0.69 - 27975 \times 3.29 \times 0.28}{27975 - 13085.03} \approx 12.06 \text{ (m)}.$$

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

Table 1.8. - Calculation of center of gravity positioning variants

Name object	Mass, kg	Coordinate CG, m	Mass moment Kg.m
equipped wing (without fuel and landing gear)	8691.83	12.75	110862.05
Nose landing gear (extended)	207.46	4.96	1029.93
main landing gear (extended)	1175.62	13.52	15899.58
fuel	3010.11	13.48	40573.59
equipped fuselage (without payload)	9426.46	13.82	130271.96
Cargo, mail	5499.89	11.15	61323.72
crew	280.00	2.20	616.00
nose landing gear (retracted)	207.46	4.46	926.20
main landing gear (retracted)	1175.62	13.52	15899.58
reserve fuel	661.61	13.41	8874.36

Table 1.9. - Airplane’s center of gravity position variants

Variants of the loading	Mass, kg	Moment of the mass, kg*m	Center of mass, m	Centerin g
take off mass (L.G. extended)	27975.0 0	360576.84	12.89	0.25
take off mass (L.G. retracted)	27975.0 0	360473.10	12.89	0.25
landing weight (LG extended)	25942.8 7	328877.60	12.68	0.19
ferry version	22791.4 8	299149.39	13.13	0.25
parking version	19501.3 7	258208.21	13.24	0.29

Conclusion to the project part

In this part of the diploma work, I have considered and calculated the main geometrical parameters for my main aircraft structure, which are wing geometrical parameters, fuselage layout, tail wing and landing gear. Then I have chosen the power unit for this aircraft.

Calculation of the aircraft gravity and determination of the center of gravity position has been carried out.

According to the geometry calculation and center of the gravity, we can conclude that the proposed prototype satisfies requirements to the planes of the transport category and have safe, reliable and economical level of the technical characteristics.

PART 2. CARGO ROLLER SYSTEM DESIGN

2.1. Analysis of existing cargo aircraft

As the air cargo quickly developing, the cargo aircraft is playing an important role. The huge air cargo market demand makes the airline company buy more cargo aircraft.

For the cargo aircraft design and product, there are three ways for designer and engineers.

Freighter is as a derivative of new or existing passenger or military.

Freighter as a development of a dedicated civilian cargo aircraft designed without consider either passenger or military.

Freighter as development of a joint civil-military air cargo plane that would satisfy both commercial and military requirements [10].

In this part, we adopt the first method.

The advantage of the derivative cargo aircraft, its cost already assessed according to the transaction of its passenger equivalent. Secondly, the financial arrangements for purchase of the aircraft have been established and the delivery time is shorter than for all new aircraft.

The disadvantage of this cargo aircraft so that they represent older technology, so, their maintenance and operation cost are more expensive than the new designed freighter. Moreover, as they in initial design didn’t consider the air cargo. The loading and unloading usually cause problems and ask some specific cargo equipment.

Due to the changing safety or noise requirement, or this aircraft become uncompetitive in passenger airliner service, most conversions used older aircraft, which is no longer suitable for passenger aircraft. But conversion can be happened because most passenger airlines are stopped because of global epidemic of infectious diseases. Cargo aircraft asks higher strength cabin floor and cargo door and less windows.

For this designed cargo aircraft, the main modification of the cargo equipment are:

- Strengthening the compartment floor structure;

- designing the new roller loading system;
- using the new and environmental unit load devices.

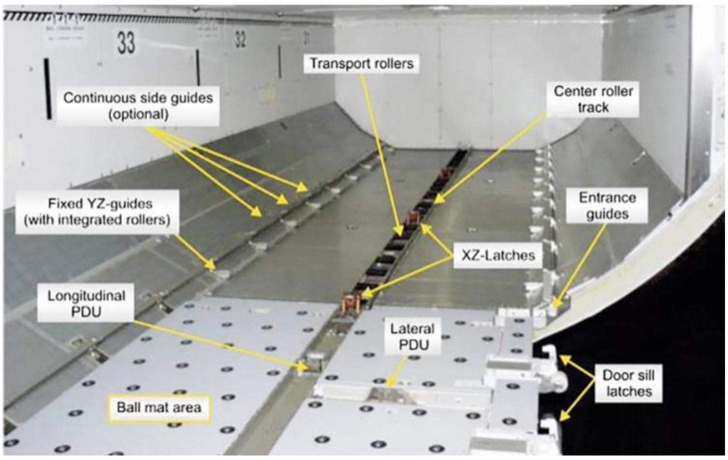

Fig 2.1. - The cargo compartment roller system

2.2. Requirements to a new floor load system

2.2.1. Fire protection requirements

This cargo hold of this aircraft is class E cargo hold which is on aircraft used only for the load cargo. According to the CS-25 of EASA and CCAR-25 of China. The equipment can't be damaged when the cargo load or unload in the cargo hold and breakage or failure of the equipment will not cause a fire hazard. Heat sources in the cargo hold must be shielded and insulated to prevent ignition of the cargo.

