МІНІСТЕРСТВО ОСВІТИ ТА НАУКИ УКРАЇНИ

НАЦІОНАЛЬНИЙ АВІАЦІЙНИЙ УНІВЕРСИТЕТ

Кафедра конструкції літальних апаратів

ДОПУСТИТИ ДО ЗАХИСТУ
Завідувач кафедри
д-р техн. наук, проф.
_____________ С. Р. Ігнатович
«_____» ___________ 2021 р.

ДИПЛОМНА РОБОТА

(ПОЯСНЮВАЛЬНА ЗАПИСКА)
ЗДОБУВАЧА ОСВІТНЬОГО СТУПЕНЯ
"БАКАЛАВР"

Тема: «Аванпроект регіонального пасажирського літака з гібридною силовою установкою»

Виконав: ___________**Є.О. Самойленко**

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Київ 2021

MINISRY OF EDUCATION AND SCIENCE OF UKRAINE
NATIONALAVIATION UNIVERSITY
Department of Aircraft Design

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

DIPLOMA WORK

(EXPLANATORY NOTE)
OF EDUCATIONAL DEGREE
«BACHELOR»

Theme: « Preliminary design of a regional passenger aircraft with a hybrid power plant»

Performed by: __________ **Y.O.Samoilenko**

Supervisor: PhD, associate professor __________ **S.S. Yutskevych**

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

Kyiv 2021

NATIONAL AVIATION UNIVERSITY

Aerospace Faculty

Department of Aircraft Design

Educational degree «Bachelor»

Major 134 "Aviation and space rocket technology"

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«___» _______ 2021

TASK

for bachelor diploma work

SAMOILENKO YELYZAVETA

1. Theme: « Preliminary design of a regional passenger aircraft with a hybrid power plant»

Confirmed by Rector's order from 21.05.2021 year № 815/ст.

2. Period of work execution:from 24.05.2021 year to 20.06.2021 year.

3. Work initial data:cruise speed V_{cr}=280 (km/h), flight range L=600 (km), operating altitude H_{op} =3 (km).

4. Explanation note argument (list of topics to be developed): introduction;the project part: choice and substantiations of the airplane scheme,choice of initial data; the calculative part: main parts geometry and aerodynamic calculation, engine selection, aircraft layout, center of gravity position, special part:description and installation of the cargo equipment.

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

6. Calendar Plan

Task	Execution period	Signature
Task receiving, processing of statistical data	15.05.2021	
Aircraft take-offmass determination	19.05.2021	
Aircraft layout	19.05.2021	
Aircraft centering determination	25.05.2021	
Special part performing	06.06.2021	
Graphical design of the parts	09.06.2021	

7. Task issuance date:21.05.2021

Supervisor of diploma work ___________ S.S. Yutskevych

Task for execution is given for ___________ Y.O. Samoilenko

ABSTRACT

Explanatory note to the diploma work « Preliminary design of the domestic passenger aircraft with hybrid-electric propulsion system» contains:

40 sheets, 2 figures, 7 tables, 12 references and 3 drawings

Object of the design is development ofthe domestic aircraft with hybrid-electric propulsion system.

Aim of the diploma work is the development of the aircraft preliminarydesign and characteristic estimation.

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

The diploma work contains drawings of design of the short-rangeaircraft with commercial cargo payload capacity up to 6000 kg, calculations and drawings of the aircraft layout, the cargo compartment and equipment.

PASSENGER AIRCRAFT, PRELIMININARY DESIGN, CABINLAYOUT, CENTER OF GRAVITY DETERMINATION, PASSENGER COMPARTMENT, HYBRID ELECTRIC PROPULSION SYSTEM, LITHIUM-AIR BATTERIES, RANGE EQUATION.

Format	№	Designation	Name	Quantity	Notes
			<u>General documents</u>		
A4	1	NAU 21 16S TT	Task of work	1	
	2		Short-range passenger aircraft	2	
A1		NAU 21 16S 00 00 00 10 GV	General view		
A1		NAU 21 16S 00 00 00 10 AL	Fuselage layout		
A4	3	NAU 21 16S 00 00 0010 EN	Short-range passenger aircraft	40	
			Explanatory note		
A1	4	NAU 21 16S 71 00 00 10 AD	Documentation for assembly units	1	
			Power Plant		

Ch.	Sheet	Doc.№	Sign.	Date	NAU 21 16S 00 00 00 10 EN		
Performed by		Samoilenko Y.O.			List of diploma work	List	Sheet	Sheets
Supervisor		Yutskevych S.S.						
Adviser						AF 402 134		
Stand. contr.		Khyzhnyak S.						
Head. of dep.		Ignatovych S.						

CONTENT

Department of Aircraft Design				NAU 21 16S 00 00 00 10EN			
Performed by	Samoilenko Y.O.			Content	List	Sheet	Sheets
Supervisor	Yutskevych S.S.						
Adviser					AF 402 134		
Stand. contr.	Khizhnyak S.V.						
Head of Dep.	Ignatovych S.R.						

Abbreviation List

MTOW- Maximum Take Off Weight

HLD- High Lift Devices

HEPS- Hybrid-Electric Propulsion System

BAT- Batteries

CRYO- Cryocooler

EM- Electric motor

INV- Inverter

PROP- Propeller

Introduction

It’s generally agreed today that climate changes and weather conditions can call into question many flights, especially in hard-to-reach areas. The need and importance of enduring aircrafts is not dropping, moreover, the demand of them is only increasing. That’s why we’ll be referring to domestic aircraft, based on An-38 features, to deal with such questions.

First and foremost, it’s reliable in operation in all flight missions, namely: safe take off due to the absence of stall at high angle of attack, great stability and maneuverability even with appearance of ice on a wing and tail unit.

High takeoff and landing characteristics, low-pressure tire landing gear allow such airplane to operate on small unpaved, iced, snow-covered airfields.

Maneuverability of such airplane permits to perform landing at steep glide path on small mountain airfields. Two high-power turboprop engines with reversible propellers and effective HLD allows to exploit the aircraft on short runways in various climate conditions-from -50°C to +45°C (including hot climate conditions and highlands), with airfield height up to 2600m (it passed the test in Yakutia up to -43° C temperature, in Uzbekistan-up to +42° C and on the highland runways in Kyrgyzstan).At the same time, the aircraft possesses low noise level, that permits aircraft to operate in airports, situated in city areas.

1.PROJECT PART

1.1.Analysis of prototypes and short description of designing aircraft

This topic is dedicated for acquaintance with a source of initial data choice. Therefore, we have the comparison of prototypes data and a brief description of main parts of aircraft.

1.1.1. Choice of the projected data

The choice of optimal aircraft design parameters is a multidimensional optimization goal aimed at forming the "shape" of a promising aircraft. By its configuration is meant the whole complex of flight-technical, weight, geometric, aerodynamic and economic characteristics. When forming the "face of the aircraft" at the first stage, the methods of transfer statistics, approximate aerodynamic and statistical dependencies are widely used. At the second stage, full aerodynamic calculations are used; the given formulas of weight calculations, experimental data.

