UDC 629.7.036 https://doi.org/10.15407/itm2020.02.036

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COMPUTER SIMULATION OF GAS-DYNAMIC PROCESSES FOR THE OPTIMIZATION OF ROCKET FAIRING DESIGN

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Обтікач необхідний для захисту корисного навантаження від впливу зовнішніх факторів під час польоту ракети. Він повинен витримувати значні силові та температурні навантаження і безпечно відстиковуватись та відводитись від ракети. В роботі розглянуто процес відстикування стулок обтікача ракети при надзвуковому обтіканні в приземних щільних шарах атмосфери. Для процесу відведення стулки від корпуса ракети запропоновано використовувати детонаційний шнуровий ракетний двигун, що розвиває значну силу тяги при малій масі та нетривалій роботі. Це суттєво дозволяє знизити масу обтікача. Визначення сил, що діють на обтікач, проводилося за допомогою комп'ютерного моделювання цього процесу. Розроблено методику розрахунку основних параметрів детонаційного шнурового ракетного двигуна для відведення стулки обтікача від ракети. В математичній моделі руху обтікача відносно ракети в процесі розстикування й відведення розглядаються два варіанти: розрахунковий варіант відриву і прискорення стулок при нерозривному механічному контакті з ракетою, а також випадок автономного руху стулки по інерції окремо від ракети. Результатами комп'ютерного моделювання є проекції аеродинамічних сил і обертаючого моменту, картина розподілу тиску повітря за найбільш характерними кутами відведення стулок обтікача від ракети. Змодельоване обтікання п'ятьох варіантів стулок з різними за формою захисними перегородками: конічною, вгнутою конічною, сферичною, вгнутою сферичною та плоскою. Оптимальним, з точки зору мінімальних затрат енергії, виявився варіант з вгнутою сферичною захисною перегородкою. Було визначено оптимальну форму, розмір та розміщення перегородки. Виявлено ефекти, що дозволяють знизити необхідну тягу двигуна для відведення стулки від корпуса ракети. У всіх випадках визначено картини розподілу тиску, що в подальшому дозволяє провести розрахунки на міцність. Визначено мінімальну необхідну тягу шнурового детонаційного двигуна для відведення стулки від корпуса ракети.

Ключові слова: *детонаційний шнуровий ракетний двигун, надзвукова течія, обертання, моделювання, розподіл тиску*.

Обтекатель необходим для защиты от воздействия внешних факторов на полезную нагрузку во время полета ракеты. Он должен выдерживать значительные силовые и температурные нагрузки, а также безопасно отстыковываться и отводиться от ракеты. В статье рассматривается процесс отделения обтекателя ракеты при сверхзвуковом обтекании в плотных слоях атмосферы Земли. Для отделения обтекателя с корпуса ракеты предлагается использовать детонационный шнуровой ракетный двигатель, который развивает значительную тягу при малой массе и коротком периоде эксплуатации. Это значительно уменьшает массу обтекателя. Определение сил, действующих на обтекатель, проводилось с помощью компьютерного моделирования процесса отведения. Разработана методика расчета основных параметров детонационного шнурового ракетного двигателя для отделения створок обтекателя от ракеты-носителя. В математической модели движения обтекателя относительно ракеты в процессе расстыковки и сброса рассматриваются два случая: расчетный случай отрыва и ускорения закрылков при неразрывном механическом контакте с ракетой, а также вариант автономного движения по инерции закрылков отдельно от ракеты. Результатами компьютерного моделирования являются проекции аэродинамических сил и крутящего момента, картина распределения давления воздуха по наиболее характерным углам отведения створок обтекателя от ракеты. Моделируются пять вариантов формы обтекателя с различными защитными перегородками: коническая, вогнутая коническая, сферическая, вогнутая сферическая и плоская. Использование вогнутой сферической защитной перегородки было оптимальным с точки зрения минимальных затрат энергии. Были рассчитаны оптимальная форма, размер и размещение перегородки. Обнаружены эффекты снижения необходимой тяги двигателя для снятия закрылка с корпуса ракеты. Во всех случаях были определены схемы распределения давления, что впоследствии позволяет рассчитывать прочность конструкции створок. Определена минимальная необходимая тяга двигателя со шнуровой детонацией для отведения обтекателя от корпуса ракеты.

Ключевые слова: *детонационный ракетный двигатель, сверхзвуковое течение, вращение, моделирование, распределение давления*.

