МІНІСТЕРСТВО ОСВІТИ І НАУКИ УКРАЇНИ

НАЦІОНАЛЬНИЙ АВІАЦІЙНИЙ УНІВЕРСИТЕТ

Кафедра конструкції літальних апаратів

«**ДОПУСТИТИ ДО ЗАХИСТУ**»
Завідувач кафедри
д-р техн. наук., проф.
______________________С.Р.Ігнатович
(підпис)
« _____»____________________ 2021 р.

ДИПЛОМНИЙ ПРОЕКТ

(ПОЯСНЮВАЛЬНА ЗАПИСКА)

ВИПУСКНИКА ОСВІТНЬО-КВАЛІФІКАЦІЙНОГО РІВНЯ

«БАКАЛАВР»

Тема: «Аванпроект літака середньої дальності вантажопідйомністю до 70 тонн»

Виконав: ____________ Чень Вейцюнь

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Київ 2021

MINISRY OF EDUCATION AND SCIENCE OF UKRAINE

NATIONAL AVIATION UNIVERSITY

Aircraft Design Department

APPROVED BY
Head of department
Professor, Dr. of Sc.
_________S.R. Ignatovych
«___» _______ 2021

DIPLOMA WORK

(EXPLANATORY NOTE)

OF EDUCATIONAL DEGREE

«BACHELOR»

Theme: «Preliminary design of a mid-range aircraft with cargo capacity up to 70 tons»

Performed by: ___________ Chen Weiqun

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

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

Kyiv 2021

NATIONAL AVIATION UNIVERSITY

Aerospace Faculty

Aircraft Design Department

Educational degree «Bachelor»

Speciality 134 "Aviation and Space Rocket Technology"

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

TASK

for bachelor diploma work

CHEN WEIQUN

1. Theme: «**Preliminary design of a mid-range aircraft with cargo capacity up to 70 tons**»
2. Confirmed by Rector's order from 2021 year №
3. Period of work execution __________________ to __________________
4. Work initial data:
   - Maximum payload – m = 70 tons;
   - flight range with maximum payload – $L_{пол}$= 5900 km;
   - cruise speed – V_{cr} = 871km/h at operating altitude H_{op}= 12000 m;
   - landing speed – V_{land} = 235 km/hour.
5. Explanation note (list of topics to be developed):
   - Choice of the project data for designing aircraft;
   - 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.

6. List of the graphical materials:

– General view of the airplane (A1×1);
– layout of the airplane (A1×1);
– assembly drawing of the cargo latch (A1×2).

7. 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		

8. Task date: «___» ___________ 2021

Supervisor of diploma work: M.V. Karuskevich

Task is given for: Chen Weiqun

ABSTRACT

Explanatory note to the diploma work «Preliminary design of a mid-range aircraft with cargo capacity up to 70 tons» contains:

75 sheets, 21 figures, 12 tables, 15 references and 4 drawings

Object of the design is development of a mid-range aircraft with cargo capacity up to 70 tons.

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

The methods of design are analysis of the prototypes and selections of the most advanced technical decisions, determination of basic geometrical parameters, determination of basic flight performance, calculation of aircraft mass.

The diploma work contains drawings of the mid-range aircraft with a carrying capacity of 70 tons, calculations and drawings of the aircraft layout, cargo latch concept, calculations and drawing.

AIRCRAFT, PRELIMININARY DESIGN, LAYOUT, CENTER OF GRAVITY POSITION, CARGO LATCH.

CONTENT

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Performed by	Chen Weiqun			CONTENT	Letter	куш	Аркушів
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Adviser					402 AF 134		
Stand.contr.	Khizhnyak S.V.						
Head of dep.	Ignatovych S.R.						

List of abbreviations

MLG	Main landing gear
NLG	Nose landing gear
TLG	Tricycle landing gear with front single-strut and two main gears
APU	Auxiliary power unit
ULD	Unit load device
CM	Center of mass
CG	Center of gravity
MAC	Mean aerodynamic chord
FAR	Federal Aviation Regulations
CS	Certification Specifications for Large Aeroplanes
CCAR	China Civil Aviation Regulations
MDCD	Main deck cargo door
LDCD	Lower deck cargo door
MTOW	Maximum take off weight
CL	Cargo Latch

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	Cargo latch (assembly drawing)	A1	1
4	Detailed view of the cargo latch	A1	1

Introduction

Air cargo is a high-value industry and plays an important role in providing fast and reliable cargo services. Cargo planes operated by airlines account for 80% of the revenue of the air cargo industry, and this proportion may increase in the future.

Most players in the industry continue to rely on and expand their cargo business by flying freighters, freighters will continue to carry most of the world's air cargo in the future.

In the future, with the growth of the world economy and the world population's demand for high-value commodities, the value of global total trade commodities is expected to rise. As the average value of traded goods increases, air freight will solve a larger proportion of trade. Air cargo will continue to be the preferred method of transporting high-value cargo.

Unlike passenger aircraft, the most important role of cargo aircraft is to transport cargo. It is not necessary to provide a comfortable environment for the goods, only to ensure that the goods are not damaged during transportation. In addition, compared with passenger aircraft, it has larger wings to generate greater lift. It has a high and wide cross section and can carry as much cargo as possible.

The purpose of the work is the preliminary design of the aircraft and the evaluation of the aircraft performance.

The objectives of the work are development of cargo mid-range aircraft with cargo capacity 70 tons.

Designed aircraft is low wing construction with a conventional tail unit, two Pratt & Whitney PW4000 engines are mounted to the wing with a thrust of 300 KN each.

Except pilot cabin, it has three cargo cabins, one main deck and two lower deck distributed at the front and rear of the lower fuselage. It has one main deck cargo door and two lower deck cargo door for loading and unloading cargo. A container pallet loader is used for loading and unloading cargo because of the large clearance between the cargo door and ground. Because of the different location of the cargo door, it is possible to loading and unloading of main deck and lower deck at the same time, which will reduce the time for loading and unloading.

According to the needs of commercial transportation in China there is a need for a cargo aircraft with medium to long range, high payload and easy landing.

In this work, I have taken into account the requirement of national and inter national airworthiness lows, rules and recommendation (FAR-25, CS-25 and CCAR-25). These are high fuel efficiency, low landing and take off requirement, short landing and take off distance, high payload efficiency, high reliability, high operability.

A special part of the work deals with the new design of the cargo latch. The key idea of the design is to design a cargo latch that can carry a load of 1.7 tons.

PART 1
PRELIMINARY DESIGN OF THE AIRCRAFT

1.1. Choice of the project data for the aircraft preliminary design

1.1.1. Technical data and geometrical parameter of prototypes

The aircraft design processes is divided into three steps: Conceptual design, Preliminary design and Detail design. The first and most important phase of aircraft design process is conceptual design. In this step, the aircraft will be designed without precise calculations, main feature of this step is that initial definition come from requirements established by customers or market.

The second step is preliminary design, in this step, initial definition come from the results from computer calculation procedures. But this result is not certain, it can be changed during the preliminary design process. The parameters are calculated in this step have an important impact on the next stage, so it's important to ensure the accuracy of parameters.

The last step is detail design, in this step, the engineer should determine the number and location of spar, rids, stringers and other structures.

Our task in this stage of diploma work is analysis of various prototypes according to our initial data. After input our approximate data that we gathered from different resources into computer program, we received more accurate initial data to begin our project. Our task was to choose a transport aircraft with cargo capacity of 70 tons to transport given payload on maximum range of 5900 km.

During the many processes in the preliminary design stage, the analysis and comparison of data related to previously successfully designed machines is very important. To design a new aircraft, the following parameters need to be considered: aerodynamics, strength of structure, safety, economy and fuel consumption.

This process is called "Analysis and Synthesis". In this process, we need to select some prototypes base on some similar parameters, such as maximum take-off weight, cargo capacity, flight range with MTOW, flight altitude, landing distance, engine thrust, etc. (table 1.1).

Table 1.1 - Operational- technical data of plane-prototypes

Name and dimensions	A330-200F	B757-200F	An-124
Max payload, kg	70000	38290	150000
Crew, numbers	2	2	6
Flight Range with MTOW,km	5900	5370	3700
Cruise speed, km/h	871	854	865
Cruise altitude, km	12	13	10
Number and type of engines	2 (PW4000)	2 (PW2037)	4 (D-18T)
Take off run at MTOW, m	2550	2728	3000
Landing distance, m	1750	1555	2800
Landing speed, km/h	235	137	-
Field length for take off, km	2.4	2.7	2.8
Thrust (each engine), kN	287	163	229
Engine pressure ratio	42.8	31.2	27.5
Mass of fuel, kg	75586	33077	270273.5
LG scheme	TLG	TLG	TLG
The form of fuselage cross-section	circular	circular	circular

Table 1.2 - Geometrical parameters of prototypes

Name and dimensions	A330-200F	B757-200F	An-124
Length of the fuselage, m	58.82	43.7	69.1
Wingspan, m	60.3	38	73.3
Wing area,m^2	361.6	185.25	628
Aspect ratio	10	7.8	8.6
Sweepback angle, degree	30	25	25
Height at tail, m	16.9	13.6	21.08
Fuselage diameter, m	5.64	3.76	10
Wing tapper ratio, m	0,25	0,25	0,25
Mean geometric chord, m	7.26	4.3	8.53

For the preliminary design, I have selected Airbus 330-200, Boeing 757-200F, and Antonov 124 as a planes-prototypes.

On the Table 1.1 and 1.2, we have operational – technical and geometric parameters of these three different transport aircraft which have similar characteristic and parameters to our desired aircraft.

The scheme of designed aircraft is determined by the relative position, number and shape of the components of the aircraft. Its aerodynamic and technical and operational properties depend on the scheme and aerodynamic layout of the aircraft. The better selection of the scheme can improve the performances of the aircraft. As for my task: preliminary design of mid-range cargo plane, it is possible to change some components that have been successfully applied on other planes. Therefore, for my designing aircraft, I will make some changes in the main components of my designing aircraft in according to the given technical, economic and other requirements.

The selection of the designing scheme will be based on the designing schemes of the selected prototype. The data in the Table 1.1 and Table 1.2 provide the possibility of selecting the best parameters of the new aircraft.

1.1.2. Brief description of prototypes

The A330-200F is a freighter version of the A330-200 which can carry 65 tons over 7,400 km or 70 tons up to 5,900 km. To solve the problem of A330's not level cargo floor, the A330F uses an improved nose gear to get a level deck. Using the same landing gears as A330-200, but it has lower attachment points in the fuselage, therefore, a unique blister fairing is required on the nose to accommodate the retracted nose landing gear.

Power is provided by two Pratt &Whitney PW4000 or Rolls-Royce Trent 700 engines.

The general view of cabin layout of A330-200F is shown in Figure 1.1 and cargo compartment is shown in Figure 1.2.

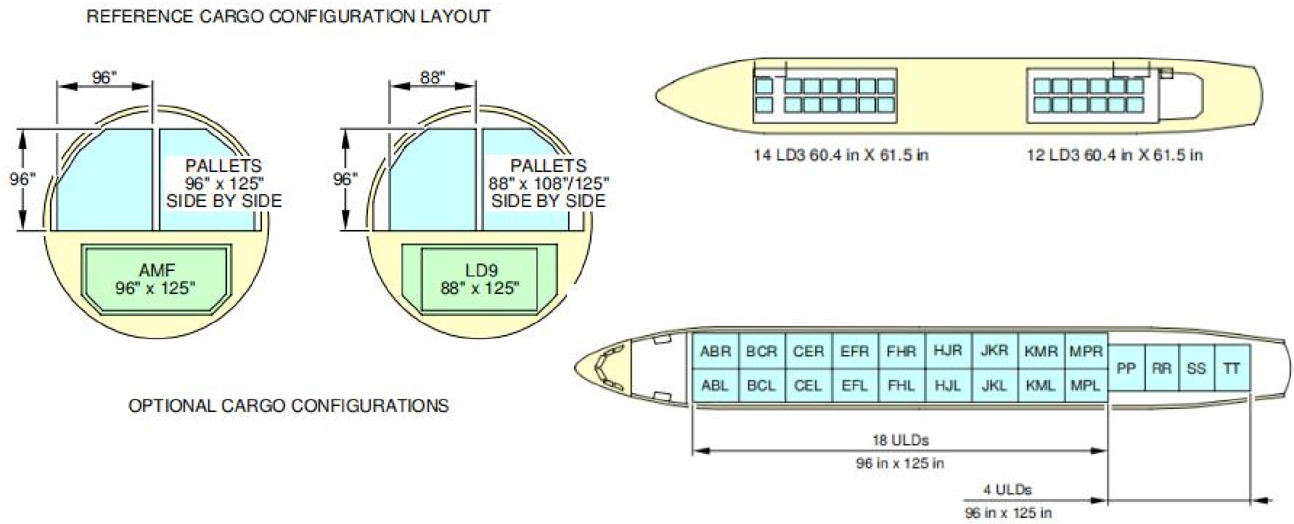


Figure 1.1 - General view and cabin layout of A330-200F

Figure 1.2 - Cargo compartment of A330-200F

Antonov An-124 Ruslan is a large strategic airlift aircraft designed by the Antonov Design Bureau of the Ukrainian Soviet Socialist Republic in the 1980s. The An-124 is still the largest military transport aircraft currently in service. The lead designer of the An-124 was Viktor Tolmachev. It allows rear cargo door to open in flight without destroying structure integrity because of the double fuselage. With a payload of 150,000kg and an initial cruise altitude of 10,000m the An-124 has an range of about 3,700km with MTOW. And it has 4 Progress D-18T high-bypass turbofan engines,which can generate a thrust of 229KN. The general view of cabin layout of A330-200F is shown in Figure 1.3 and cargo compartment is shown in Figure 1.4.