Aircraft prototypes taken for design belonged to the 15-30 passenger class. Aircraft such as Rysachok Be-132MK and Su-80 will compete with the projected aircraft in this market segment. The statistics of the prototypes are presented in Table 1.1.

Table 1.1. – Operational-technical data of prototypes; [2-4]

PARAMETER	PLANES		
1	2	3	4
	Rysachok	Be-132MK	Su-80
The purpose of airplane	Passenger	Passenger	Passenger
Crew/flight attend. persons	1	1/2	1/2
Maximum take-off weight, m_{tow},kg	5700	10000	13500
Most pay-load, $m_{к.max}$, kg	1570	1900	3300
Passenger's seat	Up to 19	26	30
The height of the flight $V_{w.\ ек.}$, m	5500	6000	10700
Range $m_{к.max}$, km	2000	885	1400
Takeoff distance $L_{зл.д.}$, m	300	1500	555
Number and type of engines	2xWalter M601F	2xPK6A-65B	2xGECT7-9B
The form of the cross-section fuselage	circular	rectangular	elliptical
Extension of the fuselage	8,3	7.2	7,6
Extending the nose and tail unit part	4,6	4.1	4.3
Sweepback on 1/4 chord, ^{0}	6 deg	10deg	30 deg

The scheme is determined by the mutual arrangement of the airplane units, their number and shape. The aerodynamic and operational characteristics of an airplane depend on the layout of the airplane and the aerodynamic scheme of the airplane. Successfully selected scheme allows increasing the safety and regularity of flights, as well as the economic efficiency of the aircraft.

1.1.2. Brief description of the main parts of the aircraft

An-38 is designed on the basis of the AN-28 short takeoff and landing aircraft and differs from it by its elongated fuselage, installed equipment and new engines. It is designed to carry passengers, mail and cargo on local airlines and can be made in the following versions: sanitary, aerial survey, geophysical, forest patrol, fishery, administrative, and as a cargo version for container transportation of goods.

1.1.3. Fuselage

An-38 is built according to the aerodynamic design of a strut-braced high-wing aircraft. It differs from the An-28 by an elongated fuselage and a modified composition of equipment. The fuselage is of the semi-monocoque type. Power set: transverse 31 frames, stringers and beams. Technologically, the fuselage is divided into three parts - nose, middle and aft. In front of the fuselage there is a two-seater cockpit with access through a door on the left side. In the middle of the fuselage is the passenger (cargo) cabin. Access to the cargo compartment, located behind the cockpit, is possible through a double-leaf door that opens outward and is located at the rear of the fuselage. Depending on the use of the aircraft, the cabin can be easily re-equipped for the needs of rescue operations or the installation of special equipment. In the aft fuselage there is an entrance hatch closed by two longitudinal latches. For passengers to enter the plane, the left wing is opened and the entrance ladder is used, which is hinged on the threshold of the hatch.

1.1.4. Wing

The distinctive feature of the aircraft is a two-spar high wing of a caisson type, with a large span and secured with profiled struts. The wing consists of a center section, rectangular in plan, and two trapezoidal consoles. Center section caissons and consoles are fuel tanks-compartments. The wing is equipped with automatic and controlled ailerons, slats, spoilers and flaps. The flaps are double-slotted and consist of a main link and deflectors, two sections are mounted on the center section, two more on consoles. Slats are installed on the leading edge of the consoles. Spoilers are located on the upper surface of the wing consoles. Ailerons are installed on the wing consoles: the left aileron is equipped with a trim tab. Ailerons are automatically deflected at an angle proportional to the angle of the flaps during takeoff and landing modes. This wing design provides the aircraft not only steep take-off and landing trajectory, but also stable cruising at low speeds and high angles of attack.

1.1.5. Tail Unit

Tail unit - Cantilever twin tail. Stabilizer and fins - two-spar, caisson design. The stabilizer is non-deflectable, the angle of installation is -2 degrees. The keels are fixed at the ends of the stabilizer and rotated 2 degrees to the axes of symmetry of the aircraft. The rudders have aerodynamic compensation and weight balancing. The elevator is made with weight balancing. The rudder and elevator are equipped with trim tabs.

1.1.6. Landing gear

It consists of nose and main landing gear. Due to the reinforced landing gearon three supports, the aircraft with this wing can also be operated on unpaved take-off and landing airfields. The front support is equipped with a non-braking wheel, which is rotatable, when the steering system is turned off, the front wheel is freely oriented. The main supports are equipped with a disc brakes and an anti-skid automatic control.

1.1.7. Power Plant

2 x 1.118 KW Allied Signal TPE331-14GR-801E turboprops with 5 blade propellers.Engines of the American company "Allied Sydnal" with take-off power of 1500 hp each.Initially, it was supposed to install the TVD-20 engine of Omsk Engine Design Bureau. But the manufacture was delayed, and American engines were installed on the first AN-38 aircraft.

1.1.8. Equipment

Due to the navigation system installed on the aircraft, radio communication equipment and a system that prevents icing of the aircraft and its functional parts, the An-38 is capable of flying under various meteorological conditions in any climatic zones, regardless of the time of day.

Power supply - the power supply system provides power supply to on-board consumers with three-phase alternating current with a voltage of 200/115 (V) at a frequency of 400 (Hz), an alternating three-phase current with a voltage of 36 (V) at a frequency of 400 (Hz) and a direct current with a voltage of 27 (V). one on each engine. The source of direct current is two rectifiers and two storage batteries.

1.2. Geometry calculations for the main parts of the aircraft

This topic implies the calculation of airplane main parts and design of the fuselage layout, depending if it's passenger or cargo aircraft.

1.2.1. Wing geometry calculation

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

Full wing area with extensions is:

$$S_{wfull} = \frac{m_0 \times g}{P_0} = \frac{11782 \times 9.8}{1585} = 72.8\ (m^2);$$

Relative wing extensions area is 0.1.

Wing area is:

$$S_w = 72.81 \times 1.4 = 52\ (m^2);$$

Wing span is:

$$l = \sqrt{S_w + \lambda_w} = \sqrt{52 \times 9.4} = 22.1 (\text{m});$$

Root chord is:

$$b_0 = \frac{2S_w \times \eta_w}{(1+\eta_w) \times l} = \frac{2 \times 52 \times 2}{(1+2) \times 22.1} = 3.13(m);$$

Tip chord is:

$$b_t = \frac{b_0}{\eta_w} = \frac{3.13}{2} = 1.565(m);$$

Maximum wing width is determined in the forehead i-section and by its span is equal:

$$c_i = c_w \times b_t = 0.15 \times 1.565 = 0.235(m);$$

On board chord for trapezoidal shaped wing is:

$$b_{ob} = b_0 \times \left(1 - \frac{(\eta_w - 1) \times D_f}{\eta_w \times l_w}\right) = 3.13 \times \left(1 - \frac{(2-1) \times 2.14}{2 \times 22.1}\right) = 2.98(m);$$

At a choice of power scheme of the wing we determine quantity of longerons and its position, and the places of wing portioning.