The fairing serves to protect the payload against exposure to external factors during the rocket flight. It must withstand considerable force and thermal loads and safely detach and move away from the rocket. This paper deals with the process of fairing flap separation from the rocket in the Earth's dense atmosphere under conditions of considerable aerodynamic loads in the range of supersonic flight speeds. It is proposed that flap removal from the rocket structure be done using a detonation corded rocket engine, which develops a considerable thrust at a low mass and a short operation period. This significantly reduces the fairing mass. The forces arising in this pro-

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cess were determined by computer simulation. A technique for calculating the basic parameters of a detonation corded rocket engine for fairing flap removal is presented. The mathematical model of flap motion relative to the rocket during the process of detachment and removal consists of two parts: the calculation of the separation and acceleration of the flaps while in mechanical contact with the rocket and the calculation of the inertial motion of the flaps separated from the rocket. The computer simulation gives the projections of the aerodynamic forces and torque and the air pressure distribution for the most characteristic angles. Five protective partition shapes were simulated: conical, concave conical, spherical, concave spherical, and flat. The concave spherical shape was found to be optimal in terms of minimum energy consumption. The optimal shape, dimensions, and placement of the partition were calculated. The minimum thrust of the detonation corded engine required for flap removal from the rocket was determined, and effects that allow one to reduce this thrust were found. The calculated pressure distributions may be used in flap strength analysis.

Keywords: *detonation corded rocket engine, supersonic flow, fairing separation, simulation, pressure distribution.*

Introduction. The fairing is necessary to protect against the effects of external factors on the payload during the missile flight. For boosters launching spacecraft on orbit, the need for this element of the design is also in the high requirements for the purity of the space in which the payload is placed. The fairing must withstand considerable force and temperature loads. Another crucial feature is the need to safely unlock and remove from the rocket. Payload is usually on top of a rocket, that’s why it causes a lot of problems.

Docking and removal takes place at altitudes where the impact of the atmosphere can’t longer harm the payload. It is usually 80 – 120 km. This altitude is chosen for solving the optimization problem, because the higher the drop point in the trajectory, the smaller the atmosphere impact on the spacecraft, but the greater the energy costs for removing this "excess" mass.

There are several options for fairing flaps:

1) spring pushers;
2) pneumatic pushers;
3) impulse of detonation cords;
4) solid propellant motors.

Sometimes it is necessary to release the payload from the fairing under conditions of considerable aerodynamic loads in the range of supersonic flight speeds. This article is devoted to solution of this task.

Formal problem statement. The initial conditions are the separating height – 1500 m, and the rocket flight speed – 650 m / s. The configuration of the main fairing is illustrated in Fig. 1, where *R* – radius.

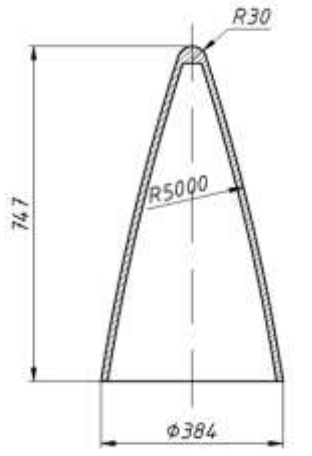


Fig. 1 – Geometric dimensions of the main rocket fairing

According to the original data and dimensions, it is clear that at supersonic flight speeds in the dense ground layers of the atmosphere there are significant aerodynamic loads. Since the mass of the structure is M = 12.5 kg and the projection F_x (Fig. 2) reaches 56350 H, it is easy to estimate the order of acceleration by the formula:

$$\frac{d^2x}{dt^2} = \frac{F_x}{M} \approx 4500\, m/s^2 . \tag{1}$$

Besides the considerable effort, it is also necessary to provide an engine thrust impulse releasing the fairing flap to fulfill the condition:

$$I_{F_t} = \int_0^{t_e} F_t(t)dt \geq I_{F_a} , \tag{2}$$

where t_e – engine running time; F_t – engine thrust; F_a – aerodynamic force; I_{F_a} – its impulse, determined by the formula:

$$I_{F_a} = \int_0^{t_e} \sum_{i=1}^{k} F_{a_i}(t)dt , \tag{3}$$

F_{a_i} – aerodynamic force at running time moment i.

Obviously, the shorter the aerodynamic forces, the smaller the total thrust momentum of the rocket engine should be applied to implement the process of removing the flap from the rocket. Therefore, it is necessary that the evacuation device operates for a very short time and develop considerable efforts to overcome the external force effects of the atmosphere.

When the rocket engine is running, combustion products are released that can adversely affect the payload. The location of the power plant must be separated from the payload area with an additional special partition. However, it is significantly alter the process of airflow during decoupling, and affect the value of aerodynamic forces. Its shape is not regulated in our case, so it is necessary to simulate the flow of the flap with different shape variants of the protective partition and to optimize the load on the fairing (flap).