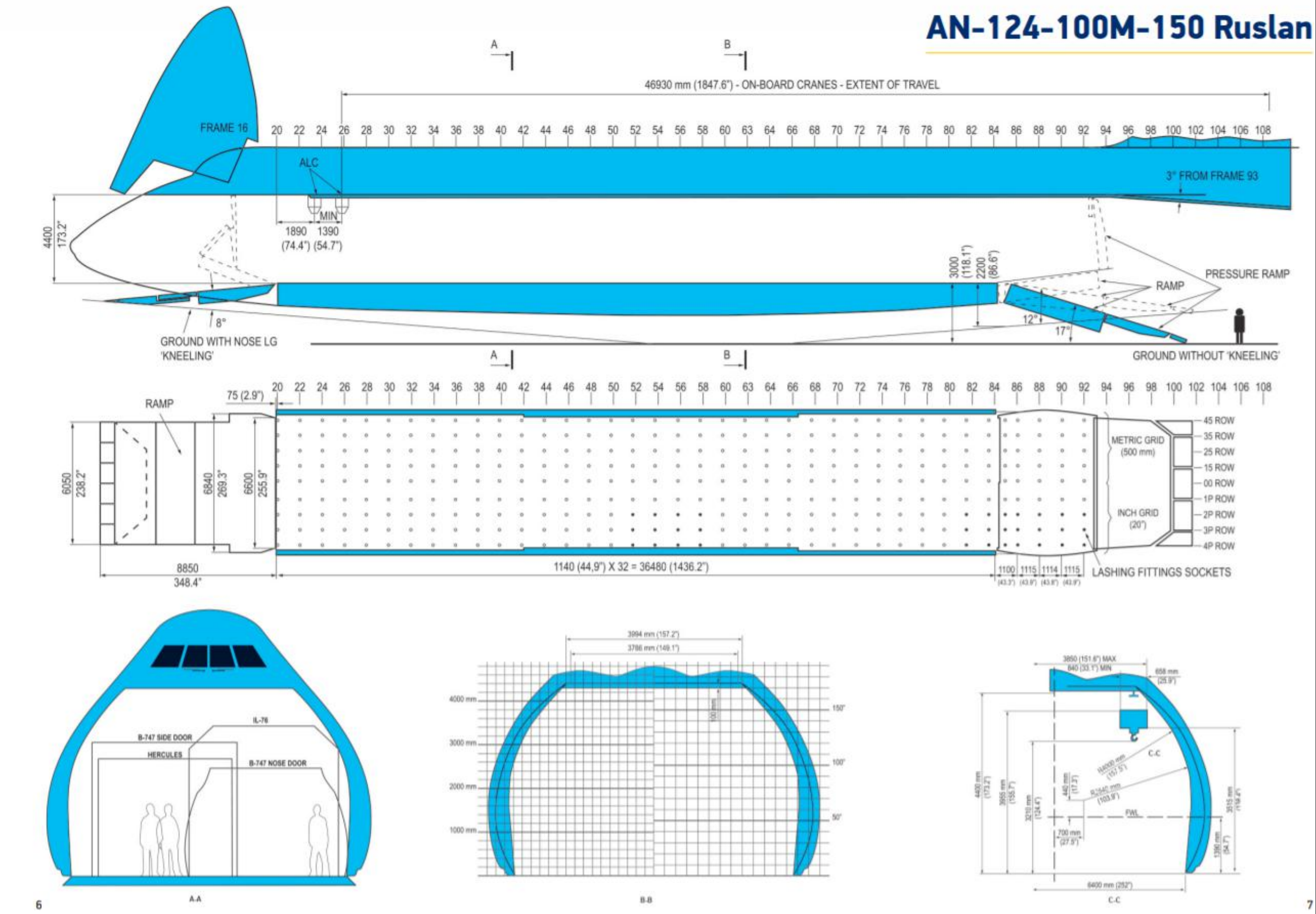


Figure 1.3 - General view and cabin layout of An-124

Figure 1.4 - Cargo compartment of An-124

The 757-200F is the freighter version of the 757-200, which served UPS Airlines in 1987. Targeted at the overnight package delivery market, it can carry up to 15 ULD containers or pallets on main deck, for a volume of up to 187 m^3, furthermore its two lower decks can carry up to 51.8 m^3 of bulk cargo. With a payload of 32,755 kg, the 757-200F ha a range of 5,435 km. Including the container weight the maximum payload capability is 39,800 kg. It is powered by two Rolls-Royce RB211-535E4 or Pratt & Whitney PW2000-37/40/43, which can generate a trust of 190 KN.

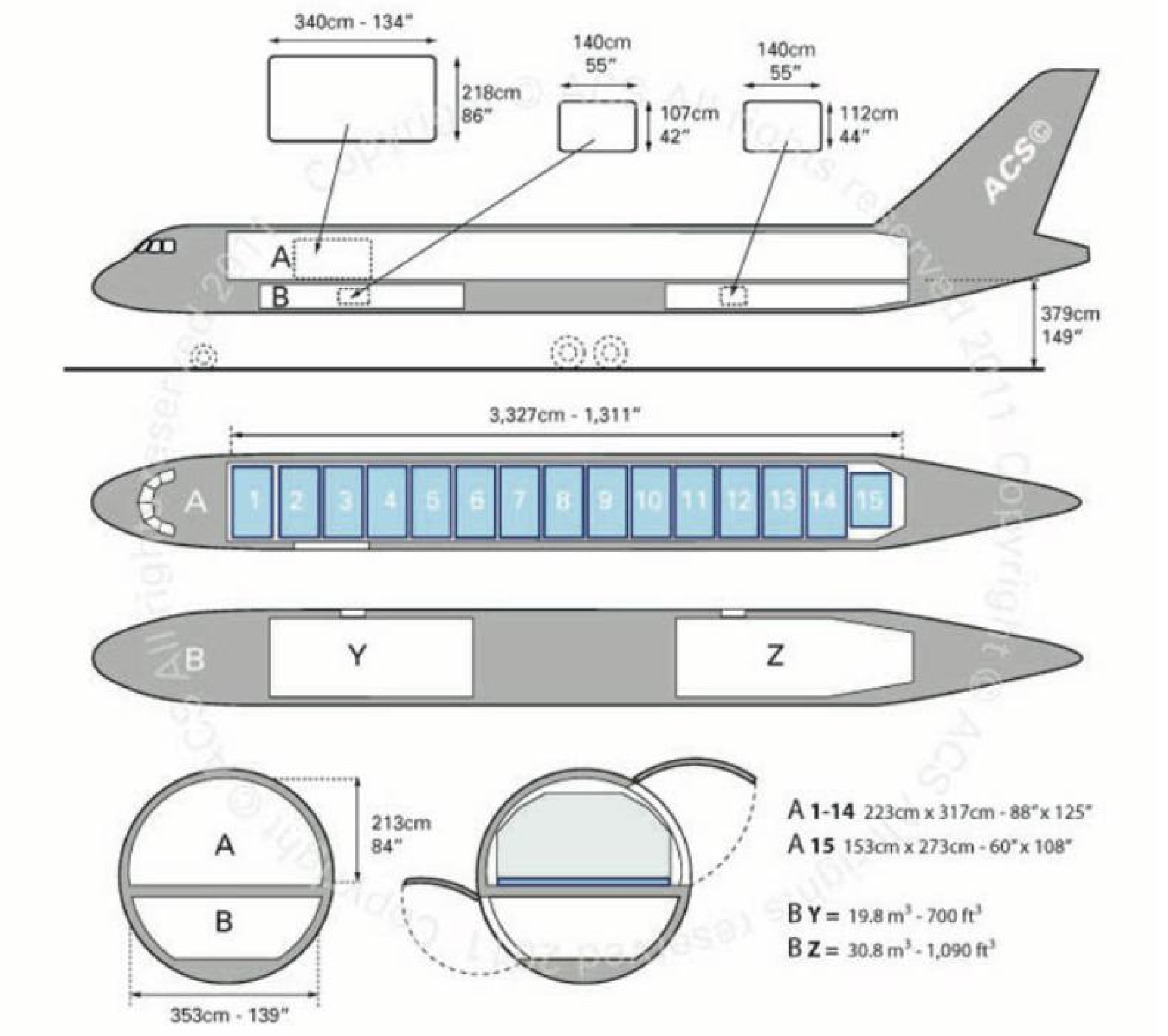


Figure 1.5 – General view and cabin layout of Boeing 757-200F

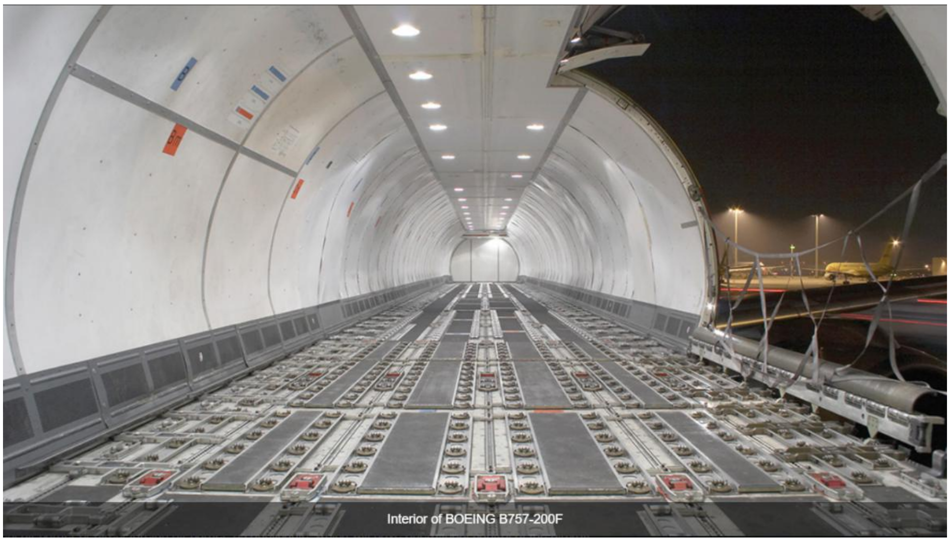


Figure 1.6 - Cargo compartment of Boeing757-200F

1.1.3. Analysis and comparison of the prototypes

The first step to start this project is to select the aircraft which is going to be analyzed as prototype. After comparing technical data and geometrical parameter of given prototypes based on given task, in my opinion the most optimal choice for this task is A330-200F. The reasons for choosing this aircraft are: firstly we consider the maximum payload, we need an aircraft have maximum payload about 70 tons, only A330-200F meets this payload inside cargo compartment, as for Boeing 757-200F, its maximum payload is 38.29 tons, it can not satisfy our require payload. About the An 124, has a maximum payload capacity of 150 tons, much larger than the 70 tons, so we can’t fully utilize the space of fuselage.

Next, we consider the range of the aircraft, only A330-200F can cover this distance perfectly with consideration of payload capacity. In addition, the cargo hold of A330-200F can accept all commonly used pallets and containers.

The aircraft considered to design in this project is mid-range commercial transport aircraft. It has low-wing, two engines, it is able to carry large cargo, equipment, and it can take off and land at a small airport with poor terrain. It has maximum payload up to 70,000kg, establishing the maximum altitude to 12.5 km settling for a range of about 5900km.

1.2. New plane description

According to my task, a cargo aircraft was designed, which is developed to carry load up to 70 tons.

As for my prototype, I choose A330-200F because of it has relatively good characteristics, large cargo capacity and parameters, which meet my requirements to carry cargo for domestic and international transportation in China. My new aircraft has one MDCD with a width of 3.58 m, two LDCD with a width of 2.7 m and one bulk cargo door with a width of 0.95 m. With the help of the huge width of MDCD and special container pallet loader, it is possible for the main deck of new aircraft to accommodate all types of pallet and container and facilitating loading and unloading.

The following methods are may used for loading and unloading:

- Container and pallet loader;
- bulk conveyor loader;
- on-board deck ball mat;
- power drive unit.

Laces and nets are used for cargo binding.

The new aircraft is designed to carry cargo at a range of 5900 km with a cruising speed of 871 km/h, at an altitude of 12000 m.

My prototype is made as a low wing with two turbofan engines, and a special wing-let is installed at the tip of the wing to reduce the induce drag. Moreover, it has a vertical stabilizer with a rudder and two horizontal stabilizers with elevator to control the aircraft yaw and pitch.

The wing has sweep-back angle on 1/4 chord 31°. The wing structures consist of ribs, stringers and two spars. In addition, two turbofan engines with a thrust of 300 kN and two main LGs are installed under the wing.

The new aircraft is designed as a semi-monocoque type of fuselage, pressurized fuselage, and circular fuselage section. The fuselage is divided into 3 compartments: forward, middle and tail part. These three compartments are connected by the skin, longerons and stringers. The area of the fuselage between frames 1-91 is pressurized. The fuselage has the largest diameter between frame 20-64.

Most of the structural elements of the fuselage are made of sheet and pressed hard aluminum and aluminum alloy. Some are may made of composite material.

As for the manufacture of the air-frame, the high-strength aluminum alloys are widly used because of its characteristic of easily manufacture, light, and good stiffness, strength and fracture toughness.

To decrease the weight of the aircraft, my new aircraft is designed to incorporate some advanced materials, such as carbon fibre, glass fibre and quartz fibre reinforced plastic, which are used in the flaps, fin and elevator, belly faring, gear doors, radome and engine cowlings.

My designed aircraft is divided into 4 parts: one pilot cabin, one main deck cargo cabin and two lower deck cargo cabins. The area of the pilot cabin is between frame 1-5. And there is a door between the pilot cabin and main cargo cabin to allow the crew to enter the main cargo cabin.

1.2.1. Fuselage

A special fairing is covered the area between the nose of the fuselage to the frame 1, and an air-navigation equipment is installed in this area. The front pressure bulkhead is installed at the frame 1. The area between frame 1-5 is the cockpit, which is pressurized. The pilot seats, flight instruments and aircraft control systems are installed here. There is a wall installed at the frame 5 to separate the cockpit and galley, and a door is installed on the wall to allow the crew enter the cockpit. The galley, lavatory and rest room are installed between frame 5-11. On both sides of the fuselage, two entrance doors for crew to enter the aircraft are installed between the frame 8-10, which are also used for emergence door. There is a small window on the entrance door to allow crew to observe the situation outside the aircraft. A wall is installed on the frame 11 to separate the main cargo cabin and pilot cabin, and also two doors are installed on the wall to allow crew to enter main deck cargo cabin from pilot cabin.

The area from the frame 19 to the frame 80 above the floor is the main deck cargo cabin. The MDCD is installed in the Port side between frame 20 to frame 29. Cargo loading system are installed on the floor of the main deck cargo cabin floor, such as ball mats, ball trays and roller trays. The ball mats are installed between frame 20 and frame 27 and extends across the entire width of the cargo hold on the main deck. It provides support and allows ULD to move in all directions within the MDCD area. Ball trays are installed between frame 27 to frame 32. It provides support and allow all-round movement of the ULD in other areas except the ball mats area. The roller pallet is installed below the cargo hold except the ball mat area to transport the ULD to its loading position.

The area from the frame 20 to the frame 39 under the floor is the forward lower deck cargo cabin. The area from frame 53 to frame 73 under the floor is the aft lower deck cargo cabin. And the forward LDCD is installed between the frame 20 to frame 26 with a width of 2.7m, the aft LDCD is installed between frame 59 to frame 65 with a width of 2.72m, the bulk cargo door is installed between frame 67 to frame 69 with a width of 0.95m.

The door is designed to withstand the internal hoop tension load pressure. They are of conventional design and have skin beams with machined frames on the outside and on the inside and internal structure of sheet metal parts. The upper end of the structure forms a hinge for the door. The door is hydraulically opened outwards and upwards.

The wing is attached to the frame 39 to frame 53 of the fuselage. And the tail unit of the fuselage carries the empennage of the aircraft, which is located behind the aft pressure bulkhead and not pressurized. The Auxiliary Power Unit is installed between the frame 90 to frame 106. The APU door for maintenance is installed between frame 95 to frame 101. The main structures of the fuselage are skin, frames, stringers, longerons, pressure bulkheads.

1.2.2. Wing

My designing aircraft is low wing with a span of 58 m and sweep-back angle of 31° . It has supercritical airfoil which minimizes the influence of aircraft's drag to its structural design. The dihedral angle of the wing is about 6°.

As for the structure of my designing aircraft, the wing consists of skin made of aluminium alloys, 22 stringers on each side of the wing made of aluminum 2024 T3, two main spars made of solid extruded aluminum or aluminum extrusions, 40 ribs on each side of the wing made of steel and aluminium alloys.

The center of the wing forms a construction of the wing box. The wing box consists of the two main wing spars, upper and lower wing skins, ribs and stringers, which form a box shape. The wing torsion box extends to the root of the wing and connects with the fuselage with bolts. One of the important function of wing box is providing structure connection and as primary load structure of the wing. For example, the wing box is the attachment point for wing components such as flap and slat, and also the main LGs and engines pylon. In addition, the area inner the wing box is used to store fuel.

Two turbojet engines are mounted under the wing rib 11 by pylon. And the main LGs are connected to the wing spar between wing rib 5 to rib 6.

About the control surface, the newly developed aircraft has a tripple sloted inboard trailing edge flaps on each side of the wing, which are used to increase lift, drag ay the low speed during take off and landing. A triple slotted outboard trailing edge flaps are mounted on the each side of the wing. It not only can increase the lift, drag and reduce the stalling speed, but also used to generate rolling moment when it is necessary.