Therefore, the materials of the cargo roller should be non-flammable, high temperature resistance and don't produce the dangerous gas when there is fire in cargo hold.

2.2.2. Strength requirement

In order to make cargo containers move into the cargo compartment, the motor of the PDU will work, then the motor drive the transmission shaft. With the help of the planetary gears, the transmission shaft drives the shaft of the roller. Lastly, the shaft of roller will make the roller rotate to move the cargo container. In this process, the shaft will load the torque of the motor and the vertical load of the cargo container.

According to the CS-25 and CCAR-25, the strength requirements are specified in terms of limit loads and ultimate loads. And for the factor of safety of 1.5 must be applied to the prescribed limit load which are considered external loads on the structure.

The structure must be able to support limit loads without detrimental permanent deformation, and the structure must be able to support ultimate loads without failure foe a least 3 seconds [2].

Additionally, the roller surface will contact with the cargo containers, the abrasion of the roller is also needed to consider.

2.2.3. Corrosion requirement

The most aircraft equipment are made of metal, the corrosion can greatly compromise the requirement strength and integrity of the aircraft equipment structure. Besides, the environment of the cargo hold is complicated because the cargo hold temperature is high or even is wet with specific conditions [11]. With the cargo move into the cargo compartment, the contact of surface with cargo container and roller, it usually makes some damage of the surface. This will make the metal expose in the air then lead to the corrosion.

Therefore, the complicated environment will make the galvanic corrosion, stress corrosion even microbiological corrosion. To prevent the corrosion, I select the aluminum alloy for the cargo compartment roller system. The materials with different electrode potential should be used for the components as well.

2.2.4 Economic requirement

The economic cargo equipment is very impartment for the airline's companies. The long service life and simple maintenance will help the airlines company spend less resources and costs in their aircraft.

2.3. Preliminary design of the cargo floor system.

2.3.1. The description of cargo holds and Unit Load Device

In this cargo aircraft, the roller system has four main structure part, rollers, trail, PDU and latch. The cargo compartment is class E, which is only for the cargo [12].

Cargo compartment sizes:

L_c =14000 (mm) – length;

W_c= 2360 (mm) – width;

H_c =1890 (mm) – height.

According to the cargo compartment size, we can determine the size of unit load device (ULD), which is for baggage and cargo carried. The ULD need to be suitable for the cargo hold. this cargo aircraft is derived from the military airplane. but the ULD which are widely used for the modern civil cargo airplane who is derived from the civil passenger airplane, the conventional ULD is not suitable for this cargo aircraft.

In this work, the Container E, (fig 2.2 Container E) can be adopted, which is IATA rate class 9 ULD [13]. The industry- standard E-Container air container is a large corrugated box, which can offer damage-free freighter storage, ship, transport and distribute. It is sturdy, and cost-effective. It is also friendly to our environment with an ever-increasing force on climate change. And it is easy recyclability helps to reduce carbon footprint.

The specific characteristics of the Container E are:

- Maximum external length (L_0): 1060 (mm);
- maximum external width (W_0): 730 (mm);
- maximum external height (W_0): 660 (mm);
- maximum gross weight (G_0): 136 (kg);
- maximum tare internal volume: 5 (m^3);
- maximum weight allowance: 8 (kg).

Fig 2.2. - Container E

In order to make full use of the cargo hold volume, the boxes will be grouped together in right-angled parallelogram form, with 8 boxes in each group. Then put this one group boxes in a pallet, using the tie and net fix this cargo boxes.

$G_b = G_0 \times 8 = 1100$ (kg);

$L_b = L_0 \times 2 = 2120$ (mm);

$W_b = W_0 \times 2 = 1460$ (mm);

$H_b = H_0 \times 2 = 1320$ (mm).

The payload of this cargo aircraft is 5500 kg, so number of group boxes is equal:

$$n_{group} = \frac{m_0}{G_b} = \frac{5500}{1100} = 5$$

The container pallet is used to Lb. load and lock the container with the help of the pallet net. In this aircraft, the PQA pallet (Fig 2.3. the PQA pallet) is applied in the cargo hold. the tare weight of this pallet is 70 kg, and maximum gross weight is 2449 kg [14]. The pallet is made of corrosion-resistant aluminum (aluminum 5052). We get from the figure 2.3., there is 4 attached rings on each long side and 3 attached rings on each short sides, and the pallet has 14 tie rings.