The design scheme used for the design optimization is shown in Fig. 2, where it's used the following keys: M_a – aerodynamic moment; l_x and l_y – the projection of the distance from the mass center of the flap to its axis of rotation (point O) on the axis x and y, accordingly; r – distance from the axis of flap rotation to the thrust vector of the engine; ψ – angle between lever from point of flap turn to mass center and projection F_y; α – angle between the axis of the rocket and the flap (this is also the angle between the base of the flap and the perpendicular plane of separation); F_x and F_y – the projection of aerodynamic force on the axis x and y, accordingly

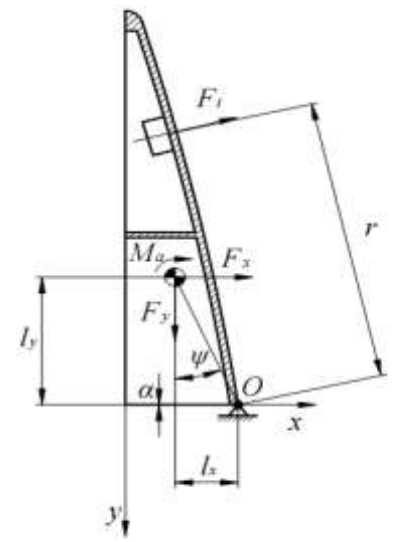


Fig. 2 – The design of the fairing flap

Literature review. Today, numerical modeling of gas-ball processes for rocket technology is widely used. In particular, the processes that occur during the rocket movement in the atmosphere in different modes: sub-, trans-, and supersonic, are simulated [1, 2]. This reduces the number of experimental studies and contributes to the economic efficiency of the missile system [3].

Numerical experiments are also conducted to identify effects that may occur in flight conditions. For example, the occurrence of resonant vibrations of the rocket body, which are caused by intense flow in the transonic speed range for flying in dense layers of the atmosphere [4]. Slipping with airflow a complex configuration of the launch vehicle introduces some difficulties in the control system operation. Turbulence and significant power factors can lead to a loss of system stability. Therefore, computer simulations are carried out not only for the purpose of forecasting, but also for analysis of already completed launches [5, 6]. This feature is interesting because it is possible to model the behavior of the system in extreme conditions, which were not specified by the initial project data. At the same time, the cost of experimental research is very high.

Auxiliary power units for the removal of separate parts of the missile have their own specificity. Usually, their operating time is limited to a few seconds and a design that is small in size and should be as small as possible. Solid propellant rocket motors are used for these tasks. However, their functions can be performed by solid propellant engines, especially when very high effort is required at low durations. In this device, instead of the combustion process at constant pressure, the detonation process is more efficient from the point of view of the thermodynamic efficiency [7]. Its peculiarity is the large pressure drop to the detonation wave and in the induction zone where the reaction takes place [8].

In addition to solid-state corded detonation rocket engines, there are installations operating on gaseous and liquid components [9]. Experimental studies in the middle of the last century confirmed the possibility of detonation combustion. However, due to the lack of powerful computing complexes, measurement tools and research materials, they remained in the initial stage. The new surge of interest was caused by the declassification of many projects and the development of computer technology. Simultaneously, the structure of the detonation front in the annular chamber of the detonation rocket engine was simulated. However, in the experiments low values of the specific impulse were obtained [9 – 10].

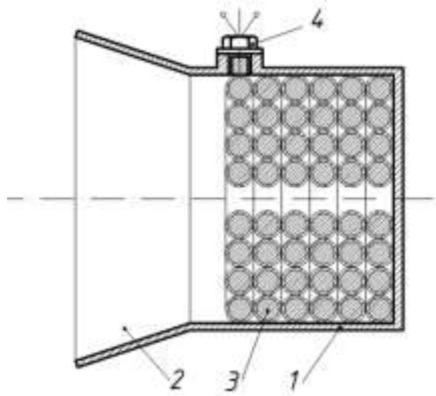
Technical solutions. Solid propellant corded detonation motor has the necessary characteristics (Fig. 3). By using explosives as a solid fuel, the running time can be very short, and it can take considerable effort. The mass of the engine is small. Experimental studies of such engines were conducted on the experimental station of the Institute of Technical Mechanics of the National Academy of Sciences of Ukraine and the State Space Agency of Ukraine in the 80s of the last century. For these purposes, bench units were created to assess the energy characteristics of solid propellant (corded) detonation rocket engines, with different types of explosives and different geometric parameters of cords. The results of some of these fire tests are shown in table 1.

Table 1 – Some results of fire tests

Mass explosives, kg	Cord length, m	Specific thrust explosives, s	Effective specific thrust, s	Traction DRD kN
0,076	3	509	201	274,6
0,2	5	448	180	241,6
0,22	5	456	182	245,1
0,136	5	452	165	246,1
0,135	5	464	172	250,2
0,135	5	581	215	313,8
0,3	10	475	158	255,9
0,576	20	575,7	200	309,9
0,576	20	381,5	132,5	205,9

The chart of solid propellant (corded) detonation rocket engine is shown in fig. 3.