Two ailerons are mounted on the tip of the wing trailing edge. When aircraft need rolling moment, ailerons on the both side of the will deflect in the opposite direction to generate rolling moment. The aileron can deflect upward to 25° and downward to 15°. Six slats are mounted on the leading edge of the wing, its main function is to increase lift during low speed operations such as take off and landing.

One speed breaker is installed near the inboard trailing edge flap to decrease the speed of the aircraft. And five spoilers are installed near the outboard flap. It not only can increase the drag, but also decrease the lift. So, it is often used during landing.

In addition, one flap track is installed under the inboard flap, three flap track are installed under the outboard flap. The flap track can able to hold the flap when the flap id extending.

Winglet with height of 2.74m are installed on the tip of the wing. It can reduce the aerodynamic induced drag caused by vortex on the tip of the wing. And then, the fuel efficiency and range are improved.

1.2.3. Tail Unit

The designed aircraft has conventional tail part, consists of horizontal and vertical stabilizers.

The horizontal tail structure consists of two spars and 20 ribs. It has a sweepback angle of 35°. And the fuel trim tank is mounted on the horizontal tail, which is used to balance the center of gravity position.

The horizontal tail consists of elevator and trim tab. The main function of the horizontal tail is to provide longitudinal trim and stability requirements. The trim tab is attached to the trailing edge of the elevator. It can counteract aerodynamic force, which can keep the altitude of elevator without the need for pilot apply constantly control force. When the pilot moves the trim tab up or down, the elevator will deflect in the opposite direction automatically.

The vertical tail consists of fin, rudder and trim tab. Rudder is mounted on the trailing edge of the fin. The main function of the vertical tail is to provide directional trim and stability requirements. For example, the rudder is used to overcome the adverse yaw caused by the engine failure. Because of the large mass of the aircraft, the rudder is controlled by hydraulic actuator to overcome the mechanical and aerodynamic loads on the rudder.

The rudder trim tab has the same function as the elevator trim tab. It can counteract aerodynamic force. The rudder has horn which can minimum the bending moment in the rudder surface.

1.2.4. Power Plant

The power plant of my designing aircraft consists of two turbojet engine (Pratt & Whitney PW4000 or Rolls-Royce Trent 700) with a engine thrust of 300kN to 400kN and Honeywell 331-350C auxiliary power unit (APU) is mounted on the tail part of the fuselage.

The engines are mounted on the wing spar under the wing, near the wing rib 11.

The engine (Rolls-Royce Trent 700) is a three shaft high bypass ratio, axial flow, turbofan engine with a length of 5.64 m, diameter of 2.5 m and dry weight of 6160kg. The compressor consists of a single stage fan with 26 blades, eight stages intermediate pressure compressor and six stages high pressure compressor.

Combustion chamber consists of Single annular and 24-off Fuel Spray Nozzles. The turbine consists of single stage high pressure turbine, single stage intermediate pressure turbine and four stages low pressure turbine.

Honeywell 331-350C auxiliary power unit (APU) is mounted on the tail part of the fuselage between the frame 90 to frame 106. the main functions of the APU are following:

- Provide additional trust when the aircraft is climbing;
- provide thrust when one of the engine is failure;
- provide hot bleed air to the aircraft;
- provide power for aircraft onboard network when aircraft is in the parking with engines not work or the starter generator is failure.

1.2.5. Landing gears

The main functions of LGs are following:

- To support the aircraft on the ground during loading, unloading and taxing;

- to provide free maneuver of the aircraft during taxing;
- to provide necessary clearance between fuselage and ground;
- to absorb shock during landing.

My designed aircraft has: Two MLGs with four wheels and related doors, a NLG with two wheels and related doors. Two MLGs are located under each side of the wing and can be retracted toward the center line of the fuselage. The NLG can be retracted forward into the fuselage. The retraction and extension of the landing gears and LG doors are operated by hydraulic and mechanical.

The NLG is installed between the frame 14 to frame 17. The MLGs are installed between frame 40 to frame 53. The main parts of the LG are: tires, break, piston, cylinder, shock absorber, torque link, shimmy damper and etc. The torque link aligns with LG and fixes the piston in the upper cylinder. Shimmy dampers help control the swing of the nose gear and prevents it from swinging violently.

The wheel base B of my designing aircraft is equal to 22.2 m, and wheel track T is equal to 11.34 m. The NLG tire has a size of 1050x395R16, ply rating of 28 and rate load of 152183.7N. The MLG tire has a size of 1400x530R23, ply rating of 40 and rate load of 333513.7N. And the clearance angle of my designing aircraft is 12°.

1.2.6. Flight Control System

The control system provides following control:
- Ailerons;
- flaps;
- rudder;
- elevator.

The ailerons provide rolling control, rudder provides yaw control, and elevator provided pitch control. Trim tabs are installed on the elevators, rudders, and ailerons.

And compared to the prototype, the aircraft I designed shares the same glass cockpit layout with the A320 and A340, equipped with electronic instrument displays instead of mechanical instrument displays. The flight deck equipped side-stick controls, six main displays, and the Electronic Flight Instrument System (EFIS), which covers navigation and flight displays. Except the flight deck, my designing aircraft also has the fly-by-wire system common to the A320 family. It is also equipped three primary and two secondary flight control systems.

1.2.7. Crew cabin

The crew cabin is installed between the frame 1 to frame 5. The crew consist of first pilot and co-pilot. And the crew cabin consists of two pilot seats, control panel, two fire extinguishers, one galley, one lavatory, one crew rest compartment, and two sofas.

The cockpit must be as small as possible to save the space and decrease the weight, but it must provide the flight crew with conditions for normal work and rest.

A wall is installed between pilot cabin and furniture compartments to separate the pilot cabin and furniture compartments. And a door is installed on the wall to allow crews to enter the pilot cabin.

A smoke detector and warning system are installed to the lavatory to detect fire. And a fire extinguisher is installed on the top of the waste compartment.

As for the emergency equipment, two emergency slides are installed on two crew entrance doors. And an emergence rope is installed on the pilot cabin window.

1.3. Substantiation of the new aircraft parameters

1.3.1. Wing geometry calculation

The geometric characteristics of the wing are determined by the take-off weight m_0 and the specific wing load P_0 (wing loading).

Wing loading value P_0 is selected based on the similar aircraft analysis.

For my designed aircraft it takes to 5223 kg/ m^2.

Full wing area with extensions is:

$$A_w = \frac{m_0 * g}{P_0} = \frac{284317 * 9.8}{5223} = 533.5(m^2).$$

Compared with full wing area of my prototype, I find my calculated wing area is too large, so l will choose the full wing area similar to my prototype, equal to 361.6 m^2.

Relative wing extensions area is 0.01.

Wing area is:

$$S_w = 361.6 \cdot (1 - 0.01) \approx 358(m^2).$$

One other important parameter for wing is aspect ratio.

The aspect ratio is often used to predict the aerodynamic efficiency of the wing because the lift to drag ratio increases with the aspect ratio. Therefore, the greater the aspect ratio of the wing, the higher the fuel efficiency of the wing.

Table 1.3 – 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 my designed aircraft, it is a high subsonic transport. The range of the aspect ratio is 8-12. And compare with the prototype, I selected the value of aspect ratio equal to 9.26.

Wing span is:

$$l = \sqrt{Aw \times \lambda} = \sqrt{361.6 \times 9.26} = 58(m).$$

where λ is aspect ratio.

To calculate the chord of the wing, we must have the taper ratio of the wing. As for my designing aircraft, I selected the taper ratio equal to 4 by the following considerations:

- The wing taper has an advantage of change the wing lift distribution;
- the wing taper will reduces the wing weight, because the mass distribution of the wing is decrease from tip to root, which cause the lower bending moment to the wing root;
- the wing taper will improve the aircraft lateral control and lateral stability;
- the main LGs of my designing aircraft are installed under the wing. So larger root chord will easily accommodate LG;
- the most important is compared with the value of the taper ratio of my prototype.

Root chord is:

$$b_0 = \frac{2 \times Aw \times \eta_w}{(1+\eta_w)l} = \frac{2 \times 361.6 \times 4}{5 \times 58} = 10(m).$$

where η_w is taper ratio.

Tip chord is:

$$b_t = \frac{b_0}{\eta_w} = \frac{10}{4} = 2.5(m) .$$

The thickness to chord value has been selected equal to 0.11 based on the following considerations:

- In subsonic range, the more airfoil thickness, the more profile drag is generated.

- the greater the thickness ratio, the smaller wing mass because bath bending and torsional stiffness are increased;

- when thickness ratio increase, the maximum lift coefficient will increase;

- the volume of the fuel tank in the wing will increase with the rise of the thickness;

- compared with the value of thickness of my prototype.

In short, the thickness to chord should be as large as possible under the premise of meeting drag requirements. So I select the value of the thickness to chord equal to 0.11.

As for the the sweepback angle on 1/4 chord, my designing aircraft has a sweepback angle of 31° base on the following considerations:

- The wing sweep permits higher cruise Mach number, or greater thickness or lift coefficient at a given Mach number;

- the sweep wing stabilizes the wing aeroelasticity;

- sweep wings can reduce turbulence by slowing down the speed of the air as it moves across the surface of the wings.

But there are some disadvantages of sweep wing, such as when aircraft flight with subsonic and supersonic speeds, the drag coefficient will quickly increase.

For wing designing, we must know quantity and position of the main structures. And I selected the two spars design.

The main structure of the wing of my designing aircraft are showing in the following figure.

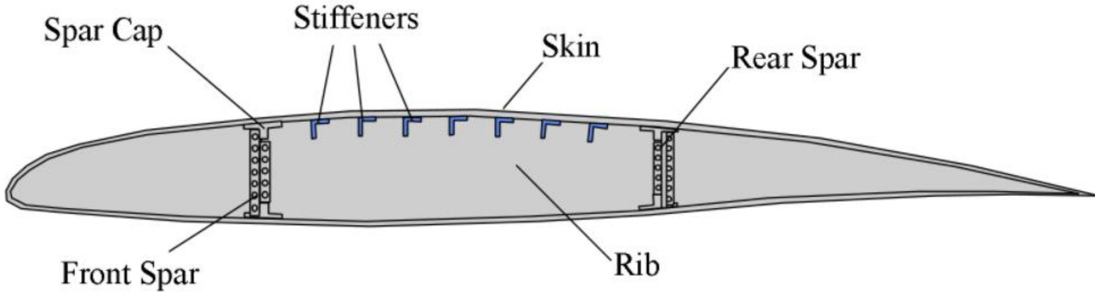


Figure 1.7 - Two spars wing design

I have used the geometrical method of mean aerodynamic chord determination (Figure 1.8.).

Mean aerodynamic chord is equal: b_{MAC}=7 m

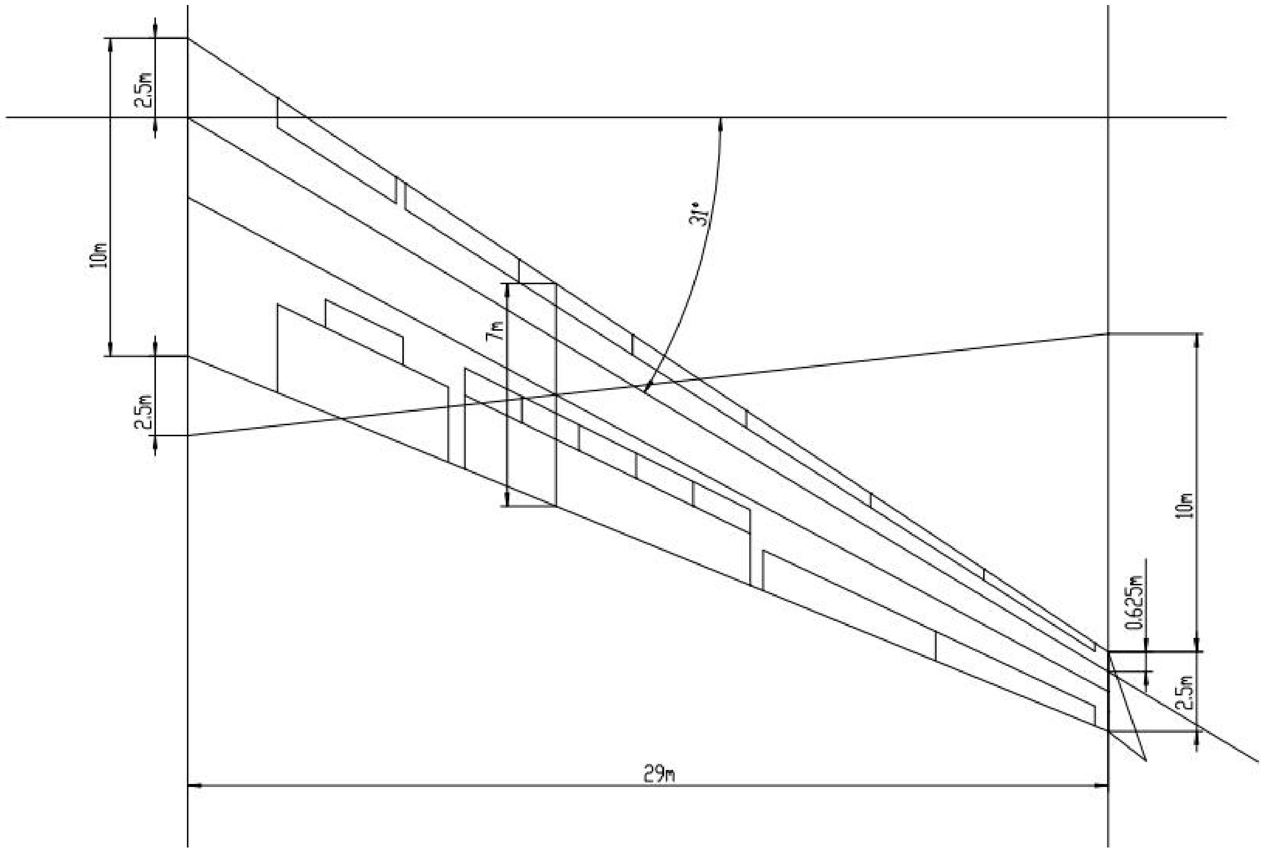


Figure 1.8 - Geometrical method of wing mean aerodynamic chord determination

The next step is to calculate geometric data of aileron and high lift devices.

Ailerons geometrical parameters are determined in next formula:

Aileron span is equal:

$$l_{ai} = 0.375 \times \frac{l_w}{2} = 0.375 \times \frac{58}{2} = 10.9(m).$$

Aileron area is equal:

$$S_{ail} = 0.065 \times \frac{S_w}{2} = 0.065 \times \frac{358}{2} = 11.6(m).$$

Aileron chord is equal:

$$b_{ail} = 0.24 \times b_t = 0.6(m).$$

Area of aileron trim tab is equal:

$$S_{tt} = 0.05 \times S_{ail} = 0.05 \times 11.752 = 0.59(m^2).$$

Range of aileron deflection:

Upward $\delta'_{ail} \geq 25^\circ$;

Downward $\delta''_{ail} \geq 15^\circ$.

To design the high lift device for a wing, the following parameters must be determined:

- High lift device location;
- the type of high lift device;
- high lift device chord;
- high lift device span;
- high lift device maximum deflection.

As for the location of the high lift device, the best location is in the inboard portion of the wing. Because the high lift device will generate a lower bending moment to the wing root, which will cause less fatigue on the wing root.

It is not necessary to design a new high lift device in the early stage of designing, because it is expensive and complex. So, it is better to use the successful design of high lift device.