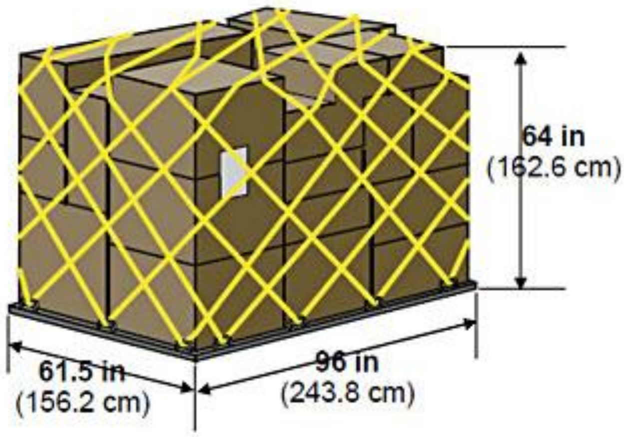

Fig 2.3. - The PQA pallet

2.3.2. The description of cargo loading system

A cargo loading system (CLS) (Fig 2.1. the cargo compartment roller system) includes many equipments to assist the cargo movement, guiding and fasten. Power drive units (PDU) are used to move ULDs semi automatically. These items are attached to the tracks and floor structure by means of tray assemblies and floor fittings. Tray assemblies provide movable restraint for locks and rollers

The designed freighters cargo floor has three trails, left, right and center trails. The left and right have continuous side guides and simple rollers, the center trail is installed the latch, transport rollers ad power drive unit (PDU). When the container is close the center trail, the PDU starts work, the rollers of PDU rotate with the motor to transport the container into the cargo hold. then the people will make the latch rise to fix the container on the floor.

Rollers allow the ULDs move freely along guild trail in the longitudinal directions. The roller’s structure consists of a rubber cylinder with a double ball bearing to load the radial force and axial force. The bearings are made of chroming steels. Then there is an axis of the rollers withstand bending, shearing and torque. The roller axis is made of the aluminum alloy (ZL204A), which is widely used in aviation files in China. This alloy is composition of Al, Cu, and Fe. ZL204A with low density and good corrosion resistance, and has a higher specific strength and stiffness, easy to process and shape [15].

Area load (AL) is the maximum load that can be accepted on any surface unit of an aircraft. It prevents the load from exceeding the capability of the aircraft structure [16].

$$Area\ Load = \frac{Weight\ of\ the\ piece}{Contour} = \frac{G_b + G_p}{S_b} = \frac{1100+70}{3.1} = 377\ (kg/m^2).$$

Running load is maximum acceptable load on the aircraft floor for any given fuselage length.

$$\text{Running Load} = \frac{G_b + G_p}{W_r} = \frac{1100+70}{0.18} = 6500\ (kg/m).$$

In the Preliminary design, there are 3 rows of rollers in the distance between 3 rows in lateral direction is 500 mm. 24 rollers are installed on the longitudinal guide trail. So, we can determine how many rollers on the 1 m^2.

$$n = \frac{24}{12} \times 3 = 6\ (\frac{rollers}{m^2})$$

The area load for one roller (G_a):

$$G_a = \frac{Area\ load \times load\ factor}{n} = \frac{377 \times 1.5}{6} = 94.25\ (kg/m^2).$$

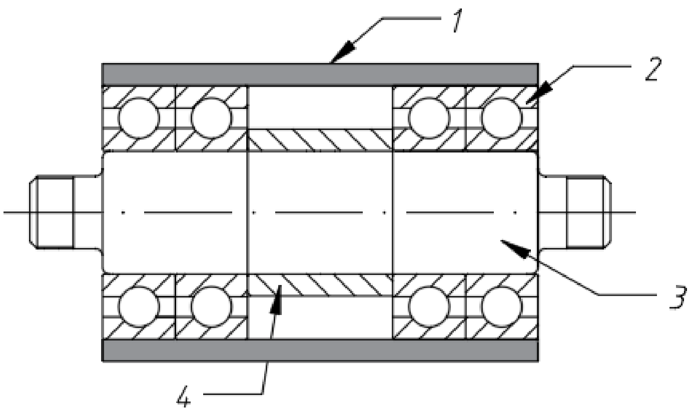

Fig. 2.4. - The roller constructure 1-roller tires, 2-ball bearings, 3-roller shaft, 4- internal cylinder.

The two bearings are combined on the shaft to withstand the load from the axial and radial direction. And the bearings are standard parts in manufacturing enterprise, in the work, the ball bearing that is produced by the Timken Company [17]. The chromium 52100 bearing steel is as the materials. This bearing steel has high and wear resistance and uniform hardness, as well as high elastic limit.

The PDU (Fig 2.5. The PDU of roller system) are installed on the cargo floor. The main function of the PDU is to provide adequate traction under all environment conditions to move containers and pallets into and out of the cargo hold. self-erecting PDUs are equipped with acting AC servo motors. The motor provides the necessary traction force to move cargo.

The PDU [18] consists of a motor and a drive roller carried for engagement with cargo to be driven by the unit. There is a planetary gear rotatably connecting the output shaft of the motor to an output gear coaxially with the output shaft of the motor. An idler pulley extends laterally from the planetary output gear and rotatably connects the planetary output gear to the roller drive shaft. A pair of angularly spaced counterforce gears are then coupled to the planetary output gear opposite the idler pulley to support the output gear against the counterforce transmitted to it by the drive rollers through the idler pulley

The PDU can offer the 0.45 m/s speed to cargo containers with the traction.