On the modern aircraft we use xenon double – or triple – longeron wing; longeron wing is common to the light sport, sanitary and personal aircrafts. Our aircraft has two longerons.

I use the geometrical method of mean aerodynamic chord determination (Figure 1.2.). Mean aerodynamic chord is equal:

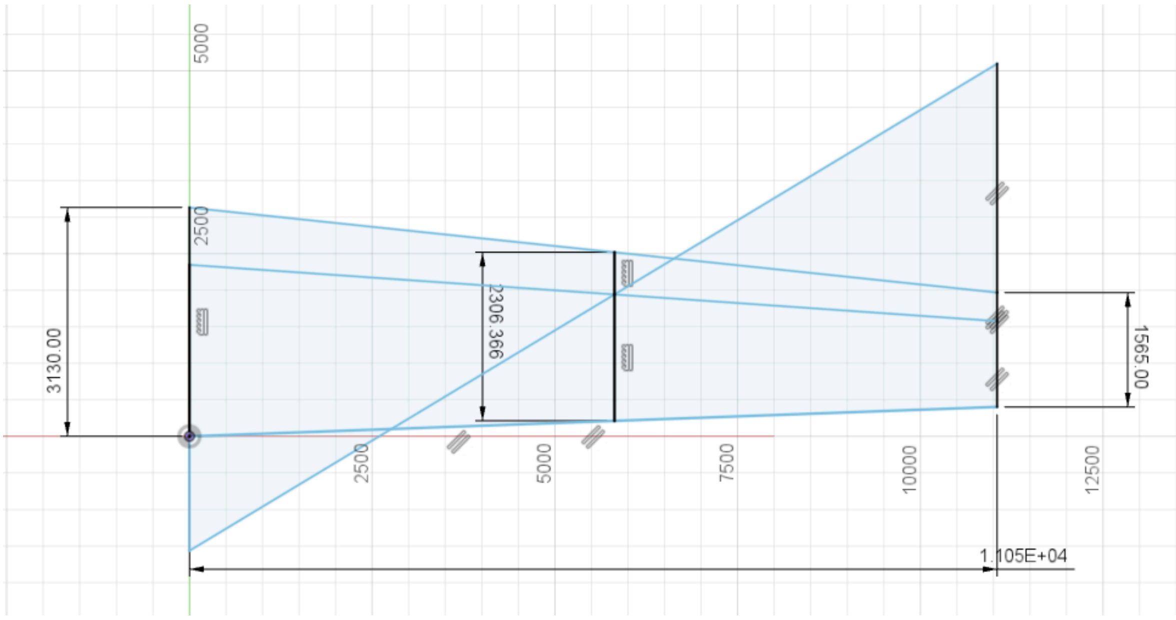

Figure 1.2. – Determination of mean aerodynamic chord

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

Ailerons geometrical parameters are determined in next consequence:

Ailerons' span:

$$l_{ai} = 0.375 \times \frac{l_w}{2} = 0.375 \times \frac{22.1}{2} = 4.14(m);$$

Aileron area:

$$S_{ail} = 0.065 \times \frac{S_w}{2} = 0.065 \times \frac{52}{2} = 1.69(m^2);$$

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

In the airplanes of the third generation there is a tendency to decrease relative wing span and ailerons area. So,l_{ail} = 0.122. In this case for the transversal control of the airplane we use spoilers together with the ailerons. Due to this the span and the area of high-lift devices may be increased, which improves takeoff and landing characteristics of the aircraft.

Aerodynamic compensation of the aileron.

Axial $S_{axinail} \leq (0.25…0.28)\ S_{ail} = 0.255 \times 1.69 = 0.43(m^2)$;

Inner axial compensation$S_{inaxinail}$ = (0.3..0.31) S_{ail};

$S_{inaxinail} = 0.3 \times 1.69 = 0.507(m^2)$;

Area of ailerons trim tab.

For two engine airplanes:

$S_{tt} = (0.04..0.06)S_{ail} = 0.04 \times 1.69 = 0.0676(m^2)$;

Range of aileron deflection

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

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

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

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

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

In the modern design the rate of the relative chords of high-lift devices is: bf = 0. 28..0.3 – one slotted and two slotted flaps;

$$b_{2slotted} = 0.3 \times b_i = 0.507(m);$$

Effectiveness of high-lift devices (C^*_{ymaxl}) rises proportionally to the wing span increase, serviced by high-lift devices, so we need to obtain the biggest span of high lift devices ($l_{hld} = l_w - D_f - 2l_{ail} - l_n$) due to use of flight spoiler and maximum diminishing of the area of engine and landing gear nacelles.

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

1.2.2. Fuselage layout

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

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

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

For transonic airplanes fuselage nose part has to be:

$l_{nfp} = 2.1 \times D_f = 2.1 \times 2.14 = 4.49(m);$

Except aerodynamic requirements consideration during the choice of cross section shape, we need to consider the strength and layout requirements.

For ensuring of the minimal weight, the most convenient fuselage cross section shape is circular cross section. In this case we have the minimal fuselage skin width. As the partial case we may use the combination of two or more vertical or horizontal series of circles.

For cargo aircrafts the aerodynamics is not so important in the fuselage shape choice, and the cross-section shape is may be close to rectangular one.

To geometrical parameters we concern: fuselage diameter D_f; fuselage length l_f; fuselage aspect ratio λ_f; fuselage nose part aspect ratio λ_{np}; tail unit aspect ratio λ_{TU}. Fuselage length is determined considering the aircraft scheme, layout and airplane center-of-gravity position peculiarities, and the conditions of landing angle of attack α_{land} ensuring.

Fuselage length is equal:

$$l_f = \lambda_f \times D_f = 7.32 \times 2.14 = 15.66(m);$$

Fuselage nose part aspect ratio is equal:

$$\lambda_{fnp} = \frac{l_{fnp}}{D_f} = \frac{4.49}{2.14} = 2.09;$$

Length of the fuselage rear part is equal:

$$l_{frp} = \lambda_{frp} \times D_f = 2.57 \times 2.14 = 5.5(m);$$

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

For passenger and cargo airplanes fuselage mid-section first of all comes from the size of passenger saloon or cargo cabin.One of the main parameters, determining the mid-section of passenger airplane is the height of the passenger saloon.

