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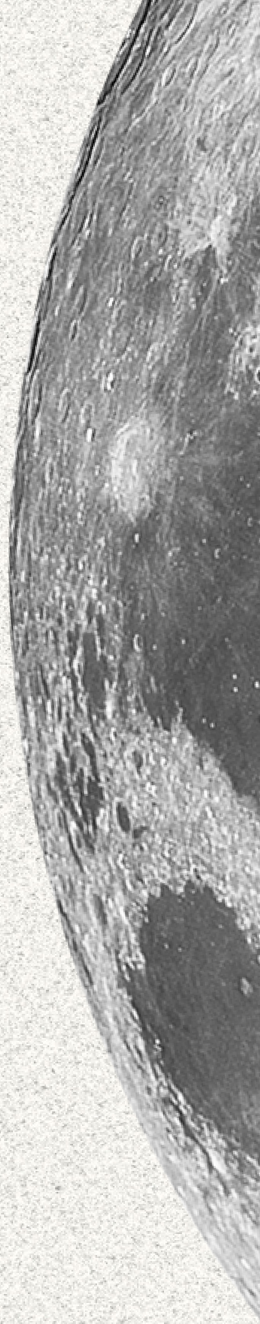

XV International Conference of Students and Young Scientists

"INTELLIGENCE. INTEGRATION. RELIABILITY"

Ukraine, Kyiv, December 7th, 2023

BOOK OF THESIS

2023

MINISTRY OF EDUCATION AND SCIENCE OF UKRAINE
NATIONAL TECHNICAL UNIVERSITY OF UKRAINE “IGOR SIKORSKY KYIV POLYTECHNIC INSTITUTE”

INSTITUTE OF AEROSPACE TECHNOLOGIES

Intelligence. Integration. Reliability

XV international students and young scientists conference 7th December 2023

Kyiv
NTUU "Igor Sikorsky Kyiv Polytechnic Institute"
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CONTENTS

UDC 629.7.

Adler R., Lukianov P.

COMPARATIVE ANALYSIS OF HYBRID PROPULSION SYSTEMS IN VERTICAL TAKEOFF AND LANDING AIRCRAFT

Introduction. In systems with vertical takeoff and landing, it is advisable to use hybrid propulsion systems, where the energy source is both high-energy fuel for long cruise flight and generated electric current for short-term vertical takeoff, which significantly increases the range and overall energy efficiency of the aircraft [1].

Results. The following types of hybrid configurations have been used in aviation technology: series, parallel, and combined. Each of these types has its own advantages and disadvantages.

The advantage of the series hybrid configuration is that the internal combustion engine (ICE) is completely mechanically separated from the propeller, and its power output is independent of the power requirement of the drive. That is, the internal combustion engine can operate in optimal conditions for it during different flight modes, and fluctuations in excess or shortage of power are compensated by a buffer battery, where excess power is converted into charging, and shortage into discharging. This hybrid configuration is conveniently used in a distributed electric propulsion (DEP) system, where the aircraft has several propellers, which in turn are driven by electric motors [2]. However, among the disadvantages of this configuration is low energy efficiency, since during a number of energy conversions, significant power losses occur, which drastically reduce the final efficiency.

In the case of a parallel hybrid configuration, the internal combustion engine and the electric motor are connected together mechanically to the propeller via a transmission, so they can either contribute to the transmission of torque separately or together. This makes it possible to combine their power and reduce the weight of each component and the system as a whole, increasing the overall efficiency, as there may be no unnecessary energy conversion [3]. The disadvantage is the need to use an additional transmission that distributes mechanical energy between the ICE and the electric motor/generator.

The combined-type configuration (series-parallel), also known as the power distributed configuration, is a combination of the above architectures [2]. This hybrid configuration not only makes the power distribution more convenient but also allows the internal combustion engine and the electric motor to operate in the most efficient modes. The combined system is the most advanced among hybrid configurations, but its design is more complex.

Rechargeable electric batteries and high-energy fuel are an integral part of an aircraft with a hybrid propulsion system. If we compare the specific energy characteristics of aviation gasoline, which has an energy value of 43-46 MJ/kg [4], with lithium-ion batteries, which have an energy density of 0.54-0.9 MJ/kg [5], the difference will be 48...85 times, which is very significant, even if we take into account the rather low efficiency of internal combustion engines compared to electric engines, which operate quite energy efficiently.

Conclusion. The comparative analysis of hybrid propulsion systems shows their advantages and disadvantages in terms of systems and convenience of electrical and mechanical energy distribution, efficiency, effectiveness, and overall technical complexity. Also, a comparative analysis of the specific energy density of electric batteries relative to high-energy fuel was carried out, according to which it is advisable to use a hybrid propulsion system if it is necessary to use an electric drive, which will reduce the weight of electric batteries by using the electrical energy generated by a generator driven by an internal combustion engine.

References:

1. Review of hybrid electric powered aircraft, its conceptual design and energy management methodologies. / Ye Xie, Savvarisal, Tsourdos Antonios, Zhang Dan, Gu Jason. // Chinese Journal of Aeronautics (CJA), Elsevier. – 2020. – p. 1-19.
2. A Review of Distributed Electric Propulsion Concepts for Air Vehicle Technology. / Hyun D. Kim; Aaron T. Perry; Phillip J. Ansell // IEEE Electric Aircraft Technologies Symposium – 2018 – p. 1-21.
3. Electric, hybrid, and turboelectric fixed-wing aircraft: A review of concepts, models, and design approaches. / Benjamin J. Brelje, Joaquim R.R.A. Martins // Progress in Aerospace Sciences, Elsevier. – 2019 – p. 1-19.
4. Hydrogen Electric Airplanes: A Disruptive Technological Path to Clean Up the Aviation Sector / Jonas Kristiansen Nøland // IEEE Electrification Magazine – 2021 – p. 1-22.
5. Design and optimization of lithium-ion battery as an efficient energy storage device for electric vehicles: A comprehensive review / F. N. U. Khan, M. G. Rasul, A.S.M. Sayem, N. K. Mandal // Journal of Energy Storage, Elseiver – 2023 – p. 108033.

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Adler R., Lukianov P.

CALCULATION OF THE MASS OF AN ELECTRIC BATTERY FOR AN AIRCRAFT WITH A HYBRID PROPULSION SYSTEM

Purpose of the study. This paper considers a method for determining the mass of an electric battery (EB) for a vertical takeoff and landing (VTOL) aircraft with a parallel hybrid electric propulsion architecture, where electric energy will be used not only for vertical takeoff but also for horizontal flight with the internal combustion engine (ICE) turned off and switched to an electric motor to reduce the thermal and acoustic footprint.

Calculations and results. The use of a hybrid propulsion system with a parallel configuration for a certain period of time allows for a special flight mode with the ICE off and the electric motor powered by the battery, which reduces the overall thermal and acoustic footprint of the ICE. In the initial sizing process, to simplify the design model, the mass of each component is usually estimated by power density and energy density, and complex off-rate characteristics are replaced by component efficiencies.

Determine the mass of the battery for an VTOL aircraft with a parallel hybrid configuration that is powered by lithium-ion batteries with the following parameters [1]:

- specific power density $PD_{batt} = 730$ W/kg,
- specific energy density $PE_{batt} = 230$ W*h/kg;

The mass of the battery must meet both the parameters of electrical power and energy capacity [2]:

$$M_{batt} = max\left\{\frac{P_{batt,max}}{\eta_b * PD_{batt}}, \frac{E_{batt,max}}{\eta_b * ED_{batt}}\right\},$$

where η_b – the efficiency of an EB, according to [1], is defined as 0.98,

$P_{batt,max}$ – required electrical power of the EB;

$E_{batt,max}$– required energy capacity of the EB.

The maximum electric power of the electric battery is required in the multi-rotor mode during vertical takeoff [3]. One of the advantages of a hybrid power system with a parallel configuration is the combination of an electric generator with an EB, which makes it possible to additionally use the generated power as support during peak electrical loads of the battery.

The required power of the battery can be determined by the following formula:

$$P_{batt,max} = \frac{P_{TO}}{FM * \eta_{motor} * \eta_{ESC}} - P_{gen} * \eta_{gen}\ ,$$

where FM is the quality indicator (Figure of Merit) of a multi-rotor system, which is equal to the ratio of the ideal required power for vertical takeoff to the real one, according to [3], we take this indicator as 0.7, η_{motor} – the efficiency of electric motors is defined as 0.9, η_{ESC} – the efficiency of the electronic speed controller is defined as 0.85, P_{gen} – power of the electric generator (15 kW), η_{gen} – the efficiency of an electric generator is defined as 0.9.

The required amount of electrical energy $E_{batt,max}$, used during the flight can be determined by the following formula:

$$E_{batt,max} = E_{TO} + E_{surv} + E_{land},$$

where E_{TO}, E_{surv}, E_{land} – EB energy capacity parameters to meet the needs of vertical takeoff, surveillance mode (or other long-duration electric flight) and landing, respectively.

$$M_{batt} = max\{M_{batt,P}, M_{batt,E}\} = \max\{20.2, 17.6\} = 20.2 \text{ kg};$$

Let us set the mass as $M_{batt} = 21$ kg.

According to the calculation, we obtained the following parameters of the electric battery:

$$\text{EB power } P_{batt} = 21 \text{ kg} * 700 \frac{\text{W}}{\text{kg}} = 14700 \text{ W};$$

$$\text{EB capacity } E_{batt} = 21 \text{ kg} * 230 \frac{\text{W*h}}{\text{kg}} = 4830 \text{ W*h};$$

EB Mass – 21 kg.

Conclusion.

The considered calculation method makes it possible to determine the mass of the electric battery of an aircraft with a parallel hybrid configuration according to the calculated initial needs for the maximum required electric power and the maximum required battery capacity to meet the needs of the aircraft flight mission.

References:

1. Initial Sizing of Hybrid Electric VTOL Aircraft for Intercity Urban Air Mobility / U. Cakin, Z. Kaçan, Z. A. Aydogan, I. Kuvvetli // AIAA Aviation 2020 Forum, Aircraft Design Optimization III – 2020 – p. 1-18.
2. Evaluation and Comparison of Hybrid Wing VTOL UAV with Four Different Electric Propulsion Systems. / J. Zong, B. Zhu, Z. Hou. // MDPI Aerospace. – 2021. – p. 1–14.
3. Design and performance analyses of a fixed wing battery VTOL UAV / Dündar Ö., Bilici M., Ünler T. // Engineering Science and Technology, Elseiver – 2020 – p. 1-12.

UDC 62.799

Artiukh M., Arkhypov O.

METEOROLOGICAL ROCKET WITH SOLID ENGINE

Introduction. Currently, the global market for meteorological rockets continues to expand. Solid Rocket Motors (SRMs) stand as the most preferable option for these rockets. One of the drawbacks of this kind of engines is the limited fuel combustion time. The objective of this study is to develop methods for controlling the solid fuel combustion process and optimizing its composition

Scientific and technical results. The concept of SRMs with extended time of work positively impacts the sustenance of continuous acceleration and delivery speed of payloads to their target points. Burn time is contingent upon the propellant type and charge geometry [1].

Elevated ignition points reduce burn time irrespective of the burn rate, by considering additional variables represented in the equation determining the overall time required for the entire combustion process [2]:

$$t_{b(total)} = \frac{d_0}{\dot{r}} \left\{ [(1+2\eta)\chi_0 + \eta] + \left(\frac{\pi(1+\chi_0)}{\eta(1+\chi_0) - \frac{\pi}{2\chi_0}} \right) \right\},$$

where d_0 – the diameter of the gas passage channel;

$\dot{r}$ – constant regression rate;

η – aspect ratio;

χ_0 – ratio of web thickness to d_0.

The analysis of the derived equation leads to the conclusion that there is a possibility to select the optimal combustion time based on desired fuel ignition points while keeping other parameters of grain geometry constant. Gradually increasing pressure reduces the combustion time until reaching a certain convergence point in the dependencies.

Approaches to address the challenge of creating long-duration solid rocket motors are offered in studies [3-5]. In the work [3], the engine runtime was extended from 50 to 90 seconds by reducing the minimum burn rate of the propellant from 5 mm/s to 3 mm/s through grain correction to limit the dynamic pressure below the permissible limit. Among various combustion rate inhibitors, oxamide demonstrated high efficiency in preserving the propellant mechanical properties. The overall content of solid components (AP+Al+oxamide) amounted to 85%, of which oxamide constituted 3%.

The distinction in the work [4] lies in the utilization of a burn-rate inhibitor alongside reducing thrust and prolonging the fuel burn time, achieved through an end-burning grain configuration, ensuring a smaller burning area compared to a channel-burning grain. The design principles of the solid rocket motor outlined in [5] also encompass an integrated engine configuration, thrust profiles utilizing segmented grains with varying burn rates, thin ablative thermal protection, and laser ignition.

Conclusion. The conducted study validates the efficacy of employing slow-burning fuel at low chamber pressures for meteorological rockets. To achieve this objective, a combustion rate inhibitor based on oxamide may be utilized. Additionally, increasing the size of oxidizer particles has been suggested. These proposed measures are economically viable and ensure the maintenance of mechanical characteristics throughout the burning process.

References:

1. Shapiro Y. M. Theory of solid fuel rocket engines / Y. M. Shapiro, G. Y. Mazing, N. Y. Prudnikov. – Moscow: Military publishing house of the Ministry of Defense of the USSR, 1968, p. 261.
2. Osheku C. A. Analytical Prediction for Grain Burn Time and Burning Area Kinematics in a Solid Rocket Combustion Chamber / C. A. Osheku, O. O. Babayomi, O. T. Olawole., 2019, p. 112.
3. Development of slow-burning solid rocket booster for RLV-TD hypersonic experiment. Current Science. 2018. Vol. 114. p. 74–83.
4. Mathesius K. J. Manufacturing Methods for a Solid Rocket Motor Propelling a Small, Fast Flight Vehicle. Massachusetts Institute of Technology, 23 May 2019.
5. Mathesius K. J., Vernacchia M. T., Hansman R. J. Low-Thrust Solid Rocket Motors for Small, Fast Aircraft Propulsion: Design and Development. AIAA Journal of Propulsion and Power. 2021, p. 1–30.

УДК 629.735.3

Astakhova O., Sukhov V.

FEATURES OF COMPOSITE MATERIALS APPLICATION IN UNMANNED AERIAL VEHICLE MANUFACTURING

Introduction. The implementation of the leading developments in the design of unmanned aerial vehicles (UAV) ensures their reliability, efficiency, versatility and significantly lower risk for human life in comparison to the piloted systems. Along with this, aerodynamic, weight and strength characteristics, as well as the application versatility, the visibility during the flight and the manufacturing cost considerably depend on the aerodynamic scheme of the UAV and its mass-dimensional parameters.

Statement of the problem. Considering the manufacturing process of the UAV of the “flying wing” scheme, which design features require the use of the airfoil with an S-shaped camber line [1], it can be noted how complex the technological process of its production with the use of aluminium is. At the same time, the application of composite materials allows to obtain the sophisticated aerodynamic shape, meanwhile, with a minimum number of joints.

The purpose of the work was to develop the technological manufacturing process of the UAV of the “flying wing” aerodynamic scheme. The Skywalker X8 was chosen as a prototype. Its main characteristics are summarized in Table 1 [2].

Table 1

The main characteristics of Skywalker X8

Parameters	Value
Wing span, *mm*	2120
Fuselage length, *mm*	790
Wing area, dm^2	80
Flying weight, *g*	2500…3000

Statement of the main materials of study. Vacuum forming is often used to solve manifold manufacturing tasks, for instance, the production of small nontypical parts or large-sized products. The essence of such type of forming is that reinforcing and bonding materials with the addition of auxiliary technological layers are laid out into the mould, which is then covered with an elastic diaphragm. Afterwards, under the diaphragm the negative pressure is created using the vacuum pump [3]. Due to the action of vacuum, the layers of the composite materials are compacted, which, in turn, displace the excess of the bonding material and air.

The prominence of vacuum forming is justified by a number of benefits such as:

- high-quality laminate as a consequence of compaction by the negative pressure and the absence of voids between the layers of the reinforcing material;
- removal of excess amount of bonding material during vacuuming;
- possibility to produce thin-walled products of complex geometry.

Nevertheless, when choosing such a type of forming, the following drawbacks should be taken into account:

- higher cost of product manufacturing in comparison to manual forming;
- multi-stage and prolonged process;
- the necessity of further mechanical processing of the product.

In view of the benefits mentioned above, it is the method of vacuum forming, which is chosen in order to produce the UAV of the "flying wing" scheme. Moreover, vacuum forming makes it possible to manufacture products of complex shapes with contour accuracy within 0.5 mm.

Conclusions. The key reason to design the UAV using composite materials is the possibility to obtain the perfect aerodynamic shape with the minimum number of joints, alongside with the low specific weight, high strength and low radio visibility. Taking into consideration the mentioned advantages, vacuum forming allows to achieve high-strength glider of a complex aerodynamic shape for the experimental model of the "flying wing" UAV.

References:

1. Pros and cons analysis of a flying-wing and a canard conceptions for a special purpose UAV in high altitude / T. Jurak et al. 2019 4th technology innovation management and engineering science international conference Bangkok, Thailand, 11–13 December 2019. 2019. URL: https://doi.org/10.1109/times-icon47539.2019.9024504

2. Skywalker X8. TECHNICAL DATA [Електронний ресурс] – Режим доступу до ресурсу: http://skywalkermodel.com/en/76.html.

3. Доценко О. К., Стельмах Н. В. Методи виготовлення деталей із композитного матеріалу. XV Всеукраїнська науково-практична конференція студентів, аспірантів та молодих вчених «ЕФЕКТИВНІСТЬ ІНЖЕНЕРНИХ РІШЕНЬ У ПРИЛАДОБУДУВАННІ». 2019. С. 113–116.

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Ortamevzi G., Atay Y.

ANALYSIS OF THE TEMPERATURE OF THE MAIN LANDING GEAR RIMS ACCORDING TO THE BRAKING SITUATION

Introduction. In this study, the temperature of the main landing gear rim of an experimental aircraft was determined under different braking situations using the finite element method. A mathematical model was created for the 3D model of the brake disc and rim of the main landing gear. Temperature values were found by performing thermal analyzes on the relevant parts using this mathematical model [1], [2], [3].

Materials and methods. In order to analyze the heat dissipation that will occur in the brake disc of the main landing gear during braking of the experimental aircraft, a 3D model consisting of the brake disc and right and left rim parts is shown in Figure 1.

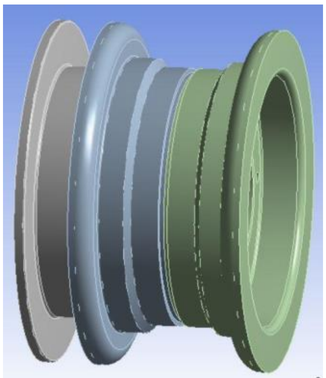

Figure 1 - 3D model of brake disc and rim parts

In addition, mathematical models have been created to perform thermal analysis of these parts using the finite element method. Mesh statistics of mathematical models 124915 nodes, 70510 elements for brake disc; 150668 nodes, 87184 elements for left side rim; For the right side rim it is 150668 nodes, 87184 elements. Mathematical models are shown in figure 2.

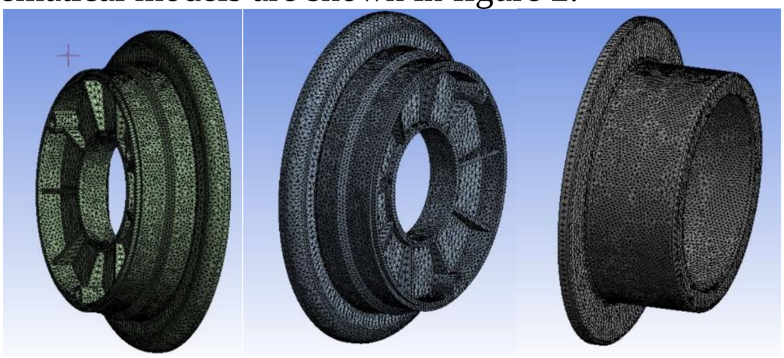

Figure 2 - Mathematical models of brake disc, left and right rim parts

The material of these parts was chosen as aluminum alloy for lightness. Temperature values of 450, 400, 350, 300, 250, 200, 150, 100, 50 °C were applied for the same period of time according to the braking force [4] [5]force on the brake pad path on the brake disc, and the heat transfer on the parts and between each other was applied and monitored. The ambient temperature is 22 °C.

Results and Discussion. According to the thermal analysis results, the temperature distribution on the parts is shown in figure 3, depending on the temperature on the brake pad being 400 °C.

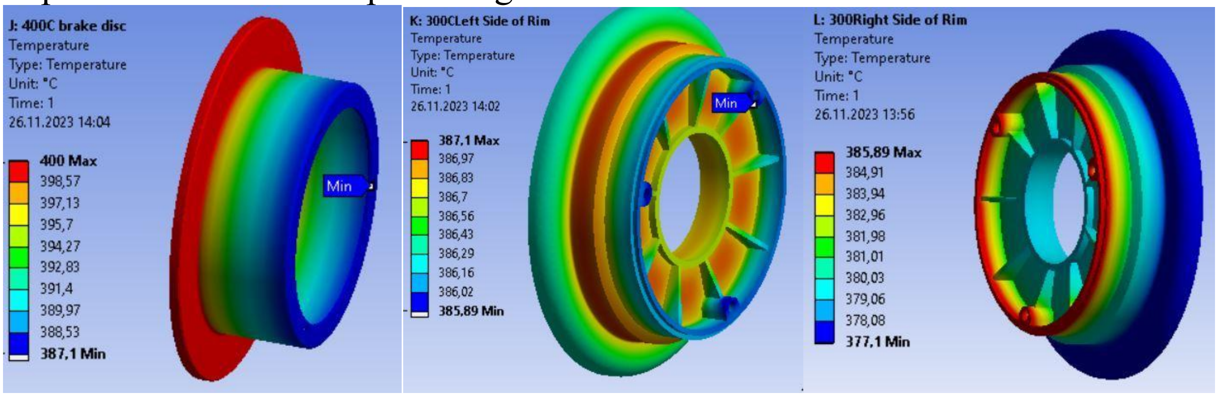

Figure 3 - Temperature Spread on brake disc, left and right rim parts when brake pad path temperature 400 °C

For different temperature values that will occur on the brake pad path, the temperatures of the tire contact surfaces of the left and right rim are shown in the graphic in figure 4. It is recommended that the operating temperature of tires not exceed 200 °C. [6]

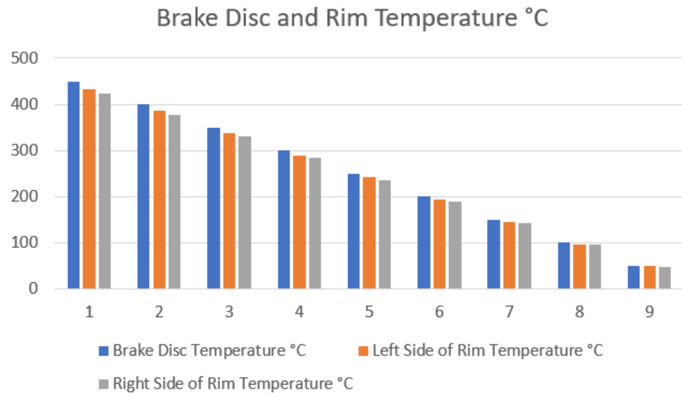

Figure 4 - Brake disc pad path temperature; left and right side rim tire contact surface temperature

Conclusion. If the brake system will be designed for extreme braking conditions, fusible plugs that will melt below the tire melting temperature should be used on the rims.

As the outdoor temperature approaches, the temperature difference between the parts decreases.

Since aluminum has high heat conductivity, the temperature difference between the parts is not high. Using a different material to reduce the heat generated in the brake disc to the rim parts, using insulators between the parts, using a cooling system, and removing the brake disc from the rims may be options to solve the problem. However, the negative effects of these applications on weight and cost should not be forgotten.

Temperature values exceeding 200°C on the tire seating surface occurred in experiments numbered 1-4. Brake dosing should be done at a level where the temperature on the brake disc does not exceed 200°C.

References:

1. T. V Manjunath and P. M. Suresh, "Structural and thermal analysis of rotor disc of disc brake," *Int. J. Innov. Res. Sci. Eng. Technol.*, vol. 2, no. 12, pp. 7741–7749, 2013.
2. A. B. ETEMOGLU, N. ETEMOGLU, B. TÜRKAN, and A. S. CANBOLAT, "Araç Fren Sistemlerinin Isil Performansinin Parametrik Olarak Incelenmesi," *El-Cezeri*, vol. 5, no. 2, pp. 556–565, 2018.
3. N. Gupta, M. Bhandwal, and B. S. Sikarwar, "Modelling and simulation of brake disc for thermal analysis," *Indian J. Sci. Technol.*, vol. 10, no. 17, pp. 1–5, 2017.
4. D. S. Hugar and U. B. Kadabadi, "Design and thermal analysis of disc brake for minimizing temperature," *Int. Res. J. Eng. Technol.*, vol. 4, no. 07, 2017.
5. E. Ölgü, "Tasit fren sistemi isil performansinin sayisal incelenmesi," Bursa Uludag University (Turkey), 2000.
6. Y. Li and D. W. Wang, "Heat generation of aircraft tires at landing," *Int. J. Aviat. Aeronaut. Aerosp.*, vol. 9, no. 1, p. 6, 2022.