Fig.2.5. The PDU of roller system

Material table of the cargo floor system:

Table 2.1. – The material of the cargo roller

Object		Material
ULD	containers	corrugated box
	pallet	aluminum alloy 5052
	pallet net	polyester
CLS	roller	rubber
	roller shaft	aluminum alloy (ZL204A)
	roller track	aluminum alloy 2024
	bearings	high chromium 52100 bearing steel, (AMS 6440).

Table 2.2. - Some aluminum alloys produced in China and can be used for the manufacturing.

Type	Shear strength (MPa)	Yield strength (MPa)	Tensile strength (MPa)
7Y69	/	874	950
7055	/	530	593
ZL204A	331	440	572
2024-T3	283	280	400

2.4. Calculation of components strength

Stress analysis comprises: Determination of the external loading from cargo container or pallet; the strength calculation of the rollers; the strength calculation of the roller shaft.

Weight on one roller:

$$M_r = 94.25\ (kg).$$

Force: F= $F = M_r \times g = 94.25 \times 9.8 = 923.7\ (N)$.

The running load of one roller:$q = \frac{F}{length\ of\ roller} = \frac{923.7}{0.04} = 23092.5\ (N/m)$.

Consider the cargo movement conditions, so there is a moment on the rollers which is produced by the PDU.

Moment: $M_i = 9549\frac{P}{n} = 9549 \times \frac{115}{8000} = 137.2\ (N \cdot m)$.

To build the shaft mathematic strength model, we need to consider the shaft is as beam, and the external force of 923.7 N and moment 137.2 $N \cdot m$ applies to the roller shaft, the bearings can be considered the hinger support. From the modern machine design handbook of China [19], we can determine the length from the hinger support (Fig 2.6. the hinger support of double ball bearing) to the end of shaft. The roller shaft loads the bending and torque [20].

Internal force calculation of the roller, determination of shear force and moment of the shaft.

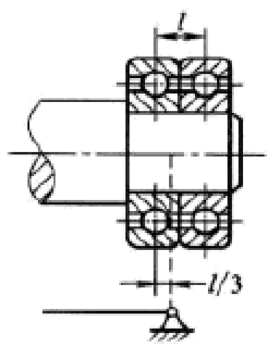

Fig.2.6. The hinger support of double ball bearings

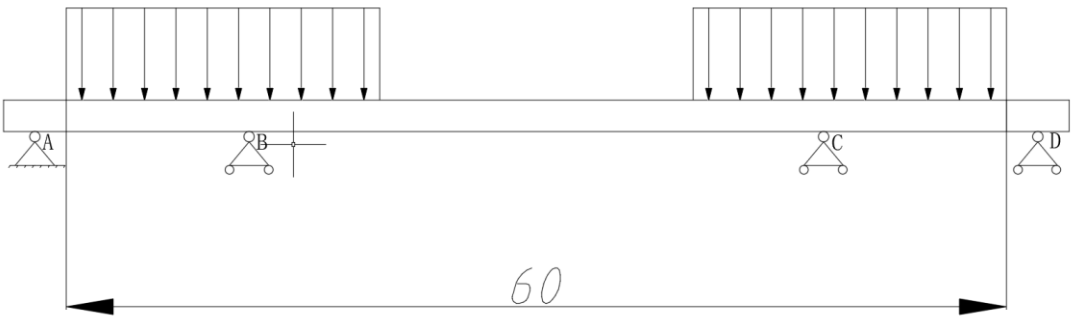

Fig 2.7. - The main view of cargo on the rollers

Consideration the bearings and guide trail, we can know the roller axis is twice statically indeterminate structure. To solve this problem, we can use force method.

AB =a=14 (mm);

BC =b= 18 (mm);

AE = e= 2 (mm)

EB= f= 12 (mm)

AF= i=22 (mm)

AG=j=32 (mm)

According to the deformation compatibility conditions, we can get following equations:

$$\Delta_1 = \delta_{11}X_1 + \delta_{12}X_2 + \Delta_{1F} = 0$$

$$\Delta_2 = \delta_{21}X_1 + \delta_{22}X_2 + \Delta_{2F} = 0$$

Because of those loading is symmetry, we can know:

$$\delta_{12} = \delta_{21} = 0$$

Determine the Δ_{1F}, Δ_{1F} means the distance of the statically system only under the F,

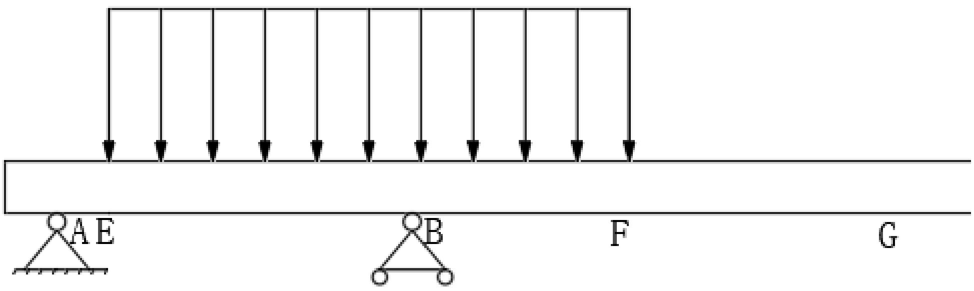

Fig 2.8. – The external force diagram of statically determinates structures

The equation of the force and moments balance have a view:

$$\sum F_y = 0;$$

$$F_A + F_B - qc - F = 0;$$

$$\sum M_A = 0;$$

$$-F_b a + qc * 0.012 + F * 0.032 = 0;$$

$$F_A = 65.7 - 1.3F\ (N);$$

$$F_B = 394.3 + 2.3F(N).$$

According to the Castigliano's method:

For the AE:

$$M_1(x_1) = F_A * x_1$$

$$\frac{\partial M_1(x_1)}{\partial F} = -1.3\ x_1$$

For the EB:

$$M_2(x_2) = F_A * x_2 - \frac{1}{2} qe(x_2 - 0.02)^2$$

$$\frac{\partial M_2(x_2)}{\partial F} = -1.3x_2$$

For the BF:

$$M_3(x_3) = F_A * x_3 + F_B(x_3 - a) - \frac{1}{2}q({x_3}^2 - e^2)$$

$$\frac{\partial M_3(x_3)}{\partial F} = -1.3x_3 + 2.3\ (x - 0.014)$$

For the FG:

$$M_4(x_4) = -F * x_4$$

$$\frac{\partial M_4(x_4)}{\partial F} = -x_4$$

$$\Delta_{1F} = \frac{\partial V_\varepsilon}{\partial F} = \int_0^e \frac{M_1(x_1)}{EI}\ \frac{\partial M_1(x_1)}{\partial F} dx_1 + \int_e^f \frac{M_2(x_2)}{EI}\ \frac{\partial M_2(x_2)}{\partial F} dx_2 + \int_f^i \frac{M_3(x_3)}{EI}\ \frac{\partial M_3(x_3)}{\partial F} dx_3 + \int_i^j \frac{M_4(x_4)}{EI}\ \frac{\partial M_4(x_4)}{\partial F} dx_4 = \frac{9.1 \times 10^{-4}}{EI}$$

Determine the δ_{11}:

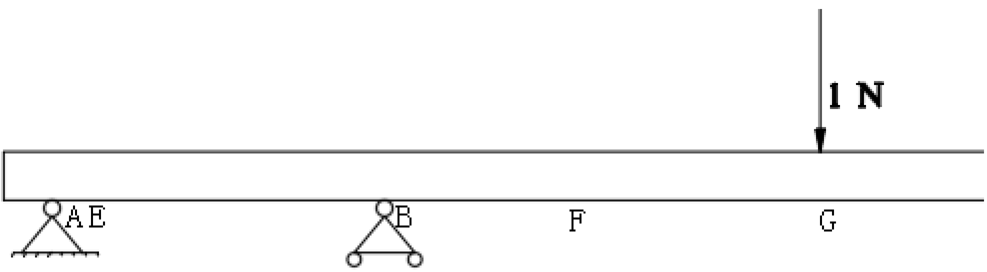

Fig 2.9. – The external force diagram of unit force

The equation of the force and moments balance have a view:

$$F_{A1} = \frac{7}{16}F'$$

$$F_{B1} = \frac{9}{16}F'$$

For the AB:

$$\overline{M_1(x_1')} = F_{A1} * x_1'$$

$$\frac{\partial \overline{M_1(x_1')}}{\partial F'} = x_1'$$

For the BE:

$$\overline{M_2(x_2')} = F' * x_2'$$

$$\frac{\partial \overline{M_2(x_2')}}{\partial F'} = x_2'$$

$$\delta_{11} = \int_0^a \frac{\overline{M_1(x_1')}}{EI} \frac{\partial \overline{M_1(x_1')}}{\partial F'} dx_1' - \int_0^f \frac{\overline{M_1(x_2')}}{EI} \frac{\partial \overline{M_1(x_2')}}{\partial F'} dx_2' = \frac{1.03 \times 10^{-5}}{EI}$$

Determine the X_1:

$$\delta_{11}X_1 + \Delta_{1F} = 0$$

$$X_1 = 88\ (N)$$

According to the symmetry condition:

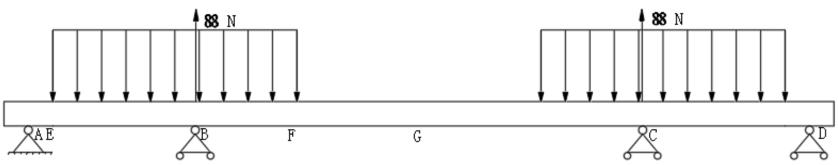

Fig 2.10. – The external force diagram which determines the over-constrained

Determination of the external force of A and D

$$F_{RA} = F_{RD} = 372\ (N)$$

$$F_{RB} = F_{RC} = 88\ (N)$$

Because of the

Only consider the torque:

$$T_e = T_i = -132.7 \ \ (N \cdot m).$$

Determine the moment of inertia of the circular section:

$$dA = 2ydz = 2\sqrt{R^2 - z^2}dz;$$

$$I_y = \int z^2 \, dA = 2\int_{-R}^{R} z^2\sqrt{R^2 - z^2}dz = \frac{\pi D^4}{64};$$

$$I = I_y + I_z = \frac{\pi D^4}{32}.$$

Then only in the bending condition, the normal stress and shear stress:

$$\sigma_{bending} = \frac{M_{max}R}{I} = \frac{7.14\times 32}{3.14\times 4913\times 10^{-9}} = 14.8(\text{MPa}).$$