Cabin height is equal:

$$H_{cab} = 1.48 + 0.17 \times 2.12 = 1.84\ (m);$$

For economic salon with the scheme of allocation of seats in the one row (2+1) determine the appropriate width of the cabin.We have to exclude armrests for the possibility of 21-passenger-accomodation.

$$B_{cab} = n_{3chblock} \times b_{2chblock} + b_{aisle} + 2\delta = 1 \times 410 + 1 \times 870 + 350 + 400 = 2.12\ (m);$$

The length of passenger cabin is equal:

$$L_{cab} = L_1 + (n_{raws} - 1) \times L_{seatpitch} + L_2 = 1200 + 6 \times 750 + 300 = 6m\ or\ 1200 + 6 \times 720 + 0 = 5.52(m).$$

1.2.3 Luggage compartment

The area of cargo compartment is defined:

$$S_{cargo} = \frac{M_{bag}}{0.4K} + \frac{M_{cargo\&mail}}{0.6K} = \frac{7 \times 20}{0.4 \times 400} + \frac{16 \times 20}{0.4 \times 400} = 2.875(m^2);$$

Cargo compartment volume is equal:

$$V_{cargo} = 0.2 \times 20 = 4(m^3);$$

Luggage compartment design is similar to the prototype.

1.2.4 Galleys and buffets

International standards provide that if the plane made a mixed layout, be sure to make two dishes. If flight duration less than 3 hours at this time of food to passengers not issued in this case provided cupboards for water and tea. Tickets to the flight time less than one-hour buffets and toilets cannot be done. Kitchen cupboards and must be placed at the door, preferably between the cockpit and passenger or cargo have separate doors.Refreshment and food cannot be placed near the toilet facilities or connect with wardrobe.

Volume of buffets(galleys) is equal:

$$V_{galley} = 0.1 \times 20 = 2(m^3);$$

Area of buffets(galleys) is equal:

$$S_{galley} = \frac{V_{galley}}{H_{cab}} = \frac{2}{1.84} = 1.08(m^2)$$

Buffet design similar to prototype.

1.2.5 Lavatories

The number of lavatories I choose according to the original airplane and it is equal:

$n_{lav} = 1;$

Area of lavatory:

$$S_{lav} = 1.5(m^2);$$

Width of lavatory:1(m). Toilets design similar to the prototype.

1.2.6 Layout and calculation of basic parameters of tail unit

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

$m_x^{Cy} = \bar{x}_T - \bar{x}_F < 0$

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

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

Determination of the tail unit geometrical parameters:

Area of vertical tail unit is equal:

$$S_{VTU} = (0.12 \dots 0.2)S_{wing} = 0.2 \times 52 = 10.4(m^2);$$

Area o horizontal tail unit is equal:

$$S_{HTU} = (0.18 \dots 0.25)S_{wing} = 0.18 \times 52 = 9.36(m^2)$$

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

Determination of the elevator area and direction:

Altitude elevator area:

$$S_{el} = 0.3 \times S_{HTU} = 2.8(m^2);$$

Rudder area:

$$S_{rud} = 0.2 \times S_{VTU} = 2.08(m^2);$$

Elevator balance area is equal:

$$S_{eb} = 0.3 \times S_{HTU} = 2.8(m^2);$$

Rudder balance area is equal:

$$S_{rb} = 0.2 \times S_{VTU} = 2.08(m^2);$$

The area of elevator trim tab:

$$S_{te} = 0.08 \times S_{el} = 0.08 \times 2.8 = 0.224(m^2);$$

Area of rudder trim tab is equal:

$$S_{tr} = 0.05 \times S_{rud} = 0.05 \times 2.08 = 0.104(m^2);$$

Height of vertical stabilizer:

$$h_{VTU} = 0.2 \times l_w = 0.2 \times 22.1 = 4.42(m);$$

Let's assume that $l_{HTU} \approx l_{VTU}$

Root chord of horizontal stabilizer is:

$$b_{0HTU} = \frac{2S_{HTU} \times \eta_{HTU}}{(1 + \eta_{HTU}) \times l_{HTU}} = \frac{2 \times 9.36 \times 2.5}{(1 + 2.5) \times 5.72} = 2.33(m);$$

Tip chord of horizontal stabilizer is:

$$b_{1HTU} = \frac{b_{0HTU}}{\eta_{HTU}} = \frac{2.33}{2.5} = 0.932(m);$$

Root chord of vertical stabilizer is:

$$b_{0HTU} = \frac{2S_{VTU} \times \eta_{VTU}}{(1 + \eta_{VTU}) \times l_{VTU}} = \frac{2 \times 10.4 \times 1.15}{(1 + 1.15) \times 5.72} = 1.94(m);$$

Tip chord of vertical stabilizer is:

$$b_{1VTU} = \frac{b_{0VTU}}{\eta_{VTU}} = \frac{1.94}{1.15} = 1.68(m);$$

1.2.7 Landing gear design

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

Main wheel axel offset is:

$$e = 0.18 \times 2.306 = 0.415(m);$$

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

$B = 0.405 \times l_f = 0.405 \times 15.66 = 6.346(m)$ or

$$B = 15.29 \times e = 15.29 \times 0.415 = 6.346(m);$$

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

Front wheel axial offset will be equal:

$$d_{ng} = B - e = 5.931(m);$$

Wheel track is:

$$T = 0.541 \times B = 0.541 \times 6.346 = 3.434(m);$$

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

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

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

The load on the wheel is determined:

K_g = 1.5...2.0 – dynamics coefficient.

Nose wheel load is equal:

$$P_{NLG} = \frac{(9.81 \times e \times K_g \times m_0)}{(B \times z)} = \frac{(9.81 \times 0.415 \times 1.5 \times 11782)}{(6.346 \times 1)} = 11337.7(N);$$

Main wheel load is equal:

$$P_{MLG} = \frac{(9.81 \times (B - e) \times m_{0)}}{(B \times n \times z)} = \frac{(9.81 \times 5.931 \times 11782)}{(6.346 \times 2 \times 1)} = 54011(N);$$

Table 1.3. – Aviation tires for designing aircraft

Main gear			Nose gear		
Tire size	Ply rating	Mass, kg	Tire size	Ply rating	Mass, kg
810x320 mm	10	23.4	600x220 mm	6	10.5

1.2.8 Choice and description of power plant

The power plant of An-38-100 includes two engines TPE 331-14GR.-801E with propellers HC-B5MA-5A/M11276 №K-3 and other supporting systems.

TPE 331-14GR-801E engine-turboprop, single shafted. Engine power on the shaft at takeoff and max. duration regimes-1500 (hp).

It consists of:

- -reducer
- -two-stage centrifugal compressor
- -three-stage axial turbine
- -unregulated exhaust unit with ejector, which provides an engine compartment ventilation.

Propeller is installed on the reducer's output shaft.

The propeller is five-bladed, with manageable feathering mechanism, 2.85 (m) diameter, rotation direction is clockwise.