UDC

Bakun V., Marynoshenko O.

MEASUREMENT OF THE THRUST OF ROCKET ENGINES

Introduction. During the process of design, development and production of rocket engines (RD) and other propulsion systems, tests on specialized stands play an important role. They make it possible to simulate working processes and the environment in which rocket engines are located in ground conditions as closely as possible. Conducting tests allows quantitative and qualitative assessment of the working processes of rocket engines, vibration modes, deformations, and confirming the operational design parameters of rocket engines.

Among the main parameters of the RD that require experimental confirmation is the confirmation of engine thrust. One of these methods is the use of strain gauges. The working principle of strain gauge pressure sensors is based on measuring the deformations of the sensitive element using wire or semiconductor strain gauges. During stretching or compression of the wire, its electrical resistance changes [1]. Fixing the signal of the change in electrical resistance, after recalculation with the help of coefficients, makes it possible to obtain the value of the thrust force created by the rocket engine as a result.

Scientific and technical results. One of the simplest methods of fixing the thrust value of a rocket engine is a rigid stand-type thrust measuring device, the general scheme of which, as well as an example of its implementation, is shown in Figure 1.

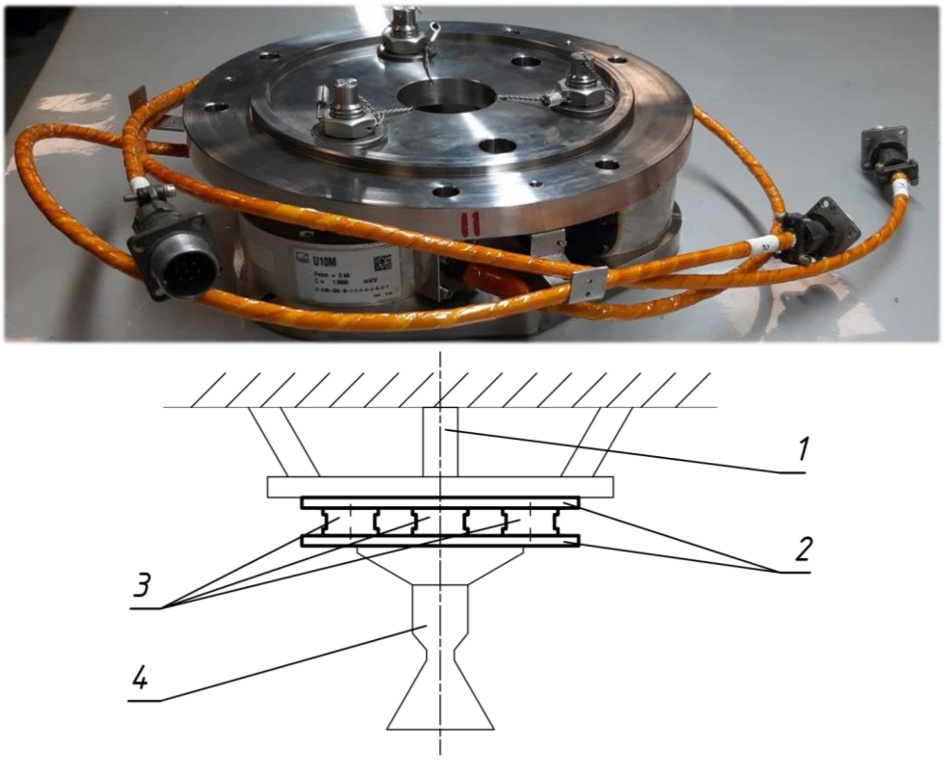

Figure 1 - Plate traction measuring device:
1 – power frame of the stand; 2 – plates of the traction measuring device; 3 – strain gauge sensors; 4- rocket engine.

From the diagram, it can be seen that, in general, the traction measuring device consists of two plates, strain-gauge force-measuring sensors and various fastening elements. The construction of this traction measuring device uses three U10M strain-gauge force-measuring sensors (2.5 kN), which work according to a bridge circuit with a balanced bridge. The zero position is fixed due to the settings in the software for recording the measurement results. A significant advantage of the selected type of sensors is that force measurement can be performed both in "plus" and "minus". This type of sensors was chosen for a small engine with a thrust of up to 5 kN. In order to eliminate the unevenness of the thrust distribution of the rocket engine, the force-measuring sensors are evenly spaced relative to each other. Exclusion of engine thrust asymmetry is ensured by the high accuracy of manufacturing, assembly and installation of the thrust measuring device on the stand, as well as by connecting the rocket engine to it.

The traction measuring device was located vertically in the axis of symmetry of the rocket engine. Options for positioning the device in other planes are also possible. In cases of positioning the traction measuring device in a horizontal plane or at an angle, a moment may occur, which in the future needs to be analyzed and taken into account, if the technical task for the development of the engine allows it. When using this type of traction measuring device, as well as others, it is important to take into account that the supply of fuel components to the engine or other auxiliary systems can affect the accuracy of the measurement of the traction measuring device. Therefore, it is necessary to use non-rigid connections, for example, using bellows for pipelines or other constructive solutions.

Conclusion. Among the advantages of the traction measuring device of the rigid stand type, the following can be noted:

- ease of manufacturing and installation at the workplace;
- ease of operation and processing of the received data (thrust values can be obtained in real time, using software),
- testing of the rocket engine can be carried out immediately, without the need for constant taring of the traction measuring device (the zero position is set by the software),
- reduction of costs for conducting tests of the rocket engine and production in general, due to the simplicity of the design;
- use for different types of engines, with different thrust, type of fuel, etc.

References:

1. Pron L.V., Podolchak S.M. Means of measurement during tests of rocket engines: Training. Manual - Д: РВВ ДНУ, 2004. - 60p.

UDC 534.1

Bekirov A., Chernyak M.

PNEUMATIC DAMPING OF SELF-OSCILLATIONS OF SENSITIVE ELEMENTS IN NAVIGATIONAL COMPENSATORY ACCELEROMETERS WITH A CAPACITIVE ANGULAR SENSOR

Introduction. Over the past century, the field of motion control systems has developed rapidly along with other areas of science and technology. A motion control system for any object is impossible without solving a navigation problem. Navigation-class accelerometers are high-precision devices that are complex to design and manufacture. Typically, devices of this class are used in long-range aircraft.

Statement of the problem. To date, pendulum accelerometers have two main schemes for the design of sensing elements: force and torque. They differ in the principle of placement of the springs.

One of the most important problems in the design of accelerometers is solving the problem of design optimization of sensing element. Damping of self-vibrations of sensing element is an integral part of the device development process, and its main need is to calculate the final output characteristic of the device in both time and frequency domains. The presence of large plate surfaces and small gaps between the moving and stationary parts in pendulous accelerometers makes gas damping the main type of damping of the self-oscillations.

Results of the study. The essence of the pneumatic damping mechanism in accelerometers is that the gas, located in the gap between the sensing plate and the base has its own viscosity, and when the plate moves at a certain speed, the gas begins to flow freely from the chamber, at which time viscous friction forces are created proportional to the plate movement speed [1]. The mechanism itself also depends on the type of gas chosen: compressed or non-compressed. A typical incompressible damping fluid would be silicone oil. For compressible fluids, it can be a gas such as nitrogen or helium. The temperature sensitivity of the dynamic viscosity of silicone oil is quite low. Helium has a higher thermal conductivity than air, which is useful for dissipating heat generated in the feedback coil. [2]

In general, the damping medium of the accelerometer is described by the system of Navier-Stokes differential equations and the continuity equation without taking into account the gas mass, dynamics (the process is considered as quasi-static) and vortex components. An analytical solution to the Navier-Stokes system of equations is impossible (but it is not proven yet), so the calculations will use the method of finding the damping coefficients by integrating the equations of gas motion in the damping chamber.

There is a limit between the dynamic range and bandwidth of MEMS accelerometers. When the input acceleration is relatively large, the damping of the compressed film will increase dramatically with the increase of the vibration amplitude, resulting in a decrease in the bandwidth. Conventional models still lack a complete analysis of the vibration response at large amplitude ratios and cannot offer appropriate guidance for optimizing such devices. Based on the solution equation of squeeze film damping, there are mainly two optimal damping schemes in MEMS accelerometers: optimization of the shape of the vibrating plate and control of the viscosity coefficient of the fluid. [3]

In terms of performance and over-regulation, the damping coefficient should be within the limits:

$\xi = 0,5...0,7$

Conclusion. Pneumatic damping of self-oscillations of sensing elements is used to achieve a finite accelerometer response time. The main design parameter that affects the value of the linear and angular damping coefficients is the initial gap.

References:

1. Chernyak M.G., Stetsenko O.V. Solution of the problem of gas damping of sensitive elements of microelectronic accelerometers. – Science journal «Technology and design in electronic equipment», №6, research institute of problems of mechanics «Rhythm», NTUU «KPI», 2001.
2. Hsu, T. R. MEMS and Microsystems. McGraw-Hill Science, Engineering & Mathematics, 2002, p. 106-133.
3. Xu, X., Wu, S., Fang, W., Yu, Z., Jia, Z., Wang, X., Bai, J., Lu, Q. Bandwidth Optimization of MEMS Accelerometers in Fluid Medium Environment. Journal "Sensors", 2022.
4. Lawrence A. Modern Inertial Technology. – Mechanical Engineering Series. Springer, New York, NY, 1998, 268p.
5. Bose A., Bhat K. N., Kurian T. Fundamentals of Navigation and Inertial Sensors. – 2014, 322 p.

UDC 629.7

Ibrahimoglu B., Kabanyachyi V.

ANGULAR MOTION CUEING ON FULL FLIGHT SIMULATORS

Introduction. Flight simulation may be defined as creating, in real-time under non-flight conditions, the performance and operation of a specific aircraft including its environment, which will respond to a pilot with the required fidelity to elicit pilot behavior as if he or she were flying the actual aircraft. Flight simulators have been applied principally to two applications: aircraft research and development and aircrew training.

An important technical means for solving complex problems of flight dynamics, researching aerobatic characteristics, and designing aircraft, which are practically not solved by other methods, is the flight simulator. Although piloting a flight simulator differs from piloting an aircraft, their use instead of an aircraft has significant advantages.

The main goal of the work is to solve, based on the current state analysis, trends in the development of simulators and progressive scientific concepts, the scientific and applied problem of ensuring adequate simulation of the entire set of angular motion cues perceived by the pilot in flight on non-maneuvering aircraft, which ensures the highest quality of motion cueing, effective ways of creating new and modernizing operated flight simulator. The method should also meet the modern requirements of engineering applications and be more efficient compared to existing ones.

Due to the high cost of motion systems and growing requirements for motion cue fidelity, it is necessary to develop an effective method of motion cueing along angular degrees of freedom on non-maneuvering aircraft. Perception of motion cueing should be as close as possible to perception of real motion cues.

Scientific and technical results. In order to achieve the set goal and solve the formulated problem, it is necessary to solve the following tasks:

- to formulate and solve the problem of angular motion cueing on full-flight simulators for non-maneuvering aircraft.
- to develop a method of angular motion cueing along individual degrees of freedom.
- to develop a method of optimal use of motion system constructive resources for angular motion cueing.

The theoretical and methodological basis of the research are methods of mathematical and simulation modeling, optimization theory, and control theory will be used as research methods.

Many investigations [1 - 4] of motion cueing were conducted in order to increase motion cueing fidelity. Motion cueing as in real flight is possible only with accurate reproduction of aircraft spatial motion. Due to limited constructive resources of flight simulators in comparison with aircraft resources, it is impossible to continuously monitor an aircraft's movement. On the other hand, only motion perception is important for pilots. Therefore, during motion cueing, it is important to consider not only the movement of the motion system itself but also the created motion cues and how accurately their perception in the flight simulator corresponds to real movements resulting from the same control actions.

So far, it has not been possible to build a mathematical model of the vestibular analyzer that would describe 100% of its reactions. A problem in the vestibular system study is the lack of a mathematical description of adaptation (a decrease in sensitivity to repeated motion cues). A mathematical model that takes into account the neurological nature of the vestibular system and the processing of receptor signals in the human brain can accurately describe human perception of motion cuein. A purely mechanical approach made it possible to create a mathematical model of the acceleration analyzer as a whole, which reflects more than 95% of vestibular reactions. However, the use of models describing only the perception threshold and the motion perception dynamics from motion cues can lead to the appearance of false motion cues. To avoid this, they need updating.

The research was conducted both on flight simulators and on non-maneuverable aircraft in real flight. Appropriate models of motion perception along linear degrees of freedom were constructed.

Conclusion. The proposed formulation of the problem of motion cueing along angular degrees of freedom shows the main directions of increasing of motion cue fidelity and, first of all, the development of an effective methodology for motion cueing along angular degrees of freedom.

References.

1. Preliminary assessment of influence of flight simulator mobility on the piloting process and analysis of compartment motion control methods: Report of the Research Institute/ TsAGI, No. GR 015110-1, inv. No. 655. - M. - 1970. - 34 p.

2. Aleksandrov V.V., Sadovnichy V.A., Chugunov O.D. Mathematical problems of dynamic flight simulation. - M.: Moscow State University. - 1986. - 181 p.

3. Bussolari S.R., Young L.R., Lee A.T. The Use of Vestibular Models for Design and Evaluation of Flight Simulator Motion // Collect. Techn. Pap. "AIAA Flight Simul. Technol. Conf. and Exhib.", Boston, Aug. 14-16, 1989. – Washington (D.C.). – 1989. – p. 86-93.

4. Hall, J.R. The Need for Platform Motion in Modern Piloted Flight Training Simulators, Tech Memo FM 35, Royal Aerospace Establishment, Bedford, UK, October 1989. – 16 p.

UDC 621.45.05

Gabrinets V., Bondarenko M.

SPEED CONTROL OF THE ROCKET AT THE END OF THE ACTIVE FLIGHT PHASE TO ENSURE THE SPECIFIED RANGE AND ACCURACY

Introduction. In the early 1990s, in Ukraine, work on the development of thrust termination systems was mainly focused on solid-fuel engines. Their feature was that the bodies of these engines were made using a solid winding scheme. In accordance with the purpose of the rocket engine thrust termination system, there are requirements, the most general of which are:

- Minimum firing time, ensuring minimum impulse afterwards;
- Absence of disturbing forces during thrust termination and harmful thermal
- impact on the rocket;
- Ability to perform thrust termination at any time interval;
- Independence of thrust termination from external conditions.

The principle of operation of the thrust termination system involves non-stationary processes of thrust reversal and extinguishing of the solid rocket fuel charge [1].

Scientific and technical results. Thrust termination in a solid-fuel engine can be achieved in several ways, including thrust termination through reversal and thrust termination through extinguishing. Thrust reversal is achieved by opening additional thrust termination nozzles that create thrust directed opposite to the engine thrust, equal to or slightly greater in magnitude. At the end of the transient process, the engine continues to operate at a new steady-state pressure level. The resulting engine thrust is subject to dependencies involving thrust termination pressure, steady-state combustion limit pressure, and pressure in the combustion chamber. The main disadvantages of thrust termination through reversal are significant forces hindering the rocket's movement and thermal effects on rocket components that occur after the thrust termination nozzles are opened [2].

Extinguishing the fuel charge of a solid rocket engine can be achieved by rapidly reducing the pressure in the combustion chamber below the steady-state combustion limit pressure. This can be achieved by opening an auxiliary critical cross-section for the outflow of combustion products or by introducing a cooling agent (coolant) into the engine chamber. The drawback of this thrust termination system is an increase in the weight of the structure and the possibility of unauthorized engine restart due to heating of the charge burning surface by radiation from the heated engine parts after termination. The following designs of the thrust termination node are proposed for use:

- A thrust termination node with nozzle attachment in a plastic shell in front of
- the bottom of the engine body;
- A thrust termination node located on the central gas manifold, which is attached
- with studs to the front flange of the body, simultaneously serving as the front
- cover of the engine;

- A thrust termination node for a plastic body located on the front metal bottom;
- A thrust termination node for a metal body connected by splined joint [3],[4].

Conclusion. The issue of controlling the speed of a rocket with solid fuel at the end of the active flight phase to achieve the specified range and accuracy has been considered. The work conducted in Ukraine in the 1990s was primarily focused on the development of thrust termination systems for solid-fuel rocket engines. The thrust termination system is of great importance for ensuring the effective flight of rockets. To optimize the thrust termination system, it is recommended to use structural nodes taking into account the specific characteristics of the particular type of rocket and the tasks it must perform.

References:

1. V.E. Alemasov, "Theory of Solid Rocket Engines" Moscow, Mashinostroenie, 1982.

2. I.Kh. Fakhrutdinov, "Solid Rocket Engines" Moscow, Mashinostroenie, 1987.

3. B.V. Orlov, G.Yu. Mazing, "Thermodynamic and Ballistic Foundations of SPR Design" Moscow, Mashinostroenie, 1985.

4. B.I. Kushnir, V.E. Plugin, "Control Elements of the Thrust Vector of SPR" Educational and Methodological Manual, YSDO, 1986.

UDC

Chernenko S., Burnashev V.

VISUAL NAVIGATION SYSTEM ALGORITHM

Introduction. In the foundation of modern navigation systems for unmanned aerial vehicles (UAVs), cameras are increasingly becoming prevalent. The diversity of such cameras in practical applications is often limited to optical and infrared cameras. They play a crucial role in navigation systems, providing a reliable source of information for onboard Strapdown Inertial Navigation Systems (SINS). An optical visual sensor is an essential component of a contemporary navigation system as it can partially or completely address the issue of error accumulation in SINS over time through specialized computational algorithms. A significant drawback of such algorithms is their computational complexity, limiting their real-time operation onboard UAVs.

In the works of Malysheva Y.O. and Zbrutsky O.V. [1, 2], an algorithm for orientation based on determining the horizon line from an optical camera is described, simulated, and experimentally validated. However, this algorithm does not allow for the determination of the UAV's roll. Additionally, its limitation lies in the inability to be used in conditions where the horizon is not visible, or it is obstructed, in mountainous terrain, etc.

The aim of this work is to develop an orientation algorithm for long-duration UAV flights using a camera capable of determining three angles in complex terrain, even in situations where the horizon line is not observable.

Scientific Results. By utilizing the block of angular velocity sensors (AVS) and the classical SINS orientation algorithm, the orientation angles can be determined. The use of a visual navigation block and an extreme algorithm for error determination helps limit errors in their calculation. Correction for each angle is performed separately. For the correction of the initially selected angle, it is first necessary to use the inertial system's information about the other two angles. Then, the current image from the camera is rotated by the selected angles. Thus, the current image remains, shifted only by one orientation angle. By comparing it with the reference (initial) image, the error in determining the SINS orientation angle can be established. The mentioned algorithm is executed for each angle in a cycle, and the interval between such corrections is determined by the specific task and errors in the AVS block.

An important aspect expediting the optimization of the chosen cost function in the algorithm is the initialization procedure based on the current orientation angle value from the SINS algorithm.

Conclusions. The proposed algorithm and error model allow determining all orientation parameters of the UAV for its long-duration flight. With the help of the extremal optimization algorithm component, it is possible to cyclically correct the error of the inertial orientation algorithm in real-time on board with limited computational resources.

References:

1. Malysheva Y.O. - Integrated navigation and orientation system of an unmanned aerial vehicle with optical instruments. Kyiv, 2016. 116 pages.

2. Zbrutsky A., Malycheva Y. - Experimental tests of an algorithm of roll and pitch estimation using only optical information / A. Zbrutsky, J.A. Malysheva // Mechanics of Gyroscopic Systems. – Kyiv: NTUU "KPI", 2015. - Issue 29. – P. 5-14. – ISSN 0203-3771.

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Dymarchuk Y., Marynoshenko O.

DEVELOPMENT OF A UAV FOR VERTICAL TAKE-OFF AND LANDING

Introduction. In today's world, structural changes in the field of aviation technology necessitate the search for innovative solutions to improve the functionality and efficiency of airborne platforms. In particular, unmanned aerial vehicles (UAVs) define a new stage in the development of aviation systems, providing extreme flexibility and unique capabilities in a number of applications.

The relevance of studying VTOL (vertical take-off and landing) UAVs is determined by the growing need for high-precision, manoeuvrable and remote systems that provide real-time monitoring, data collection and processing.

Scientific and technical results. Having analysed the classical schemes of UAVs [1], namely, the aircraft type and the multi-rotor type, a number of advantages and disadvantages were identified. Aircraft-type UAVs perform well in long-term missions, can carry a fairly heavy payload, and have the ability to fly long distances at relatively high speeds. The most painful issue is take-off and landing, as there is a need for either a runway or special launch equipment.

Multi-rotor UAVs are actively used in almost every field, gaining popularity for their relatively low cost and ease of use. Among their advantages is the ability to hover at high altitudes, they are always manoeuvrable and easy to use. However, due to the way they operate, multirotor have a relatively short operating time, low cruising speed, and limited payload.

A good alternative solution is the VTOL UAV, as it combines the advantages of both aircraft and multirotor UAVs. There are three basic schemes [2]: helicopter type, wing with a rotary power unit mechanism, and wing without a rotary power unit mechanism. The third scheme is the most popular, as it is relatively simple and reliable, and it was chosen as the main scheme. This scheme involves an aircraft-type airframe, four electric motors responsible for the vertical component of the flight and one motor responsible for cruising.

The potential of VTOL UAVs can be unlocked by three fundamentally different schemes for implementing the propulsion system [3]. The first is fully electric, i.e. based on electric motors powered by a single battery. The second is a hybrid system, which includes an internal combustion engine, a generator for charging the battery, and four electric motors for take-off and landing, respectively, and can be powered by either the same internal combustion engine or an additional electric motor. The third scheme is based on the principle of parallel power plants, i.e. separate horizontal (ICE) and vertical (electric motors).

The analysis of the efficiency of each scheme showed that it is advisable to use the all-electric scheme at a distance of up to 50 km and with a payload not exceeding 5 kg, since with an increase in these indicators, the weight of the airframe and batteries increases, which leads to low efficiency of this type of UAV. The parallel scheme is relevant if the operating modes do not involve long hovering or long ascent using the part of the power plant responsible for take-off and landing, as this requires a battery of greater capacity and, accordingly, weight. The hybrid propulsion system is the most efficient, as the generator charges the battery if necessary, the main thing is to design the internal combustion engine so that there is no power shortage during cruise flight, the feasibility of this scheme is maximum when the take-off weight is more than 25 kg and the payload share is within 15-20%.

According to the analysis of existing VTOL UAV power plant schemes, the features of individual components and assemblies were taken into account to ensure the most successful layout of the aircraft.

Conclusions. The research on this topic plays an important role in the aircraft industry, since the effective selection of the type of UAV, its power plant and layout is a key factor in finding an economically successful and practically effective solution when designing a UAV.

As a result of this study, it can be argued that the introduction of a hybrid power plant in VTOL-type UAVs opens up new prospects for the development of unmanned aerial vehicles, providing high efficiency, versatility and safety in a single complex.

References:

1. Boon, M. A., Drijfhout, A. P., and Tesfamichael, S.: COMPARISON OF A FIXED-WING AND MULTI-ROTOR UAV FOR ENVIRONMENTAL MAPPING APPLICATIONS: A CASE STUDY, Int. Arch. Photogramm. Remote Sens. Spatial Inf. Sci., XLII-2/W6, 47–54

2. Yu, S. , Heo, J. , Jeong, S. and Kwon, Y. (2016) Technical Analysis of VTOL UAV. Journal of Computer and Communications, 4, 92-97. doi: 10.4236/jcc.2016.415008.

3. H. Gu, X. Lyu, Z. Li, S. Shen and F. Zhang, "Development and experimental verification of a hybrid vertical take-off and landing (VTOL) unmanned aerial vehicle(UAV)," 2017 International Conference on Unmanned Aircraft Systems (ICUAS), Miami, FL, USA, 2017, pp. 160-169, doi: 10.1109/ICUAS.2017.7991420.

4. McEvoy JF, Hall GP, McDonald PG. 2016. Evaluation of unmanned aerial vehicle shape, flight path and camera type for waterfowl surveys: disturbance effects and species recognition. PeerJ 4:e1831

UDC

Havaza O., Sukhov V.

SYSTEM ENGINEERING AND MULTIDISCIPLINARY OPTIMIZATION AT PRELIMINARY STAGE DESIGN OF FLIGHT VEHICLE LIFTING SURFACE.

Introduction, Background and Objectives: The relentless pursuit of enhanced performance and efficiency within the aerospace industry necessitates the adoption of advanced methodologies throughout the design process. This thesis is dedicated to the crucial integration of system engineering principles and multidisciplinary optimization (MDO) techniques, with a specific emphasis on aeroelasticity.

The complexity of flight vehicle design is evident in the dynamic interactions of various disciplines, each contributing to the overall performance of the aircraft. Lifting surfaces, particularly wings, hold a central role in determining aerodynamic efficiency and overall flight characteristics.

The research aims to demonstrate the effectiveness of incorporating aeroelastic considerations in the early stages of flight vehicle design including system engineering principles and MDO techniques.