Shear stress calculation:

$$\tau_{bending} = \frac{4F_s}{3\pi R^2} = \frac{4\times 372}{3\times 3.14\times 72.25\times 10^{-6}} = 2.2 \ (MPa)$$

Strength check:

According to the Tresca criterion:

$$\sigma = \frac{1}{W}\sqrt{M^2 + T^2} = \frac{1}{W}\sqrt{M^2 + T^2} = 137 \ (MPa);$$

$$[\sigma] = \frac{\sigma_{Yield}}{n} = \frac{440}{1.5} = 293 \ (MPa);$$

$$\sigma < [\sigma].$$

So, we can conclude the roller axis is safety.

The strength calculation of the guide trail.

Considering the guide trail strength condition under the cargo container fastened on the rollers.

There are 4 rollers support the cargo container in longitudinal direction. This calculation is object to determine the bearing stress of the contract surface of the roller axis and trail surface.

The next tasks are:

- Determination of the shear force of the contract surface of the axis and guide trail.
- The bearing stress calculation of the hole.

According to the equilibrium equation, the shear force is equal to the force on the roller shaft.

$$F_{CM} = \frac{ql}{2} = 407\ (N);$$

$$\sigma_{CM} = \frac{F_{CM}}{td_a} = \frac{407}{2\times10\times10^6} = 20\ (MPa);$$

$$[\sigma_t] = \frac{\sigma_t}{1.5} = \frac{280}{1.5} = 186\ (MPa);$$

$$\sigma_{CM} < [\sigma_t].$$

Therefore, it means the guide trail will not break with the maximum load.

Conclusion for the second part

The main conclusions are:

- The preliminary design of the roller system has been developed;

- according to the cargo hold size, determined the length of the guide length;

- there are 3 guide trails which are 14 m on the cargo hold floor, 25 rollers and 6 latches are on the one guide trail. The distance between two rollers is 500 mm; and the distance between two latches are 2375 mm.

- requirement for the roller system design are formulated; according to the CCAR-25, CS-25 and FAR-25, and environment of the cargo hold, determination of the specific requirements to the cargo roller system.

- material for the design are selected. In this work, chosen the materials for the cargo hold roller system according to the specific requirements. Selected Chinese aviation aluminum alloy ZL204A for the roller axis; high chromium 52100 bearing steel, (AMS 6440) for the bearings and aluminum alloy 2024 for the guide trail.

- external force analysis for the roller and stress calculation of the roller axis has been conducted; when the cargo containers move into the cargo hold, the roller axis withstands the bending and torque. There are 923.7 N and a moment 137.2 $N \cdot m$ applies to the roller axis. Then, considered the internal normal stress and shear stress of the roller axis. Besides, considered the contract surface of the axis and guide trail, in this condition, the shear force loads on the contract surface, and determined the bearing stress of contract surface. Finally, checked the strength of the roller axis.

General conclusions

In my work, the 2 main aims and objectives have been achieved:

1. Preliminary design short-range cargo aircraft with payload 5.5 tons:

- the geometrical calculation of the wing, landing gears, tail unit, fuselage.

-applied of turboprop engines (AI-24VT) for this aircraft which has a trust about 1900-2100 kW；

-The range of center of gravity position in different load situations is from 0.19 to 0.25;

2. The conceptual design of the cargo compartment roller system with load 5.5 tons had been achieved：

- analysis of existing cargo compartment roller system；

- determination of materials of the cargo compartment rollers according to the cargo hold environment;

- calculated the external force and internal stress of the roller axis and contract surface of axis and guide trail;

- check the axis and guide trail strength to determine cargo hold rollers and guide trail are in the allowable stress.

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Appendix A

INITIAL DATA AND SELECTED PARAMETERS

Passenger Number Flight	0
Crew Number	2
Flight Attendant or Load Master Number	4
Mass of Operational Items	854.64 kg
Payload Mass	5500kg
Cruising Speed	440km/h
Cruising Mach Number	0.3844
Design Altitude	5.8 km
Flight Range with Maximum Payload	1100 km
Runway Length for the Base Aerodrome	1.68km
Engine Number	2
Thrust-to-weight Ratio in N/kg	0.2350
Pressure Ratio	7.65
Fuel-to-weight Ratio	0.2450
Aspect Ratio	11.30
Taper Ratio	2.8
Mean Thickness Ratio	0.128
Wing Sweepback at Quarter Chord	8 degree
High-lift Device Coefficient	0.580
Relative Area of Wing Extensions	0