Rotational frequency during flight is set by the propeller control lever in 100-95% range (100% corresponds to rotational frequency 1552 (rpm)).

1.3 Determination of the aircraft center of gravity position

In given topic we ascertain the center of gravity positions of the wing and fuselage for four different load situations.

1.3.1 Determination of centering of the equipped wing

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

The example list of the mass objects for the aircraft, where the engines are located under the wing, included the names given in the table 3.1. The example list of the mass objects for the aircraft, where the engines are located in the wing, included the names given in the Table 1.4. Coordinates of the center of power for the equipped wing are defined by the formulas:

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

Table 1.4. - Trim sheet of equipped wing

N	Name	Mass		C.G. coordinates x_i (m)	Moment $m_i\, x_i$ (kgm)
		Units	total mass m_i (kg)		
1	Wing (structure)	0,19368	2281,9	1	2262,7
2	Fuel system, 40%	0,0027	31,8	1	31,5
3	Control system, 30%	0,0045	53,0	1,4	73,4
4	Electrical equip. 10%	0,00299	35,2	0,2	8,1
5	Anti-icing system 50%	0,0122	143,7	0,2	33,1
6	Hydraulic system, 70%	0,02401	282,9	1,4	391,4
7	Power units	0,08758	1031,9	0,7	696,5
8	Equipped wing without fuel and LG	0,32766	3860,5	0,7	2800,3
11	Fuel	0,09445	1112,8	1	1103,4
	Equipped wing	0,42211	4973,3	0,8	3903,7

1.3.2 Determination of centering of the equipped fuselage

Origin of the coordinates is chosen in the projection of the nose of the fuselage on the horizontal axis. For the axis X the construction part of the fuselage is given. The example list of the objects for the AC, which engines are mounted under the wing, is given in Table 1.5.

The CG coordinates of the FEF are determined by formulas:

$$X_f = \frac{\Sigma m_i^{/} X_i^{/}}{\Sigma m_i^{/}};$$

After we determined the C.G. of fully equipped wing and fuselage, we construct the moment equilibrium equation relatively to the fuselage nose:

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

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

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

where $\boldsymbol{m_0}$ – aircraft takeoff mass, kg; $\boldsymbol{m_f}$ – mass of fully equipped fuselage, kg; $\boldsymbol{m_w}$ – mass of fully equipped wing, kg; $\boldsymbol{C}$ – distance from MAC leading edge to the C.G. point, determined by the designer.

C = (0,23...0,32) $\mathbf{B_{MAC}}$ – high wing; X_{MAC} =7.860

Table 1.5. – Trim sheet of equipped fuselage

№	Objects	Mass		Coordinates of C.G.	Moment (kgm)
		Units	Total (kg)		
1	Fuselage	0,10526	1240,2	7,8	9710,6
2	Horizontal TU	0,0278	327,5	14,3	4683,8
3	Vertical tail unit	0,02624	309,2	14,3	4420,9
Equipment					
4	Anti-icing system, 50%	0,0122	143,7	5,8	833,7
5	Control syst 70%	0,0105	123,7	7,8	2482,5
6	Hydraulic sys30%	0,01029	121,2	11	1328,9
7	Electrical eq, 90%	0,02691	317,1	7,8	968,7
8	Radar	0,0048	56,6	0,6	33,9
9	Air-navig. system	0,0072	84,8	0,8	67,9
10	Radio equipment	0,0036	42,4	1,1	46,7
11	Instrument panel	0,0084	99,0	1,2	120,7
Passenger aircraft					
Passenger eq+ Non typical eq+ Additional equipment+ Service equipment					
12	Seats of pass. economical class	0,016	188,5	4,5	843,6
13	Seats of crew	0,00047	5,5	1,7	9,3
14	Seats of flight attendance	0,00037	4,4	10,6	46,3
Furnishing (Lavatory, Galley/buffet)					
15	Lavatory	0,00185	21,8	10,6	229,9
16	Galley	0,00185	21,8	2,5	54,5
	Equipped fuselage without payload	0,34549	4070,6	7,7	31245,6
Payload					
17	Mail/*Cargo*	0,0294	346,4	12	4156,7
18	*Crew*	0,020371	240,0	2,1	504,1
19	Baggage	0,0514825	606,6	11,8	7157,5
20	Meals	0,00377	44,4	2,5	111,1
21	Passengers	0,1273765	1500,7	6,8	10205,1
	Total	0,57789	6808,7	7,8	53379,9

1.3.3 Calculation of center of gravity positioning variants

The list of mass objects for center of gravity variant calculation given in Table 1.6. and Center of gravity calculation options given in table 1.7., completes on the base of both previous tables.

Table 1.6. – Calculation of C.G. positioning variants

Name	Mass, kg	Coordinates	Moment
Object	m_i	C.G. m	Kgm
Equiped wing without fuel and L.G.	3860,5	0,7	2800,3
Nose landing gear (retracted)	132,3	0,8	103,6
Main landing gear (retracted)	396,9	7,1	7,1
Fuel	1112,8	1	1103,4
Equiped fuselage	4070,6	7,7	31245,6

Table 1.7. – Airplanes C.G. position variants

№	Variants of the loading	Mass, kg	Moment of the mass, кg*m	Centre of the mass, m	Centering, %
1	Take-off mass	11782,0	99193,9	8,4	38,6
2	Landing variant	11630,4	93869,6	8,1	23,5
3	Transportation variant (without payload)	9283,9	74741,3	8,1	22,7
4	Parking variant (without fuel and payload)	8460,3	67320,9	8	18,6

1.4. Conclusion to the project part

In the process of given designing work I've obtained the next results:

- preliminary design of the short-range aircraft with 21 passengers;

-the center of gravity position for four different load situations in the range from 18,6 to 38,6, listed in Table 1.7.;

-the calculation of the main geometrical parameters of the landing gear, wing, fuselage, tail unit;

-the choice of the tires, which satisfy the requirements;

The chosen layout of high-wing aircraft provides accommodation for 21 passengers.

Installation of turboprop engines type, TPE 331-14GR.-801E ensures good operation and suitable characteristics (take-off power of 1500 hp each) for such type of aircraft.

2. DEVELOPMENT OF HYBRID PROPULSION SYSTEM

2.1. Introduction

Nowadays, the importance of hybrid-electric technologies is raising with each day, covering ground-based transports, cars, buses, trains and also ships. Same tendency is intruded in aviation industry, portending a realm of new propulsion system architectures and even new aircraft configurations. The replacement of aircraft systems (pneumatic and hydraulic) by electric carries forwarded seeks to maintain all present-day demands, such as the reduction of pollution, noise lowering and increase of fuel efficiency. At the same time, all basic design criteria must be taken into account. This substitution is a question of current interest either in Unmanned Aerial Vehicles used for specific missions or conventional regional aircraft.