Scientific Results: Applying the core tenets of system engineering [1], our research initiated a meticulous decomposition of the aircraft wing design process, yielding a detailed flowchart (refer to Figure 1). This method unveiled three distinctive design loops, each iteration strategically incorporating the exchange of aerodynamic and structural elasticity data. The efficacy of this exchange, crucial for optimal results within specified requirements, was facilitated by contemporary software tools.

Upon examination of existing calculation method and tools [3] for aeroelasticity, we identified a reliance on digital calculations and observed a significant gap: a lack of explicit dependencies between wing performance and design parameters. While these tools excel in facilitating data exchange, their limitation lies in providing a transparent link between design choices and resulting performance metrics. This observation underscores the necessity for a paradigm shift to address this critical gap and enhance the comprehensibility of aeroelastic design processes.

Shifting focus to the second part of our results, our investigation emphasized the integral role of MDO in enhancing flight vehicle performance. Despite the modernity of existing tools [2], their effectiveness is hampered by the absence of explicit dependencies between design parameters and wing performance.

To overcome this limitation, we propose a groundbreaking shift towards a new mathematical model based on a computer algebra system. This model aims to rectify the shortcomings of current digital calculation-based software and elevate the efficiency of MDO. By providing explicit dependencies, our proposed model strives to unlock new frontiers in the precision and effectiveness of the optimization process. This approach addresses the current gaps in MDO, making it more effective and transparent by explicitly linking design parameters to the performance of flight vehicle designs.

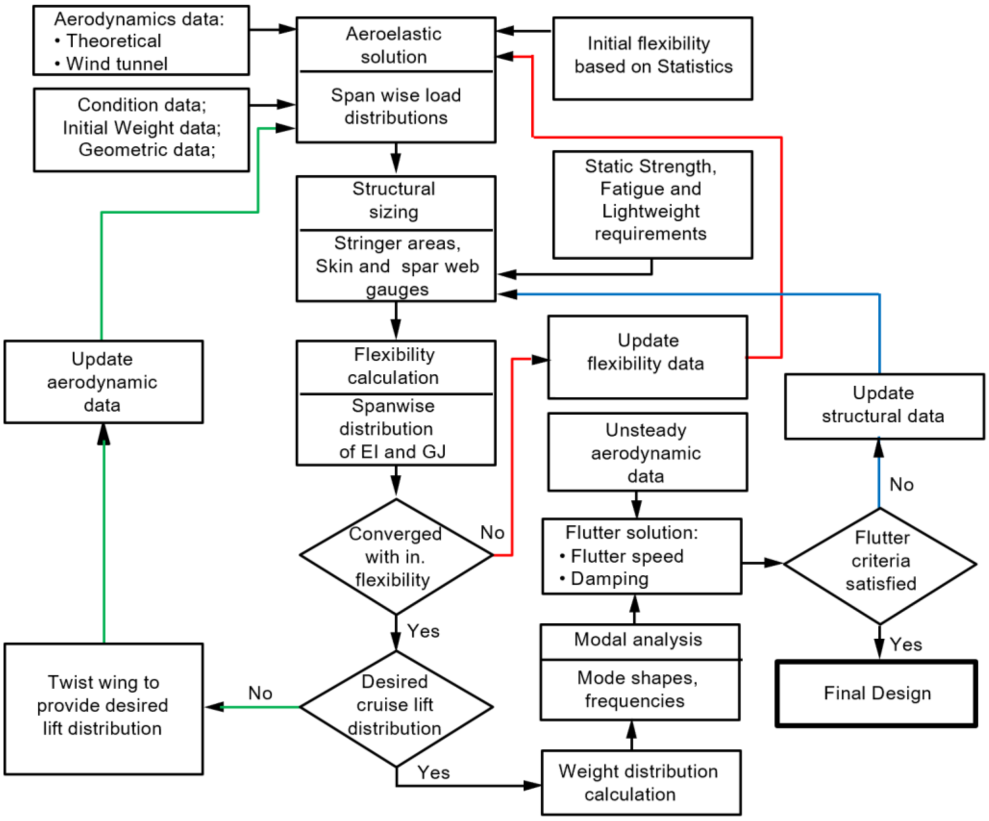

Figure 1

Conclusion: In conclusion, our study underscores the need for a paradigm shift in flight vehicle design. Integrating system engineering and MDO reveals crucial insights. Existing aeroelasticity tools lack explicit dependencies, urging a more transparent approach. Proposing a new mathematical model for MDO using a computer algebra system holds the promise of revolutionizing precision and efficiency in the design process. This integrated approach has the potential to enhance transparency and effectiveness by explicitly linking design parameters to flight vehicle performance.

Reference:

1. Mohammad H. Sadraey., Aircraft Design. A Systems Engineering Approach. Daniel Webster College, New Hampshire, 2013.

2. Abdelkader Benaouali, Stanisław Kachel, Multidisciplinary design optimization of aircraft wing using commercial software integration, Warsaw, Volume 92, September 2019, Pages 766-776

3. M.J.Rycroft, Introduction to Structural Dynamics and Aeroelasticity. Cambridge University, 2005

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Ince B., Ortamevzi G.

ANALYSIS OF THE LIFE OF THE MAIN LANDING GEAR WHEEL BEARINGS OF AN EXPERIMENTAL AIRCRAFT ACCORDING TO THE USE OF FLAPS

Introduction. In this study, bearing life analyzes were carried out in order to predict the bearing replacement periods of the main landing gear wheels of an unmanned aerial vehicle produced for an experimental purpose. The operating conditions of the bearing were simulated after the loads on the landing gear with and without flaps and landing gear of different weights were defined as bearing radial and axial loads and the average bearing rotation speed.

Materials and Methods. Figure 1 shows the structure of the landing gear and the location of the ground load. Ball bearing sizes 6204 and 6205 were used to fit tightly onto the landing gear shaft. Life analyzes of bearings subjected to ground loads were carried out.[1][2].

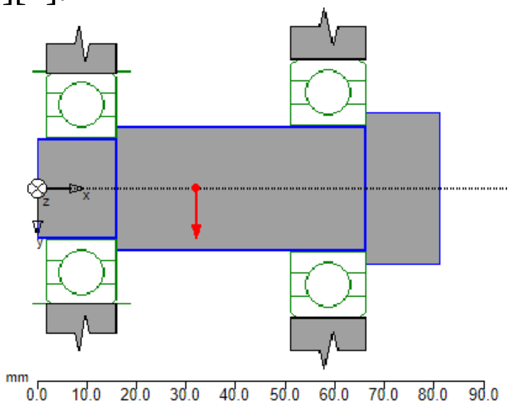


Figure 1 -The structure of the landing gear shaft and the location of the ground load

Theoretical bearing life calculation. There are some theories for predicting bearing life calculations. The most important of these theories is known as the Hertz Contact Stress theory. However, according to the Weibull Analysis, the prediction of bearing life calculations can be predicted statistically[3] According to Formula 1, bearing life can be predicted simply.

L_{10} is million revolving number of total at 90% fatigue life estimate of the bearing that calculation as follows:

$$L_{10} = \left(\frac{C_D}{P_{eq}}\right)^3 \quad (1)$$

Empirically modified the dynamic load capacity CD for ball bearings as follows:

$$C_D = f_c \frac{-\mathrm{i}d^2 Z^{2/3} cos\beta}{1+0.02\mathrm{d}} \quad (2)$$

where,

f_c - material-geometry coefficient;

i - number of rows of rolling elements (balls or rollers);

d - ball or roller diameter;

Z - number of rolling elements (balls or rollers) in a row;

β - bearing contact angle.
Equivalent Load as follows;

$$P_{eq} = XF_r + YF_a \tag{3}$$

where,
P_{eq} - the equivalent load;
F_r - the radial component of the actual load;
F_a - the axial component of the actual load;
X - a rotation factor;
Y - the thrust factor of the bearing.

Although bearing life calculation programs give adequate results for theoretical comparison. Preloading is applied to the deep groove bearings to hold under control the vibration and noise performance [4][5][6][7].

Results. Average rpm values and ground loads of the bearings were calculated according to the aircraft's landing speed with and without flaps. In the light of these values, bearing life analyzes were carried out for both the 6204 bearing and the 6205. Figures 2 and 3 show the program analysis results. The data is presented graphically in Figure 4.

Bearing	Lh10 (xy) h	L10 (xy) 10^6 U	S0_xy_min	n 1/min
6204	4495	1049	4.932	3891.0
6205	9038	2110	6.714	3891.0

Table Explanations:
Lh10 (xy): Catalog rating life per DIN ISO 281
L10 (xy): Catalog rating life per DIN ISO 281
S0_xy_min: Static safety (Catalog)
n: Equivalent speed

Bearing	Lh10 (xy) h	L10 (xy) 10^6 U	S0_xy_min	n 1/min
6204	2601	607.3	4.110	3891.0
6205	5230	1221.0	5.595	3891.0

Table Explanations:
Lh10 (xy): Catalog rating life per DIN ISO 281
L10 (xy): Catalog rating life per DIN ISO 281
S0_xy_min: Static safety (Catalog)
n: Equivalent speed

Figure 2 - Landing simulation with flaps 110 km/h; 2500N ground load on the left, 3000N on the right

Bearing	Lh10 (xy) h	L10 (xy) 10^6 U	S0_xy_min	n 1/min
6204	3803	1049	4.932	4599.0
6205	7647	2110	6.714	4599.0

Table Explanations:
Lh10 (xy): Catalog rating life per DIN ISO 281
L10 (xy): Catalog rating life per DIN ISO 281
S0_xy_min: Static safety (Catalog)
n: Equivalent speed

Bearing	Lh10 (xy) h	L10 (xy) 10^6 U	S0_xy_min	n 1/min
6204	2201	607.3	4.110	4599.0
6205	4425	1221.0	5.595	4599.0

Table Explanations:
Lh10 (xy): Catalog rating life per DIN ISO 281
L10 (xy): Catalog rating life per DIN ISO 281
S0_xy_min: Static safety (Catalog)
n: Equivalent speed

Figure 3 - Landing simulation without flaps 130 km/h; 2500N ground load on the left, 3000N on the right

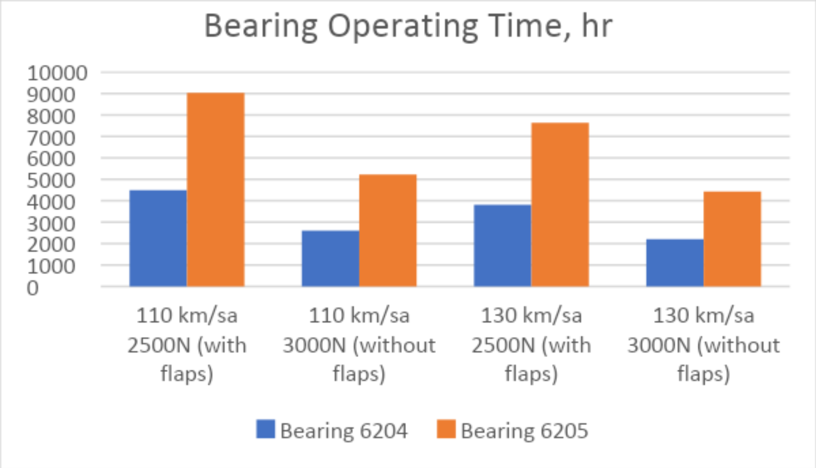


Figure 4 - Bearing operating time

Conclusion. Bearing life is longer in landings with flaps than in flapless landings.

Increasing ground loads dramatically reduce bearing life.

If different bearings of sizes 6204 and 6205 are used in the landing gear shaft, the life of the larger bearing is twice that of the smaller bearing. Therefore, it is envisaged that the smaller bearing can be replaced more easily during the design.

References:

[1] O. Asi and Ö. Yesil, "Failure analysis of an aircraft nose landing gear piston rod end," *Eng. Fail. Anal.*, vol. 32, pp. 283–291, 2013.

[2] D. Turan, B. Karabayrak, S. Baskut, and S. Dalkilic, "Failure analysis of landing gears strut bearings," *Eng. Fail. Anal.*, vol. 82, pp. 525–532, 2017.

[3] E. V Zaretsky, "Rolling bearing life prediction, theory, and application," *NASA Tech. Reports*, no. November 2016, p. 66, 2013, [Online]. Available: http://www.sti.nasa.gov.

[4] T. A. Harris, *Rolling bearing analysis*. John Wiley and sons, 2001.

[5] T. IGARASHI, "Noise of ball bearing in electric motor," *Bull. JSME*, vol. 7, no. 25, pp. 200–208, 1964.

[6] M. Jafarboland and H. B. Farahabadi, "Optimum design of the stator parameters for noise and vibration reduction in BLDC motor," *IET Electr. Power Appl.*, vol. 12, no. 9, pp. 1297–1305, 2018.

[7] Y. Fang and T. Zhang, "Vibroacoustic characterization of a permanent magnet synchronous motor powertrain for electric vehicles," *IEEE Trans. Energy Convers.*, vol. 33, no. 1, pp. 272–280, 2017.

UDC 536.24

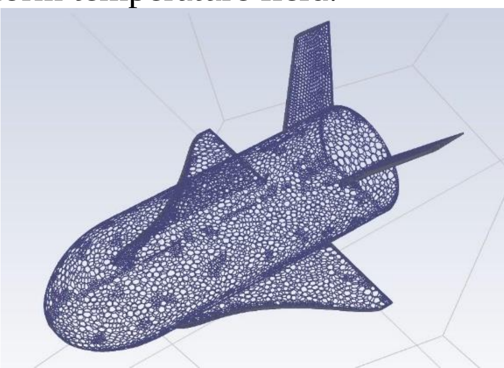
Iskra O., Lobunko O.

CALCULATION OF THE PASSIVE THERMAL PROTECTION SYSTEMS ELEMENTS FOR THE REUSABLE SPACECRAFT

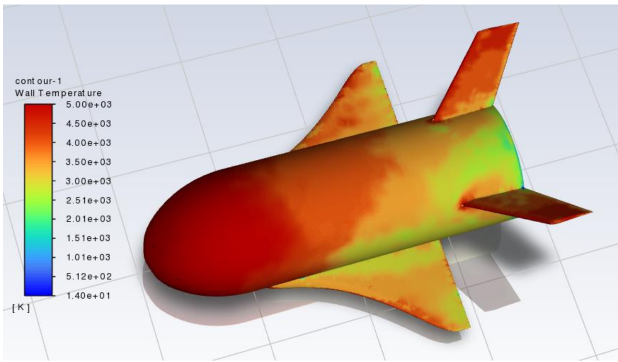
Introduction. A necessary part of the spacecrafts' designing is its thermal designing, which goal is to ensure maintaining necessary thermal conditions during the spacecraft's operational time. For the reusable spacecraft, it is especially important to ensure protection against aerodynamic heating during the atmospheric reentry. To solve essential problems in the thermal protection systems designing, mathematical and computer modeling is widely used, a common method of which is the finite element method.

Scientific and technical results. Based on the [1, 2, 3, 4, 5, 6] experience generalization, the following algorithm for calculation of the passive thermal protection systems elements for the reusable spacecraft during the atmospheric descent is formulated: 1) input of geometrical and physical parameters of spacecraft's elements, its protection systems (finite element model) and the environment; 2) calculation of the heat flows and spacecraft's surface temperature T_w; 3) if $T_w \geq$ to the ablation removal temperature T_{abl}, calculation of the insulation thickness $\Delta\delta_i$ and mass Δm_i change due to the ablation; 4) calculation of covering thermal conditions $T_i(x, y, z, \tau)$; 5) output of the covering temperature fields in the calculation time $T_i(t)$; 6) analysis of the thermal protection effectiveness ($N_i(T_i)$, m_i, $P_i(T_i)$); 7) if the effectiveness is considered unacceptable, change of the spacecraft's protection systems parameters and recalculation.

The study of the perspective models of the spacecraft's protection systems elements is performed using computer and mathematical modeling. The spacecraft is presented as a finite-element model (Fig. 1), discretized into elements with a uniform temperature field.



a)



b)

Figure 1 - Examples of spacecraft's thermal conditions modeling (author's work):
a) finite element model of the spacecraft;
b) thermal conditions of the spacecraft's surface during atmospheric reentry

The optimal model's variant should meet the requirements of the necessary thermal fields in the structural elements $T_i(x, y, z, \tau) \rightarrow min$, required number of load cycles $N_i \rightarrow max$, minimal mass $m_i \rightarrow min$ and minimal damage $P_i \rightarrow min$ by various ways in the expected operating conditions during the flight time τ.

Based on the research, the main requirements for thermal protection materials of prospective aircraft were formulated: high heat resistance; the ability to radiate a significant part of the convective heat flow; minimum thermal conductivity coefficient; minimum coefficient of thermal expansion; minimum mass; resistance to the external climatic influences.

Conclusion. The evaluation of the effectiveness of possible ways to ensure the thermal operation conditions for the reusable spacecraft and its systems is implemented using mathematical and computer modeling, the main method of which is finite element modeling. The algorithm for calculation of the passive thermal protection systems elements for the reusable spacecraft includes calculations of the heat flows near its surface, temperature of the covering and the amounts of the thermal protection insulation's ablative removal, which allows to analyze the effectiveness of the spacecraft's thermal protection insulation.

References:

1. Brociek, R., Hetmaniok, E., Słota, D. (2022). Reconstruction of aerothermal heating for the thermal protection system of a reusable launch vehicle. Applied Thermal Engineering, 219. URL: https://doi.org/10.1016/j.applthermaleng.2022.119405

2. Dang, D.Z. (2021). Thermal and Structural Response Modeling of a Woven Thermal Protection System. The University of Michigan.

3. Fortescue, P.W., Swinerd, G., Stark, J. (2011). Spacecraft systems engineering (4th ed.). John Wiley & Sons, Ltd.

4. Piacquadio, S., Pridöhl, D., Henkel, N., Bergström, R., Zamprotta, A., Dafnis, A., & Schröder, K.-U. (2023). Comprehensive Comparison of Different Integrated Thermal Protection Systems with Ablative Materials for Load-Bearing Components of Reusable Launch Vehicles. Aerospace, 10, 319. URL: https://doi.org/10.3390/aerospace10030319

5. Xu, Q., Li, S., & Meng, Y. (2021). Optimization and Re-Design of Integrated Thermal Protection Systems Considering Thermo-Mechanical Performance. Applied Sciences, 11. URL: https://doi.org/10.3390/app11156916

6. Lobunko O., Iskra O. Mathematical Modeling of the Thermal Conditions of Aerospace Products' Protection Systems. *Міжнародна науково-технічна конференція «Датчики, прилади та системи – 2023»*: матеріали Міжнар. наук.-техн. конф., Україна, Черкаси, 12-14 вересня 2023. С. 8-10.

UDC 629.7

Ismahilova B., Arkhipov O.

ROCKET FOR LAUNCHING NANOSATELLITES

Introduction. Space is increasingly taking a significant place in our lives, especially during times of war. Satellites now serve not only for communication, exploring landscapes, and weather forecasting but also play a crucial role in vital intelligence aiding our soldiers. During times of major conflicts, they play a crucial role in communication and transmission of essential data. Therefore, Ukrainians need to find a solution to better, faster, and safer utilization of satellites. Creating a satellite constellation by deploying several small but powerful space vehicles - nanosatellites - into orbit can address this demand. The key question is how to achieve this, and for that, we need a domestically produced rocket.

Scientific and Technical Results. In rocket design, a crucial factor is the payload that needs to be delivered, the orbit altitude, and the desired characteristics of the rocket we aim to achieve.

Payload. The valuable payload will be nanosatellites – satellites weighing from 1 to 10 kilograms, capable of forming constellations, sometimes comprising up to 100 units. They can remain in orbit for up to 25 years. As noted by Ben Pilkington [1], nanosatellites have already contributed to several significant research endeavors, from daily tracking of Earth's surface for natural disaster prediction to understanding the impact of radiation on human DNA. Furthermore, the risk of losing research data is minimized, as if one satellite in the constellation fails, it can be replaced by another, ensuring the functionality of the overall constellation. Additionally, their cost is relatively low, only around £500,000, compared to conventional satellites that can cost approximately £500 million.

Modeling. When modeling a rocket, it is crucial to understand the risks it will face during its launch into orbit. The primary propulsive force for the rocket is its engine, tasked with generating the necessary thrust for the rocket to easily overcome Earth's gravity. This can be influenced by both the size of the engine and the fuel it carries. Additionally, the engine undergoes loads from internal pressure, thermal stresses, thrust loads, and loads from centrifugal forces. Analyzing all these risks, Ibrahim Hossam, Sherif Saleh, and Hisham Kamel [2] concluded that the best solution to increase payload and flight range is for the rocket engine to have sufficient strength and light weight, achievable by replacing conventional materials with composites. Despite some drawbacks, they can reduce weight by 23-28%, a significant improvement.

Economy. When modeling a rocket, it's crucial to remember that besides its primary mission of deploying payload into orbit, it must be economically viable to create and use. In their article, Tina C. Highfill and Alexander C. MacDonald [3] noted that the manufacturing gross output in the US space economy from 2012-2018 was approximately 50 billion dollars annually, while Ukraine's space agency received only 88.5 million dollars in 2018. However, the creation and deployment of small satellites led to price reductions in space-related sectors.

Conclusions. Considering the above, one can conclude that rocket modeling is quite complex from both an economic and technological standpoint. However, substituting conventional materials with composites allows for a reduction in the engine's mass by 23-28%. Nano-satellites, due to their size, are a thousand times cheaper than traditional ones, making them economically more advantageous and an ideal replacement for many existing satellites.

Therefore, it is essential to improve technological processes, explore more cost-effective ways of material usage, and investigate innovative methods to enhance the efficiency of rocket manufacturing.

References:

1. Pilkington B. What are nanosatellites?. AZoNano. 2021. URL: https://www.azonano.com/article.aspx?ArticleID=5840.
2. Hossam I., Saleh S., Kamel H. Review of challenges of the design of rocket motor case structures. IOPscience. 2019. URL: https://doi.org/10.1088/1757-899X/610/1/012019.
3. Highfill T. C., MacDonald A. C. Estimating the united states space economy using input-output frameworks. ScienceDirect. 2022. Vol. 60. URL: https://doi.org/10.1016/j.spacepol.2021.101474.

UDC

Kaczmarek K., Kopyt A.

URBAN AIR MOBILITY TRAFFIC ANALYSIS TOOL

Introduction. The Urban Air Mobility (UAM) concept might be one of the solutions to the transportation issues arising from constantly growing cities and increasing numbers of citizens. It includes various types of aircrafts, varying from small unmanned aerial vehicles (UAVs), carrying small packages, to personal air vehicles (PAVs), able to quickly transport passengers from the origin to the destination. As opposed to traditional means of transport, aircrafts are not dependent on the current road traffic which allows them to get to their destination more quickly and with less disruptions. It is a very attractive concept to the inhabitants and can highly improve their quality of life. However, it brings a lot of new challenges when compared to traditional aviation which has over 100 years of experience in creating a safe and efficient environment for its users. Therefore, there exists a need to test new concepts of UAM traffic control. The Urban Air Mobility Traffic Analysis Tool (UAM TAT) aims to aid in this task by allowing users to simulate the envisioned environment of UAM vehicles.

Scientific and technical results. UAM TAT was created with the use of MATLAB/Simulink software. The usage process consists of four main stages:

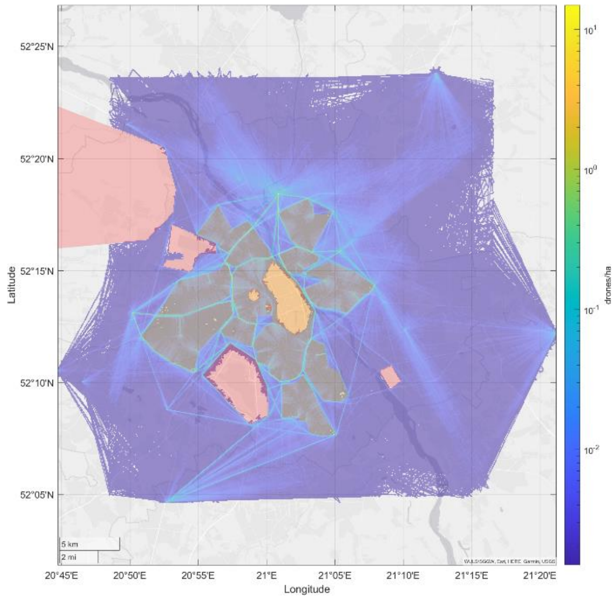
- input data generation,
- input data processing,
- simulation,
- output data generation.

The first stage focuses on manual preparation of a JSON file with the declarations of drones' families' ranges of parameters. These ranges are then used to generate data of individual aircrafts by random draws based on the applied distributions. Examples of parameters that need to be present in the input files are size, velocity, range, altitude, autonomy level, failure rate, etc.