Wing Airfoil Type - supercritical
Winglets - no
Spoilers - no

Fuselage Diameter	3 m
Fitness Ratio	9.52
Horizontal Tail Sweep Angle	17.0 degree
Vertical Tail Sweep Angle	30.0 degree

CALCULATION RESULTS

Optimal Lift Coefficient in the Design Cruising Flight Point 0.49484
Induce Drag Coefficient
0.1008
ESTIMATION OF THE COEFFICIENT $Dm = M_{critical} - M_{cruise}$
Cruising Mach Number
0.38442

Wave Drag Mach Number
0.67147
Calculated Parameter D_m
0.28705

Wing Loading in kPa (for Gross Wing Area):
At Takeoff 2.596
At Middle of Cruising Flight 2.495
At the Beginning of Cruising Flight 2.544

Drag Coefficient of the Fuselage and Nacelles	0.00703
Drag Coefficient of the Wing and Tail Unit	0.01009

Drag Coefficient of the Airplane:

At the Beginning of Cruising Flight	0.03142
At Middle of Cruising Flight	0.03123
Mean Lift Coefficient for the Ceiling Flight	0.49484

Mean Lift-to-drag Ratio 15.84727

Landing Lift Coefficient	1.551
Landing Lift Coefficient (at Stall Speed)	2.326
Takeoff Lift Coefficient (at Stall Speed)	2.067
Lift-off Lift Coefficient	1.488
Thrust-to-weight Ratio at the Beginning of Cruising Flight	0.091
Start Thrust-to-weight Ratio for Cruising Flight	0.132
Start Thrust-to-weight Ratio for Safe Takeoff	0.156
Design Thrust-to-weight Ratio	0.161
Ratio $D_r = R_{cruise} / R_{takeoff}$	0.846

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

Takeoff	0.2855
Cruising Flight	0.2451
Mean cruising for Given Range	0.2470

FUEL WEIGHT FRACTIONS:

Fuel Reserve	0.02365
Block Fuel	0.08395

WEIGHT FRACTIONS FOR PRINCIPAL ITEMS:

Wing	0.14459
Horizontal Tail	0.01800
Vertical Tail	0.01849
Landing Gear	0.04944
Power Plant	0.13083
Fuselage	0.17880
Equipment and Flight Control	0.12329
Additional Equipment	0.00179
Operational Items	0.03055

Fuel 0.10760
Payload 0.19660

Airplane Takeoff Weight 27975 kg
Takeoff Thrust Required of the Engine 2249.4 kN

Air Conditioning and Anti-icing Equipment Weight Fraction 0.0180
Passenger Equipment Weight Fraction 0.0017
(or Cargo Cabin Equipment)
Interior Panels and Thermal/Acoustic Blanketing Weight Fraction 0.0112
Furnishing Equipment Weight Fraction 0.0038
Flight Control Weight Fraction 0.0099
Hydraulic System Weight Fraction 0.0246
Electrical Equipment Weight Fraction 0.0299
Radar Weight Fraction 0.0045
Navigation Equipment Weight Fraction 0.0067
Radio Communication Equipment Weight Fraction 0.0034
Instrument Equipment Weight Fraction 0.0078
Fuel System Weight Fraction 0.0031

Additional Equipment:
Equipment for Container Loading 0.0000
No typical Equipment Weight Fraction 0.0018
(Build-in Test Equipment for Fault Diagnosis,
Additional Equipment of Passenger Cabin)

TAKEOFF DISTANCE PARAMETERS

Airplane Lift-off Speed 188.81 km/h
Acceleration during Takeoff Run 1.92 m/s^2
Airplane Takeoff Run Distance 715 m
Airborne Takeoff Distance 578 m
Takeoff Distance 1293 m

CONTINUED TAKEOFF DISTANCE PARAMETERS

Decision Speed 179.37 km/h
Mean Acceleration for Continued Takeoff on Wet Runway 0.26 m/s^2
Takeoff Run Distance for Continued Takeoff on Wet Runway 1148.42 m
Continued Takeoff Distance 1692.27m
Runway Length Required for Rejected Takeoff 1752.00m