2.1.1. Hybrid-electric propulsion system implementation

The attention of Aviation industry is dedicated to hybrid electric and subsequently full electric aircrafts, considering different approaches to reach the goal. The basic of them are:

- Batteries as energy source to (partly) replace fossil fuels.
- Provide electric power at specific flight phases to limit the variation in the gas turbine operability.
- Novel aircraft configurations (e.g., distributed propulsion).

The difficulty in first approach is that modern batteries have the specific energy around 100-200 (Wh/kg) at the cell level, while the approximate specific energy of gasoline is 13,000 (Wh/kg). The best revealed lithium-air batteries have the value of 363 (Wh/kg), it means that this is far from possible, and it is supposed that batteries with an energy density of 750 (Wh/kg) will probably not be approachable until 2035. Moreover, even with considerable battery technology improvements, aircraft size will be limited.

The second choice would be to apply batteries for propulsion to deliver extra power during certain phases of the flight, like takeoff and climb. In such case, ordinary gas turbines could be enhanced for one flight mode (cruising). Therefore, they can have less weight and be more efficient.

Promising future configurations, namely the blended wing body, allow separation of the engine from the propulsor. A distributed propulsion can be applied to improve efficiency through wake-filling and boundary layer absorption. In addition, a higher degree of power plant and airframe integration can result in lighter structure.

2.1.2. Hybrid electric propulsion system architecture

In addition to targets for future innovations, a hybrid-electric design for actual regional turboprop aircraft is being considered. The parallel configuration may be chosen as the solution for hybrid-electric regional aircraft due to the higher efficiency compared to the series configuration, that is used for aerospace applications. Fuel is utilized to power the turboshaft engine while batteries are utilized to power the electric motor. Either the turboshaft engine or the electric motor drives a train connected to the propellers. We use lithium-air batteries as a power source incorporated with conventional fuel.

The considerable batteries weight in comparison with the fuel will have a crucial impact on the design. The scheme of the system is shown in Figure 2.1.

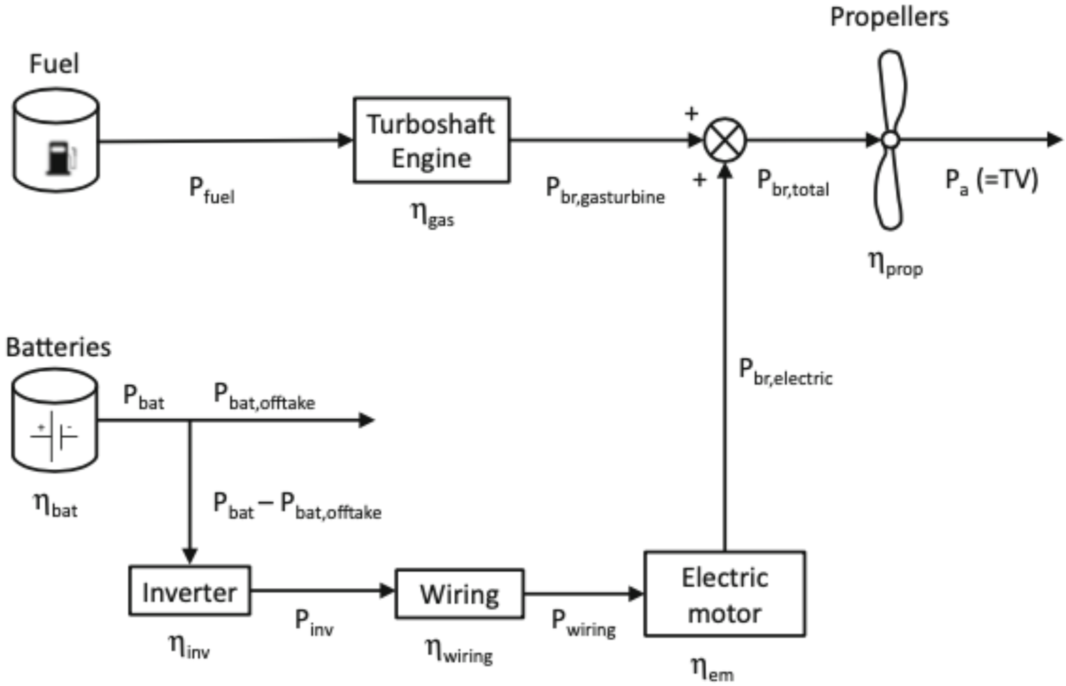

Figure 2.1. Parallel hybrid electric propulsion system architecture

A distinctive feature of these batteries is their weight rise somewhat upon t he operation. If to talk about cell voltage, it is supposed that their high voltage is ar ound 10 (kV), for transmission losses lowering. The current intensity is assumed to be 50-
100 (A). It should be noted that cell voltage as well as current intensity do not have an impact on the design process presented in this study. The basic hybrid electric p ropulsion system parameters are shown in Table 2.2.

Table 2.1. – Hybrid electric propulsion system parameters

Parameter	Value
Lithium–air battery-specific energy range (Wh/kg)	750-1500
Lithium–air battery mass variation [kg/Wh]	0.000192
Efficiency batteries η_{bat} (-)	0.95
Efficiency electric motor (excluding cryocooler), η_{em} (-)	0.98
Efficiency inverter, η_{inv} (-)	0.98
Efficiency wiring, η_{wiring} (-)	0.99
Power density electric motor,p_{em} (kW/kg)	15
Power density inverter, p_{inv} (kW/kg)	20
Power density cryocooler, p_{cryo} (kW/kg)	0.33
Cryocooler power requirement (% of max. electric motor power output)	0.45
$U_{cable,}$ (kV)	6

Correspondingly to NASA, the power of the cryocooler should be 0.16% of the maximum resultant power of the electric motor. Despite that, a more moderate value of 0.45% is chosen. High-temperature superconducting motors require alternating current, it means that an inverter is needed. High voltage aluminum cables (6 (kV)) are applied for wiring. This rather high voltage is necessary in cable diameter reduction. In addition, aluminum cables for the same voltage and current have less weight than copper cables. The weight rate is a function of the current for a 6 (kV) rated cable and is based on commercially available products.