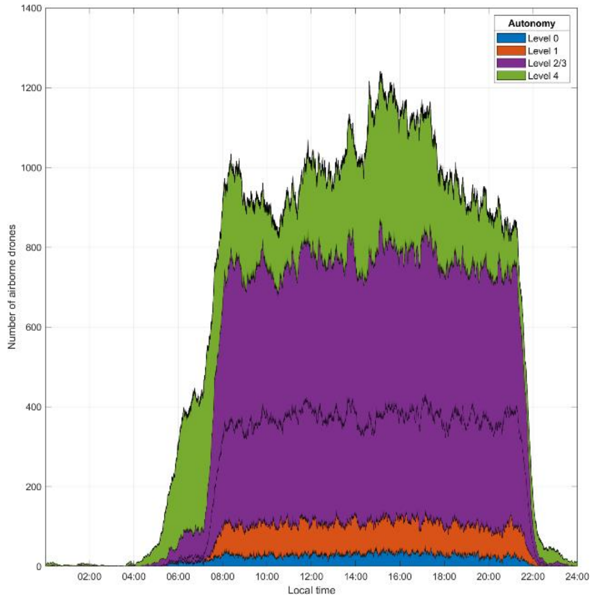
The second step is done automatically by the program. Its most important part is the path planning algorithm A* which considers no-fly zones in which the drone is not allowed to fly at all and restricted zones. The drone is allowed to enter the restricted fly zone in a way that minimizes the time it spends there but only if one of its waypoints is in the zone.

The simulation module consists of a simple 3DOF drone model. The simulation is performed with a constant time step. The program can detect collisions between drones which can be prevented if at least one of the drone's autonomy levels is high enough. Another functionality of UAM TAT is the ability to simulate drones' malfunctions based on the failure rate of individual aircrafts. There are several types of malfunctions implemented: free fall, autonomy level decrease and transition from waypoints following to loitering.

Output data consists of several types of plots, one video and one text file. The program outputs a heatmap of the drones' presence overlayed on top of a map of the simulated area (see Fig. 1a). The second plot shows the number of airborne drones as a function of time (with different autonomy levels included) (see Fig. 1b). The third plot shows a histogram of drones' collisions and avoidance manoeuvres. The video is an animation of the aircrafts' flight overlayed on top of the map. There is also one text file which contains the most important data about the simulation such as the time when it was started, how long it took to perform it, how many drones were included, how many collided, etc.



a. Heatmap of airborne drones



b. Plot of the number of airborne drones as a function

Figure 1 - Examples of output files

Conclusion. The developed program – UAM TAT – due to its capabilities can help in developing new air traffic management rules, as well as in testing the influence of introducing tens of thousands of aircrafts into the urban airspaces. It presents the user with aggregated data regarding the simulation that aims to provide a concise information about the aircrafts' performance in the simulated case.

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Khylenko V., Bobkov Y.

CONTROLLING THE MOVEMENT OF A MULTICOPTER IN PAIRS

Introduction. At the present stage, the range of tasks solved by unmanned aerial vehicles (UAVs), particularly of the multi-copter type, is constantly expanding. The expansion of the scale of tasks leads to the necessity of employing entire groups of UAVs that perform coordinated actions [1]. This requires addressing the issue of their organized movement for the correct interaction in space during the execution of the assigned task.

The conducted analysis indicates that the decisive factors include the selection of an appropriate method for group control, as well as the spatial arrangement of multi-copters relative to each other and maintaining it throughout the entire flight [1]. The basic element of the group is a pair consisting of a leader and a follower [1,2].

The objective of the work is to develop a control system that ensures coordinated movement of multi-copters in a pair.

Scientific and technical results. The leading multi-copter is guided either by control signals from the ground station or by a predefined algorithm. The follower should move in coordination with the leader along a predetermined trajectory, considering the pre-defined spatial positioning relative to it.

The primary challenge that needs to be addressed lies in identifying the spatial position of the leader and generating appropriate control signals for the follower. To determine specific environmental features, the work [3] suggests employing additional sensors for unmanned aerial vehicles (UAVs). Furthermore, for identifying target objects and obstacles in the control of small unmanned aerial vehicles, it is advisable to utilize computer vision systems [4].

The conducted analysis allows us to conclude that, to ensure the movement of multi-copters in pairs, it is advisable to implement:

- Special markers on the leader for identifying its position relative to the follower.
- Computer vision system for tracking the leader, located on the follower.
- Rangefinder for precise determination of the distance to the leader, placed on the follower.

The main parameters of motion for the follower include pitch, yaw, roll angles, and the distance to the leader. To determine the angles, a computer vision system is proposed, while the distance is measured using a rangefinder. In this case, the optical axis of the camera and the measuring axis of the rangefinder practically coincide. The relative position of the follower to the leader is set on the ground by pre-adjusting the camera at the corresponding angles.

The analysis showed that it is advisable to use infrared LEDs as markers on the leader [5]. This allows for recognizing the leader in changing lighting conditions and in the presence of external atmospheric disturbances. In the process of analyzing the possible placement of infrared LEDs, the decision was made to position them on the frame of the multi-copter with an extension beyond the main structure.

A camera sensitive to infrared light and a microcomputer for image processing are installed onboard the follower. Additionally, a stabilizing gimbal with feedback is employed, housing both the camera and rangefinder.

Using images of the infrared markers captured by the camera, the microcomputer's image processing system determines the deviation of current directional angles from the specified ones. Simultaneously, the rangefinder measures the distance to the leader. The obtained information is utilized to adjust the parameters for controlling the motion of the follower behind the leader, specifically the pitch, yaw, roll angles, and speed.

Conclusion. The use of groups of unmanned aerial vehicles enhances efficiency and speed in performing various tasks, making the development of group control systems a highly relevant direction.

In this study, a control system for the movement of multi-copters in pairs was developed, using infrared markers on the leading vehicle to determine directional angles through a computer vision system. Computer modeling and practical experiments using a quadcopter frame simulator with placed markers confirmed the viability of the proposed solutions.

References:

1. Tal Shima, Steven Rasmussen, UAV Cooperative Decision and Control Challenges and Practical Approaches – SIAM, 2009

2. Mercado D. A. Castro R. Lozano R. Quadrotors Flight Formation Control Using a Leader-Follower Approach. 2013 European Control Conference (ECC) July 17-19, 2013, Zürich, Switzerland. – P.3858-3863

3. Floreano D., Wood R. J. Science, technology and the future of small autonomous drones //nature. – 2015. – T. 521. – №. 7553. – C. 460-466.

4. Neto M. F. S. et al. Computer Vision Based Solutions for MAV Target Detection and Flight Control //10th International Micro-Air Vehicles Conference, Melbourne, Australia. – 2018.

5. CUTLER, Mark; MICHINI, Bernard; HOW, Jonathan P. Lightweight infrared sensing for relative navigation of quadrotors. In: 2013 International Conference on Unmanned Aircraft Systems (ICUAS). IEEE, 2013. p. 1156-1164.

UDC 531.768

Kolesnyk V., Chernyak M.

IMPROVING STRAPDOWN INERTIAL NAVIGATION SYSTEM PERFORMANCE BY SELF-COMPENSATION OF INERTIAL SENSOR ERRORS

Introduction. Satellite navigation systems (SNS) are the most accurate source of navigation, but their signal can be obstructed by natural (terrain relief) and artificial (electronic interference) obstacles. In such cases, the role of the inertial navigation system, particularly the strapdown inertial navigation system (SINS), becomes crucial. The accuracy of the navigation system directly depends on the precision of its primary measurement devices in the inertial measurement unit (IMU), specifically gyroscopes and accelerometers. Thus, the resulting navigation accuracy degrades over time and limits the autonomy of the application, especially when microelectromechanical (MEMS) sensors are used [1]. The undeniable advantages of these sensors, such as small size, low power consumption, and, importantly, low cost, are devalued by substantial zero bias errors and low-frequency flicker noises, leading to signal drift in the sensors over time. While overcoming the first problem (zero bias offset) is a relatively straightforward task that can be resolved during IMU calibration before deployment, mitigating noises within the frequency range coinciding with the valuable signal range is challenging and currently actively researched. One way to solve this problem is the rotation modulation method.

Method. Rotation modulation (RM) was initially described by Heller in 1967 [2]. This method has proven effective with precision sensors, so the inevitable question became whether it could improve the quality of applications based on MEMS sensors. One of the first studies on this issue by Sun and colleagues [3] showed that rotating the IMU can modulate the errors of inertial sensors and consequently reduce navigation errors in a static case. The coordinate determination error was reduced by more than two times, but the rotational platform significantly increased the size and complexity of the overall navigation system, inevitably affecting its cost. But, despite the many challenges, such implementation of RM for MEMS IMU has enormous potential, which calls for a thorough investigation of the approach.

Assume the IMU is rotated at a constant rate around the Z axis. Thus the orientation of the IMU in the body frame after time t is expressed as:

$$C_S^B = \begin{pmatrix} \cos\omega t & -\sin\omega t & 0 \\ \sin\omega t & \cos\omega t & 0 \\ 0 & 0 & 1 \end{pmatrix}. \quad (1)$$

Therefore, the estimation of body acceleration aIBB and angular velocity IBB relative to the inertial frame has the following form:

$$\begin{gathered} \tilde{a}_{IB}^B = C_S^B \tilde{a}_{IS}^S - a_{BS}^S = a_{IB}^B + C_S^B \Delta a^S - a_{BS}^B; \\ \tilde{\omega}_{IB}^B = C_S^B \tilde{\omega}_{IS}^S - \omega_{BS}^S = \omega_{IB}^B + C_S^B \Delta\omega^S - \omega_{BS}^B. \end{gathered} \quad (2)$$

where $\tilde{a}_{IS}^S$ and $\tilde{\omega}_{IS}^S$ are accelerometers and gyroscopes output; $a_{BS}^S = (0;0;0)^{\mathrm{T}}$ and $\omega_{BS}^S = (0;0;\omega)^{\mathrm{T}}$ are acceleration and angular velocity of the IMU relative to the object in sensor frame; Δa^S and $\Delta\omega^S$ are sensor errors.

The integrals of the harmonically modulated quasi-static errors Δa^S and $\Delta\omega^S$ are practically eliminated through the integration of (2) during the full rotation cycle with period T. As a result, the estimates for acceleration and angular velocity for one rotation cycle are as follows:

$$\hat{a}_{IB}^B \approx \frac{1}{T}\int_0^T a_{IB}^B dt + \begin{pmatrix} 0 \\ 0 \\ \Delta a_z^S \end{pmatrix};\ \hat{\omega}_{IB}^B \approx \frac{1}{T}\int_0^T (\omega_{IB}^B - \omega_{BS}^B) dt + \begin{pmatrix} 0 \\ 0 \\ \Delta\omega_z^S \end{pmatrix}. \quad (3)$$

Results. The effectiveness of the method was primarily verified during static laboratory tests of the IMU prototype built on the Xsens MTI-1 with a single-axis rotary stand MPU-1 as a modulator. A reed switch and magnets were used to control the rotation angle in a 90-degree step around the moving platform. The magnitude of the modulating rotation BSS, which was subtracted from the output of the axis gyroscope, was estimated indirectly from measurements of the rotation period. Compensation for gravitational acceleration was made based on the estimation obtained at the initial alignment IMU.

Static tests of the modified INS took place with two variants of the orientation of the axis of RM relative to the gravitational acceleration vector: vertically and horizontally.

The IMU modulates sensors' errors perpendicular to the rotation axis, and corresponding orientation and velocity errors are independently eliminated after a complete rotation cycle. As we can see from the results, the method allows us to reduce the growth rate of errors by order of magnitude, excluding sensors with axes along the RM axis, so the position and speed errors along the rotation axis are propagated in the same way as in the classical SINS.

Conclusions. The rotation of the IMU is capable of harmonically modulating the quasi-static errors of inertial sensors, which are practically eliminated during the navigational algorithm processing estimation, which, in our opinion, paves the way for a significant reduction of its navigation errors and increases its autonomous operation time. However, a substantial limitation of the proposed RIMU computational scheme is that the RIMU's output sample rate is a multiple of the rotation frequency that must be at least twice the maximum frequency of body maneuvers to ensure their observability.

References

1. Savage, P. G., 1998, “Strapdown Inertial Navigation Integration Algorithm Design Part 2: Velocity and Position Algorithms,” J. Guid. Control Dyn., 21, pp. 208–221.

2. E. S. Geller, 1968, “Inertial System Platform Rotation,” IEEE Trans. Aerosp. Electron. Syst., AES-4(4), pp. 557–568.

3. Du, S., Sun, W., and Gao, Y., 2016, “MEMS IMU Error Mitigation Using Rotation Modulation Technique,” Sensors, 16(12).

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Kovalenko O., Arkhipov O.

THE MUTUAL INFLUENCE OF THE GLIDING CONTAINER AND THE AIRCRAFT ON THEIR AERODYNAMIC CHARACTERISTICS IN THE PROCESS OF SEPARATION

Introduction. The rapid development of unmanned aerial vehicles in the world primarily aims to reduce the risks to human life as much as possible. During emergency situations, the issue of delivering various cargoes to hard-to-reach places without risking the lives of pilots is a pressing issue.

Gliding cargo containers, which are considered in the work, are launched from the carrier aircraft at a distance from the landing zone and do not need to be returned [1]. They are made of the simplest materials, can be easily manufactured in large quantities in a short period of time, are easily assembled with handy tools, are inexpensive, inconspicuous and do not give away the launch site.

Scientific and technical results. The purpose of the study is the aerodynamic characteristics of the aircraft in the process of separation of aircraft, and the conditions of safe separation of the container. The flight test experience of Logistic Gliders and Yates Electrospace Corporation (YEC) in the development of LG-1K and Silent Arrow gliders, which are direct prototypes of a gliding container, is considered and taken into account [2], [3], [4].

To carry out the research, models were created in several configurations. For all models, calculations were made for speeds of 95.8 m/s, 50 m/s and 30 m/s, angles of attack -3°, 2°, 3° for the glider container and 0°, 3°, 6° for the aircraft system - carrier with gliders.

It was found that the lift coefficient of the carrier aircraft at an angle of attack of 6° and a speed of 95.8 m/s decreases by about 6% with one and both gliding containers, and the drag coefficient with one glider increases by 5%, and with by two by 15%, the aerodynamic quality of an aircraft with gliders increases with an increase in the angle of attack. When moving the glider to a distance of 50 mm from the wing suspension system of the carrier aircraft, the lift coefficient of the aircraft in the speed range of 50-95.8 m/s decreases by 1.5% - 2.4%.

It was established that under the wing the lift coefficient of the glider increases, but remains negative and when separated from the carrier aircraft until the moment of deployment of the wings, the glider container will safely move away from the wing of the carrier aircraft. When the glider is moved away by 160 mm, that is, at such a distance that the carrier aircraft does not interfere with opening the wings, the influence of the glider container on the characteristics of the carrier aircraft and the influence of the carrier aircraft on the glider container with the wings spread is much less than one percent, and it can be assumed that with at a distance of 160 mm, the two aerial vehicles do not affect each other further.

According to the Breguet equation, the flight range of a carrier aircraft with two containers decreases by almost 3 times at angles of attack of 0°, and by 1.3 times at angles of attack of 6° [5].

With the help of visualization tools, an analysis of the pressure distribution on the aircraft body, the flow line, the isometry of the flow change, both of the glider container separately, and of the carrier aircraft together with the glider containers was performed. ANSYS software was used for finite element calculations. Figure 1 shows the streamlines in the plane of symmetry of the airframe of the two-aircraft system.

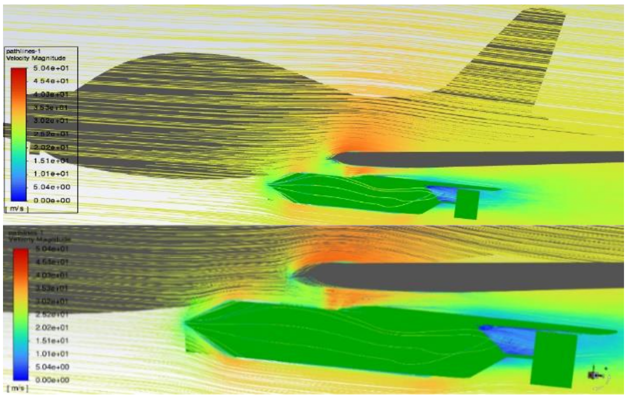

Figure 1 - Flow lines in the plane of symmetry of the glider

Conclusion. Aerodynamic characteristics of aircraft separately from each other, in a system of two and three aircraft in various flight configurations and in the process of separation were determined and analyzed.

On the basis of the obtained data, it is possible to improve the design of the glider container, conduct additional aerodynamic studies of the influence of gliders on the controllability of the carrier aircraft, and additional calculations of the flight dynamics of the carrier aircraft system and the glider container.

References:

1. Selection of a Carrier Aircraft and a Launch Method for Air Launching Space Vehicles / Marti Sarigul-Klijn, Ph.D, Nesrin Sarigul-Klijn, Ph.D, Gary C. Hudson та ін.]. // SPACE Conference & Exposition. – 2008. – C. 12.
2. Marti M. Sarigul-Klijn, Ph.D. Technology Demonstration of a 1-ton Single Use Disposable Glider / Marti M. Sarigul-Klijn, Ph.D, Maurice P. Gionfriddo, Nesrin Sarigul-Klijn, Ph.D. // SciTech Forum. – 2019. – №7. – C. 9.
3. Military resupply and humanitarian relief / Silent Arrow –Available at ithttps://silent-arrow.com/
4. Rebecca Jeffrey. Silent Arrow drone completes first overseas deployment / Rebecca Jeffrey // Air Cargo News. – 2022. - Available at https://www.aircargonews.net/airlines/drones/silent-arrow-drone-completes-first-overseas-deployment/.
5. E. Torenbeek. Synthesis of Subsonic Airplane Design / E. Torenbeek., 1976. –648 c.

UDC 629.7.085

Kozey Y.M., Kozey Y.S., Sukhov V.

CATAPULT WITH POLYPASS SYSTEM FOR UNMANNED AIRCRAFT

Introduction. The launch catapult of an unmanned aerial vehicle (UAV) is used for forced acceleration of UAVs on takeoff. At the same time it is necessary to receive speed of separation at admissible overload. Energy for this is obtained from devices that do not belong to the UAV. The catapult must be autonomous when launching the UAV, have a simple design, light weight and dimensions in the transport position. It must be easy to maintain, quickly deploy to the starting position and provide dynamic stability when starting the UAV. Catapult can be placed on a vehicle or trailer required permeability and mobility. The design of the catapult must ensure a reliable launch of the aircraft, and the launch parameters must be constant with each subsequent launch. In order to start efficiently and to avoid damage to expensive equipment, it is necessary to minimize the probability of failure. These include: the influence of external conditions, including weather, and human factors.

The purpose of the work. The purpose of this work is the design calculation of a pneumatic catapult for an unmanned aerial vehicle. To achieve this goal, the following tasks were set:

- Analysis of existing versions of catapults for UAVs and literature on modern principles of calculation of such structures.
- Determination of mass-geometric and dynamic parameters of the catapult.
- Development of a mathematical model of a catapult.
- Calculation of the design of the catapult. Checking the proposed design option for strength.

Results of work. The first part of the study is based on the analysis of existing UAV launchers. Collected and analyzed information on the characteristics, design schemes and type of launch of common catapults: pneumatic MC0315L, pneumatic ARCTURUSUAV, pneumatic PL-40, pneumatic C400P, badge (rubber) C200R. And UAVs for launch: RQ-21A Blackjack, Bayraktar TB2, Krunk 25-1, People's Drone PD-1. It is established that the type of construction mostly depends on the mass and speed characteristics of the UAV[1].

Mathematical modeling of the entire system was divided into two main parts: the air cylinder and the movement of the UAV located on the guide [2]. With the help of appropriate mathematical models, the required force, speed, and acceleration to achieve the minimum height for UAV launch have been determined[3].

During start-up, the UAV is attached to the cradle. It is acted upon by the force of gravity, the aerodynamic force due to the aerodynamic forces, the weight of the system itself and the force that drives the cradle. The purpose of mathematical modeling is to determine the required force required for the movement of the cradle and its acceleration. The assumption is that the force F will remain constant during start-up, as this simplifies the calculations[4]. The Newton-Euler equation is the basis of mathematical calculation.

As a result of mathematical calculations a number of dependences is received (Figure 1).

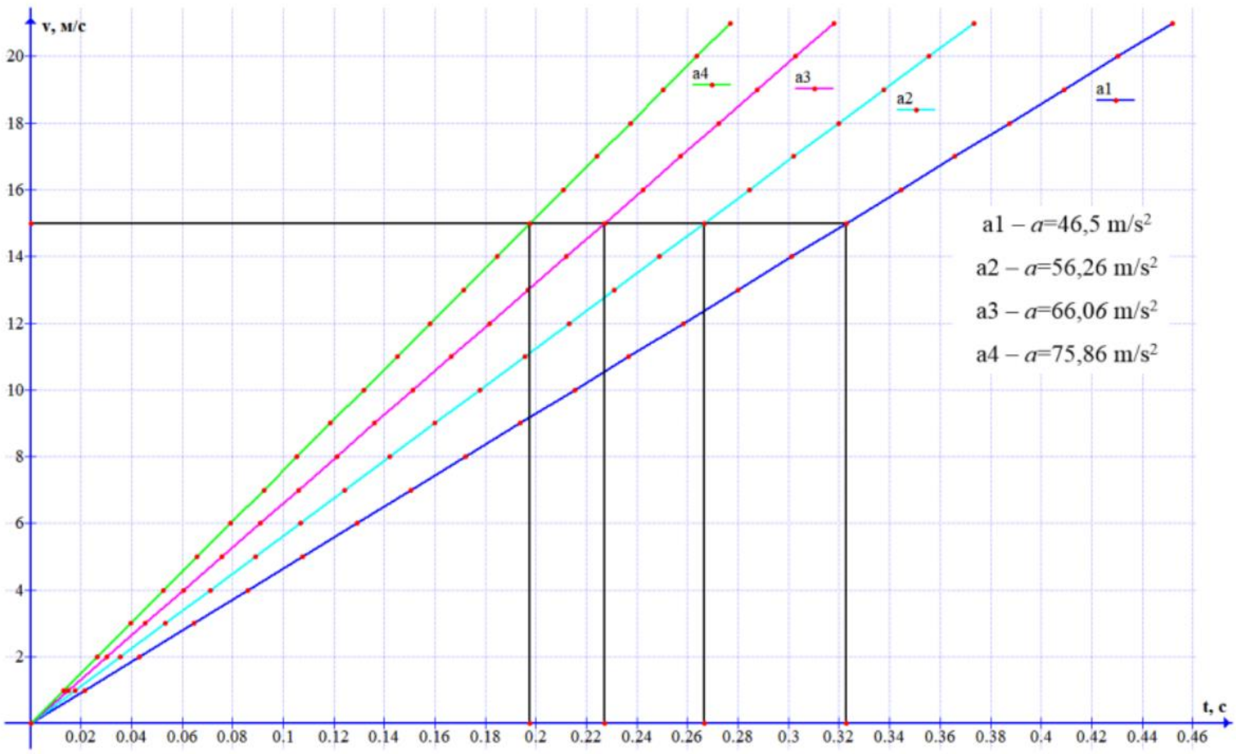

Figure 1 - Graph of the dependence of the velocity v on time t at different values of acceleration a.

After which the design of the catapult was calculated. At the finishing stage, a spatial model of the catapult is created, shown in Figure 2.

Figure 2 - Spatial model of the pulley and catapult

The model is tested for strength in the Simens NX 12 software. After simulation, the results shown in Figures 3 were obtained.

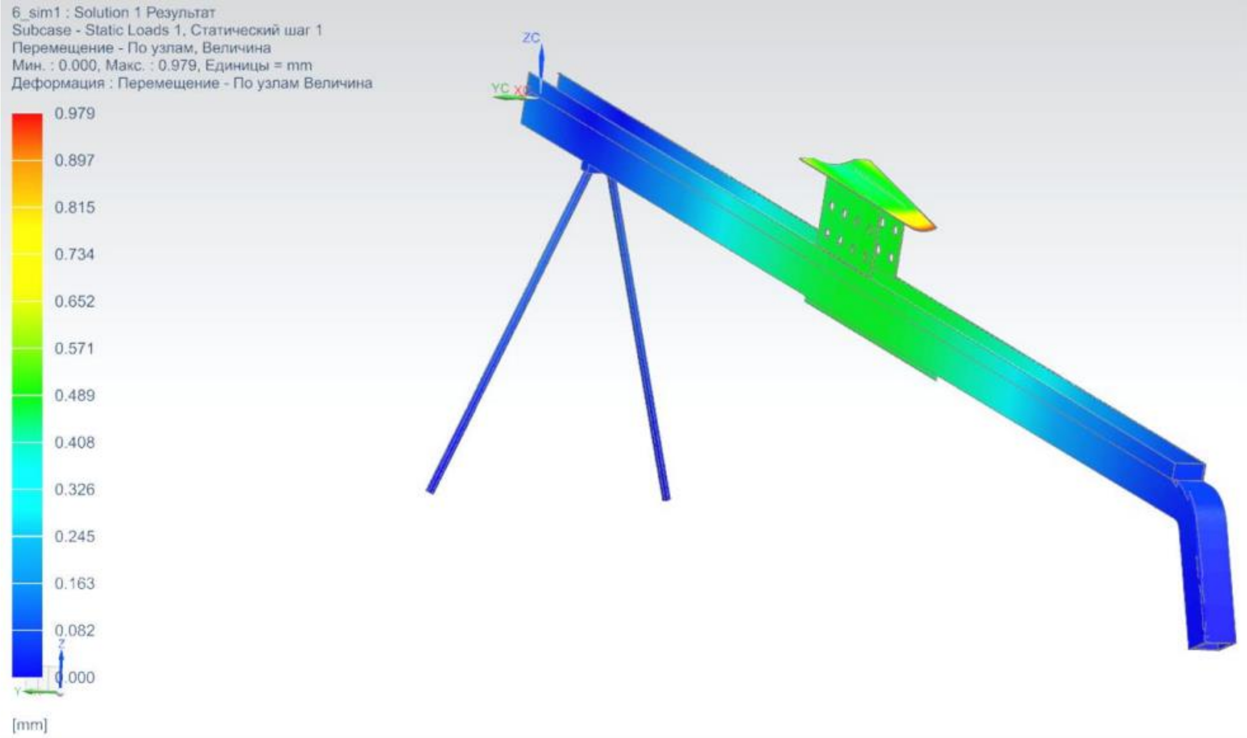

Figure 3 - Modeling of the stress-strain state

Conclusion. In the process of work the following stages were performed: analogues of catapults and UAVs were inspected and analyzed; researched and selected mathematical model of calculation; calculated mass-geometric and dynamic parameters of the catapult and pneumatic cylinder; the design calculation of the design of the polypass catapult is cradle out; developed a spatial model of the catapult; the analysis of the tensely deformed state is made. Simens NX 12 software was chosen to design the spatial model.