LANDING DISTANCE PARAMETERS

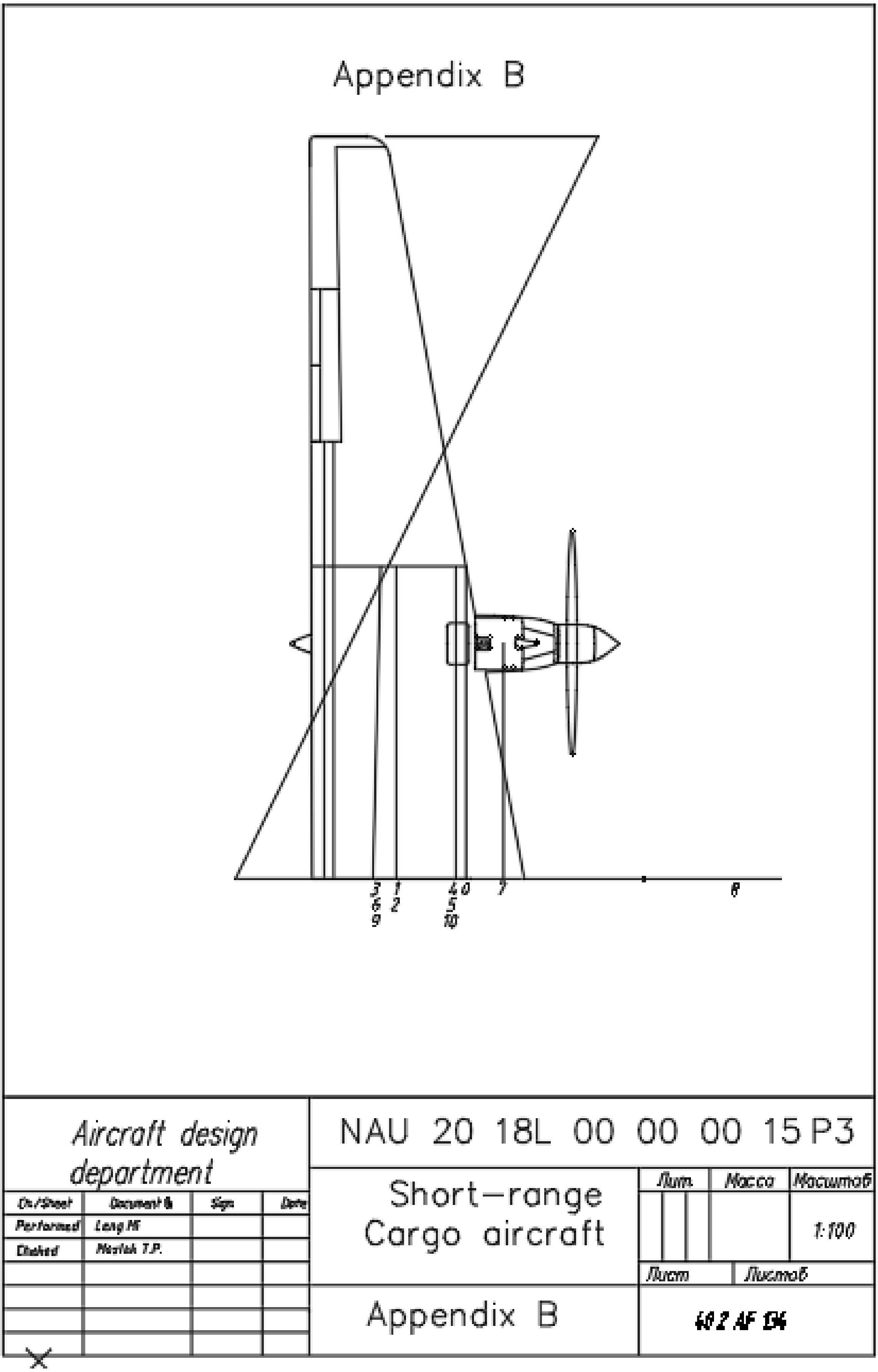
Airplane Maximum Landing Weight 26922 kg
Time for Descent from Flight Level till Aerodrome

Traffic Circuit Flight	12.1 min.
Descent Distance	14.83 km
Approach Speed	196.46 km/h
Mean Vertical Speed	1.67 m/s
Airborne Landing Distance	500 m
Landing Speed	181.16 km/h
Landing run distance	492 m
Landing Distance	992 m
Runway Length Required for Regular Aerodrome	1657 m
Runway Length Required for Alternate Aerodrome	1409 m

ECONOMICAL EFFICIENCY

These parameters are not used in the project

Appendix B



Appendix B
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Aircraft design department
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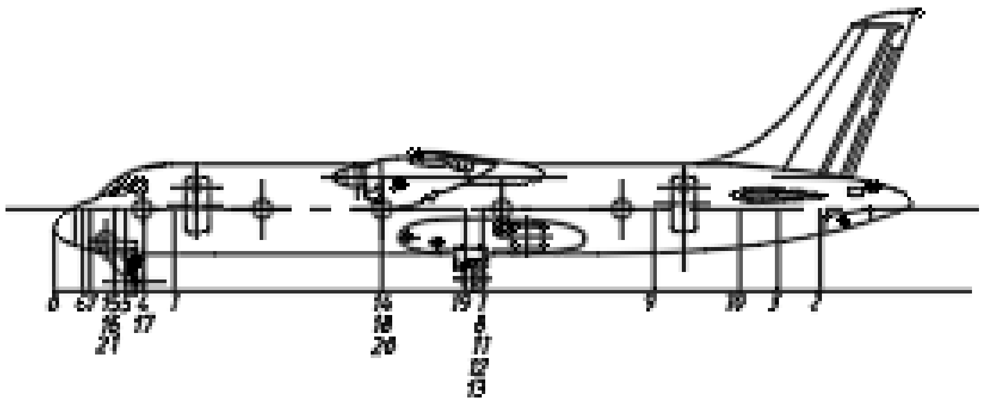
Appendix B

Appendix C

Appendix C



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	Appendix C

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