I selected the three slotted flaps for my designing aircraft based on the following consideration and compared with the type of high lift device of my prototype:

- A heavier aircraft requires a more powerful high lift device;
- the more powerful high lift device, cause a slower stall speed, which means high security;
- the more powerful high lift device cause a shorter runway length during take-off and landing.

As for the chords of wing high life device calculation. In the modern design the rate of the relative chords of wing high-lift devices is:

b_f = 0.3..0.4 – for three slotted flaps and Fowler's flaps;

b_s = 0.1..0.15 – slats.

So, the chord of the wing high lift devices are:

$$b_f = 0.3...0.4\ b_i; \quad b_s = 0.1...0.15\ b_i.$$

where b_i is the chord of the wing in the current location.

1.3.2. Fuselage Layout

The main function of fuselage is provision of volume for payload and provide overall structural integrity. My designing aircraft is a cargo aircraft, so payload is cargo. Just like a traditional cargo aircraft, my designing aircraft is semi-monocoque construction, this construction has many advantages: not only the skin can carry load, but also the longitudinal elements such as stringers and longerons. It can increase the skin stability and provide multiple load paths for fail-safe design. About the material of fuselage, usually we use Aluminum alloy, but some composite materials will be used too.

When selecting a fuselage section, not only the aerodynamic requirement must be considered, but also the layout of the fuselage to provide as much volume as possible for the payload. For my designing plane, i select Double-Bubble section.

For transonic airplanes fuselage nose part length is:

$$l_{nfp} = 1.73 \times D_f = 1.73 \times 5.64 = 9.76(\mathrm{m}) .$$

To design the fuselage, the following geometrical parameters should be calculated:

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

Because there is no other initial data to calculate all parameters, so I have to compare my designing plane to prototype and select. Some of the following parameters are compared with the prototype value.

Fuselage diameter D_f=5.64m;

Fuselage length lf=58.82m;

So fuselage fineness ratio is equal:

$$\lambda_f = \frac{l_f}{D_f} = \frac{58.82}{5.64} = 10.43 .$$

Fuselage nose part fineness ratio is equal:

$$\lambda_{np} = \frac{l_{np}}{D_f} = 1.73 .$$

Tail unit fineness ratio is equal:

$$\lambda_{tu} = \frac{l_{tu}}{D_f} = 3.64 .$$

For the height of main deck cargo cabin and lower deck cargo cabin, I will select h_{main}=2.45m and h_{low}=1.68m based on the large cargo capacity and compared the value of my prototype.

As for the the stringers and frames of the fuselage, the distance between two stringers is about 20 cm, and the distance between two frames is about 50cm.

New designed aircraft has a maximum load capacity of 70 tons, cabin length of 45m and maximum cabin width of 5.26m. Compared with my prototype, the main deck of my designing aircraft can accommodate up to 22 pallets(96”x125”x96”), the lower deck of my designing aircraft it can accommodate 8 pallets(96”x125”x64”) +2LD3 containers.

1.3.3. Lavatory and galley

Although my designing aircraft is a cargo plane, but it is a mid-long range plane, which has a block time with maximum payload of 9 hours, so it is necessary to provide a lavatory for the crews. My designing plane has two crew, so the lavatory number is equal: $n_{lav} = 1$.

Area of lavatory:

$$s_{lav} = 1.5(m^2).$$

The location of lavatory and galley is show in Figure 1.9.

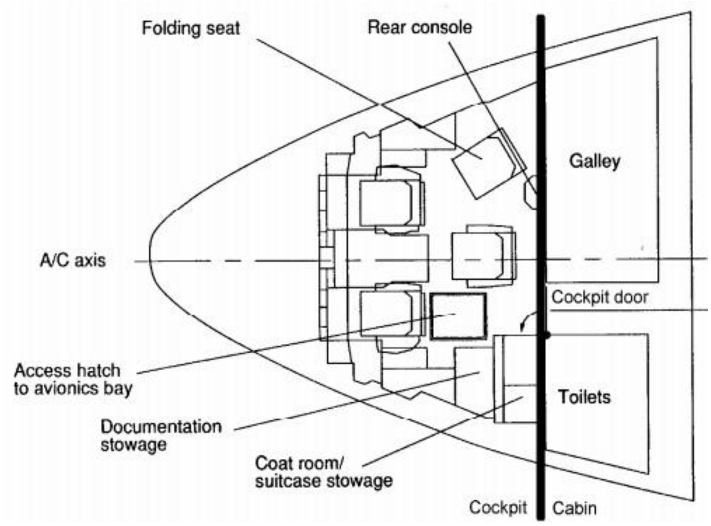


Figure 1.8 - The location of lavatory and galley

Toilets design similar to the prototype.

About the galley, because my designing plane has a block time with maximum payload of 9 hours, it's necessary to provide a galley for crews to provide water, tea and food.

Area of galley:

$$s_{galley} = 3(m^2).$$

1.3.4. Layout and calculation of basic parameters of tail unit

The tail unit of a conventional aircraft has two components of horizontal tail and vertical tail and has two main functions:

- Trim during pitch and yaw;
- stability during pitch and yaw.

The designed plane has conventional tail unit with a horizontal tail sweep angle of 35° and vertical tail sweep angle of 45°.

For static moment coefficient: horizontal A_{HTU} =0.8, vertical A_{VTU}=0.1.

Determination of the tail unit geometrical parameters:

The arm for horizontal and vertical tail from center of gravity is calculated by:

$$L_{VTU} \approx L_{HTU} = (3.2...3.3) \cdot b_{MAC} \approx 30(m) .$$

Area of vertical tail unit is equal:

$$S_{VTU} = \frac{l_w \times S_w}{L_{VTU}} \times A_{VTU} = \frac{58 \times 361.6}{30} \times 0.1 = 69.9(m^2) .$$

This value meets recommendation: $S_{VTU} = (0.12...0.2) \times S_w = 43...72\mathrm{m}^2$.

Area of horizontal tail unit is equal:

$$S_{HTU} = \frac{b_{MAC} \times S_w}{L_{HTU}} \times A_{HTU} = \frac{7 \times 361.6}{30} \times 0.8 = 67.5(m^2) .$$

This value meets recommendation: $S_{HTU} = (0.18...0.25) \times S_w = 65...90\mathrm{m}^2$.

Determination of the elevator area and rudder area:

Elevator area:

$$S_{el} = 0.35 \times S_{HTU} = 0.35 \times 67.5 = 23.6(m^2) .$$

Rudder area:

$$S_{ru} = 0.4 \times S_{VTU} = 0.4 \times 69.9 = 28(m^2) .$$

Determination of the balance area: $M = 0.82 \geq 0.75$.

Then $S_{abea} \approx S_{abed} = (0.18..0.2) S_{e}$.

Elevator balance area is equal:

$$S_{ed} = 0.26 \times S_{HTU} = 0.26 \times 67.5 = 17.55(m^2).$$

Rudder balance area is equal:

$$S_{rb} = 0.22 \times S_{VTU} = 0.22 \times 69.9 = 15.38(m^2).$$

The area of altitude elevator trim tab:

$$S_{te} = 0.1 \times S_{el} = 0.1 \times 23.6 = 2.36(m^2).$$

The area of rudder trim tab:

$$S_{tr} = 0.05 \times S_{ru} = 0.05 \times 28 = 1.4(m^2).$$

Tail unit span calculation:

Span of the horizontal tail unit is:

$$L_{HTU} = 0.4 \times l_w = 0.4 \times 58 = 23.2(m).$$

Height of the vertical tail unit is:

For low wing, M<1: h_{VTU}=(0.14...0.2)L_W, where L_W is the wing span.

$$h_{VTU} = 0.16 \times l_w = 0.16 \times 58 = 9.28(m).$$

Tail unit taper ratio calculation:

For aircraft M<1: η_{HTU}=2...3; η_{VTU}=1...3.3.

Compared with my prototype, I selected:

Taper ratio of horizontal tail unit is:

$$\eta_{HTU} = 2.9 .$$

Taper ratio of vertical tail unit is:

$$\eta_{VTU} = 3.1 .$$

Determination of horizontal and vertical tail unit chords:

Root chord of horizontal tail is:

$$b_{rootHTU} = \frac{2 \times S_{HTU} \times \eta_{HTU}}{(1+\eta_{HTU}) \times L_{HTU}} = \frac{2 \times 67.5 \times 2.9}{(1+2.9) \times 23.2} = 4.33(m) .$$

Tip chord of horizontal tail is:

$$b_{tipHTU} = \frac{b_{rootHTU}}{\eta_{HTU}} = \frac{4.33}{2.9} = 1.49(m) .$$

Root chord of vertical tail is:

$$b_{rootVTU} = \frac{2 \times S_{VTU} \times \eta_{VTU}}{(1+\eta_{VTU}) \times h_{VTU}} = \frac{2 \times 69.9 \times 3.1}{4.1 \times 9.28} = 11.39(m) .$$

Tip chord of vertical tail is:

$$b_{tipVTU} = \frac{b_{rootVTU}}{\eta_{VTU}} = \frac{11.39}{2.77} = 3.67(m) .$$

Horizontal tail unit mean aerodynamic chord:

$$b_{MACHTU} = 0.66 \cdot \frac{\eta_{HTU}^2 + \eta_{HTU} + 1}{\eta_{HTU} + 1} \cdot b_{tipHTU} = 3.1(m).$$

Vertical tail unit mean aerodynamic chord:

$$b_{MACVTU} = 0.66 \cdot \frac{\eta_{VTU}^2 + \eta_{VTU} + 1}{\eta_{VTU} + 1} \cdot b_{tipVTU} = 8.1(m).$$

Relative thickness (t/c) of stabilizer is equal:

$$\overline{C}_{TU} = 0.8 \times \overline{C}_w = 0.8 \times 0.118 = 0.0944.$$

1.3.5. Landing gear design

The main functions of LG are to absorb and dissipate landing shock energy during landing, keep the plane stable on the ground and allow the plane to freely move and maneuver during taxing. For my designing plane, it is tricycle type LG. It has two wheels in nose LG, and four wheels in each leg of main LG.

Main wheel axel offset is:

$$B_m = 0.2 \times b_{MAC} = 0.2 \times 7 = 1.4(m).$$

LG wheel base comes from the expression:

$$B = 0.3 \times l_f = 0.377 \times 58.82 = 22.2(m).$$

Front wheel axial offset will be equal:

$$B_n = B - B_m = 22.2 - 1.4 = 20.8(m).$$

Wheel track is:

$$T = 0.7 \times B = 11.34\,(m)\,.$$

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. For the low wing CG is placed relatively lower than fuselage horizontal, on the distance:

$$H = 0.19 \times d_f = 0.19 \times 5.64 = 1.07(m)\,.$$

To choose the size of the LG tire, we must calculate the load on nose LG and main LG.

The load on the wheel is determined:

K_g = 1.5..2.0 – dynamics coefficient.

The load on main wheel is:

$$P_{MLG} = \frac{m_0 \times g \times (B - B_m)}{B \times n \times z} = \frac{284317 \times 9.81 \times 20.8}{22.2 \times 2 \times 4} = 326657.18(N)\,.$$

The load on nose wheel is:

$$P_{NLG} = \frac{m_0 \times g \times k_g \times B_m}{B \times z} = \frac{284317 \times 9.81 \times 1.7 \times 1.4}{22.2 \times 4} = 74754.24(\mathrm{N})\,.$$

Compared the tire type and the maximum load of the tire of the prototype, i choose the tire size of 1400x530R23 for main LG and 1050x395R16 for nose LG. The parameters of the tire is in the table 1.4.

Table 1.4 - The parameters of the tires

Main gear			Nose gear		
Tire size	Ply rating	Rated load(N)	Tire size	Ply rating	Rated load(N)
1400x530R23	40	333513.7	1050x395R16	28	152183.7

1.3.6. Power plant

My designed plane has two engines in total, which are installed under the wing. According to the initial data, the bypass ratio is 5.5 and pressure ratio is 32.8. And compared the engine of prototype, I select The Pratt & Whitney PW4000 and Rolls-Royce Trent 700 for my designing aircraft.

The parameter of the engine is in the table 1.5.

Table 1.5 - The parameters of the engines

Model	Trust (KN)	Bypass ratio	Pressure ratio	Weight (Kg)
PW4000	287–311	4.9	32-34.1	5851
Trent 700	300.3-316.3	5	36	6160

1.4. Determination of the aircraft centre of gravity position

The CG calculation of the aircraft is the most important process of designing a new aircraft, because the CG position has a deep influence on the aircraft longitudinal stability and controllability.

However the center of gravity position will change during the operation of aircraft due to the fuel consume, loading and unloading cargo and even the crew weight. So, we can just control the center gravity in a controllable range, while not a fixed point. When the center of gravity position is out of this range, serious aircraft control problems will appear.

The CG position affects the longitudinal stability of the aircraft, if the center gravity moves forward, its stability will increase. In contrast, its stability decrease. And the maximum front allowable center of gravity position is determined by the efficiency of the elevator. The stronger of the elevator, the larger the range of the front center of gravity position is accepted.

The main requirements for aircraft layout are as follows:

- The layout must meet the requirements of aircraft operations;
- the layout of the aircraft must make all unit successfully realize its function;
- the layout of the aircraft must make it is easily to install, repair, maintenance and remove.

The operational and technical requirements for the aircraft layout are reflected in their importance to the design of the aircraft. First, we have to meet the most important requirements and meet other requirements as much as possible. In addition, the aircraft layout uses the principle: single structure element performs multiple functions. The application of this principle provides not only decrease the weight of the aircraft, but also save space.

The center of gravity (CG) is calculated according to the procedure:

- Determine the weight and arms of all objects in the aircraft;
- multiply the weight of the arm by all the masses to calculate the torque;
- add the moments of all masses together;
- divide the total torque by the total mass of the aircraft to get the total arm.

1.4.1. Determination of centering of the equipped wing

The quality of the equipped wing includes the quality of wing structures, the quality of nose LG and main LG, the quality of equipment installed at the wing and the quality of the fuel. The quality register includes the name of the object, the quality itself and its center of gravity coordinates. The origin of coordinates of those quality center is the projection of leading edge of the mean aerodynamic chord onto the axis of the fuselage, and if the coordinate of mess center is in front of the origin, the coordinate will be negative.

The list of the mass objects for the wing, where the engines are located under the wing, included the names given in the table 1.6. The MTOW of aircraft is 284317 kg.

The CG coordinates of the equipped wing is determined by formulas:

$$x_w = \frac{\sum m_i x_i}{\sum m_i} = 1.76(m).$$

Table 1.6 - Trim sheet of equipped wing

N	object name	Mass		C.G. coordinate X_i ,(m)	Mass moment $X_i m_i$,(kg.m)
		units	total mass m_i,(kg)		
1	wing (structure)	0.12185	34644.03	3.01	104278.52
2	fuel system	0.0096	2729.44	3.01	8215.62
3	flight control system, 30%	0.00096	272.94	4.2	1146.37
4	electrical equipment, 30%	0.00372	1057.66	0.7	740.36
5	anti-ice system , 70%	0.00742	2109.63	0.9	1898.67
6	hydraulic systems , 70%	0.007	1990.22	4.2	8358.92
7	power plant	0.07326	20829.06	-5.26	-109560.87
8	equipped wing without LG and fuel	0.22381	63632.99	0.24	15077.59
9	Nose LG	0.005682	1615.49	-18	-29078.81
10	Main LG	0.032198	9154.44	4.2	38448.64
11	fuel	0.29797	84717.94	3.01	255000.99
	total	0.55966	159120.85	1.76	279448.41

1.4.2. Determination of centering of the equipped fuselage

The origin of the coordinate of mass center of fuselage is the nose point of fuselage, so the coordinate of mass center of fuselage will be all positive. The list of the objects for the fuselage is given in table 1.7.