References:

1. Davi R. Mechanical design and optimization of swarm-capable UAV launch systems / Raymond L Davi. – California: NAVAL POSTGRADUATE SCHOOL, 2015. – 139 c.

2. Voogd A. Robird autonomous take-off: pneumatic launching system / Voogd Alan, 2017. – 34 c.

3. Разон В. Ф. Учебно-методическое пособие по курсовому и дипломному проектированию «Расчет пневматических силовых приводов средств механизации производственных процессов» / Владимир Федорович Разон. – Гомель: Белорусский государственный университет транспорта. 2014. – 94 с.

4. UAV Catapult / Jody Cearns, Sean Huxley, David Garcia, Ben Lacasse., 2019.

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Krasnov R., Lukyanov P.

THE PRINCIPLES OF MODULAR DIVISION OF UNMANNED AERIAL VEHICLES

Introduction. In the modern high-tech world, the development of unmanned aerial vehicles (UAVs) is becoming increasingly relevant and important. Innovative technological systems open up new perspectives and possibilities in the aviation industry. The aim of my research is to analyze and apply the significant role of a modular approach in the creation of UAVs.

Considering the rapid evolution of technologies, UAVs play a crucial role in addressing various tasks, from transportation to monitoring emergency situations. The use of modular construction will streamline and simplify the improvement and development of new UAVs.

Technological possibilities provided by modularity in the process of UAV development and operation have been explored.

Scientific and technical results. The research paper examines the technical and functional features of unmanned aerial vehicles, their applications, and the advantages and disadvantages of existing systems.

The technological capabilities of modular structures, the elements that make up the modular architecture, as well as the relationship between modules and their functions are discussed in detail.

In the process of preliminary design of the unmanned aerial vehicle (UAV) under study, a number of calculations were carried out to determine the optimal characteristics and parameters of the structure. Particular attention was paid to selecting the optimal aerodynamic scheme and determining the main geometric dimensions of the UAV airframe.

One of the key stages was the aerodynamic calculation, which involved determining the optimal parameters to ensure maximum performance and functionality of the aircraft. An important aspect of this was the ability to integrate modular systems into the selected aerodynamic scheme, which contributed to the efficiency and flexibility of the project.

Based on the calculations and analysed data, a model of the airframe was created in the NX CAD CAM CAE system and the airframe was divided into modular compartments (see Figure 1.) The division of the airframe into compartments was carried out to meet the needs of the equipment that would be deployed in this area in accordance with the layout of the UAV and the conditions for maintaining the geometry of the airframe.

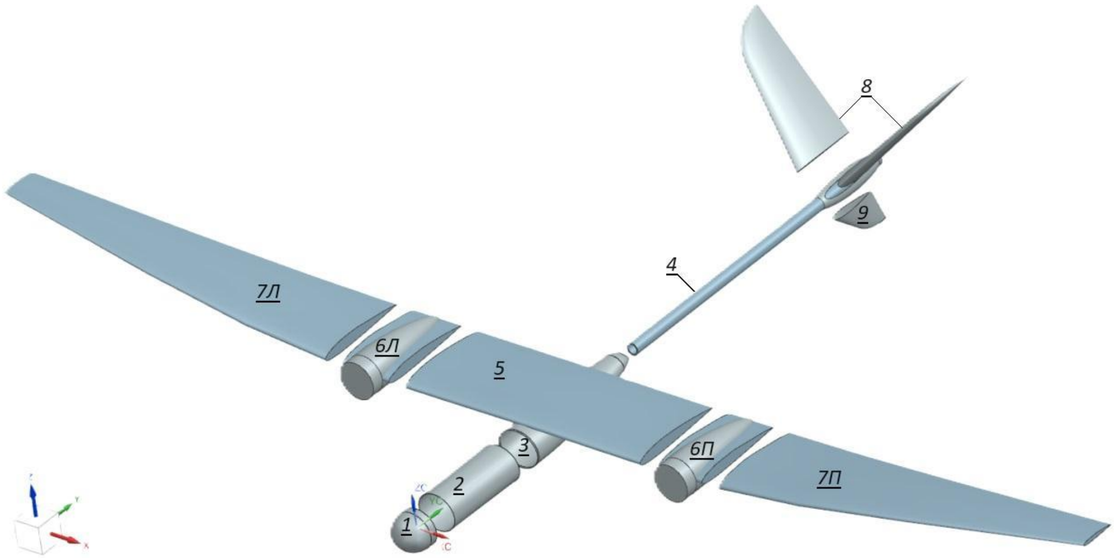

Figure 1

Conclusion. The modular design of such a UAV is a key factor in achieving high efficiency and flexibility in their development, production and operation. The flexible system of module arrangement and replacement not only allows for the adaptation of the technical characteristics of the PSU to various tasks, but also helps to optimise the integration of new equipment.

This approach to UAV construction not only improves maintainability and the ability to implement upgrades, but also makes the vessels more adaptable to changing conditions and tasks, creating prospects for creating highly efficient UAV families.

References:

1. U. S. Department of Defense Unmanned Systems Roadmap: 2007 – 2032
2. Frew, E. W., & Pounds, P. E. I. Unmanned Aircraft Systems: UAVS Design, Development and Deployment. 2017
3. Kendrick, D. Advances in UAV Technologies. 2019

UDC 536.24

Liakhovoi K., Lobunko O.

THERMAL MANAGEMENT IN SPACE

Introduction: In space there is no heat-conductive environment. This leads to a problem that must be solved – how can we effectively cool on-board electronics, science instruments and inhabited modules of a spacecraft? This task is somewhat easier to solve for big spacecrafts, while small satellites and CubeSats require more compact but still effective heat management systems. A wide variety of materials and mechanical principles, used and developed to ensure reliable and effective solutions, are presented on the market.

Scientific and technical results: The heat sources can be situated in a satellite itself (PCBs, scientific instruments etc.). The Sun gives a considerable amount of heating. Planet (if any) also emits some heat as well as reflects its star's radiation. A spacecraft can also lose its thermal energy by radiating it into space or store it in special systems (Fig.1) [1].

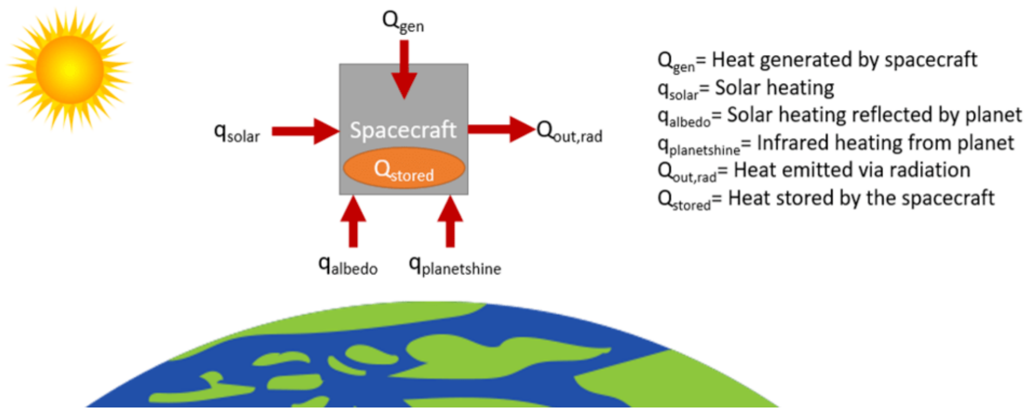

Figure 1 – Simplified view of orbiting spacecraft heat sources and sinks.

Now we can differentiate methods to manage heat, coming from different origins. As spacecrafts are designed for specific temperature ranges [2], Q_{solar}, Q_{albedo} and $Q_{planetshine}$ can be reduced to fit them, using several methods. The simplest way to reduce radiation heat from these sources is to minimize exposed silhouette. This method can be used if spacecraft does not have to constantly maintain its orientation. Otherwise, special coatings, sunshields and Multi-layer Insulators (MLI) can be used. MLI is a passive thermal control component that can reflect space radiation, as well as trap heat inside of the satellite. MLI is composed of multiple layers of thin plastic sheets, covered by vapor-disposed metal. Layers are separated by low conductive materials. The problem is that MLIs require special mounting system resulting in space-consuming design. Small satellites are limited by size, that is why they use specialized coatings. Those are designed to match the scope of use – for example, if spacecraft will often be heated by any radiation, coating must have high α (albedo) and high emissivity to reflect as much emissions as possible. Different coatings are sometimes mixed. Sunshields are deployable devices which can hide spacecraft from thermal radiation source by reflecting it. They are often contained on-board in folded state to save available volume. If we want to use even less volume, new folding technics must be used. Japanese "origami" art contains many technics which can be used to fold sunscreens incredibly efficiently and drastically reduce their size in folded state.

Talking about heat sinks, the only available heat transfer path in space is radiation. All structural elements of satellites and spacecraft are radiators themselves, but some of them use separate modules to cool down more effectively. Passive radiators are metal surfaces with high emissivity and low α, which get heat from the spacecraft by conduction. Active ones are fitted with fluid loop, which carries heat from the craft with higher efficiency. There is a concept of droplet generator, which showers huge foldable radiator with heat carrier droplets. Cooled fluid then returns into the system [3].

In prior to radiating out, excessive heat can also be stored in TSU – Thermal Storage Unit. These components feature heat-capacitive material (usually phase-changing) to smooth out sudden changes of temperature and then safely conduct excessive heat to the radiator or warm up instruments in need. TSUs are hard to use on small satellites as they are relatively heavy and big [1].

To transfer heat to needed zone, heat pipes or HPs are used. They consist of metal or carbon strap of required form, which is mounted to transfer heat from directly sources to sinks. Some HPs can have fluid inside, which evaporates, travels to the cold end, cools down and then goes back using the capillary effect. The problem lies in testing – capillary effect in zero-G environment differs from one-G and HPs may not function as expected [4].

Conclusion: A wide variety of thermal management methods exist, but they must be chosen to strictly match design requirements in size and mass. All systems have to be reliably manufactured, as craft's science instruments and other modules greatly depend on compliance of real working conditions with the expected ones.

References:

1. Sonja Caldwell, Brian Dunbar (2023, May 27). *State-of-the-Art of Small Spacecraft Technology. 7.0 Thermal Control.* National Aeronautics and Space Administration.
https://www.nasa.gov/smallsat-institute/sst-soa/thermal-control/
2. M.N. De Parolis & W. Pinter-Krainer (1996, August). *Current and Future Techniques for Spacecraft Thermal Control 1. Design drivers and current technologies.* European Space Agency.
https://www.esa.int/esapub/bulletin/bullet87/paroli87.htm
3. Abraham Hertzberg (1992, January 1). *Thermal management in space.* National Aeronautics and Space Administration. Johnson Space Center. https://ntrs.nasa.gov/api/citations/19930007720/downloads/19930007720.pdf
4. Col. John E. Keesee (2003). *Spacecraft Thermal Control Systems.* Massachusetts Institute of Technology.
https://ocw.mit.edu/courses/16-851-satellite-engineering-fall-2003/e3a84cc153960fff8d55480fe228bbcc_l23thermalcontro.pdf

UDC 629.7.023.46

Lytvynenko I., Bondar Y.

PROGRAMS FOR CREATING TYPICAL FRAMES IN THE SIEMENS NX SYSTEM

Introduction. The engineering and technical sphere of the modern industrial environment is constantly evolving, requiring specialists to constantly improve and implement new technologies to increase the efficiency of design and production. One of the key components of this process is the development of software aimed at facilitating and optimizing simple and repetitive tasks. The use of standard templates makes the creation of new projects much easier and faster than ever before. The Siemens NX system is one of the most advanced platforms for design and manufacturing, allowing users to customize NX for specific industry and technology requirements.

Scientific and technical results. Frames are a structural and load-bearing element of an aircraft fuselage that is widely used in modern aircraft construction. Having studied the methods of building frames in the Siemens NX system, we can identify 4 most optimal and convenient ones. For the future program, two methods will be sufficient as they will be able to create the largest number of frame variants. In addition, depending on the user's skills, one of the available methods will be simpler and will allow the user to work with already created frame sections.

The first method is the simplest, but in complex cases it will require additional construction and modification of the frame. Firstly, we get the intersection curve of the plane of the frame and the surface of the skin. After that, we build a sketch of the frame profile and, by sweeping along the intersection curve, we get the frame body.

The second method gives the user more options. Firstly, select or create a new sketch of the frame web and set its thickness. Then you can build the cap frame. The second method allows you to create non-standard frames with a complex web and different cap (or no cap at all).

Both methods will allow you to build cutouts for the stringer in the frame. You can choose a standard cutout for a specific stringer profile or a custom cutout.

When it came to choosing a CAD system for this work, I opted for the Siemens NX system. It allows the user to customize NX for specific tasks and add their own tools. The NXOpen library is available for this purpose. NXOpen is a set of API (Application Programming Interface) tools that enable flexible integration of custom applications with NX using an open architecture that can be used by third party developers, clients and users. The purpose of NXOpen is to automate complex and repetitive tasks.

Benefits of NXOpen:

- The ability to customize NX for specific industry or technology requirements;
- Reduced time to market by automating complex and repetitive processes;
- No need for special user training due to full compliance with the NX interface;

- Full NX functionality;
- Concurrent development with interactive features;
- Used by Siemens PLM developers.

NXOpen has a wide range of programming languages, including C#, VB.NET, Java, Python, C++. Recorded macros can be easily viewed as program code, allowing you to use the most complex NX functions.

Conclusion. In the rapidly evolving landscape of the modern industrial environment, the imperative for continuous improvement and the implementation of cutting-edge technologies underscores the essence of progress. As specialists strive to increase the efficiency of design and production, software development is emerging as a key component in simplifying and optimizing tasks.

The scientific and technical research in this study focused on frames, which are critical structural elements in aircraft fuselage design. By delving into Siemens NX, we identified several optimal methods for constructing frames. In the context of developing a future program, two methods emerged as particularly robust and capable of generating a wide range of frame variants. Depending on user skills and project complexity, each method offers a unique advantage, simplifying the creation and modification of frame sections.

UDC 629.7

Mohamed Naleem Y. F., Grishanova I.

STRENGTH INVESTIGATION IN AEROSPACE CRAFTS

Introduction. During whole development of aerospace crafts strength of its structure played key role in its design. This role imposes considerable limitations on all aerospace craft`s characteristics. As a result, it was necessary to crate ways to create reliable and durable structure with sufficient economical efficient and desired characteristics.

To transport payload to required location aerospace craft needs to sustain different external factors such as: gravitational loads, take off overload, aerodynamic resistance, maneuver loads and others. Also, structure need to be durable to sustain these loads during whole flight.

There are many methods to calculate strength of structure, but all of them have their own limitations and assumptions. Those need to simplify process of computations without produce false resalts, which can lead to catastrophic failure.

Scientific and technical results. To start with, strength of evaluation can be divided into two parts: evaluations related to structure and choosing material for it. These parts play even role in design effective aerospace craft.

As Crane F. A. and Charles J.A. show in their work [1], choosing of material need to be essential design part and should be numerically facilitated. To achieve this, firstly it is need to establish requirements to the material, which includes cost and mass effectiveness, manufacturing and service requirement. Main considerations are:

1. Mechanical Properties:

Strength: The ability to withstand loads without failure.

Elasticity and Plasticity: How much the material can stretch (elastic deformation) or permanently deform (plastic deformation).

Toughness: The material's ability to absorb energy before fracturing.

Hardness: Resistance to surface deformation.

Fatigue Strength: The material's ability to withstand repeated or fluctuating stresses.

Ductility and Brittleness: The material's capacity for plastic deformation before rupture.

2. Thermal Properties:

Thermal Conductivity: How well the material conducts heat.

Thermal Expansion: How much the material expands or contracts with temperature changes.

Heat Resistance: The ability to retain properties at high temperatures.

3. Chemical Properties:

Corrosion Resistance: The ability to resist chemical degradation.

Chemical Stability: Resistance to reactions with other materials or environments.

4. Electrical Properties:

Electrical Conductivity or Resistivity: Important for electrical and electronic applications.

5. Manufacturing Considerations:

Machinability: Ease with which the material can be machined.

Formability: The ease of forming the material into desired shapes.

Weldability and Joinability: Compatibility with welding and other joining techniques.

6. Cost and Availability:

Economic Feasibility: Material cost, including raw material and processing costs.

Availability: Accessibility of the material in desired quantities and forms

According to Niu C. Y. and Bruhn F. E. [3,4] structure analysis, this part provides engineers with data, that can describe how structure will behave under different external loads, and provides different methods of evaluating strength of different parts of this structure. This part comprises of analysis of external loads, designating internal loads and using selected material evaluating margins of safety of parts alone and whole structure. Key aspects of structure analysis are:

1. Material Properties

2. Types of Loads:

Static Loads: Constant loads applied to the structure.

Dynamic Loads: Time-varying loads, including impact and shock loads.

Cyclic or Fatigue Loads: Repeated loading and unloading cycles.

Thermal Loads: Stresses due to temperature changes.

3. Load Magnitude and Distribution:

Magnitude: The size of the loads in appropriate units (N, lb, etc.).

Point, Distributed, and Uniform Loads: How the load is applied over the structure.

Combination of Loads: Considering the superposition of different types of loads.

4. Boundary Conditions and Constraints:

Fixed, Pinned, or Free Supports: These determine how the structure is supported and can move or deform.

Contact Points: Areas where structures interact with other components or supports.

5. Geometric Factors:

Shape and Size: The geometry of the component or structure, including any holes, notches, or other stress concentrators.

Thickness: Especially for plates or shells under load.

6. Stress Concentrations:

Discontinuities: Areas like holes, sharp corners, or sudden changes in cross-section where stress can be amplified.

7. Analysis Method:

Analytical Calculations: For simpler structures or as a preliminary check.

Numerical Methods: Finite element analysis (FEA) for complex geometries and load conditions.

8. Safety Factors:

Factor of Safety (FoS): Applied to account for uncertainties in load estimations, material properties, and environmental conditions.

9. Environmental Conditions:

Temperature, Corrosion, and Wear: These can significantly affect the long-term performance and integrity of the structure.

10. Compliance with Standards and Regulations:

Building Codes and Engineering Standards: Ensure the design meets all relevant industry and safety standards.

11. Failure Modes:

Understand Potential Failure Modes: Such as yielding, buckling, fatigue, or fracture.

Conclusion. This works describes process of design using conventional methods, which were created during developing of aerospace crafts with more specific date about static strength. This situation emerges due to lack of understanding of dynamic and cyclic loads during those day. More resent works take deeper approach to toughness and fatigue durability, basing their methods on previous exploitation experiences.

Also, regarding materials, recently composites and ceramics began to be widely used in primal structure of aerospace crafts due to their versatility and superior weight to strength ratio. But the problem is in difficult production and maintenance compare to metals. And with appearing of metallic 3d printers this difference even increased.

To reach the goal of more efficient aerospace crafts more researches need to be conducted connected with complex behaviors of structure, like diagonal tension field to eliminate imperfections, that can led to excessive material. Also it is quite promising practice to use AI in creation of complex parts with very sophisticated load pass.

References:

1. Crane, F.A. and Charles, J.A. (1984) Selection and Use of Engineering Materials, Butterworths, London.

2. Ashby, M.F. and Jones, D. R. H. (1980) Engineering Materials – An Introduction in to Their Properties and Applications, Pergamon, Oxford

3. Niu, C. Y. AIRFRAME STRESS ANALYSIS AND SIZING. Hong Kong: Wanchai Post Office, 1997

4. Bruhn, F. E. ANALYSIS AND DESIGN OF FLIGHT VEHICLE STRUCTURES. USA: TriState Offset Company, 1973

5. Vasiliev, V. V. and Morozov, E, (2005) Mechanics and Analysis of Composite Materials, Elsevier.

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Mostovyi O., Arkhipov O.

THE USE OF ADDITIVE TECHNOLOGIES IN THE MANUFACTURE OF LIQUID-FUEL ROCKET ENGINES

Introduction. Additive manufacturing involves manufacturing techniques that add material to produce metal components, typically layer by layer. The significant growth of this technology is partly due to its commercial and operational advantages in the aerospace industry. The fundamental capabilities of additive manufacturing in the aerospace industry include: significant reductions in manufacturing costs and time, new materials and unique design solutions, component weight reduction through highly efficient and lightweight designs, and consolidation of multiple components to improve performance or manage risk, for example by internally cooling thermally stressed components or eliminating traditional joining processes. These capabilities are being commercialized in a wide range of aerospace applications, including liquid-fueled rocket engines.

Scientific and technical results. The use of heat exchange devices has significant potential to disrupt traditional aerospace manufacturing processes through the use of additive manufacturing methods due to their complex design and required internal characteristics. Manufacturing these devices using additive manufacturing can significantly reduce the number of parts, shorten lead times, lighten the design, and lower costs. For example, the GE9X heat exchanger boasts a 40% reduction in weight and a 25% reduction in cost, all while combining 163 components into a single part using additive manufacturing methods compared to traditional manufacturing methods [1].

The production of heat exchangers using additive manufacturing also has productivity advantages over traditional manufacturing methods. A demonstration sample of additive manufacturing of a rocket engine by Cellcore demonstrates the possibility of manufacturing a complex thrust chamber assembly in the IN718 engine with integrated lattice internal cooling channels [2]. NASA has also established a significant use of additive manufacturing for heat exchangers, such as combustion chambers of liquid rocket engines and channel-cooled nozzles.

A variety of applications have been reported, including combustion chambers using copper alloys such as GRCop-42, GRCop-42, C18150, as well as Inconel 718 and bimetallic copper-superalloy structures that have accumulated over 30,000 s and 400 hot fire launches. Channel-cooled nozzles made of various alloys, including JBK-75, NASA HR-1, Inconel 625, Haynes 230, and bimetallic (copper-superalloy) structures, have also been demonstrated by NASA and have accumulated more than 11,000 s and 250+ hotfire tests [3]. While these combustion chambers and nozzles meet performance requirements, they have also demonstrated significant cost savings and reduced equipment delivery times [4].

Typically constructed using multiple materials and thousands of components, combustion chambers, nozzles, and injector assemblies have recently attracted great interest for use in additive manufacturing for the production of single parts. An example of the use of additive manufacturing methods for the manufacture of single parts is the Aeon 1 rocket engine created by Relativity Space. The Aeon 1 engine, which will be launched on the future Terran 1 launch vehicle, has an injector, igniter, combustion chamber, and nozzle manufactured as a single component using additive manufacturing methods.

Conclusion. In conclusion, the integration of additive manufacturing techniques in the aerospace industry has proven to be a transformative force, revolutionizing traditional manufacturing processes and unlocking unprecedented capabilities. The fundamental advantages of this technology, including substantial reductions in manufacturing costs and time, the introduction of novel materials, and the realization of intricate design solutions, have positioned it at the forefront of innovation within the aerospace sector.

References:

1. General Electric - Additive, GE9X Engine Additive Parts, , https://www.ge.com/additive/sites/default/files/2020-08/GE9X Additive parts.pdf.
2. S. Donath, 3D printing a rocket engine, 2019. https://www.etmmonline.com/3d-printing-a-rocket-engine-a-886960/.
3. P.R. Gradl, C.S. Protz, Technology advancements for channel wall nozzle manufacturing in liquid rocket engines, Acta Astronaut. 174 (2020) 148–158, https://doi.org/10.1016/j.actaastro.2020.04.067.
4. P.R. Gradl, C.S. Protz, K. Cooper, D. Ellis, L.J. Evans, C. Garcia, GRCop-42 development and hot-fire testing using additive manufacturing powder bed fusion for channel-cooled combustion chambers, 2019. doi: 10.2514/6.2019- 4228.

UDC 629.7

Novak Y., Zinchenko D.

PROPERTIES OF BIO-COMPOSITE INTERIOR PANELS FOR PASSENGER AIRCRAFT

Introduction. The modern aviation industry is actively adapting to the challenges of environmental sustainability and requirements for increased efficiency, which leads to the search for innovative materials and technologies. Improving the internal equipment of passenger aircraft is becoming a key aspect aimed at ensuring safety, comfort and reducing the environmental impact of aviation.