2.2 Design approach

The design scheme is further enlarged with modules for sizing electrical co mponents and a new mission analysis. Different approaches can be used in the liter ature to measure the level of hybridization. Two of them are mentioned in this stud y: the degree of hybridization in (2.1) the source and (2.2) on the shaft driving the propeller. Either ways have their pros and cons, so they are used in parallel. The po wer split (S) is determined as the ratio of the shaft power provided by the electric motor and the gas turbine, while the hybridization factor (w) is specified as the rati o of the energy stored in the batteries to the total energy stored in the fuel and batte ries.

$$S = \frac{P_{br.electric}}{P_{br.total}} \quad (2.1)$$

$$\psi = \frac{E_{bat}}{E_{total}} \quad (2.2)$$

The power distribution can, indeed, change during flight, and because of that another criterion is presented: the supplied power ratio. It is determined as the total motor power integrated during the mission in relation to the total shaft power integrated during the mission.

$$\Phi = \frac{\int P_{br.electric}}{\int P_{br.total}} \quad (2.3)$$

The range of the common aircraft powered by fuel (zero wind conditions, thrust vector parallel to the airspeed vector) can be calculated based on the Bre´guet range equation.

$$R = \int_{W_{final}}^{W_{start}} \frac{V \cdot C_L \cdot 1}{C_T \cdot C_D \cdot W} dW \quad (2.4)$$

During integral solution many factors can make an impact on it, namely:flight strategy(for example, gradual climb at constant speed and angle of attack),models, illustrating the propulsion and aerodynamic performance. Initial data for the regular flight strategies can be found in many standard aircraft performance textbooks. As a result of gradual change of weight during flight, the range equation is based on that fact, therefore we have the integration over the aircraft weight. Hence, the range equation is not used for aircraft that (partially) use batteries as an energy source. Theoretically, range can be obtained by integrating speed over time.

$$R = \int_{t_{start}}^{t_{final}} V\,dt \quad (2.5)$$

To solve this problem, you can relate the energy stored in the fuel and batteries to time. Over time, the energy stored in the batteries or fuel decreases. The power applied (energy per second) is the sum of the fuel power and battery power.

$$\frac{dE}{dt} = \frac{dE_{fuel}}{dt} + \frac{dE_{bat}}{dt} \quad (2.6)$$

The time rate of change of the energy stored in the fuel can be connected to the fuel flow rate and the fuel energy density. In addition, the power on the shaft of a turboshaft engine is linked to the fuel consumption through the power-specific fuel consumption.

$$F = -\frac{dW_{fuel}}{dt} = -\frac{g}{H_{fuel}}\frac{dE_{fuel}}{dt} \quad (2.7)$$

$$F = C_P \cdot P_{br.gasturbine} \quad (2.8)$$

Power split determines the ratio of shaft power transmitted by the turboshaft engine to the total shaft power delivered to the propeller. Propeller efficiency determines the amount of the shaft power converted to available power.

$$P_{br.gasturbine} = (1-S) \cdot P_{br.total} = (1-S)\frac{P_a}{\eta_{prop}} \quad (2.9)$$

Therefore, the change in energy stored with fuel can be expressed as a function of available power.

$$\frac{dE_{fuel}}{dt} = -\frac{c \cdot E}{\eta_{prop}}\frac{H_{fuel}}{g}\,(1-S) \cdot P_a \quad (2.10)$$

Connecting Eqs. 2.9 and Eqs. 2.10, we acquire an equation for the change in total energy with time.

$$\frac{dE}{dt} = -\left(\frac{C_P}{\eta_{prop}}\frac{H_{fuel}}{g}(1-S) + \frac{S}{\eta_{elec}\cdot\eta_{prop}}\right)\cdot P_a \qquad (2.11)$$

In a quasi-steady, quasi-rectilinear flight, the thrust should be similar to the drag and the lift should be equal to the weight. Supposing that the thrust vector is parallel to the airspeed vector, Equation 2.12 can be defined.

$$P_a = DV = \frac{C_D}{C_L}WV \qquad (2.12)$$

Here the basic range equation for the cruising flight of a hybrid electric vehicle can be composed by summing up Eqs. 2.5, 2.11 and 2.12.

$$R = \int_{E_{final}}^{E_{start}} \frac{1}{\left(\frac{C_P}{\eta_{prop}}\frac{H_{fuel}}{g}(1-S) + \frac{S}{\eta_{prop}\cdot\eta_{elec}}\right)}\frac{C_L}{C_D}\frac{1}{W}\,dE \qquad (2.13)$$

The power distribution and angle of attack are supposed to remain constant throughout the cruise flight. In addition, the specific fuel consumption, propeller efficiency and combined electrical efficiency are assumed to be constant for the speed and altitude range we are interested in. In the matter of constant airspeed flight, this results in a gradual gain in altitude. It should be noted that this climb is less steep than for conventional aircraft. In the situation of a full electric airplane, this becomes a horizontal flight profile.

$$R = \frac{1}{\left(\frac{C_P}{\eta_{prop}}\frac{H_{fuel}}{g}(1-S) + \frac{S}{\eta_{prop}\cdot\eta_{elec}}\right)}\frac{C_L}{C_D}\int_{E_{final}}^{E_{start}}\frac{1}{W}\,dE \qquad (2.14)$$

The weight term is remaining in the integral because it is a function of energy. Both battery weight and fuel weight can be expressed in energy terms based on their respective energy densities.

$$W = W_{empty} + W_{payload} + W_{bat} + W_{fuel} \qquad (2.15)$$

$$W = \frac{E_{bat}\cdot H_{fuel} + E_{fuel}\cdot H_{bat}}{H_{bat}\cdot H_{fuel}}g + W_{empty} + W_{payload} \qquad (2.16)$$

Applying the hybridization coefficient, the equation can be revised as a total energy function.

$$W = \frac{(\psi \cdot H_{fuel} + (1-\psi) \cdot H_{bat})}{H_{bat} \cdot H_{fuel}} \cdot g \cdot E_{start} + W_{empty} + W_{payload} \quad (2.17)$$

The energy density, weight of the payload, empty weight, and hybridization factor are all constant. Consequently, the final range equation can be obtained by solving the integral.

$$R_{hybrid} = \frac{\eta_{prop}}{g \cdot \left(C_P \cdot \frac{H_{fuel}}{g} \cdot (1-S) + \frac{S}{\eta_{elec}}\right)} \frac{C_L}{C_D} \frac{H_{bat} \cdot H_{fuel}}{\left(\psi \cdot H_{fuel} + (1-\psi) \cdot H_{bat}\right)} \cdot \ln\left(\frac{\frac{(\psi \cdot H_{fuel} + (1-\psi) \cdot H_{bat})}{H_{bat} \cdot H_{fuel}} \cdot g \cdot E_{start} + W_{empty} + W_{payload}}{W_{empty} + W_{payload}}\right) \quad (2.18)$$

2.2.1. Calculations

For the development and research of electric and hybrid electric propulsion architectures, it is very important to select a reference aircraft to determine the required propeller propulsive power. A preliminary comparative analysis has shown that hybrid propulsion systems can possibly be developed and are reviewed practically applicable on relatively small light aircraft because of their low power requirements. Characteristics such as lift and drag coefficients, MTOM, we can take from the initial data of our aircraft:

$C_L = 0.55878$

$C_D = 0.01275$

$MTOM = 11782\ (kg)$

Moreover, energy at start condition (before flight) can be obtained:

$E_{start} = 0.99 \cdot MTOM = 11664.18\ (kg \cdot f)$, converting to Joules:

$E_{start} = 114386.531\ (J)$

Next step was the selection of the turboprop engine TPE 331-14GR-801E , in which the efficiency of the propeller was taken as $\eta_{prop} = 0.81$. As for the electric motor, batteries are used as an energy source substituting the fuel. Since lithium-air batteries have by far the highest theoretical specific energy, given type was selected for the design of the aircraft. Design studies will be carried out for a battery specific energy range of 750-1500 (Wh/kg). The lower limit of 750 (Wh/kg) ($H_{bat} = 2700000\ (\frac{J}{kg})$ is chosen because miscellaneous studies have shown that this value is necessary, at least for a hybrid electric aircraft to be a realistic option. As a result, the electric system efficiency, considering all the parameters in Table 2.1. will be as $\eta_{elec} = 0.95$.