The CG coordinates of the equipped fuselage is determined by formulas:

$$x_w = \frac{\sum m_i x_i}{\sum m_i} = 28.35(m).$$

Table 1.7 - Trim sheet of equipped fuselage

N	objects names	Mass		CG coordinates X_i, (m)	mass moment m_iX_i
		units	total mass		
1.	fuselage	0.08645	24579.20	33.56	824878.11
2.	horizontal tail	0.01168	3320.82	55.62	184704.15
3.	vertical tail	0.01206	3428.86	55.62	190713.36
4.	radar	0.004	1137.27	2.21	2513.36
5.	radio equipment	0.003	852.95	2.21	1885.02
6.	instrument panel	0.0071	2018.65	3.01	6076.14
7.	aero navigation equipment	0.0061	1734.33	1.05	1821.05
8.	operational Items	0.00581	1651.88	50.08	82726.24
9.	lavatory1, galley 1	0.0398	11315.82	6.84	77400.19
10.	aircraft control system 70%	0.00224	636.87	30.41	19367.22
11.	hydro-pneumatic sys 30%	0.003	852.96	41.17	35119.40

Continue to table 1.7

12.	electrical equipment 70%	0.00868	2467.87	30.41	75047.97
13.	not typical equipment	0.0011	312.75	49.56	15499.83
14.	furnishing and thermal equipment	0.0043	1222.56	33.56	41029.22
15.	cargo cabin equipment	0.0002	56.86	33.56	1908.34
16.	anti ice and air conditioning system 30%	0.00318	904.13	30.41	27494.53
19.	seats of pilot	0.00011	31.27	2.21	69.12
20.	additional equipment	0.00107	304.22	49.56	15077.1
21.	equipped fuselage without payload	0.19988	56829.28	28.21	1603330.35
22.	on board meal	0.00018	51.18	4.46	228.25
24.	cargo, mail	0.2404	68349.81	28.56	1952070.48
25.	crew	0.0007	199.02	2.21	439.84
	total	0.44116	125429.29	28.35	3556068.92

After we determined the CG of fully equipped wing and fuselage, we establish the moment balance equation for all mass with the nose of the fuselage as origin：

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

where X_{MAC} is the wing MAC leading edge position relative to fuselage, it is calculated by the formula:

$$X_{MAC} = \frac{m_f x_f + m_w x_w - m_0 C}{m_0 - m_w} = \frac{3556068.919 + 279448.4126 - 284317 \cdot 0.32 \cdot 7}{284317 - 159120.8522} = 25.55(\text{m}) .$$

where C is distance from MAC leading edge to the CG point. For the designed plane, it has swept wing of 31°, so C equal to $0.32*B_{MAC}=2.24$ (m); m_0 – aircraft takeoff mass, kg; m_f – mass of equipped fuselage, kg; m_w – mass of equipped wing, kg.

1.4.3. Calculation of center of gravity positioning variants

The table 1.8 shows the list of mass objects for center of gravity variants calculation and table 1.9 shows center of gravity calculation option, both tables are calculated base of previous tables.

Table 1.8 - Calculation of CG positioning variants

Name object	Mass m_i (kg)	Coordinate X_i (m)	Mass moment (kg*m)
equipped wing (without fuel and LG)	63632.99	25.79	1640842.32
Nose LG (extended)	1615.49	6.67	10775.31
main LG (extended)	9154.44	28.85	264105.56
total fuel	84717.94	27.16	2300861.79
equipped fuselage (without payload)	56829.28	28.21	1603330.35
on board meal	51.18	4.46	228.25
cargo	68349.81	28.56	1952070.48
crew	199.02	2.81	559.25
nose LG (retracted)	1615.49	5.67	9159.82
main LG (retracted)	9154.44	28.85	264105.56
reserve fuel	3425.15	28.67	98198.94

Table 1.9 - Airplanes CG position variants

№	Variants of the loading	Mass, m_i (kg)	Moment of the mass, m_i*X_i (кg*m)	Centre of the mass, X_i (m)	Centering
1	Take-off mass (LG opened)	284550.14	7772773.32	27.32	0.25
2	Take-off mass (LG retracted)	284550.14	7771157.83	27.31	0.25
3	Landing variant (LG opened)	203257.35	5570110.46	27.4	0.27
4	Transportation variant (without payload)	216149.16	5818859.09	26.92	0.2
5	Parking variant (without fuel and payload)	131232.2	3519053.54	26.82	0.18

Conclusion to the project part:

As a result of the presented work, I have learned the general steps for designing an aircraft. The aim of the work has been achieved by the performing following tasks:

- Preliminary design of the middle range cargo aircraft with a payload of 70 tons;
- wing geometry of my designing aircraft;
- the fuselage layout of my designing aircraft ;
- geometry and location of the furnishing compartment of my designing aircraft;
- the layout and calculation of basic parameters of tail unit;
- the geometrical calculation of the LG and the chose of the tire;
- the chose of the power unit;
- the center of gravity calculations of the designed aircraft.

Proposed aircraft meets requirements to the planes of the transport category and have contemporary level of the technical characteristics.

PART 2

CARGO LATCH DESIGN

2.1. Analysis of existing cargo latches

A typical load loading system consists of the necessary devices to conveyance, guide and restrain the ULD. The conveyance components included the ball mat, ball transfer unit, roller tracks and power drive unit, which allow the container and pallet to move and turn direction with low friction.

The guide components consist of pivot guide and side guide. The main function of the guides is to help to align the container or pallets during loading and unloading.

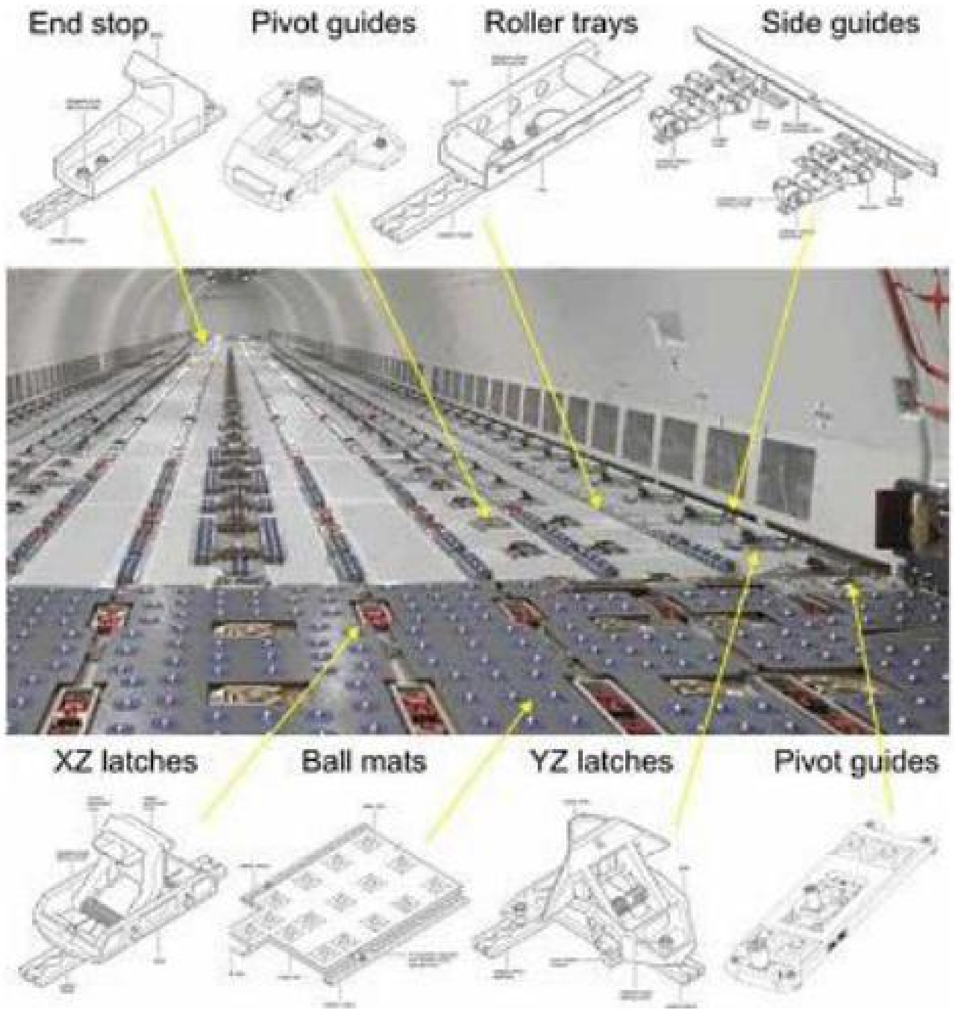


Figure 2.1 – Components of cargo loading system

As for the lock system of ULD, it consists of XZ latch, YZ latch and end stops. These devices are installed in aircraft cargo compartment floor to restrain aircraft ULD and withstand against the cargo loads. Before we describe the latch, it is necessary to know there are 3 axes for aircraft:

- Longitudinal or X axis, which is the same as the fuselage symmetry axis.

- lateral or Y axis, which is the perpendicular line of the X axis, and from left wing to the right wing.

- vertical or Z axis, which is line from the upward to the downward.

The YZ - latches are used to fix the ULD in Y- and Z- direction. If the YZ latch is installed at the door sill, it is used to fix the ULD in the door area. It is manually operated, we can press the red handle on the side of the device to make it fall, and pull up with hand to make it rise. Override-able YZ latch is installed at the fuselage floor center line, which provide to lock half-size containers in Y and Z direction.

End stops are installed at the end of the forward and after cargo cabin, which are used to fix the ULD in X and Z direction.

The XZ - latches are used to fix the ULD in X and Z direction. They are installed at the roller tracks. They can be retracted under the cargo cabin floor to allow the loading and unloading, and be manually extended to fix the ULD. They can be also moved along the track and then the latch can be fixed by shear pins.

The aim of my special part design is to design a XZ latch.

The methods of design are analysis of existing CL and selections of the most advanced technical decisions, determination of basic geometrical parameters, calculation of the component strength.

The following are the analysis of some existing cargo latches.

The first type of XZ latch is retractable XZ latch, as shown in the Figure 2.2. It is spring loaded and manually operated. It is installed between the ULD devices, used to fix the ULD in X and Z direction. Each latch composed of an aluminium base, inner and outer locking pawls.

After the ULD is in place, the latch is locked by manually pulling the outer pawl upwards, and then the the inner pawl extends and locks automatically. To unlock the XZ latch, the ULD which is fixed by inner pawl must unloading at first. And then press the inner pawl by hand or foot downwards, the outer pawl will unlock automatically. When it is not necessary to use the latch, it can be manually lowered to the cargo floor. The Figure 2.3 shows the three positions of the inner and outer pawl.

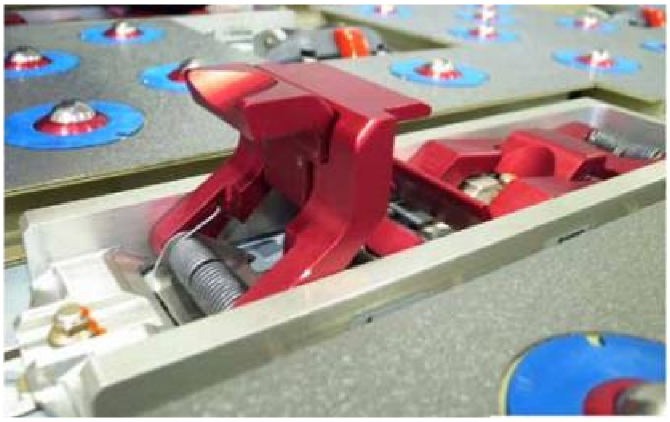


Figure 2.2 - Retractable XZ latch

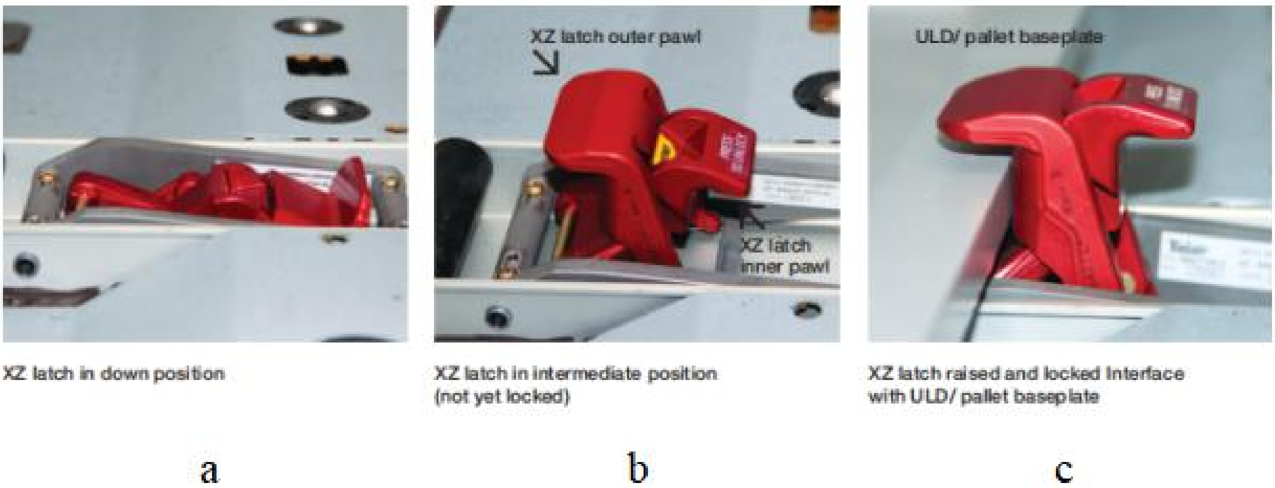


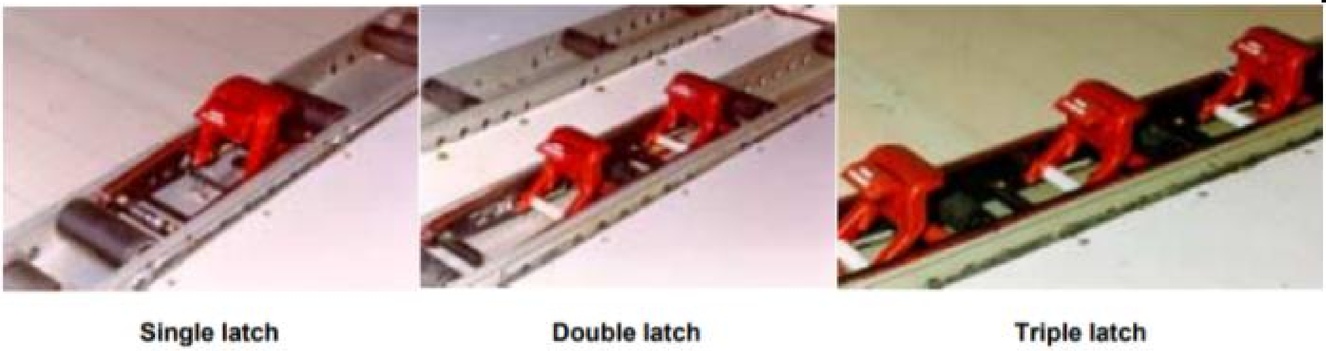
Figure 2.3 - Three positions of the inner and outer pawl: a - XZ latch in down position; b - XZ latch in intermediate position; c - XZ latch is locked interface with ULD baseplate.