Biocomposite materials, especially in the context of domestic equipment, are identified as a promising area of research, as they use natural resources and meet the standards of the aviation industry, contributing to sustainable development and environmental responsibility.

Problem statement. The study aims to in-depth analyze the properties of biocomposite panels for potential use in passenger aircraft. The main tasks include analyzing existing materials in aircraft, comparing the characteristics of biocomposites with traditional materials, studying the impact on the weight and energy efficiency of aircraft, studying the production and recoverability of biocomposites with a focus on environmental friendliness, as well as evaluating their aesthetic appearance and ability to integrate into modern aircraft design. The work also includes the development of recommendations for the use of biocomposites in aircraft construction and conclusions about possible advantages and challenges in their implementation[1].

Main part. Analyzing technologies and materials in aircraft construction, it was found that plastics, metals and wood are used taking into account their unique properties. Plastics give lightness and strength, metals - stability and strength, and wood - aesthetic appearance.

Research on biocomposites shows that they are competitive in strength, lightness and aesthetics compared to traditional materials. Biocomposites are made from biological and polymer components, have a lower density and can have a diverse aesthetic appearance.

Studying the effect of using biocomposite panels, it was found that their lightness leads to a significant reduction in the weight of the aircraft. This helps to improve energy efficiency and reduce CO_2 emissions. It is important to keep in mind that biocomposites are highly resistant to corrosion and can improve thermal insulation[2].

In general, the use of biocomposites can contribute to the creation of efficient, sustainable and environmentally friendly aircraft, but requires careful consideration of all technical and economic aspects before implementing them in the aircraft industry.

The study of biocomposite production processes and their environmental sustainability after operation highlights the importance of using renewable materials and energy-efficient technologies. The results of the study show the prospects of biocomposites in sustainable production and their possibility of recycling, which increases their environmental benefits[3].

Evaluation of the aesthetic appearance of biocomposite panels emphasizes their naturalness and modularity, which contributes to the ease of integration into the modern design of passenger aircraft. A high level of decor and a wide color palette provide opportunities for creative design of aircraft interior equipment.

The development of recommendations indicates the need to improve technologies and standardize the use of biocomposites in aircraft construction. The importance of large-scale testing and certification to ensure high standards of safety and durability is noted[4].

The advantages of using biocomposites, such as reduced aircraft weight, environmental sustainability and aesthetic appearance, highlight their potential significance. However, the challenges associated with durability, cost and production require careful consideration before widespread implementation in the aircraft industry[5].

Conclusion. The use of biocomposition panels in the interior of passenger aircraft opens up new prospects for the aviation industry, providing an aesthetic appearance, lightness and environmental sustainability. Despite these advantages, it is important to address challenges such as strength, cost, and certification to ensure sustainable and efficient development of the use of biocomposites in passenger aviation.

References:

1. Kondrashchenko O. V., Zhiglo A. A. newest finishing materials, products and constructions: textbook. – 2016.

2. Avdeeva N. Yu.methodological foundations of functional organization of the process of designing residential complexes in territories close to airports.

3. Kashitsky V. P. et al. Development of technology for forming biocomposite decorative products //scientific notes. – 2022. – №. 74. - pp. 134-139.

4. Kashitsky V. P. et al. Development of biocomposites filled with products of processing secondary raw materials of plant origin //Vestnik VPI.№ 1: 95–102. – 2022.

5. Savchuk P. P. et al. Investigation of the properties of biocomposites filled with discrete fibers of natural origin //scientific notes. – 2020. – №. 69. - pp. 75-81.

UDC 621.7

Pashistiy B., Arkhipov O.

THE PROBLEM OF INTRODUCING ADDITIVE TECHNOLOGIES IN AVIATION PRODUCTION

Introduction. In recent years, additive technologies have been making their way into the private market, and now literally anyone with a small amount of money and a desire can have an almost industrial machine on a miniature scale at home. Industrial engineering should have adopted these technologies long ago, but it's not that simple. Having analyzed the work of leading Ukrainian enterprises, it is not difficult to conclude that engineers still rely more on traditional manufacturing solutions. However, additive technologies are growing and improving significantly and have unprecedented advantages that no other technology has, and the cost of parts made using these new methods is falling significantly. But what is it that prevents this seemingly intuitive development?

Scientific and technical results. The aerospace industry has always been the first to introduce new technologies, so in the global market 10 years ago, the aerospace industry was the leader in the use of additive technologies, and today about 20% [1] of the total volume of additive manufacturing use is in this area, but Ukrainian enterprises are lagging in this matter. And as in any technical field, the distance is increasing every year. Companies that implement additive manufacturing are already using it very actively on serial samples, for example, Boening produces about 1000 parts in its 787 Dreamliner aircraft using this technology and thus saves 2-3 million dollars [2].

To begin with, additive technologies offer advantages, such as a variety and complexity of shapes that are often difficult and expensive to achieve with traditional methods, and sometimes impossible, as well as increased production efficiency, which can ensure a much higher environmental friendliness of production. However, all of this can only be achieved if design approaches change, and unlike conventional metalworking, for example, additive technologies require much lower initial investment in equipment, but the cost of the output part is often higher than in conventional manufacturing.

Usually, experienced engineers choose the technology they are used to, which has been tested over the years, as engineers are not very risk averse. The Ukrainian engineering school is very critical in this regard, as low budgets and tight deadlines rarely give engineers room to experiment and select the best new solutions. It is also worth noting that such equipment requires appropriate specialists, namely operators of this equipment, who are also in short supply, and training in this area usually takes place abroad at friendly enterprises or at the equipment manufacturer's plant, which is mostly very expensive.

Additive technologies require new approaches to design. Designing for this equipment often differs from conventional technologies, as well noted in [3] "New tools, new rools". This requires additional knowledge and skills, and most importantly, the emergence of additional errors that will need to be resolved, which, accordingly, takes time, which is often what the management of enterprises is afraid of. Another big advantage of computer-controlled systems is a significant reduction in paperwork, which is currently a big problem for many state-owned enterprises, but this will require deep structural changes, which are even more difficult to implement during the war.

New calculation methods. When using additive manufacturing methods, the actual structure of most parts and assemblies must be changed. New parts will need to be shaped in a way that is favorable for manufacturing on the equipment chosen by the engineer for the additive technology. Also, the most important change is the change in the strength and reinforcing elements of structures and assemblies, since additive technologies offer the greatest advantage when using topological optimizations. Accordingly, a sufficiently deep scientific and more applied basis is needed for such a design. Therefore, it will be necessary to modernize both old solutions and create new ones to meet new requirements, which also takes additional time for engineers.

Introduction of new production standards or their deep modernization. The current production standards are already quite outdated, having been written decades ago, and focused on traditional manufacturing methods. So now we have to adjust a lot of things, including the accuracy parameters for manufacturing, and implement the latest global practices in this area [4].

Conclusion. In conclusion, additive manufacturing has many advantages over traditional methods, and while it makes no sense to completely abandon classical production methods, it makes sense to combine different methods to get the best result. Aviation companies in Ukraine are hiring more and more young and ambitious engineers who are ready to implement new technologies and solve the problems mentioned above.

References:

1. Najmon, J. C., Raeisi, S., & Tovar, A. Additive Manufacturing for the Aerospace Industry. 2019. Vol.15(2). pp. 7–31. https://doi.org/10.1016/B978-0-12-814062-8.00002-9

2. Badiru, A. B., Valencia, V. V., & Liu, D. Additive Manufacturing Handbook: Product Development for the Defense Industry. Air Force Institute of Technology (AFIT) – Dayton, Ohio, 2017 by Taylor & Francis Group, LLC. Vol.1 pp. 31–36.

3. Pillet M., Adragna P-A., Germain F., Inertial Tolerancing: "The Sorting Problem", Journal of Machine Engineering: Manufacturing Accuracy Increasing Problems, optimization, Vol. 6, No. 1, 2006, pp. 95-102.

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Pavlova V., Korobko I.

ANALYSIS OF AVAILABLE METHODS FOR DETERMINING THE THRUST ECCENTRICITY OF A ROCKET PROPULSION SYSTEM

Introduction. A rocket engine must create the proper thrust with the appropriate characteristics, so the question of its evaluation arises. Is it of the proper size, was it symmetrical relative to the center of mass of the spacecraft? The asymmetry of the thrust force of the rocket power plant determines the accuracy of bringing the device to a given point in space, its controllability, the direction of flight, etc. Therefore, solving the problem of estimating the thrust asymmetry of the power plant of the space vehicle will provide the possibility of making changes in the operation of the rocket engine in advance.

Scientific and technical results. Methods for determining the thrust vector (ETV) of a rocket engine can be divided into two categories - numerical and experimental (direct and indirect). Sometimes the computational and experimental groups are distinguished as an intermediate group.

When the engine is running, it is necessary to know whether the TV and its characteristics differ from the designed value in the given conditions. One of these parameters is the eccentricity of the thrust vector (ETV), which is one of the defining characteristics of rocket engines (RE).

The asymmetry of the TV is influenced by many factors, among which the main ones are: the design features of the nozzle and the gas-dynamic characteristics of the spatial flows in it.

Various numerical methods are used to study continuous supersonic spatial gas flows in nozzles [1].

The layer-by-layer method of characteristics. In the classical scheme of the method, nodes of the difference grid are determined in the process of numerical solution as points of intersection of characteristics. Its main advantage is that a grid of this type allows maximum consideration of the current structure, in particular, calculation of rarefaction waves, identification of lines of weak breaks, and determination of areas of hanging shock waves.

Disadvantages include the inconvenience of calculating the value that must be found at the nodes of the previously unknown characteristic grid. [2].

Method of small perturbations. This is an asymptotic method of studying gas flows in the transonic region of the nozzle. In 1965, the solution of the equations obtained by the method of small perturbations for the general case of spatial unsteady flow in the transonic region was given, and the conditions leading to the occurrence of shock waves were also formulated [2, 3].

The difference method of the second order of accuracy. This method is based on the technique of the difference scheme with the agreed approximation of convective flows [3]. The flow is described in Euler variables. Alignment by flow approximation is an element of the construction of completely conservative difference schemes and imposes sufficiently strict restrictions on the form of recording the difference equations.

Method of end-to-end calculation. Many problems of gas dynamics are characterized by the presence of surface discontinuities in the flow. This method of calculating discontinuous solutions of gas dynamics equations consists in the fact that discontinuities are distinguished in the calculation process [2, 4]. At the same time, the Rankine-Hugonio conditions are satisfied at the discontinuities, and in the region of the smooth solution, the differential equations are integrated using any sufficiently accurate difference scheme.

Experimental methods. Most experimental methods involve the use of special test stands of various types. Depending on the parameter registered on the stand, direct and indirect (mediated) methods are distinguished.

With the indirect method, the pressure in the RE combustion chamber is recorded on the test bench, and the thrust value of such a power plant is determined based on its values [5]:

$$P = k_T p_{кз} F_{кр}, \tag{1}$$

where k_T is the traction coefficient;

$p_{кз}$ - pressure in the combustion chamber;

$F_{кр}$ - the critical cross-sectional area of the nozzle.

However, the indirect method gives low reliability of information about the $p_{кз}$ value due to the unevenness of the fuel combustion process over the entire volume of the combustion chamber. It is also worth taking into account the impossibility of measuring the thrust of an RE with an adjustable nozzle and determining the deviation of the line of action of the thrust vector from the axis of the combustion chamber.

With the direct method of measurements, the traction force is determined directly using bench equipment. Test stands are equipped with special traction measuring devices and consist of a machine, measuring and tare system (Fig. 1).

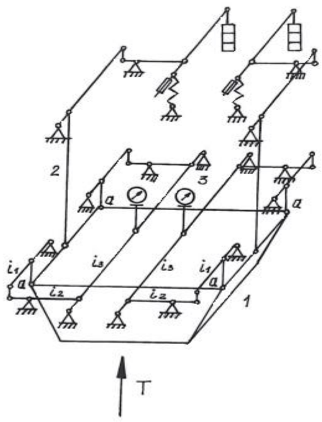

Figure 1 - Diagram of a lever traction-measuring device

The machine consists of two main elements: a fixed bed and a frame. The frame, which can be movable or stationary, perceives the thrust of the power plant and transmits it to the force measuring device and the frame. When calibrating a rigid-type machine, the simulation of engine traction is achieved by creating pressure in the hydraulic cylinder, which transmits force to the frame using a rigid connection.

In rigid machines, mechanical, hydraulic and electrical force measuring transducers (FMT) are used as thrust measuring transducers (TMT).

Mechanical and hydraulic FMTs are significantly inferior in accuracy to electric FMTs when measuring rapidly changing values of thrust in transient and impulse modes of engine operation.

The mechano-electrical converter itself (tensor resistor) has much smaller dimensions and weight compared to other converters.

Conclusion. Summing up, it is worth noting that the method of determining ETV is chosen according to the available resources and the set tasks. Experimental methods give better results and should be used wherever possible. The indirect measurement method is used only when the direct method cannot be applied.

Therefore, to determine the instantaneous value of the parameters (module and coordinates of the line of action) of the LPRE thrust vector in any mode of operation, the experimental method is the most appropriate - a direct method of measurements using electrical force measuring transducers.

References:

1. Rathakrishnan, E. (2010), Applied Gas Dynamics, John Wiley & Sons (Asia) Pte, Ltd.

2. Canosa, A. and Jiménez, E. (2022), “Laval Nozzles: Design, Characterization, and Applications”, Uniform Supersonic Flows in Chemical Physics, June, pp. 63-172

3. Richtmayer, R. D. and Morton, K. W. (1967) Diflerence Methods for Inilial- Value Problems, Interscience, New York

4. Johnson, R. C. (1965). Real-gas effects in critical-flow-through nozzles and tabulated thermodynamic properties. Washington, D.C., National Aeronautics and Space Administration.

5. Martins, C. A, (2011), “Development of test stand for experimental investigation on of chemical and physical phenomena in Liquid Rocket Engine” , Journal of Aerospace Technology and Management, vol.3, pp. 159-170.

6. Kovalenko, N., Kukushky`n, V., Ky`ry`chenko A., Y`gnat`ev A., (2014) “Eccentricity of the thrust vector of the adjustable large-sized SPRE with the injection of chamber gas into the supersonic part of the nozzle. № 2(107). pp. 23-30

7. GOST 102732-93. Aviation gas turbine engine test stands. General requirements for the compilation, content, design and approval of the technical task for the test stand. Valid from 01.07.93, 10 v.

UDC 536.24

Petrichenko O., Lobunko O.

USAGE OF SOLID PROPELLANT REACTIVE ENGINES AND PROBLEMS ASSOCIATED WITH IT

Introduction. Solid rocket engines (SPREs) are the oldest method of launching rockets. The first prototypes of this type of chemical rocket engine appeared 2,000 years ago in China [2]. Since then, solid rocket engines have been used in numerous aerospace systems, both civilian and military.

The main objectives of this work are to determine modern approaches to the organization of thermodynamic processes and structural solutions for SPRE elements, based on which conclude current trends, advantages, and possible risks in the use of such chemical rocket engines.

Scientific and technical results. As the name implies, a solid rocket engine is a chemical engine that converts the chemical energy of solid fuel combustion into thrust. Throughout the entire period of use, solid rocket engines have been distinguished by their low cost of production, reliability, ease of development, and ease of achieving multi-stage multiple engines. One of the design variants of the SPRE is presented below (Fig. 1)[1].

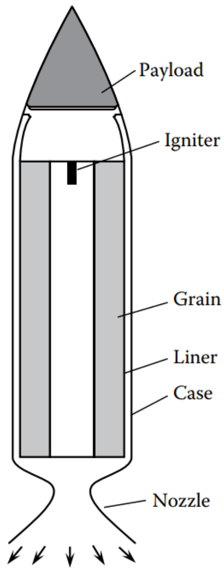

Figure 1 – Design of SPRE

In this version of the design of the SPRE, an igniter is provided to start the fuel combustion process. During this process, the resulting high-temperature working fluid (gas) passes through the jet nozzle at high speed and generates thrust.

It should be noted that the combustion process of solid fuels is quite difficult to control [4]. The walls of the combustion chamber and the jet nozzle are constantly (from the beginning to the end of the engine operation) subjected to a rather high thermal load. In addition, the design and use of active cooling of this engine are problematic due to the specifics of the engine structure (due to the agregate state of the fuel). Accordingly, one of the problems in the operation of the SPRE is the possible excessive deformation of the nozzle walls due to overheating.

Another problem when using a solid rocket engine is the possibility of fuel explosion if there are cracks in it. This is due to the chemical process of explosion. An explosion is a rapid chemical reaction. When fuel is burned, the air in the crack acts as an oxidizer and accelerates the chemical reaction of combustion, which causes detonation. The consequences of such a reaction depend on the individual characteristics of a particular engine and range from a shift in course to a full-ledged rocket explosion [3].

It is proposed to focus the search for concepts to improve the technical characteristics of the SPRE to meet the needs of advanced aircraft in the following areas: research and justification of the choice of thermogas-dynamic parameters in the combustion chamber and jet nozzle; choice of type and form of fuel, calculation and experimental studies of charge and progressivity characteristics; modeling of the stress-strain state of especially critical components of the SPRE; calculation and experimental studies of alternative materials for heat shielding.

Conclusion. Due to its advantages and disadvantages, it can be noted that SPREs are suitable for short-term launches. Currently, these engines are used as "boosters" for launching civilian rockets and satellites into orbit, and in the defense sector as a way to provide the initial acceleration and corresponding speed of a weapon at the lowest possible cost. The size of the engine can be improved (selected) as a result of a set of research and development activities.

References:

1. D.P. Mishra (2017). *Fundamentals of Rocket Propulsion.* CRC Press. https://ftp.idu.ac.id/wp-content/uploads/ebook/tdg/DESIGN SISTEM DAYA GERAK/Fundamentals of Rocket Propulsion.pdf

2. Tom Benson (2021, May 13) *Brief History of Rockets*. https://www.grc.nasa.gov/www/k-12/TRC/Rockets/history_of_rockets.html

3. Adam Augustyn (2023, October 16) *Challenger disaster* https://www.britannica.com/event/Challenger-disaster

4. Joseph Banks (2023, April 17) *Chemical Propulsion Systems* *https://www1.grc.nasa.gov/research-and-engineering/chemical-propulsion-systems/*

UDC 536.24

Plisetska E., Lobunko O.

THERMAL ANALYSIS AND PROTECTION OF MODERN AEROSPACE SYSTEMS

Introduction. In the process of flying at supersonic speed in the dense layers of the atmosphere, the temperature fields in some parts of the outer surface of aerospace objects reach more than 1600 °C, while the operating temperature of their constituent parts should not exceed 150 °C. The use of materials based on stable high-temperature metals, typical for supersonic aircraft, does not satisfy this requirement, as their thermal and stress-strain state changes significantly. Degradation of the mechanical and chemical properties of metal construction materials under the influence of high temperatures is observed. The study of the mechanisms of thermal destruction of components of aerospace systems and the justification of the choice of thermal protection materials is an urgent scientific task. The materials from which the thermal insulation is made must reliably protect the main materials of the structure, have appropriate optical characteristics for effective "management" of heat flows perceived by the coating of the aerospace object, and be heat-resistant in the process of repeated use for at least 100 cycles [1].

Scientific and technical results. Depending on the expected conditions of thermomechanical influence during the flight, the outer surface of the aerospace object is conditionally divided into several parts that require rational thermal protection. Each of the selected parts must have individual properties of protective equipment, be distinguished by a certain set of characteristics and structure.

Carbon fibre reinforced carbon (CFRC) is a composite material consisting of reinforced carbon fiber in a graphite matrix. It is developed for ballistic missile launchers and is widely known as a material for the nose cone and wing leading edges of the Space Shuttle [2]. It is well suited for structural applications at high temperatures (up to 2000 °C) or where resistance to thermal shock and (or) a low coefficient of thermal expansion is required [3].

High-temperature reusable surface insulation (HRSI) is a spacecraft insulation tile that is 90% air and the remaining 10% is 99.8% amorphous fiber [4]. This protection is designed to withstand the transition from regions of extremely low temperature (the vacuum of space, about −270 °C) to high temperatures (about 1600 °C) on re-entry (caused by the interaction, mainly compression in the hypersonic shock, between the gaseous medium of the upper atmosphere and spacecraft body). Therefore, this insulating tile is a poor conductor, preserves the integrity of the structure and is crack-resistant [5].

Low-temperature Reusable Surface Insulation (LRSI) – it is a variation of the HRSI. This type of coating has a high heat reflectivity, which minimizes heat flows from the Sun in the orbital section of the flight. They are used in parts where the temperature ranges from 400˚C to 650˚C. Due to the lower loading temperature in these areas, LRSI tiles are larger and thinner [6]. According to the same thermal characteristics, it is possible to distinguish advanced flexible reusable surface insulation (AFRSI) – it is a fibrous silicate batting of low density, which consists of high-purity silica and 99.8% of amorphous silica fibers. They are used to reduce the weight of the space system [3].

On the basis of the conducted research, it is proposed to continue the search for concepts for the reuse of aerospace systems, including the necessary technologies of thermal analysis and ensuring the protection of the most scarce, high-cost and responsible structural elements. Determination of the technological possibility and scope of their recovery (repair), checks of compliance with norms before a new start. Initiative research work corresponds to the individual goals of the "Horizon Europe" program [7].

Conclusion. As a result of the conducted preliminary stage of thermal analysis, it was found that the use of traditional high-temperature metals for thermal protection of aerospace objects is ineffective. Alternatives in the form of non-metallic coatings are presented. It is these coatings that withstand high operating temperatures and their extreme changes and have poor thermal conductivity, due to which internal mechanical structures are not subjected to excessive heating. The highlighted data emphasize the importance of in-depth thermal analysis of aerospace objects and the informed choice of materials and insulation technologies to ensure the safety and efficiency of aerospace missions under conditions of extreme thermal loads.

References:

1. Borodai M. V., Kolomiets I. D. Study of the effect of temperature on the optical characteristics of thermal insulation of a reusable spacecraft, 2011. 6 p.

2. Carbon Fiber-Reinforced Carbon: properties, uses, forms supplied. URL:http://edge.rit.edu/content/P07109/public/Design%20I/Published%20Research%20Documents/CARBON%20FIBER%20PROPERTIES.pdf

3. Wei Heok Ng. Thermomechanical analysis of a damaged thermal protection system. Austin: 2007. 161 p.

4. Jim Dumoulin. Space shuttle orbiter systems: thermal protection system. URL:https://www.globalsecurity.org/space/library/report/1988/sts_sys.html

5. Ryan Oakes. Space Shuttle Ceramic Tiles. URL:https://depts.washington.edu/matseed/mse_resources/Webpage/Space%20Shuttle%20Tiles/Space%20Shuttle%20Tiles.htm

6. Moo Y. Lee, John H. Hofer, David R. Bronowski and Robert D. Hardy Mechanical Properties of Thermal Protection System Materials. USA: 2005. 60 p.

7. Reusability for European strategic space launchers - technologies and operation maturation including flight test demonstration. European Union Framework Program for Research and Innovation "Horizon Europe" (2021-2027).

UDC 629.735.33

Radchenko S., Konotop D.

THE SPECIAL-PURPOSE QUADROCOPTER CARGO DROP SYSTEM

Introduction: The research focuses on developing a safe, reliable, and efficient cargo drop system for special-purpose quadcopters. Despite the capabilities of modern special-purpose quadcopters, existing load shedding systems face reliability, accuracy, and load capacity limitations. The meta-research aims to create a download reset system meeting specific requirements: load capacity, unloading accuracy, reset speed, and safety.

Project Goals:

1. Overview of Modern Special Purpose Quadcopters: Conduct a comprehensive review of modern special-purpose quadcopters, outlining basic requirements for a cargo drop system, including types of quadcopters, cargo types, and drop conditions.
2. Development of an Experimental Model of the Unloading Mechanism: Create an experimental model of the unloading mechanism meeting the defined requirements for the cargo unloading system.
3. Determination of Requirements for Reliability and Operational Quality: Establish requirements for the reliability and operational quality of the reset mechanism based on the review of modern quadcopters and experience with similar systems.
4. Development and Installation of Sensors for Aircraft Position: Install position sensors, such as gyroscopes, accelerometers, and magnetometers, to ensure the accuracy of the cargo drop system by receiving information about the aircraft's position in space.
5. Selection of Parameters to Control Functional Systems: Select parameters that respond sensitively to deviations from the normal operation mode to ensure safe operation of the reset system.
6. Description of Maintenance of the Unloading Mechanism: Develop a maintenance plan, including external inspection, performance checks, and replacement of worn parts, to ensure uninterrupted operation of the cargo unloading system.