For a typical fuel-driven airplane, the range can be easily found using the Breguet equation. The Breguet equation calculates the distance on which an airplane can fly for a certain set of parameters, and consequently has a great influence on the design of modern propulsion systems and airframes. The derivation of this equation can be found in paragraph 2.2.

$H_{bat} = 2700000\ (\frac{J}{kg})$ and $H_{fuel} = 43488000\ (\frac{J}{kg})$ and are the energy densities of the fuel and battery respectively. Power specific fuel consumption we assume as $C_p = 0.00703\ (m^{-1})$.

If to talk about the weights, the empty weight of the airplane is $W_{empty} = 77755.73\ (N)$, the weight of the payload is $W_{payload} = 25242.3171\ (N)$. We will suppose the fuel weight 788 (kg) with 1364 (kg) battery instead of 1100 (kg) of pure fuel in conventional aircraft.

Accordingly,

$$E_{bat} = H_{bat} \cdot m_{bat} = 3682800000\ (J)$$

$$E_{fuel} = H_{fuel} \cdot m_f = 3426854400\ (J)$$

Energy and power split factor with the discussed specification will be $\psi = 0.1$ and $S = 0.6$. Thus, if S is equal to 0, the power used to run the propeller is completely from the combustion engine, whereas if S is 1, it is completely the opposite. Then, if S is located between 0 and 1 the airplane is behaving as a HEP aircraft.

As a result, final range equation in compliance with the procedure described in topic 2.2 can be found:

$$R_{hybrid} = \frac{\eta_{prop}}{g \cdot \left(C_P \cdot \frac{H_{fuel}}{g} \cdot (1 - S) + \frac{S}{\eta_{elec}}\right)} \frac{C_L}{C_D} \frac{H_{bat} \cdot H_{fuel}}{\left(\psi \cdot H_{fuel} + (1 - \psi) \cdot H_{bat}\right)} \cdot$$

$$\cdot \ln\left(\frac{\frac{(\psi \cdot H_{fuel} + (1-\psi) \cdot H_{bat})}{H_{bat} \cdot H_{fuel}} \cdot g \cdot E_{start} + W_{empty} + W_{payload}}{W_{empty} + W_{payload}}\right) = 910.4\ (km).$$

2.3. Conclusions to the special part

During this analytical aircraft design were defined next achievements:

- the potential environmental benefits of a hybrid electric propulsion system for regional turboprop aircraft are investigated;

-different approaches to reach such aircraft configuration were described;

-the architecture of parallel HEPS was depicted and explained;

-the main parameters of HEPS were shown;

- a new range equation suitable for hybrid electric aircraft was introduced and the procedure to obtain such equation was set out;

-the range equation for turboprop aircraft with hybrid-electric propulsion system was solved, with a result of 910,4 (km) and fuel weight 788 (kg) with 1364 (kg) battery instead of 1100 (kg) of pure fuel in conventional aircraft.

General conclusions

The following accomplishments were identified during this analytical aircraft design:

- the preliminary design of a 21-passenger short-range aircraft;

-determination of the center of gravity position for four different loads ranging from 18.6 to 38.6;

-determination of basic geometric parameters of landing gear, wing, fuselage, tail unit;

-selection of tires to meet the requirements;

- investigation of the potential environmental advantages of a hybrid electric propulsion system for regional turboprop aircraft;

-description of various approaches for achieving given aircraft configuration;

-explanation of parallel HEPS architecture;

-the basic parameters of the HEPS were revealed;

- introduced a new range equation suitable for hybrid electric aircraft the procedure was outlined for obtaining such an equation;

-the range equation for a hybrid-electric propulsion turboprop aircraft with a result of 910.4 km and a fuel weight of 788 (kg) with a battery of 1364 (kg) instead of 1100 (kg) of pure fuel in a conventional aircraft is solved.

A hybrid-electric propulsion system was investigated and analyzed for an airplane to develop its feasibility for improved performance and reduced fuel consumption and emissions.

Since pure electric propulsion concepts are not fully feasible today, a normalized range for a parallel hybrid configuration was calculated to determine the potentiality of the propulsion system. The analysis showed that the parallel hybrid configuration demonstrates a high range (910.4 (km)) in the research due to lower mass and lower power transmission losses.

References

1) An-38 [Digital source] — Access mode: URL: http://www.airwar.ru/enc/craft/an38.html
2) An-38 [Digital source] — Access mode: URL: https://aviado.ru/airlines/jets/an-38/
3) An-38 [Digital source] — Access mode: URL: https://ru.wikipedia.org/wiki/%D0%90%D0%BD-38
4) AN-38. Light Multipurpose Airplane (n.d.). Antonov [Electronic source] — Access mode: URL: https://www.antonov.com/en/history/an-38
5) Antonov An-38. Photo. Video. Interior layout. Specifications. Reviews [Digital source] — Access mode: URL: https://avia.pro/blog/antonov-38-foto-video-shema-salona-harakteristiki-otzyvy
6) Antonov An-38 [Digital source] — Access mode: URL: https://www.airlines-inform.ru/commercial-aircraft/an-38.html
7) Aircraft AN-38-100. Flight manual. Book 1 / Karetskiy Y, *1997. — 394p.*
8) Aircraft AN-38-100. Flight manual. Book 1 / Karetskiy Y, *1997. — 448p.*
9) Aircraft AN-38-100. Loading and centering guide / Karetskiy Y, *1997. — 88p.*
10) Analysis and design of hybrid electric regional turboprop aircraft / Voskuijl, M., van Bogaert, J., & Gangoli Rao, A. : *CEAS Aeronautical Journal, 2017. — 12p.*
11) Hybrid Propulsion for Regional Aircraft: a Comparative Analysis based on Energy Efficiency / Jérome Thauvin, Guillaume Barraud, Xavier Roboam, Bruno Sareni, Marc Budinger, Dimitri Leray : *ESARS-ITEC, 2016. — 8p.*

12) Performance analysis of hybrid electric and distributed propulsion system applied on a light aircraft [Digital source] — Access mode: URL: https://www.sciencedirect.com/science/article/pii/S0360544220319307#bib26