The latch can move along the track, and than be fixed by tension studs and the shear studs. To meet the situation of different cargo configurations, single, double or tripe latch may be selected to use. The Figure 2.4 shows the single, double and triple latch.

The advantages of this kind of latch are:

- Simple structure. The latch is composed of three main parts: aluminum base, inner and outer pawl.

- simple operation. Pulling up the outer pawl to lock the latch, and pressing down the inner pawl to unlock the latch.



a b c

Figure 2.4 - Single, double and triple latch: a - single latch; b - double latch; c - triple latch

The second type of the latch is rotatable XZ latch. The figure 5 shows the construction of the rotatable latch [14] .

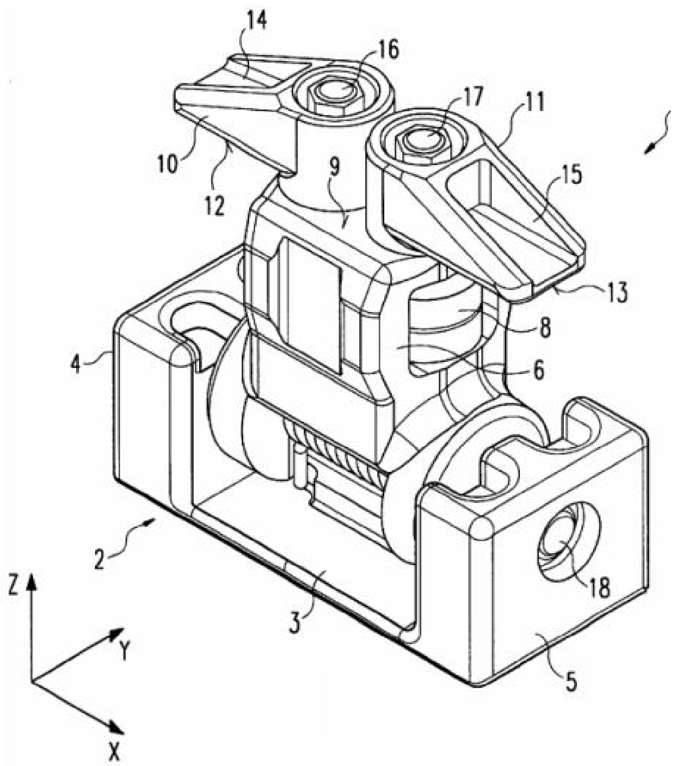


Figure 2.5 - Rotatable latch [14]

The rotatable latch comprises of a U-shaped frame, a base, a main shaft, three springs, two deflecting rollers, two sub shafts and two restraining hooks. The lower part of the base crosses the main shaft, which allow the base to rotate around the main shaft. The main shaft is surrounding by a spring. The spring is connected to the lower part of the base and the frame, which is used to prevent the base from deviating from the vertical operating position.

Two sub shaft are rotatably installed in the base. Two restraining hook are installed at the upper part of two sub shaft, which can rotate around the sub shaft and then be fixed by nut to prevent the hook separate from the sub shaft. Two deflecting rollers are installed on two sub shaft, which is used to provide a guide for ULD in lateral direction.

When the ULD is pushed into the final storage position, the latch can be locked by rotating the base into vertical operating position and rotating the restraining hook into operating position. On the contrary, to unloading the cargo, we need to rotate the restraining hook into unlocking position and then rotate the latch to under the floor.

The Fig. 2.6 shows three positions of the restraining hook. The 11a in the picture is the operating position of the restraining hook, while 11b shows the position of the restraining hook rotated 45° from the operating position, and 11c shows the restraining hook rotated 90° from the operating position [14].

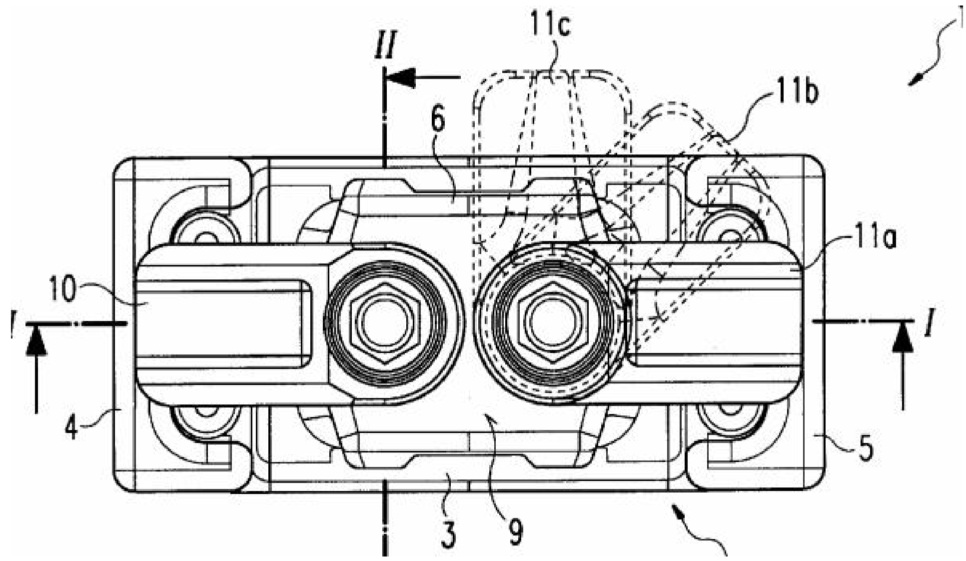


Figure 6 - Three positions of the restraining hook [14].

The advantages of the rotatable latch are:

- Easily operational: Only a few manual operations are needed when we need to lock and unlock the latch.

- easily maintenance.

- light weight and of small size.

- high security and reliability. The complex constructions of the latch provide the high security and reliability.

2.2. Requirements to a new design

The first requirement for new CL design is that it should be able to fix the standard containers and pallets which are used on my designing aircraft.

For my designed aircraft, it has a maximum load capacity of 70 tons. It is designed to accommodate the following standard containers and pallets: LD3, AAY, AAD, AMJ, AMV, AYY, SAA, AMA, 96"x125" pallets and 88"x125" pallets. The table 2.1 show the parameters of the mentioned ULD.

Table 2.1 - The parameters of the mentioned ULD

Type	Volume (m^3)	Tare weight(kg)	Dimensions (WxLxH)/in	Load capacity (kg)
LD3	4.3	91	61.5x60.4x64	1587
AAY	11.6	225	88x125x82	6033
AAD	14.6	288	88x125x96	5772
AMJ	16.6	253	96x125x96	6804
AMV	15.1	246	96x125x96	6804
AYY	5.7	135	62x88x79	3016
SAA	12.6	263	88x125x110	6030
AMA	17.5	323	96x125x96	6800
96"x125" pallets	11.52	130	96x125x96	6804
88"x125" pallets	10.53	120	88x125x96	6804

The second requirement is the selection of the optimum materials for the CL manufacturing. I have selected aluminum alloy for CL manufacturing. It has the following advantages:

- Low density and light weight;

- high corrosion resistance;
- good plasticity and processability;
- high strength.

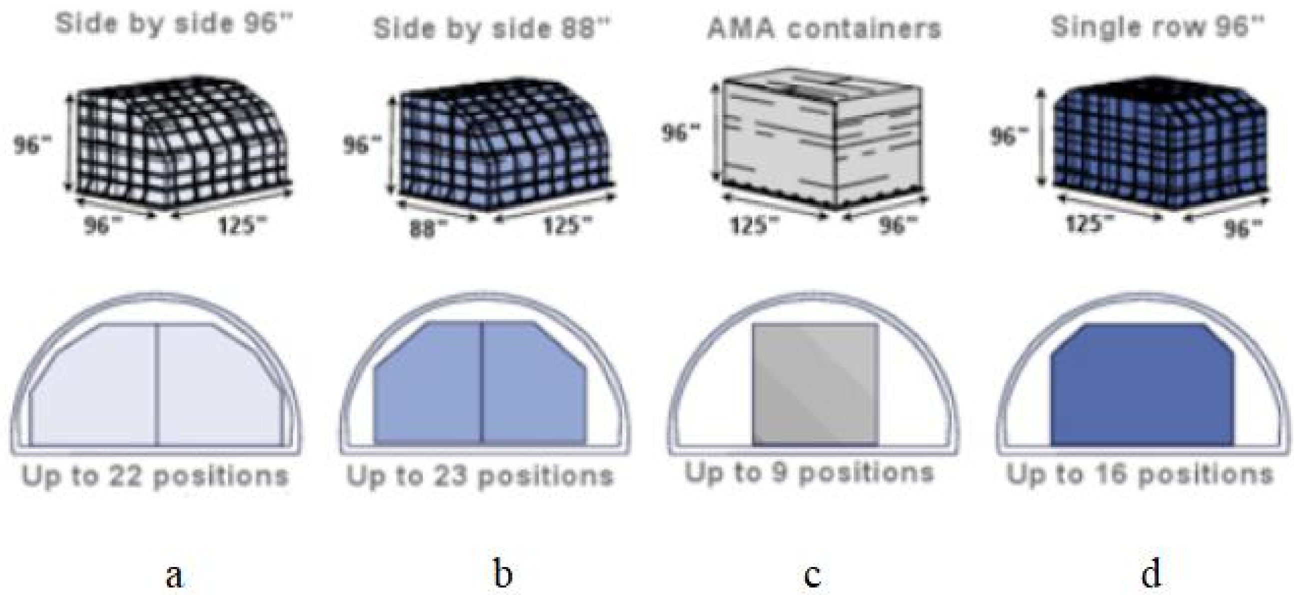


Figure 2.7 - Some mentioned ULD and four kinds of cargo configuration on the main deck: a - 22 side by side 96"x125" pallets; b - 23 side by side 88"x125" pallets; c - 9 AMA containers; d - 16 single row 125"x96"x pallets.

I have selected aluminum alloys which can be produced and used in China. The following table 2.2 shows the materials produced in China.

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

Type	Shear strength(MPa)	Yield strength(MPa)	Tensile strength(MPa)
7Y69	/	874	950
7055	/	530	593
7075	331	503	572
2024-T3	283	280	400

The finally requirement for the new design is that the latch latch should be able to withstand loads on the ground and on the air. Especially when the aircraft is taking off and lading, the latches have to withstand the impact due to the acceleration and deceleration of the aircraft. To make sure the CL can be able to fix all usable containers and pallets, we need to calculate strength of the lock under ultimate load. It means that if the latch can withstand the heaviest ULD, then the CL is able to withstand all usable ULD.

As for my designed aircraft, the heaviest ULD it can accommodate is 96"x125" pallet. 96"x125" pallet has a tare weight of 320 kg and a maximum gross weight of 6804 kg.

2.3. Preliminary design of the cargo latch

This new design is designed to hold the ULD in X and Z direction. It is simple, reliable, retractable and easily to install.

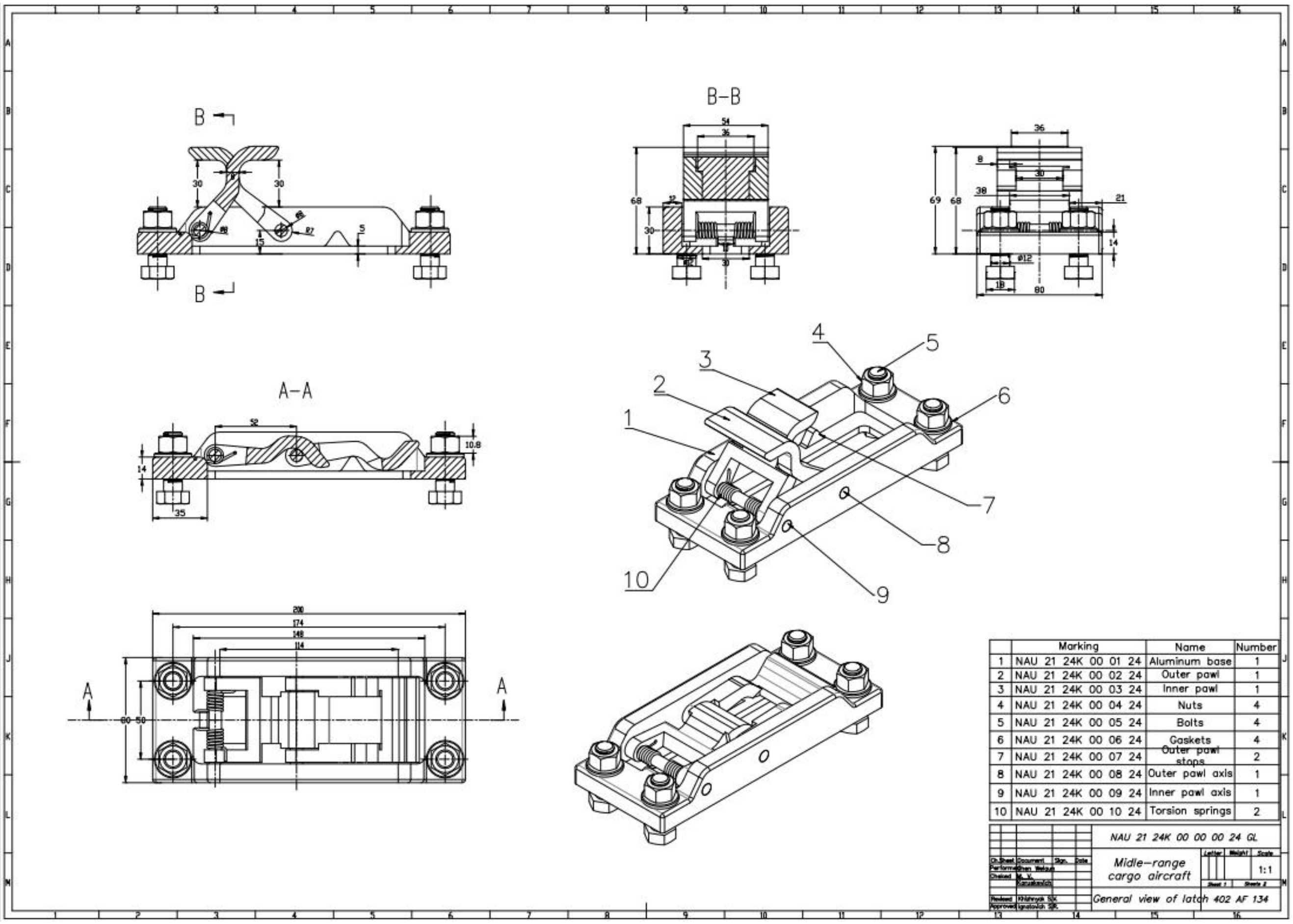


Figure 2.8 - Preliminary sketch of the CL

The main parts of the new design are: one aluminum base, one outer pawl, one inner pawl, two outer pawl stops, one outer pawl axis, one inner pawl axis, two torsion spring, four gaskets, four bolts and nuts. The figure 2.8 shows the preliminary sketch of the CL.

The CL can slide along the track, and then to be fixed by four bolts and nuts at both ends of the aluminum base. When it is not needed, it can be retracted under the cargo floor.

Working principle: when one ULD is in place, we can manually pull up the outer pawl. At the same time, the inner pawl begins to erect automatically due to the torsion spring. Once the inner face of the outer pawl contacts with the ULD base plate, the inner pawl erect to the limit position. Locking surfaces on the outer pawl and inner pawl contact with each other, which keeps the outer pawl fix. Then we just loading the next ULD, and make sure the its base plate is contact with the inner face of the inner pawl. Finally the ULD are locked.