Specific Stages of Project Implementation:

1. Stage 1 (6 months):
   - Review of modern special-purpose quadcopters;
   - Development of an experimental model of the reset mechanism;
   - Determination of requirements for reliability and operational quality.
2. Stage 2 (12 months):
   - Development and installation of aircraft position sensors;
   - Selection of parameters for controlling functional systems;
   - Development of a startup project.
3. Stage 3 (6 months):
   - Testing and evaluating the download reset system;
   - Preparation for the launch of the startup project;

Total Project Budget: 1 million USD

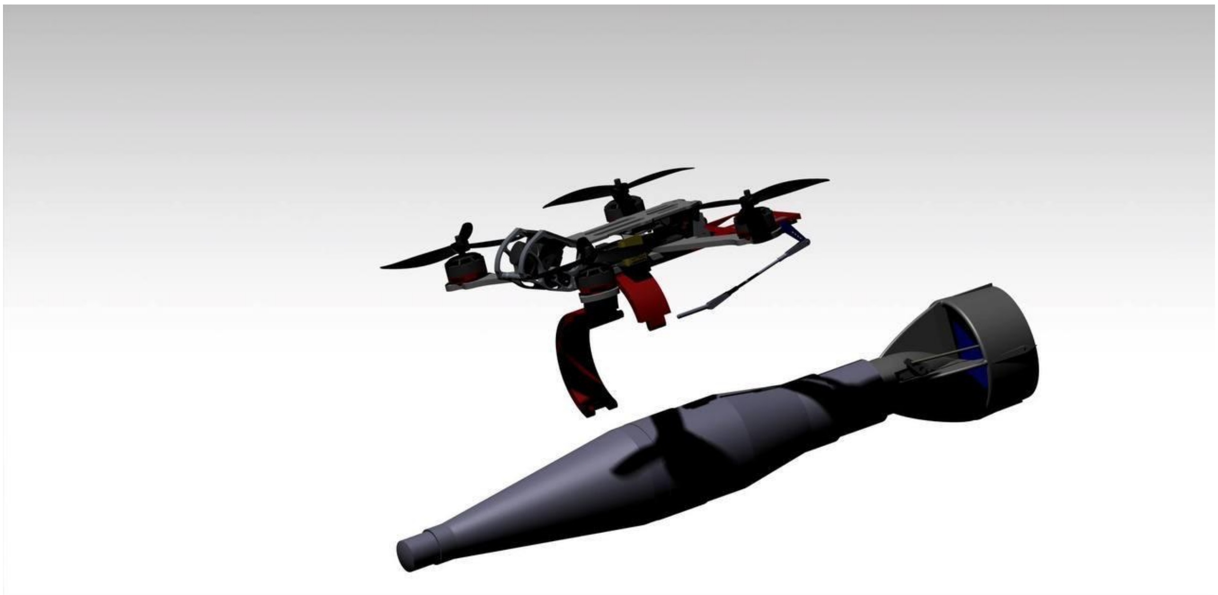

Figure 1 – Basic model of the system for discharging the vantage of a quadcopter for special purposes

Conclusion:The implementation of the project, "Special Purpose Quadcopter Cargo Drop System," aims to create a safe, reliable, and effective cargo drop system. This has significant potential for application in various fields, including rescue operations, firefighting, military operations, and more. The project addresses current shortcomings, promising a system that meets all specified requirements, thereby potentially revolutionizing the field of special-purpose quadcopters for increased efficiency and safety.

References:

1. Quadcopters: theory and practice / Ed. V.S. Klyueva. – K.: COURSE, 2022.
2. Quadrocopters: design and production / S.V. Girchakov. – K.: COURSE, 2022.

UDC 681.52

Savenko L., Bobkov Y.

CONTROLLING THE MOVEMENT OF THE QUADCOPTER USING A TECHNICAL VISION SYSTEM

Introduction. In recent years, unmanned aerial vehicles (UAVs) have been widely used in both military and civilian fields. Receivers of global navigation satellite systems (GNSS) in combination with a unit of inertial sensors of spatial orientation [1] are the basis of UAV navigation systems. However, inertial navigation systems have the disadvantage of increasing error in determining coordinates over time, and GNSS signals may be unavailable in a certain area or may be distorted by natural or artificial interference.

The usage of technical vision systems (TVS) is one of the possible ways to solve these problems. In general, a TVS can be used either as a supplementary system to traditional navigation systems or as the main UAV movement control system. In both applications, the structure and functions of the TVS are in most cases the same.

This work aims to develop a quadcopter movement control system based on the TVS signals.

Scientific and technical results. The basis of the autonomous movement of the quadcopter is the observance of a certain predefined route, which can be set in various ways.

The following are the main options for using TVS to control the movement of a quadcopter when flying along a predetermined route. Among them are the application of reference photos [2], navigation landmarks (NL) [3], the stereo effect for flying over terrain [4], and the SLAM algorithm [5] ("Simultaneous localization and mapping").

The analysis has shown that the application of the SLAM algorithm requires a significant amount of additional information and high computing power, which is not always possible for the onboard system of a conventional quadcopter.

The disadvantages of using reference photos are the high amount of information required to store them on board, as well as the significant computing power to compare the reference photos and the received images in real-time.

There are two scenarios when using NL: flying by artificial or natural landmarks. Flying with artificial NL requires the development of a special system of aids and its maintenance. When using natural landmarks in the form of landscape features or general-purpose artificial objects (buildings, roads, etc.), the disadvantage is the influence of seasonal and weather factors and man-made impacts. The disadvantages of terrain-based flight are similar to those of natural landmarks.

Flying according to the previously defined system of a limited number of NLs makes it possible to reduce the amount of information about a predetermined route and the requirements for onboard information and computing resources. For this reason, this approach will be considered for controlling the movement of a quadrotor using TVS.

At the preflight stage, all the NLs along the route are loaded into the onboard system's memory. A certain preselected system of features is used to describe the UAS. Various machine-learning algorithms are frequently applied at this stage.

Throughout the flight, at each part of the route, the direction of movement of the quadcopter is selected as the one towards the nearest NL. When the current NL goes out of the frame, the TVS searches for the next NL and then adjusts the direction of the quadcopter according to that NL's location.

For the initial description and further detection of the navigation landmark on the images from the TVS camera, it is proposed to use feature extraction methods, for example, SURF or ORB algorithms. The next step is to determine the centre of mass of the NL and its location in the frame concerning the direction of the UAV's movement. Based on the coordinates of the NL's image in the frame, the direction of the quadcopter's heading is determined by calculating the angle between the current direction of movement and the direction toward the NL using simple trigonometric relations. The calculated angle is transmitted to the control system to change the heading. After the last navigation landmark is detected, the quadcopter proceeds to the final goal of the flight task, for example, landing at a given point, dropping cargo, etc.

Conclusion. This paper considers the application of a technical vision system for controlling the movement of a quadcopter while flying according to the system of navigation landmarks, which is loaded into the memory of the onboard system at the stage of preflight preparation. There are proposed algorithms for determining the position of the NL in the frame of the TVS and the following changes in the heading of the quadcopter.

References:

1. Zhang, Y., Gao, Y. A method to improve the alignment performance for GPS-IMU System. GPS Solutions, 2007. No.11 (2), pp.129–137.

2. Conte, G., Doherty, P. An Integrated UAV Navigation System Based on Aerial Image Matching. Aerospace Conference, Montana, USA, 1–8 March 2008. 10p.

3. Zhang, T., Kang, Y., Achtelik, M., Kühnlenz, K., Buss, M. Autonomous Hovering of a Vision-IMU Guided Quadrotor. Proceedings of the International Conference on Mechatronics and Automation, Changchun, China, 9–12 August 2009, 6 p.

4. Stefanik, K. V., Gassaway, J. C., Kochersberger, K., Abbott A. L. UAV-Based Stereo Vision for Rapid Aerial Terrain Mapping. GIScience & Remote Sensing, 2011. No.48 (1), pp. 24–49.

5. Munguía, R., Urzua, S., Bolea, Y., Grau, A. Vision-Based SLAM System for Unmanned Aerial Vehicles. Sensors. 2016. No.16 (3), 372p.

UDC 629.783

Shkolnyi V., Bondarenko O.

REDUCING LOADS ON THE WING USING CONTROL SURFACES AT CRITICAL SPEEDS

Introduction. The reduction of wing loads is a big deal in today's aircraft engineering, especially when it comes to aeroelastic flutter situations. This study is all about creating a new way to figure out the best angles to move control surfaces, so we can lower the loads on wings, especially when they're going at speeds that might cause flutter. As planes get faster and more efficient, having really good aeroelastic design becomes important.

Scientific and technical results. This research represents a groundbreaking approach to aeroelastic design. It introduces a carefully crafted technique that thoroughly considers the complexities associated with critical speeds during aeroelastic flutter. Importantly, the method presented in this study isn't just for theoretical discussions, however directly into practice.

The main idea of this research is to deflect control surfaces to align the center of lift with the stiffness center of the structure, see figure 1.

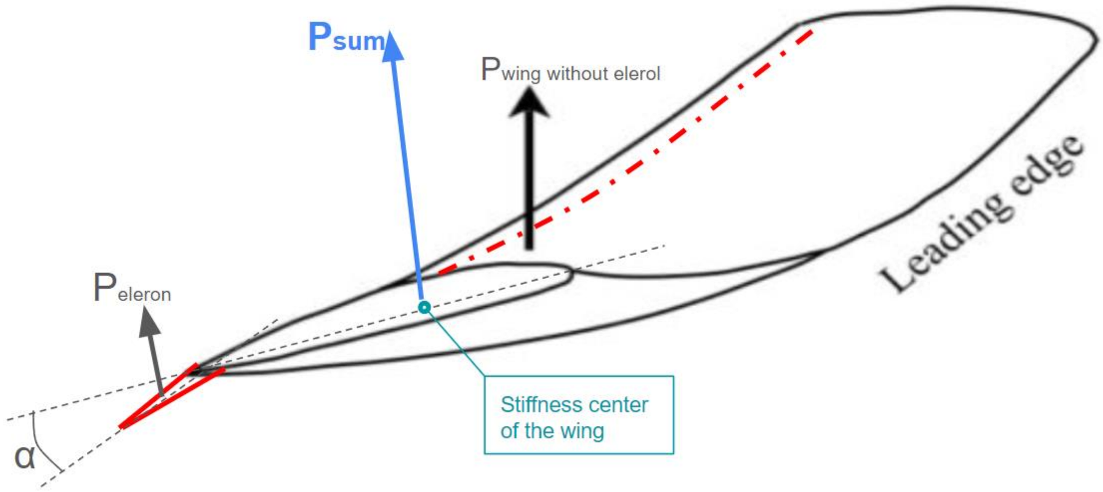

Figure 1

There will be installed vibration sensors and linear strain gauges on the wing which transmits data on the autopilot. An algorithm will be developed that will make it possible to work synchronously and provide correct signals to servos which deflect control surfaces.

The method shows great promise for being integrated into modern fly-by-wire systems. This could enhance the aircraft's ability to respond quickly to different aerodynamic conditions, almost in real-time.

Conclusion. The goal of the research is to reduce loads on wings, particularly during speeds that could trigger flutter. As the aviation industry continues to seek and embrace innovative solutions, the implementation of this method holds the promise of substantial advancements in aeroelasticity and improvement of aviation safety.

UDC 629.19

Sudyma Y., Pysarets A.

THE USE OF THE HOHMANN TRANSFER AS THE SIMPLEST INTERORBITAL MANEUVER

Introduction. The Hohmann transfer is an orbital maneuver used by a spacecraft to travel between two circular orbits of different altitudes along an elliptical trajectory cotangent to the initial and final orbits in the central gravitational field. Since this concept was introduced by Walter Hohmann [1] scholars have been mainly focused on proving that it is the most fuel-efficient two-impulse transfer. The analytical proofs were provided by Palmore [2] and Barrar [3]. Marec [4] presented a proof using graphical construction and Hazelrigg [5] used Green's theorem. The researchers also addressed the best calculation methods, time, and energy performance.

In our century, the Hohmann transfer is considered the most fundamental space maneuver. However, there are a few issues that still concern engineers. The gravitational interaction of celestial bodies creates challenges in fuel assumptions and launch time estimates, and many researchers are trying to solve these problems with modeling.

Scientific and technical results. Today, there is no doubt that the Hohmann transition is the most energy-efficient since it is designed in such a way that fuel is used only to change the magnitude of the vehicle's velocity vector, but not its direction. It is also the easiest in terms of trajectory calculation [6], but its use is complicated by the fact that it requires some preconditions. For the Hohmann transfer between two celestial bodies to be possible, the launch and target bodies must be in a certain position relative to each other, defined by an angle. Missions must wait for the launch window, i.e. the time when such an alignment happens. For travel between Earth and Mars, for example, the launch window occurs about every 25 months [7]. The Hohmann transfer orbit also determines the invariable time required to travel between the start and end points, which is half of a period of the transfer orbit.

The algorithm described in this article is designed to determine the launch windows on the time scale requested by the user for spacecraft traveling between the Earth and other planets of the Solar System, in both descending and ascending cases; to calculate and generate a visual representation of what the Hohmann transfer might look like.

In the idealized case, the initial and target orbits are circular and coplanar. Therefore, the Solar System is assumed to be two-dimensional in this paper: all orbits belong to the same plane. Each orbit is considered circular. We also operate under the assumption that the origin and destination planets have no gravitational influence on the spacecraft to transform the model into a two-body problem.

For the calculations, we used the gravitational constant, the mass of the Sun, and the characteristics of the planets' orbits [8].

The most optimal solution was to compute the orbits of all the planets at once to use the obtained coordinate arrays, rather than performing calculations for each case individually. These arrays are easy to determine since the orbits are defined as circular. Assigning a date to an array element is done by simply counting up from the date previously defined as the start date.

Launch windows are determined by the angle between the Earth and destination planet at the moment of spacecraft departure. This is the angle the target planet will cover in its orbit over the course of the spacecraft's flight. The windows are found by checking the positions of both planets each day within the specified time frame.

The created model plots the motion of the planets involved in the mission and the spacecraft simultaneously, from the selected launch date up to arrival, using the arrays of pre-calculated coordinates.

Conclusions: Based on the simple algorithm discussed in this article, we wrote a code in the MATLAB software that allowed us to model interplanetary missions that utilize the Hohmann transfer, and to predict launch windows and flight trajectories of a hypothetical spacecraft. The potential for further work may include incorporating time and energy estimates into the algorithm and specifying elliptical orbits instead of circular ones.

References:

1. Hohmann W. The Attainability of Heavenly Bodies, National Aeronautics and Space Administration, 1960.

2. Palmore J. An elementary proof of the optimality of Hohmann transfers. Journal of Guidance, Control, and Dynamics, 1984.

3. Barrar R. B. An analytic proof that the Hohmann-type transfer is the true minimum two-impulse transfer. Astronautica Acta, 1963.

4. Marec J.P. Optimal space trajectories, Studies in Astronautics, vol. 1, Elsevier, Amsterdam, 1979.

5. Hazelrigg Jr. G. A. Globally optimal impulsive transfers via Green's theorem. Journal of Guidance, Control, and Dynamics, 1984.

6. Prussing J. Simple proof of the global optimality of the Hohmann transfer. Journal of Guidance, Control, and Dynamics, 1992.

7. NASA. Basics of Space Flight, Section 1, Chapter 4, "Trajectories", 2017.

8. Dr. David R. Williams, NASA. Planetary Fact Sheets, NASA Goddard Space Flight Center, 2016.

UDC 629.052.7

Tarash H., Cherniak M.

CALIBRATION OF INERTIAL MEASUREMENT UNIT ON LOW PRECISION EQUIPMENT IN OPERATION

Statement of the problem. It is necessary to determine metrological model (MM) passport coefficients of the navigation accuracy class IMU. At the same time, it is known that MM coefficients change over time. This leads to a deterioration in the accuracy of determining the apparent acceleration and angular velocity projections during the IMU operation, so its recalibration becomes necessary. The calibration in the manufacture is a well-established procedure, which is usually carried out by setting certain test positions using precision rotary stands. The disadvantage of this approach is its limited applicability due to the large size, special conditions of use, high cost of the test bench equipment and the need for skills to work with it [1]. All of these facts make it impossible to carry out such a precise calibration using only the facilities of the IMU operator, which rarely include such equipment for setting high-precision test positions. This raises the problem of developing a method of high-precision calibration using low-precision equipment available to the operator. The literature describes such a method. It consists in measuring the norm of the gravity vector (for accelerometers) or angular velocity (for gyroscopes) with iterative [2,3,5], and non-iterative (using a specific mathematical apparatus) methods [4]. But it has already become traditional to use the MM, which is described by three zero offset coefficients (ZOC), three transformation coefficients (TC) and three combinations of non-orthogonality angles (NOA). The combinations of NOA cannot be separated, because they refer to different elements of the IMU sensor triads. Such a model requires physical alignment of one of its axes with the building axis of the object on which the IMU is installed. However, the MM based on 12 coefficients is used in operation. Therefore, the problem is to separate the values of the six NOAs from the values of the NOAs combinations obtained during in-service calibration.

Statement of the main materials of the study. The proposed calibration method consists in setting not less than nine random calibration positions relative to the gravity vector and the angular velocity vector, as well as six additional positions that provide for a high-precision 180° rotation around each of the IMU measuring axes. The basic mounting surface on which the IMU will be installed can have a deviation from the horizon plane of up to 3°.

After receiving the output signals via the channels of linear acceleration and angular velocity of each of the IMU axes, calculations are performed according to the following algorithm:

1. Initialisation of the predefined by the manufacturer passport (hereinafter – the initial) values of MM coefficients.

2. Formation of a SLNHAE in relation to the desired differences of unknown actual and initial MM coefficients passport values based on the received output signals in at least 9 random positions.

3. Solving the SLNHAE: determining the increments to the initial values of the MM coefficients and combinations of the NOAs.

4. Determination of the sums, which consist of the increments (obtained in step 3) and the initial (step 1) MM passport values for the ZOC and TC.

5. Determination of the NOAs on the basis of the received output signals in 6 additional positions.

6. Formation of the condition statement, which consists in comparing the NOAs combinations (obtained in step 3) with the NOAs combinations calculated with the NOAs values obtained in step 5.

6.1. If the difference between the NOAs combinations obtained in step 3 and step 5 does not exceed the acceptable value, we consider that the MM coefficients were determined correctly in accordance with step 4.

6.2. If the difference exceeds the acceptable value, repeat steps 1 – 6, using in step 1 the MM coefficients obtained in step 4 instead of the initial ones.

After determining all the MM parameters, the quality of the calibration is checked by the complex accuracy parameter at a random orientation of the fixed IMU relative to the horizon plane.

Conclusions. The proposed use of the six additional calibration positions makes it possible to unambiguously determine the real values of all twelve MM coefficients on non-precision equipment with an accuracy that corresponds to the primary manufacture calibration. So, this method can be recommended for calibration in operation without restrictions on the accuracy of the bench equipment.

References:

1. Cherniak, M., Rybak, V., & Palushok, V. Calibration with specified accuracy of triaxial block of navigation accelerometers on the uniaxial rotary stand. MECHANICS OF GYROSCOPIC SYSTEMS, 28, 5–15.

2. Skog, I., Nilsson, J. O., Handel, P., & Nehorai, A. Inertial Sensor Arrays, Maximum Likelihood, and Cramér–Rao Bound. IEEE Transactions on Signal Processing, 64(16), 4218–4227.

3. Hassan, M. U., & Bao, Q. A Field Calibration Method for Low-Cost MEMS Accelerometer Based on the Generalized Nonlinear Least Square Method. Multiscale Science and Engineering, 2(2–3), 135–142.

4. Cherniak, Kryvonozhenkov, & Chornomorets. Calibration of the triaxial navigation accelerometers unit using the mathematical apparatus of elliptic functions. MECHANICS OF GYROSCOPIC SYSTEMS, 0(37), 51–60.

5. Yang, J., Wu, W., Wu, Y., & Lian, J. Improved Iterative Calibration for Triaxial Accelerometers Based on the Optimal Observation. Sensors, 12(6), 8157–8175.

UDC 629.783

Edanur T., Lobunko O.

HYBRİD PROPULSION SYSTEMS OF AİRCRAFT

Introduction. A hybrid electric aircraft (HEA) uses a combination of traditional fossil fuel-powered engines and electric motors to provide propulsion. HEAs typically use electric motors for takeoff and landing, while conventional engines give power while in the air. Batteries or other energy storage systems, such as fuel cells, power electric motors [1, 2].

HEA technology is still in the early stages of development and will be an urgent scientific task in the future.

Scientific and technical results. *Charge.* Hybrid electric aircraft (HEA) technology is still in the early stages of development, but some figures are available from companies doing research and testing. That said, all HEAs are different based on size and structure. Those planes are expected to take 30 to 50 minutes to charge.

In an earlier 2015 test, engineers from the University of Cambridge tested a plane with a "parallel hybrid-electric propulsion system," suggesting that their design could also recharge its batteries during flight.

Range. Like charge time, a HEA's range depends on various factors, including the size and weight of the aircraft, the type and capacity of the energy storage system, and the specific operating conditions.

The Heart ES-30's 30-50 minute charge time is estimated to give a 250-mile range with an onboard turbine generator running on sustainable aviation fuel. With a full charge, Heart says the HEA can move 30 passengers and their bags 125 miles without the turbine generator.

HEAs use both traditional fossil fuel-powered engines and electric motors to provide propulsion, relying on batteries or other energy storage systems.

The primary fuel source for HEA varies based on design, but most use aviation fuel, including sustainable aviation fuel.

Electric motors, however, are commonly powered by batteries. Other energy storage systems, such as fuel cells, can be used, but batteries can be charged from various sources, so many options exist. These include renewable energy sources, such as solar or wind power, and conventional electricity from the grid.

Using electric motors and alternative fuel sources may reduce emissions compared to conventional fossil fuel-powered aircraft.

Main Components.

- Traditional engines are used to provide primary propulsion during cruising.
- Electric motors are used for takeoff, landing, and as an auxiliary power source during cruising.
- An energy storage system stores and delivers electrical energy to the electric motors.
- The energy storage system can be batteries, fuel cells, or a combination of both.
- Electronics are used to manage and control the flow of electrical energy between the energy storage system, electric motors, and traditional engines.

- Control systems manage the operation of the hybrid electric aircraft and ensure that the traditional engines and electric motors work together efficiently and safely.

Propulsion. The specific propulsion system can vary depending on the design of the aircraft, based on using electric motors for takeoff and landing, while the traditional engines provide power during cruising.

During takeoff and climb, the electric motors provide additional power to the aircraft, reducing the need for the traditional engines to operate at full power. Once the aircraft reaches cruising altitude, the conventional engines take over, and the electric motors are either turned off or used to assist the engines if needed.

Passenger Load. The previously mentioned Heart ES-30 is a 30-passenger electric aircraft, replacing the company's earlier 19-seat design, the ES-19. However, it has yet to be thoroughly tested.

Heart's engineering team conducted testing with a ground-based prototype of the ES-19's complete electric propulsion system in 2021. As for the ES-30, a full-scale integrated test facility with a full-flight simulator of the ES-30 has been built. A proof-of-concept aircraft is expected to be rolled out in 2024, with flight testing planned to start in 2026.

In 2020, a modified Cessna Caravan 208B with just a seat installed for the pilot flew for 30 minutes solely powered by electricity. VoltAero's Cassio, with options for four-, six- and nine-seat configurations, also completed a test flight.

Conclusion. Researchers are developing more efficient electric motors, optimizing the integration of electric and traditional propulsion systems, and creating aerodynamic designs that can reduce drag and improve efficiency. Materials research and structural engineering are ongoing in the aviation and aerospace field, creating materials that can withstand high temperatures and pressures.

On the certification, regulation, and infrastructure side, regulatory bodies must research and develop new infrastructure, standards, and certification processes for HEAs.

References:

1. Amprius (2023) The future of hybrid electric aircraft. Heart Aerospace, California, USA. https://amprius.com/hybrid-electric-aircraft/#:~:text=How%20do%20hybrid%20electric%20aircraft,or%20other%20energy%20storage%20systems.

2. Manuel A. Rendón, Carlos D. Sánchez R., Josselyn Gallo M., Alexandre H. Anzai. Aircraft Hybrid-Electric Propulsion: Development Trends, Challenges and Opportunities Journal of Control, Automation and Electrical Systems (2021) 32:1244–1268 https://doi.org/10.1007/s40313-021-00740-x

UDC XXX.XXX

Tomko D., Arkhypov O.

DESIGN AND DIAGNOSIS OF TANKS AND PIPELINES OF THE FUEL SYSTEM OF THE LAUNCH

Introduction. One of the main directions of the rocket and space industry of Ukraine is the modernization of existing and creation of new models of rocket and space technology in international cooperation. The implementation of this direction allows reducing the budget burden, getting rid of dependence on traditional partners, combining new technologies of the parties with the aim of creating rocket and space technology with higher characteristics, and obtaining financial support at the expense of commercial launch services.

The purpose of the study is to design tanks and pipelines of the fuel system of the launch vehicle, which combine high reliability at a low cost, as well as to determine and apply optimal non-destructive diagnostic methods that will ensure high reliability indicators of the fuel system of the launch vehicle.

Scientific and technical results. The fuel tanks of modern space vehicles are the main aggregates of products designed to accommodate liquid engine fuel and supply it to the combustion chamber. In addition, the power housings of the fuel tanks can perform a load-bearing function - to accept the load from the elements of the space vehicle located above. Special fuel displacement systems in zero-gravity conditions significantly improve tank design and manufacturing technology.

Rocket fuel system tanks are made of various materials: aluminum alloys, high-strength steel, composite materials and their combinations. Approaches to the selection of the parameters of the power elements of the tanks for the minimum mass are determined by their load schemes and the possible types of destruction associated with them. As noted in the source [1], most of the elements of missile bodies work on stretching or compression, which was implemented during the design.

In the designed rocket, the carrier tanks are inflated with gas generator gas: the fuel tanks are inflated with gas with excess fuel, the oxidizer tanks are inflated with gaseous oxygen. Thus, the tanks work on stretching, and not on the loss of stability, which makes their construction much easier.

Conclusion. On the basis of the calculations and justifications, the choice of aluminum alloy as a material for the manufacture of tanks of the fuel system of the launch vehicle is optimal in terms of the "price-quality-strength" ratio.

The choice of the method of checking tanks for tightness by manometric-differential and deformation methods is substantiated, which allows to increase the accuracy of the measurements.

References:

1. Duhopelnikov D.V., Ryazanov V.A., Shilov S.O. Comparison of the characteristics of a model engine with an anode layer when operating on xenon, argon and nitrogen. Proceedings of higher educational institutions. Machine building. Moscow, 2018. No.7, p. 69-76.

2. Zakharchenko M. G., Arkhypov O. G.. "Minimization of the mass of fuel tanks of a liquid rocket engine"; Deutsche Internationale Zeitschrift für zeitgenössische Wissenschaft 20 (2021). p. 63-68.

3. Cerro Jeffrey A., Wallace Matthew L., Wu K. Chauncey, Development of Non-Optimum Factors for Launch Vehicle Propellant Tank Bulkhead Weight Estimation (2012).

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Torokhtii M., Bondar Y.

OPTIMIZATION OF RESOURCE CHARACTERISTICS OF THE MAIN LANDING GEAR OF AN-TYPE AIRCRAFT CHASSIS

Introduction. When designing a new aircraft, it is necessary to provide the given resource characteristics. The service life of the structure, the reliability of the structure, the amount of maintenance costs and productivity depend on the accuracy of the calculations**.** Optimizing the characteristics of the structure at the design stage will further reduce the risks of structural failure due to premature initiation and growth of cracks, ensure optimal use of materials, and reduce maintenance costs.

Analytical, numerical and experimental methods can be used to determine resource characteristics.

When determining the number of cycles to failure of a part, it is important to know what stresses occur in the part under load, where the analytical method is difficult to apply for parts of complex shapes. The experimental method will give the most accurate result, but this method is expensive and requires more time. In such cases, the numerical method using finite element analysis can help.

The purpose of this work is to determine the critical support zones of the An-32 aircraft chassis, to optimize these zones, and to determine the number of cycles to failure.

Scientific and technical result. According to the certification standards [1], the loads acting on the chassis support are defined. To determine critical areas of chassis support, a typical flight program has been created, according to the Egbert Torenbeek methodology [2], in which flight modes are defined - start, taxiing before take-off, take-off, landing, run, taxi after run.

The resulting landing gear support loads were used to create a finite element model of the support and a finite element analysis was performed using the NASTRAN system using analysis type 101 Linear Statics in Simcenter 3D software.

After obtaining the result of the stress-strain state, the greatest stresses were found in the upper zones of the left and right traverse tubes where the radial transition is located. The method [3] of determining the number of cycles to failure for these stressed zones was used.

The SHERPA algorithm [4] was chosen for the optimization of critical zones. This algorithm is presented in the HEEDS software. For optimization, a range of radii values from 15 mm to 55 mm was set, and as a result of optimization, the most optimal radius is 48.5 mm. Larger values of the radius do not lead to a significant decrease in stress.

The results of shape optimization were calculated to determine the number of cycles to failure according to V.M. Dmitriev's method. [5]. As a result, for the upper zone of the left pipe of the traverse, the number of cycles to failure increased by 4.9 times, and for the right pipe by 2.26 times.

Conclusion.

1. Determination of critical zones in the structure of the chassis support of the An-32 aircraft and optimization of the geometry of the support improves resource characteristics up to 4.9 times.

2. Optimizing the geometry made it possible to reduce the risks of structural failure supports due to premature initiation and growth of the crack at the expense of the optimum use of materials, which allows to reduce maintenance costs.

3. Optimization of critical zones by selecting the optimal radius helps to increase the number of cycles to failure, which when done correctly selection of parameters to improve resource characteristics many times.

4. The obtained results showed that the optimization of the shape geometry can be significant to improve the weight efficiency of the support due to the improvement of resources characteristics.

References:

1. Certification specifications and acceptable means of compliance for large Aeroplanes CS-25. // European Union Aviation Safety Agency. – 2020.
https://www.easa.europa.eu/sites/default/files/dfu/cs-25_amendment_26_0.pdf.

2. Egbert Torenbeek. Advanced aircraft design. Conceptual design, analysis, and optimization of subsonic civil airplanes. Delft university of technology, the Netherlands,1988, Typeset in 10/12pt Times by Aptara Inc., New Delhi, India. ISBN: 9781119969303

3. Sabitov N.G. Design and strength of aircraft. MGA, KIIGA, Book, Kyiv, 1988. 169-230.

4. Red Cedar Technology. SHERPA – An Efficient and Robust Optimization/Search Algorithm. ISBN 978-1-118-56811-8 (cloth)

5. Dmitriev V. M. Scientific and technical report: Methodological recommendations for calculation assessment of the durability of chassis support parts at the design stage, -TSAGY, 1983. 76-78 p.

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Vorobei M., Ponomarov O.

IN-FLIGHT PRESSURIZATION OF HYDROGEN PEROXIDE TANKS

Introduction: The pressurization system of fuel tanks for rocket carriers is designed to maintain excess pressure in the gas cushion of the tanks in accordance with the requirements for fuel component delivery and tank construction. The use of the fuel components hydrogen peroxide + kerosene opens up new possibilities for optimizing the pressurization system.

Scientific and Technical Results: The type of pressurization system significantly shapes both the structural complexity of the rocket and the structure of the launch position, stand testing base, and, to a large extent, production capacities.The modern fuel component delivery system is the most expensive part of the rocket, following the liquid rocket engine. This necessitates the development of efficient and innovative pressurization methods to enhance the performance and flexibility of space missions.

Our research begins with a review of existing methods of tank pressurization, such as mechanical, thermal, and chemical approaches. We must have a clear understanding of the advantages and limitations of these systems to effectively apply advanced technologies. We will then explore innovative technologies, such as electrical pressurization and automated systems. The integration of hydrogen peroxide and kerosene provides us with the opportunity to further optimize energy efficiency and fuel delivery flexibility in-flight.

We will discuss the thermodynamic and aerodynamic aspects of implementing innovative pressurization systems into rocket structures. It is necessary to analyze engineering solutions aimed at optimizing weight, stability, and system safety, as well as the impact of innovative pressurization systems on overall performance, fuel efficiency, and maneuverability of rocket carriers. Finally, we will consider the prospects for cost reduction and mission flexibility.

References:

1. Brian J. Cantwell - Aircraft and Rocket Propulsion - 2010
2. Smith, A. et al. "Innovative In-Flight Fuel Pressurization Techniques." Proceedings of the International Astronautical Congress, 2021.
3. Miller, R. "In-Flight Pressurization Techniques for Rocket Propulsion Systems." Proceedings of the International Astronautical Congress, 2019.

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Vorobiov M., Bobkov Y.

ALGORITHM FOR SEMI-AUTOMATIC ANALYSIS OF UAV IMAGES

Introduction. At the current stage of technological development, unmanned aerial vehicles (UAVs) with various cameras are widely used in almost all industries. They allow to obtain up-to-date images of sufficiently high resolution of certain objects, territories, etc. At the next stage, the task of processing the images to solve the tasks arises.

The most common and urgent task is to search for and select certain objects of interest in UAV images. The complexity of solving this problem depends, in particular, on when the images are processed: after the UAV mission is completed or directly during the flight.

Problem statement. The most effective way to analyze the images obtained from UAVs is to use human or artificial intelligence. In the first case, the task is quite simple, but obtaining the results of information analysis from a human operator and its subsequent description in the form of digital data requires significant time and a sufficient number of specially trained personnel, which is not always acceptable. These drawbacks are absent in the case of artificial intelligence [1]. But then the problem of creating appropriate processing algorithms and databases for training an artificial intelligence system comes to the fore. It also requires very powerful computing systems that cannot be placed on board small UAVs.

The aim of this work is to develop a semi-automatic algorithm for analyzing objects in UAV images during its flight based on information received from the vision system and with the operator's preliminary selection of objects of interest.

Research methodology. To achieve this goal, we used pattern recognition methods and computer vision algorithms.

Various algorithms based on object boundary detection and various image processing algorithms, such as filtering, are used to detect objects. For structured objects in the form of urban development, this approach is very effective. In [2], image processing based on the Roberts boundary extraction algorithm with subsequent filtering and formation of the required elements is proposed. In [3], the algorithm of boundary extraction based on the Canny method and subsequent processing with the Hough filter is applied. The disadvantage of these works is that they are rigidly tied to a specific algorithm, which may be suboptimal when the image changes. Processing the entire frame requires significant computational resources, which limits the use of onboard UAV systems.

The essence of the proposed method is a two-stage processing. At the first stage, the operator points a certain mask at the object of interest, changes the scale of the mask to the desired size in order to cover the entire object. After that, the selected area of the image is "captured". At the second stage, within the selected mask, the existing objects are automatically analyzed using the specified boundary detection algorithms.

Results. The most information about any object for further analysis is provided by its structured image with the selection, first of all, of the contours (borders) of the object.

The paper analyzes the most common methods of object image contour extraction, namely: Sobel, Prewitt, Roberts, Laplace-Hauss, Kenny, and Otsu. The first five methods are based on spatial differentiation using certain filters, followed by detection of brightness changes. The Otsu method uses segmentation algorithms to extract object contours. The research has shown that the effectiveness of each method significantly depends on many factors that characterize the quality of the images and the individual characteristics of the objects. Therefore, it was decided that the choice of a specific algorithm that is optimal for a given image is made by the operator during preliminary analysis within the mask applied to the object of interest. The developed algorithm was programmatically implemented using MATLAB.

Conclusions. An effective semi-automatic two-stage algorithm for detecting an object of interest in UAV images has been developed, which does not require significant computing power and can be implemented using onboard data processing tools at the second, automatic stage.

References

1. Girshick R., Donahue J., Darrell T., Malik J. Region-based convolutional networksfor accurate object detection and semantic segmentation // IEEE Trans Pattern Anal Mach Intell. 2016. 38(1). P. 142–158.

2. Hnatushenko V. V. Algorithm for identifying houses on satellite images / V. V. Hnatushenko, E. O. Obidennyi, Y. I. Shedlovska // Scientific Bulletin of Tavria State Agrotechnological University. - 2016. - Issue 6, Vol. 2. - P. 157-162.

3. Canny J. F. A Computational Approach to Edge Detection / J. F. Canny // IEEE Transactions on Pattern Analysis and Machine Intelligence. — 1986. — № 8. — P. 679–698.

UDC 528.5-233.1

Ortamevzi G., Yilmaz M. I.

STABILITY ANALYSIS OF LANDING GEAR SHAFT ACCORDING TO FLAP AND NON-FLAP LANDING SITUATION

Introduction. In this study, the loads on the landing gear shaft of an experimental aircraft landing with and without flaps were simulated and the behavior and durability of the shaft were examined. Ground loads on bearing seating surfaces during descent have been defined. Equivalent stress, total deformation, equivalent elastic strain and safety factor were used as comparison criteria. These criteria were calculated using the finite element method for both forces, landing with flaps and landing without flaps.

Materials and Methods. The landing gear shaft has two bearing seating surfaces of suitable dimensions fortight fit mounting of bearings 6204 and 6205. The landing gear shaft 3D model is shown in Figure 1. Nevertheless, the shaft material is defined as structural steel [1].

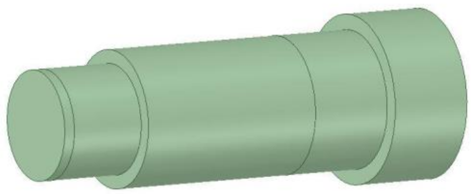

Figure 1 - 3D Geomerty of Shaft

In order to calculate the strength criteria of the landing gear shaft, Equivalent Stress, Strain, Total Deformation and Safety Coefficient values, with the FEM method, the mathematical model of the shaft was created as shown in Figure 2. Mesh statistics consist of 261077 node sand 152764 elements.[2][3][4].

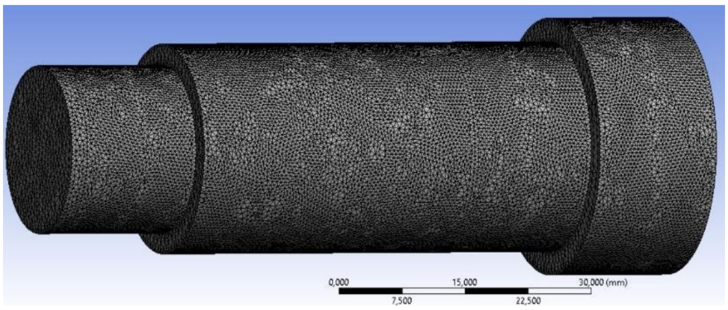

Figure 2 - Geometry Mesh Statistics Nodes: 261077 Elements: 152764

Remote Force 2500 N and 3000 N were applied to the bearing as radial and axial forces to seating surfaces, simulating the ground load as landing with and without flaps, as shown in Figure 3.[5][6]**.**

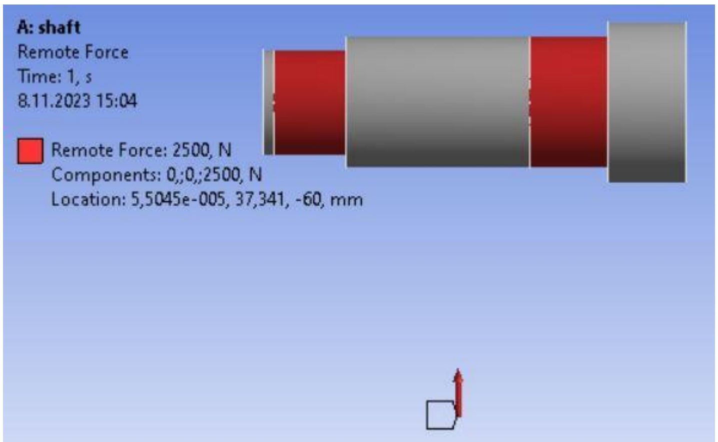

Figure 3 - Remote force application

Results and disscusion. In order to compare the behavior of the landing gear shaft at different ground loads, the strength criteria Equivalent Stress, Strain, Total Deformation and Safety Coefficient values were calculated. The values in the graphic sare the value soccurring in the weakest part of the shaft. The results calculated with the FEM method are shown in Figure 4-7.

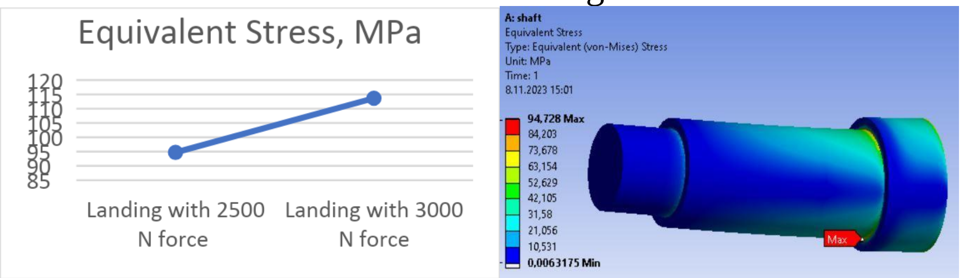

Figure 4 - Equivalent Stress, MPa

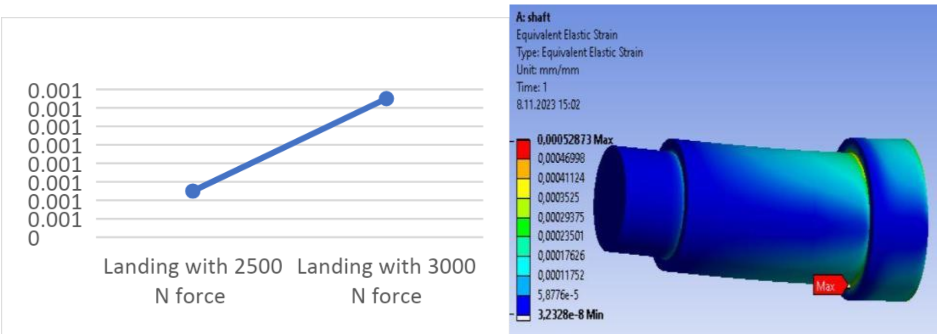

Figure 5 - Equivalent Strain, mm/mm

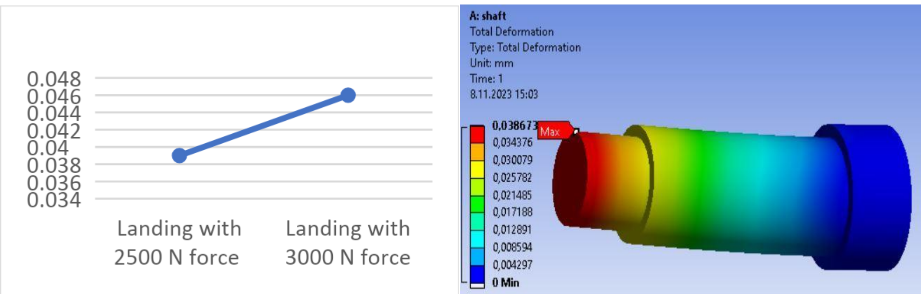

Figure 6 - Total Deformation, mm

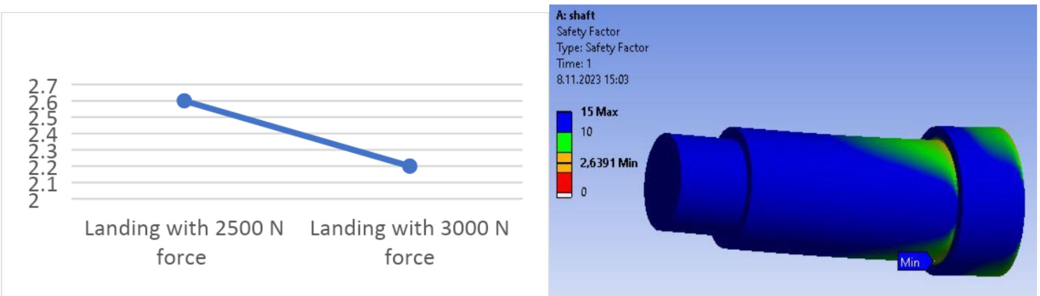

Figure 7 - Safety Factor

Conclusion. It has been observed that equivalent stress is concentrated in the bearing seating flange. Distributing the stress by applying radius in these regions is a solution option.

The harder the landing has been caused the higher the stress and the lower the safety factor.

It has been observed that the safety factor drops to 2.2 in the bearing support area during landings with a 3000 N impact on the landing gear. If the landing gear will be exposed to higher forces, the bearing diameters must be increased, and the shaft diameter must be increased in order to avoid endangering flight safety. It should not be forgotten that this situation will have a negative impact on weight and cost.

References:

1. D. A. Osage and M. Prager, "Technical Basis of Material Toughness Requirements in the ASME B\&PV Code, Section VIII, Division 2," in *Pressure Vessels and Piping Conference*, 2010, vol. 49200, pp. 635–652.

2. M. Imran, R. M. S. Ahmed, and M. Haneef, "FE analysis for landing gear of test air craft," *Mater. Today Proc.*, vol. 2, no. 4–5, pp. 2170–2178, 2015.

3. S. Timoney, E. Timoney, and C. Lynch, "Correlation of fatigue test results and finite element analysis for a prototype independent suspension axle housing," in *International Truck \& Bus Meeting \& Exhibition*, 2001, no. 2003-01–3434.

4. J. Zhang, D. Guo, W. Wu, and P. Guo, "A Time-Discontinuous Galerkin Finite Element Method for the Solution of Impact Problem of Gas-Saturated Coal," *Shock Vib.*, vol. 2020, 2020.

5. E. J. Miller *et al.*, "Evaluation of the hinge moment and normal force aerodynamic loads from a seamless adaptive compliant trailing edge flap in flight," in *54th AIAA Aerospace Sciences Meeting*, 2016, p. 38.

6. M. P. Kaplan, T. Willis, and T. A. Wolff, "Damage Tolerance Assessment of CASA Landing Gear," *SAE Trans.*, pp. 1250–1260, 1997.

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Zavorotynskyi L., Konotop D.

DESIGNING OF A LIGHT AIRCRAFT WITH SHORT TAKEOFF AND LANDING CAPABILITIES

Introduction. Today's light aircraft are a type of universal workhorse that meets a variety of needs in various aviation sectors. These aircraft play a key role in local airlines, agriculture, and more, all because of their high maneuverability and compact size. Particularly notable are short takeoff and landing (STOL) aircraft, which have the unique ability to operate efficiently in limited space, including unprepared runways. This feature expands the range of their usage to include regional aviation, rescue missions, pilot training, reconnaissance, and various other tasks.

STOL can be achieved by using factors such as a high-curvature wing profile, a powerful engine, a large wing area, complex wing mechanization, a robust landing gear design, and an efficient braking system.

Scientific and technical results. Aircraft design is a complex process that includes the development and improvement of the initial idea, multi-iteration performance calculation, component design, and 3D modeling.

The design of a light STOL aircraft includes the creation of a product requirements (PR) for the development of the product, the selection of modern analogues in accordance with the PR, and the creation of a statistical database based on these analogs. This database is used to start designing the aircraft in the first approximation, namely, to obtain mass and geometric characteristics for further design stages.

Based on the information received from analog aircraft and the PR, the aerodynamic profile of the aircraft is chosen to have the highest aerodynamic characteristics at the set flight modes. The powerplant is also selected to match the specified conditions. Usually, light-class aircraft are powered by a piston or turboprop engine, which requires an additional propeller selection.

Next, the aircraft fuselage is designed and calculated to match the required PR characteristics and standards set out in regulatory documents such as АП-23, FAR-23, etc. We also begin to create a master geometry of the aircraft model. Mechanization, plumage, and control surfaces of the aircraft are designed and calculated. As a first approximation, the center of mass of the aircraft is calculated based on the required lengths of the horizontal wing arm, which provides enough information to complete the master geometry of the aircraft model (Fig. 1).

After receiving all the necessary data, a mathematical model is created for preliminary calculation of aerodynamic characteristics and further optimization of the model for their improvement. After that, the energy and ballistic characteristics of the aircraft are estimated, which makes it possible to understand if the designed model meets the characteristics specified in the PR. This is followed by an iterative process to obtain the most accurate preliminary aircraft characteristics.

The main feature of this project is the wing, as it has a complex mechanization consisting of a controlled flap, a single-slot Fowler flap and an interceptor. A novelty is a flap that is placed along the entire trailing edge of the wing, this location of the flap makes an aileron at the end of the wing impossible, but significantly improves the takeoff and landing characteristics of the aircraft, as the area of wing mechanization increases compared to a wing with ailerons. Since there are no ailerons, their roll control function will be performed by interceptors, which will additionally create resistance during landing, reducing the required distance for landing.

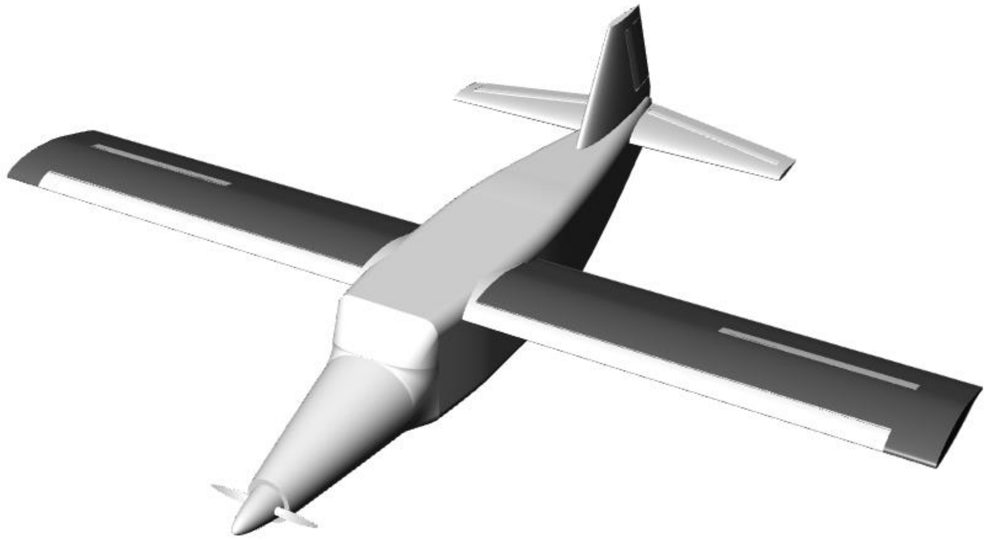

Figure 1 - Master geometry of a lightweight STOL aircraft

To do this work, the following software is used: Microsoft Office suite, PANSYM and Xfoil for aerodynamic calculations, Rhino 7 for creating sketches and 3D models, Origin 2023b for working with graphs, and MATLAB 2023b for automating calculations.

Conclusion. This project aims to demonstrate the stages of development of a light aircraft. The scientific novelty of this project is the use of a wing without ailerons, which improves the takeoff and landing characteristics of the aircraft.

References:

1. Roskam J. Airplane Design Part I - VIII. Kansas: DARcorporation, 1997.

2. Torenbeek E. Synthesis of Subsonic Airplane Design. Delft University Press, 1976.