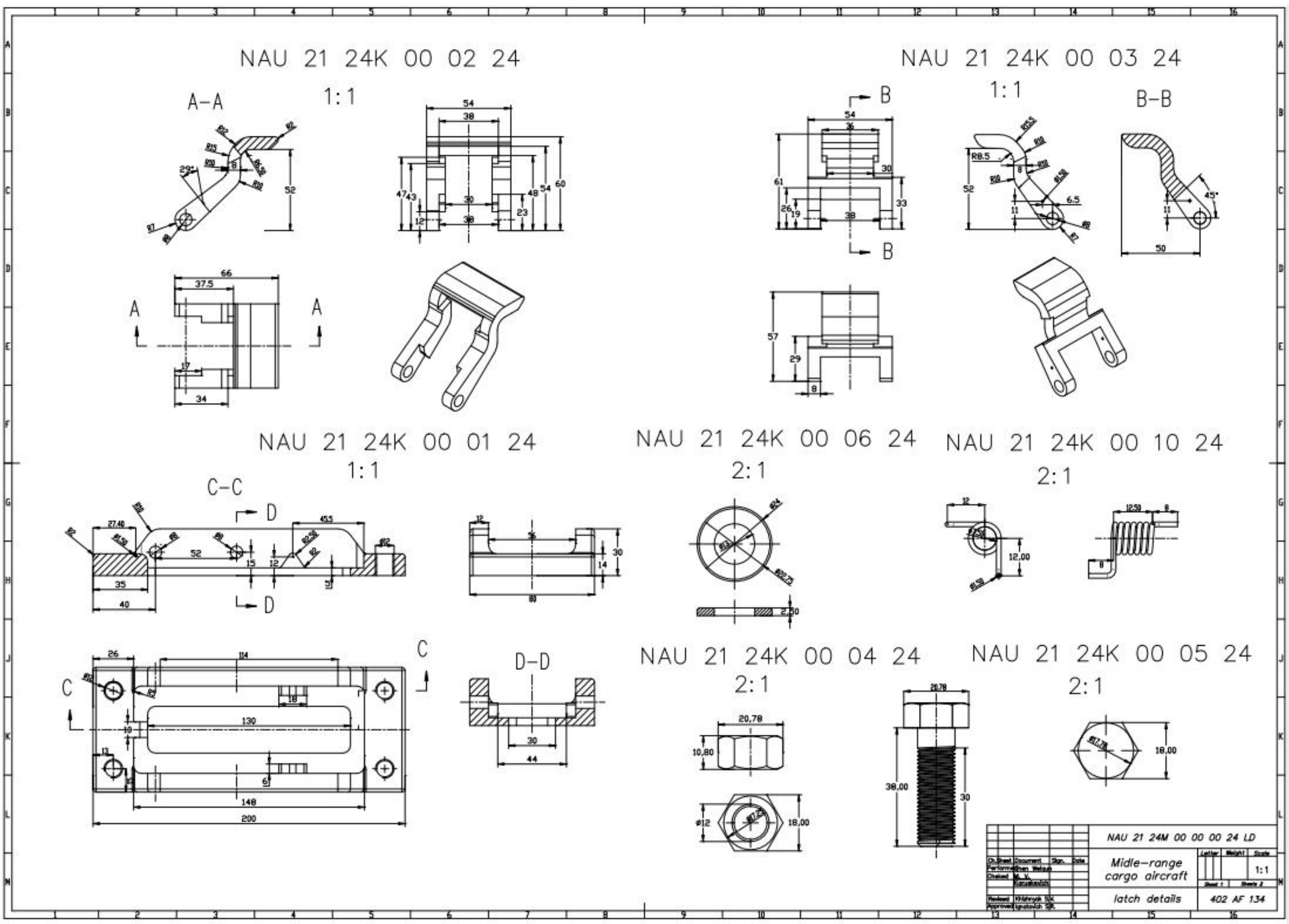


Figure 2.9 - Detail sketch of the cargo latch

Once the ULD on both sides of it is in the place, two ULD are locked together. As long as the outer ULD is not unloaded, the locking surface and two pawl axis will form a triangle. As we all know that triangles are stable. When inner ULD presses the latch in X direction. The force applies to the outer pawl is transferred to the inner pawl, which drives inner pawl rotates down. But the outer ULD base plate prevents this rotation. When inner ULD presses the latch in Z direction, the force in X direction applies to the outer pawl is transferred to the inner pawl through the lacking surface. Also the outer ULD will prevent this rotation.

When the outer ULD presses the latch in X direction, the force applies to the outer pawl to drive outer pawl rotate down. But the limiting surface on inner pawl and base plate will prevent the rotation of the outer pawl.

When the outer ULD press the latch in Z direction. The force applies to the inner pawl to drive the inner pawl rotate. The limiting surface on outer pawl prevent the rotation of the inner pawl.

When we need to unlock the latch, we should unload the outer ULD at first, and press down the inner pawl by hand or foot. Then the outer pawl is unlocked and can be rotated down to under the cargo floor. The outer pawl stops are used to Stop it from over-rotating. The Fig. 2.9 shows three positions of the latch pawls.

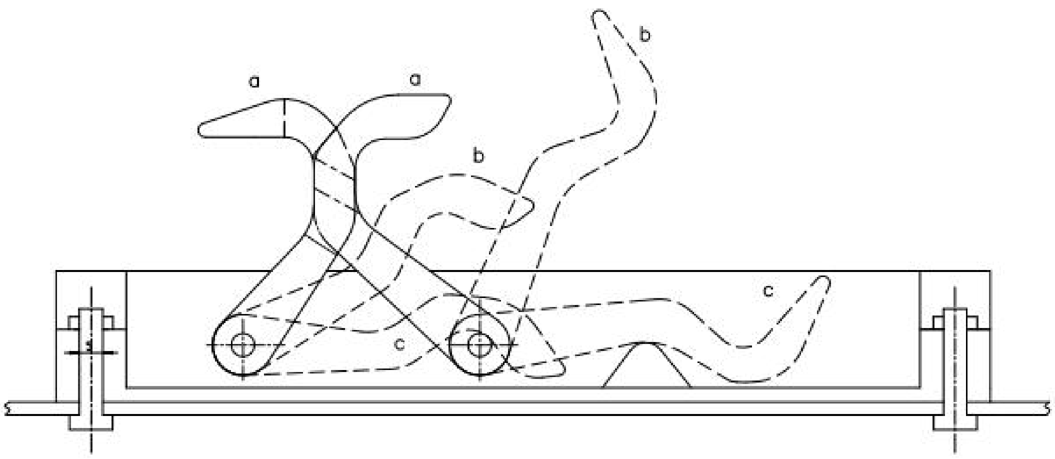


Figure 2.10 - Three positions of the CL pawls: a - designing latch is locked; b - designing latch in intermediate position; c - designing latch in down position interface with ULD base plate.

2.4. Calculation of components strength

To ensure long service life and durability of the latch it is necessary to know:

- Weight of containers and pallets;
- the values of load factor in all directions;
- to find critical components of the design.

Weight of containers with cargo is provided by the table 2.1.

The value of the load factors in all directions are:

- Along the X axis: +9;
- along the Y axis: -4 and +2;
- along the Z axis: -1.5 and +1.5.

The critical parts of the latch are attachment units. As for my designing CL, it is attached to the track with two bolts. The main force which exerts on the latch is shear force. The following table 2.3 shows the some materials of the bolts and its performances.

Table 2.3 - Materials and performances of the bolts

Standar for Bolt	Grade	Tension Strength(MPa)	Yield Strength (MPa)	Proof Load Stress(MPa)	Hardness
	4.6	400	240	225	B67.95
ISO 898-1 2009	4.8	420	340	310	B71.95
	5.6	500	300	280	B79.95
	5.8	520	420	380	B82.95
	6.8	600	480	440	B89
	8.8	830	660	600	C23-34
	10.9	1000	940	830	C32-39
	12.9	1080	1100	970	C39-44

As for my designing CL, I will select bolt grade 8.8 to manufacture the CL, and it is usually made of alloy steel. The alloy steel has the following advantages:

- High wear resistance, which provides a longer service life.
- high corrosion resistance. The surrounding of CL makes it easy to be corroded, so the corrosion resistance of the lock is very important.

- high strength. The CL used in my designing aircraft need to withstand big stress, so the material for CL must be high strength.

2.4.1. Strength calculation for cargo latch attachment units

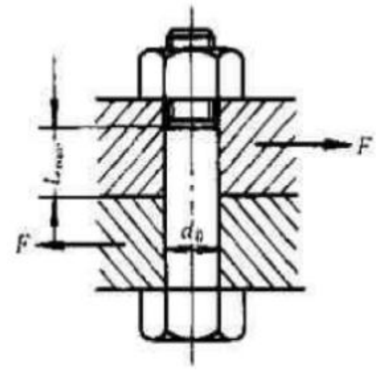


Figure 2.11 - Schematic diagram of bolt connection

Before calculating the strength of the CL, we need to find the maximum shear force F_{max} applied to one bolt. F_{max} is calculated by the formula:

$$F_{max} = \frac{m_{max}*g*n_X}{N} = \frac{(6804+320)*9.81*9}{4*4} = 39311(N).$$

where m_{max} is the weight of one heaviest ULD and cargo, as for my designed aircraft, the heaviest ULD is 96"x125" pallet, kg;

-g is gravitational acceleration, equal to 9.81;

-n_x is the load factor in X direction;

-N is the number of latches which secure one ULD and are subjected to shear forces. As for my designed latch, I need 8 latches to secure 96"x125" pallet in total, only 4 latches of them are subjected to shear forces. And each latch has 4 bolts.

To make ensure that the bolt has the enough shear strength, it must meet the following formula:

$$\tau = \frac{F}{i*\frac{\pi}{4}*d_0^2} \leq [\tau].$$

where τ is shear stress of the bolt, MPa;

F is shear force on one bolt, N;

i - is the number of the shear plane;

d_0 - is the diameter of the bolt, mm. For my selected bolt, the diameter of bolt is 12mm;

[τ] is allowable stress of the bolt, MPa. And it is calculated by following formula:

$$[\tau] = \frac{\sigma_s}{[S]_\tau} = \frac{640}{1.5} = 427(\text{MPa}).$$

where σ_s is the yield limit of the alloy steel, which is given on table 2.3;

$[S]_\tau$ is the safety factor, according to the CCAR requirement, it takes 1.5.

So

$$\tau = \frac{F}{i*\frac{\pi}{4}*d_0^2} = \frac{39311}{1*\frac{\pi}{4}*12^2} = 348(\text{MPa}).$$

$$\tau = 348(\text{MPa}) \leq 427(\text{MPa}).$$

So this means that the diameter of the bolt and the choice of material meet my requirements.

2.4.2 Strength calculation for cargo latch axis

The main function of the CL axis is to provide a rotation axis for latch pawl, it is mainly subjected to shear forces and.

Before calculating the strength of the CL, we need to find the maximum shear force F_{max} applied to one latch axis. F_{max} is calculated by the formula:

$$F_{max} = \frac{m_{max}*g*n_X}{N} = \frac{(6804+320)*9.81*9}{4*2} = 78622(\text{N}).$$

Figure 2.12 - Force analysis of the axis

To make ensure that the axis can withstand the maximum shear force, it must meet the following formula:

$$\tau = \frac{4F}{i*\pi*d_0^2} \leq [\tau].$$

where d_0 is the diameter of the axis, mm. For my selected bolt, the diameter of bolt is 8mm.

The shear stress is calculated by:

$$\tau = \frac{4F}{i*\pi*d_0^2} = \frac{4*62897.8}{2*\pi*8^2} = 782(\text{MPa}).$$

Allowable stress of the axis is calculated by following formula:

$$[\tau] = \frac{\sigma_s}{[S]_\tau} = \frac{870}{1.1} = 790(\text{MPa}).$$

where σ_s is the yield limit of the alloy steel, which is given on table 2.3

where $[S]_\tau$ is the safety factor.

So

$$\tau = 782\ (MPa) \leq 790\ (MPa).$$

It means that the diameter of the axis and the choice of material meet my requirements.

And the crushing force acting on the eyelet of the pawl is equal to shear force acting on the axis, then the it must meet the following strength condition :

$$\sigma_t = \frac{F}{A} \leq [\sigma_t].$$

The shear force is calculate:

$$F = \frac{m_{max}*g*n_X}{N} = \frac{(6804+320)*9.81*9}{4*4} = 39311(N).$$

so

$$\sigma_t = \frac{F}{A} = \frac{39311}{8*8} = 614(MPa).$$

where A is the contact area, mm.

And

$$[\sigma_t] = \frac{\sigma_t}{n} = \frac{870}{1.4} = 621(MPa).$$

Then

$$\sigma_t = 614(MPa) \leq [\sigma_t] = 621(MPa).$$

It means that the eyelet on pawl will not break under the maximum load.

2.4.3 Strength calculation for cargo latch pawl

The latch pawl is mainly subjected to compress, bending and shear force duo to the shock of the ULD base plate.

The calculation of tensile strength on outer pawl locking surface:

The area of the outer pawl locking surface A_{sl} is:

$$A_{sl} = L * W * n = 54 * 8 = 432(mm^2).$$

The force applied on the locking surface F is:

$$F_{max} = \frac{m_{max}*g*n_X}{N} = \frac{(6804+320)*9.81*9}{5*2} = 62897.8(N).$$

The compressive strength calculation is as follows:

$$\sigma_{bs} = \frac{F_{max}}{A_{sl}} \leq [\sigma_{bs}].$$

where $[\sigma_{bs}]$ is allowable compressive stress.

$$[\sigma_{bs}] = \frac{k_{CM}*\sigma_t}{k} = \frac{0.8*950}{3.5} = 217(MPa).$$

where k_{CM} is coefficient taking into account the properties of the material and stress concentration, k_{CM} = 0.8 - 0.9;

K is safety factor (k = 3 - 4).

Compressive stress is calculated by:

$$\sigma_{bs} = \frac{F_{max}}{A_{sl}} = \frac{62897.8}{432} = 145.6(MPa).$$

So

$$\sigma_{bs} = 145.6(MPa) < 217(MPa).$$

It means that the latch pawl can withstand the compressive stress.

2.4.3 Strength calculation for torsion spring

The torsion spring is installed at the inner pawl axis, which is used to keep the inner pawl erect. The aim of the strength calculation are to calculate the torque of the spring when inner pawl is rotated down.

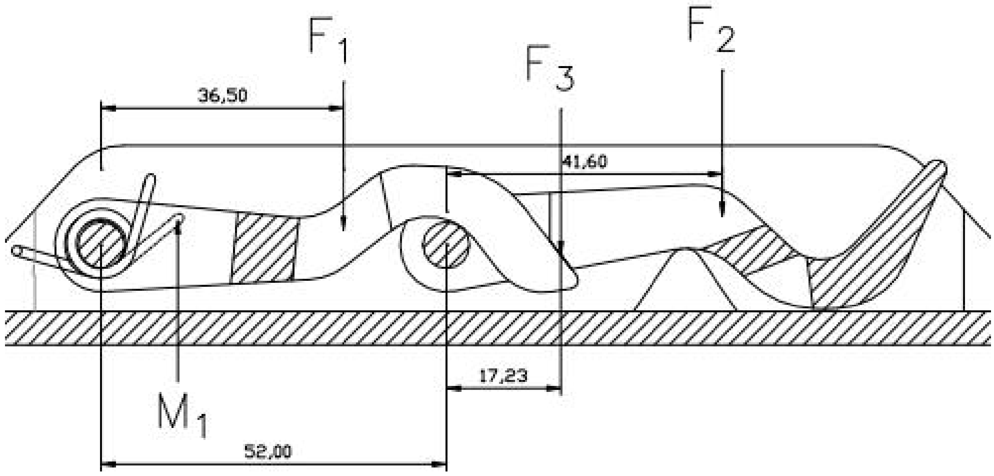


Figure 2.13 - Force analysis of the pawls

And to make sure the locking surface on outer pawl root can lock the inner pawl due to the outer pawl weight, it must meet the following requirement:

$$F_2 * r_2 > F_3 * r_3.$$

where F_2 is weight of the outer pawl, N;

F_3 - is the force inner pawl applies to the outer pawl at locking surface, N;

r_2 - is distance between center of gravity of outer pawl and the outer pawl axis, m;

r_3 - is distance between locking surface on outer pawl and the outer pawl axis, m;

And

$$M_1 = F_1 * r_1 + F_3 * R_3.$$

where M_1 is the torque moment of the torsion spring, N*m;

F_1 - is the weight of the inner pawl, N;

r_1 - is distance between center of gravity of inner pawl and the inner pawl axis, m;

R_3 - is distance between locking surface on outer pawl and the inner pawl axis, m;

The torque moment of spring is calculated by following formula:

$$M_1 = \frac{E*d^4*\varphi}{3670*D*N*L} * g * l = \frac{19400*1.5^4*30}{3670*9*6.25*12} * 9.81 * 0.012 = 0.12(N * m).$$

where E is the shear modulus of elasticity, for my selected spring, it takes 19400;

d - is diameter of the spring wire;

φ - is the angle of spring torsion, °;

D - is mid diameter of the spring, mm;

N - is the number of coils of the spring;

L - is the load arm of the spring, mm;

g - is gravitational acceleration.

The weight of inner pawl and outer pawl are calculated:

$$F_1 = \rho_1 * V_1 * g = 2.86 * 27.5 * 9.81 * 10^{-3} = 0.77(N).$$

$$F_2 = \rho_2 * V_2 * g = 2.86 * 32.1 * 9.81 * 10^{-3} = 0.9(N).$$

So

$$F_3 = \frac{M_1 - F_1 * r_1}{R_3} = \frac{0.12 - 0.77*0.037}{0.069} = 1.33(N).$$

And

$$F_2 * r_2 = 0.9 * 0.0416 = 0.037(N * m).$$

$$F_3 * r_3 = 1.33 * 0.017 = 0.023(N * m).$$

So

$$F_2 * r_2 = 0.037(n * m) > F_3 * r_3 = 0.023(N * m).$$

It means the torsion spring meet my requirement.

Conclusion to special part

As a result of the special part work, we have designed the CL which can fix the ULD in X and Z direction and applicable to all cargo configurations of my designing aircraft. The methods of special part design are: analyze the existing CL, selection the most advanced technical decisions, determination the basic geometrical parameters and calculations the component strength.

My designed CL has a maximum load capacity of 1.7 tons. The attachment units has a diameter of 12 mm and can withstand a maximum force of 39311 N. Aluminum alloys has been selected to manufacture the latch pawl, which can be produced in China. Alloy steel has been selected to manufacture the attachment unit.

The strength condition of latch components had been calculated, it meets the requirement of CS-25, FAR-25 and CCAR-25.

My designed latch is attached to the floor track by 4 M12 bolts and nuts. The latch is locked by the attachment of the locking surface on inner and outer pawl. Unless the outer ULD is unloaded, the latch is fixed.

GENERAL CONCLUSION

Based on the information obtained during the diploma work, the following goals were achieved:

- First we compared the parameters of three prototypes with desired parameters, and selected A330-200F as the prototype for new plane because it has the most similar parameters to our initial data.

- preliminary design of middle range cargo aircraft with cargo capacity 70 tons was performed.

- the main parameters of wing, landing gears, tail unit, fuselage, lavatory and galley were calculated.

- turbo fan engine PW 4000 and Trent 700 had been selected, which has a thrust about 300 KN.

- center of gravity position has been calculated, which has a range between 18% to 25%.

- conceptual Cargo latch design, the cargo latch with a load capacity of 1.7 tons had been designed. At first I analyzed the existing cargo latch and selected the most advanced technical decisions. Then selection of the dimensions and material of latch components was conducted. Then I found the maximum force applied at one bolt, calculated the shear stress on one bolt and compared the allowable stress of the selected material and compared the strength condition with the allowable stress of all latch components.

All calculated parameters of proposed aircraft meet the requirement of CS-25, FAR-25 and CCAR-25 and have contemporary level of the technical characteristics.

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Appendix A-initial data

INITIAL DATA AND SELECTED PARAMETERS

С А М О Л Е Т А С Т Р Д Д

НАУ, кафедра К Л А

ПРОЕКТ diploma Расчет выполнен 16.10.2020

Исполнитель Chen Weiqun Руководитель Karuskevich

ИСХОДНЫЕ ДАННЫЕ И ВЫБРАННЫЕ ПАРАМЕТРЫ

Количество пассажиров	0.
Количество членов экипажа	2.
Количество бортпроводников или сопровождающих	4.
Масса снаряжения и служебного груза	1651.22 кг.
Масса коммерческой нагрузки	68600.00 кг.
Крейсерская скорость полета	871. км/ч
Число "М" полета при крейсерской скорости	0.8186
Расчетная высота начала реализации полетов с крейсерской экономической скоростью	12.00 км
Дальность полета с максимальной коммерческой нагрузкой	7400. км.
Длина летной полосы аэродрома базирования	3.30 км.
Количество двигателей	2.
Оценка по статистике тяговооруженности в н/кг	2.7100
Степень повышения давления	40.00
Принятая степень двухконтурности двигателя	4.50
Оптимальная степень двухконтурности двигателя	4.50
Относительная масса топлива по статистике	0.2900
Удлинение крыла	9.26
Сужение крыла	4.00
Средняя относительная толщина крыла	0.110
Стреловидность крыла по 0.25 хорд	31.0 град.
Степень механизированности крыла	1.050
Относительная площадь прикорневых наплывов	0.010
Профиль крыла - Суперкритический	
Шайбы УИТКОМБА - установлены	
Спойлеры - установлены	
Диаметр фюзеляжа	5.64 м.
Удлинение фюзеляжа	10.50
Стреловидность горизонтального оперения	35.0 град.
Стреловидность вертикального оперения	45.0 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 подъемной силы в расчетной точке крейсерского режима полета Cy 0.48274

Значение коэффициента Cx.инд. 0.00895

ОПРЕДЕЛЕНИЕ КОЭФФИЦИЕНТА Dм = Мкрит - Мкрейс

Число Маха крейсерское	Мкрейс	0.81861
Число Маха волнового кризиса	Мкрит	0.82389
Вычисленное значение	Dм	0.00529

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

при взлете 5.223
в середине крейсерского участка 4.401
в начале крейсерского участка 5.025

Значение коэффициента сопротивления фюзеляжа и гондол 0.00617
Значение коэфф. профиль. сопротивления крыла и оперения 0.00896

Значение коэффициента сопротивления самолета:
в начале крейсерского режима 0.02606
в середине крейсерского режима 0.02479
Среднее значение Cy при условном полете по потолкам 0.48274

Среднее крейсерское качество самолета 19.47307

Значение коэффициента Cy.пос. 1.557
Значение коэффициента (при скорости сваливания) Cy.пос.макс. 2.335
Значение коэффициента (при скорости сваливания) Cy.взл.макс. 1.926
Значение коэффициента Cy.отр. 1.406
Тяговооруженность в начале крейсерского режима 0.473
Стартовая тяговооруженность по условиям крейс. режима Ro.кр. 2.223
Стартовая тяговооруж. по условиям безопасного взлета Ro.взл. 2.555

Расчетная тяговооруженность самолета Ro 2.683

Отношение Dr = Ro.кр / Ro.взл Dr 0.870

УДЕЛЬНЫЕ РАСХОДЫ ТОПЛИВА (в кг/кН*ч):

взлетный	36.1475
крейсерский (характеристика двигателя)	58.4044
средний крейсерский при заданной дальности полета	61.3152

ОТНОСИТЕЛЬНЫЕ МАССЫ ТОПЛИВА:

аэронавигационный запас	0.02834
расходуемая масса топлива	0.26963

ЗНАЧЕНИЯ ОТНОСИТЕЛЬНЫХ МАСС ОСНОВНЫХ ГРУПП:

крыла	0.12185
горизонтального оперения	0.01168
вертикального оперения	0.01206
шасси	0.03788
силовой установки	0.08286
фюзеляжа	0.08645
оборудования и управления	0.10097
дополнительного оснащения	0.00107
служебной нагрузки	0.00581
топлива при Lрасч.	0.29797
коммерческой нагрузки	0.24128

Взлетная масса самолета "M.o" = 284317. кГ.
Потребная взлетная тяга одного двигателя 381.45 kH

Относительная масса высотного оборудования и
противообледенительной системы самолета 0.0106

Относительная масса пассажирского оборудования (или оборудования кабин грузового самолета)	0.0002
Относительная масса декоративной обшивки и ТЗИ	0.0043
Относительная масса бытового (или грузового) оборудования	0.0398
Относительная масса управления	0.0032
Относительная масса гидросистем	0.0100

Относительная масса электрооборудования	0.0124
Относительная масса локационного оборудования	0.0040
Относительная масса навигационного оборудования	0.0061
Относительная масса радиосвязного оборудования	0.0030
Относительная масса приборного оборудования	0.0071
Относительная масса топливной системы (входит в массу "СУ")	0.0096
Дополнительное оснащение:	
Относительная масса контейнерного оборудования	0.0000
Относительная масса нетипичного оборудования	0.0011

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

ХАРАКТЕРИСТИКИ ВЗЛЕТНОЙ ДИСТАНЦИИ

Скорость отрыва самолета	277.41 км/ч
Ускорение при разбеге	1.99 м/с*с
Длина разбега самолета	1491. м.
Дистанция набора безопасной высоты	578. м.
Взлетная дистанция	2069. м.

ХАРАКТЕРИСТИКИ ВЗЛЕТНОЙ ДИСТАНЦИИ
ПРОДОЛЖЕННОГО ВЗЛЕТА

Скорость принятия решения	263.54 км/ч
Среднее ускорение при продолженном взлете на мокрой ВПП	0.14 м/с*с
Длина разбега при продолженном взлете на мокрой ВПП	3323.12 м.
Взлетная дистанция продолженного взлета	3901.50 м.
Потребная длина летной полосы по условиям прерванного взлета	4045.40 м.

ХАРАКТЕРИСТИКИ ПОСАДОЧНОЙ ДИСТАНЦИИ

Максимальная посадочная масса самолета	222170. кг.
Время снижения с высоты эшелона до высоты полета по кругу	22.2 мин.
Дистанция снижения	53.77 км.
Скорость захода на посадку	250.63 км/ч.
Средняя вертикальная скорость снижения	2.02 м/с
Дистанция воздушного участка	517. м.
Посадочная скорость	235.63 км/ч.
Длина пробега	763. м.
Посадочная дистанция	1280. м.
Потребная длина летной полосы (ВПП + КПБ) для основного аэродрома	2138. м.
Потребная длина летной полосы для запасного аэродрома	1818. м.

ПОКАЗАТЕЛИ ЭФФЕКТИВНОСТИ САМОЛЕТА

Отношение массы снаряженного самолета к массе коммерческой нагрузки	1.9047
Масса пустого снаряженного с-та приход. на 1 пассажира	0.00 кг/пас.
Относительная производительность по полной нагрузке	469.68 км/ч
Производительность с-та при макс. коммерч. нагрузке	57712.7 кг*км/ч
Средний часовой расход топлива	8715.281 кг/ч
Средний километровый расход топлива	10.36 кг/км

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Средний расход топлива на тоннокилометр              151.011 г/(т*км)
Средний расход топлива на пассажирокилометр         0.0000 г/(пас.*км)
Ориентировочная оценка приведен. затрат на тоннокилометр0.2085 $/(т*км)
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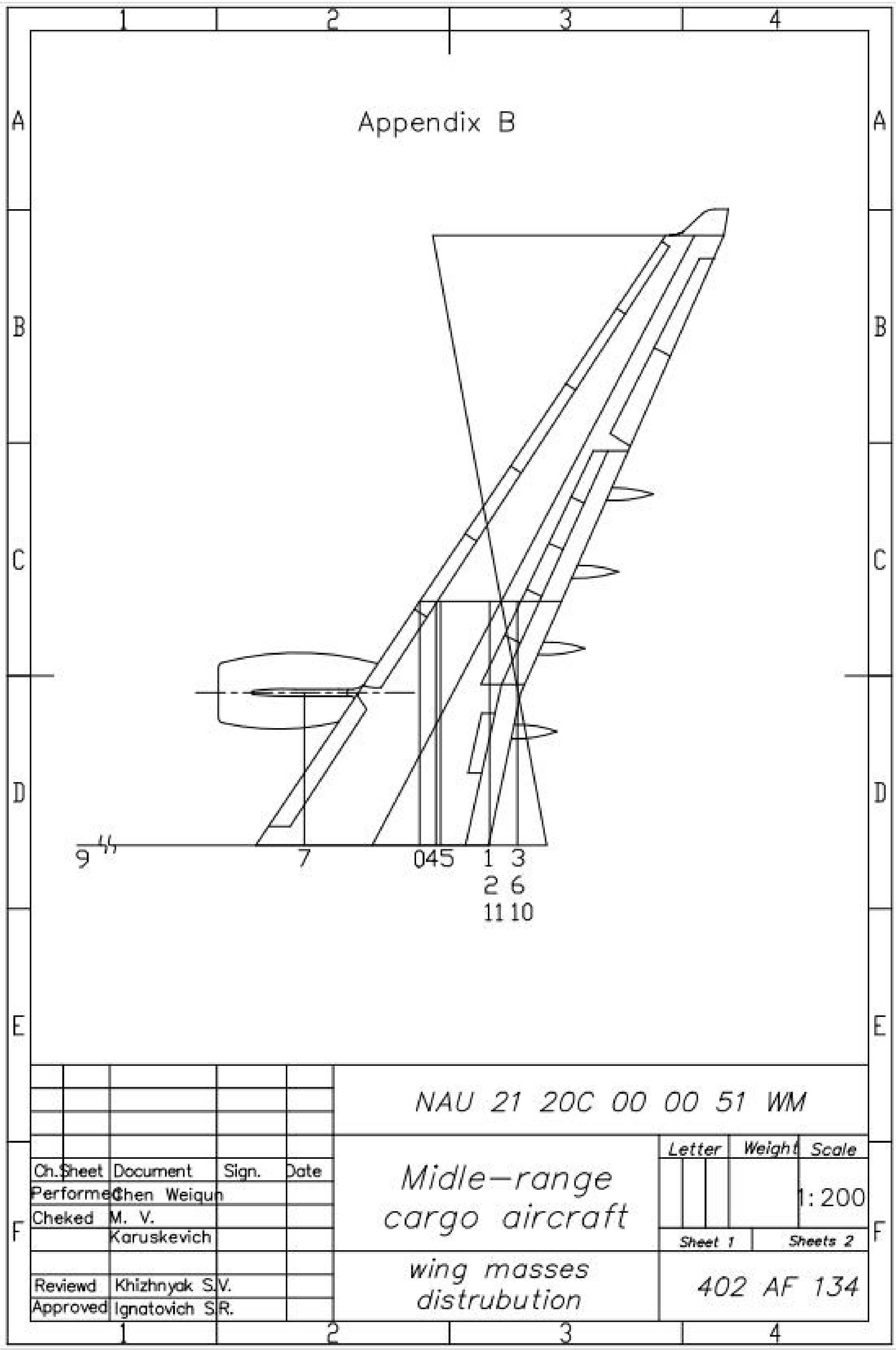
Appendix B-wing masses position



Appendix C-fuselage masses position