It consists of a combustion chamber (1) into which the explosive cord (3), initiated from the nozzle (2) by the detonator (4), is placed in spiral layers.



1 – housing, 2 – nozzle, 3 – explosive cord, 4 – detonator

Fig. 3 – Solid propellant (corded) detonation rocket engine

Since the detonation velocity of the explosive is within 6 – 8 km / s, the charge of several kilometers burns almost instantly. To select the output parameters of the solid propellant rocket-propulsion rocket engine, it is necessary to determine, first of all, the loads acting on the flaps in flight.

Numerical modeling of aerodynamic loads. The mathematical model of the flap movement relative to the rocket during the process of undocking and removal consists of two parts: 1) the calculated case of separation and acceleration of the flaps with non separated mechanical contact with the rocket; 2) the calculated case of autonomous movement by the inertia of the flaps separately from the rocket. The second design case is necessary to check the absence of the hit on the rocket body, since it becomes larger in diameter than the main part.

At the moment of flap unlocking the withdrawal engine thrust and the aerodynamic force are actuated, which can be designed on the axis and called respectively as the drag force and the lateral force.

In a plane-parallel statement, the rotational motion equation is:

$$\begin{cases} I \cdot \dfrac{d^2\alpha}{dt^2} = F_t \cdot r + F_x \cdot \sqrt{l_x^2 + l_y^2} \cdot \cos(\phi - \alpha) - F_y \cdot \sqrt{l_x^2 + l_y^2} \cdot \sin(\phi - \alpha) \\ \phi = \mathrm{arctg}\left(\dfrac{l_x}{l_y}\right), \end{cases} \tag{4}$$

where I – the moment of inertia of the flap; $\ddot{\alpha}$ – angular acceleration of the flap.

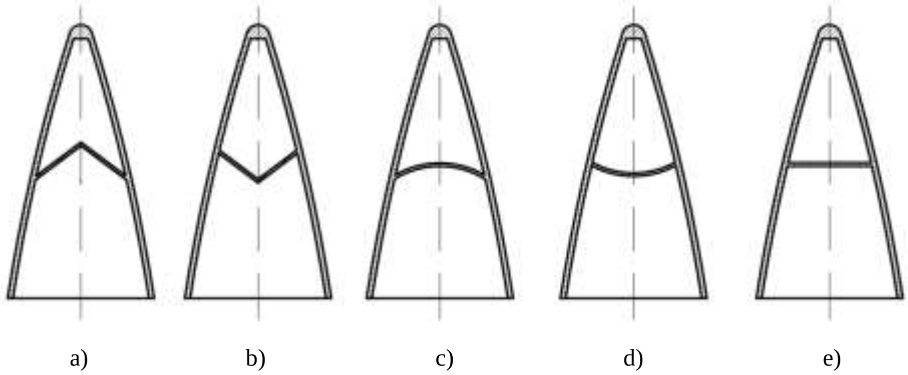
However, these equations do not make it possible to estimate the total thrust of the rocket engine and other important characteristics, so they need to be supplemented.

These parameters can be determined by knowing the work done by external power factors. Since the flap rotates about the axis O, then the work is determined by the formula:

$$A = \int_0^{\alpha} F_x(\alpha) \cdot l_x(\alpha) d\alpha + \int_0^{\alpha} F_y(\alpha) \cdot l_y(\alpha) d\alpha + \int_0^{\alpha} M_a(\alpha) d\alpha \,. \tag{5}$$

The aerodynamic force projections F_x and F_y, as well as the torque M_a, were determined from computer simulation in the SolidWorks application package. It allows performing hydro-gas simulations using the Navier-Stokes system [6]. The projections of the distance from the mass center of the flap to the axis O on the axis of the coordinate system also depend on the angle α. Here the initial data, as already mentioned, is the height at which the docking occurs – 1500 m and speed 650 m/s. The atmospheric parameters are taken for the standard model of the atmosphere [11, 12] at this height: T = 278,4 K; P = 84559 Pa; ρ = 1,058 кg/m^3. The simulation is performed in a stationary setting. Since it is not known yet what accuracy is provided in the producing of the fairing surface, it was considered that it is ideally smooth with zero height of micro-irregularities. The forces from the pressure of combustion products were also disregarded. This quite a difficult question and the interaction of the incoming flow with the flow field from the engine can be a topic of further research.

Five variants of the partition shape were selected, illustrated in Fig. 4. For the flap of the fairing with a flat partition, the flow field was simulated with a discrepancy of the rotation angle α equal to 1˚. It has been determined the significant changes in the flow pattern occur during depressurization when flap opens at 1˚. Further, all the flow fields are similar to each other up to the angle 22˚, which was taken as the extreme one for the transition to autonomous flap movement. So it can be taken smaller samples and interpolate intermediate values by known ones.



a) conical; b) concave conical; c) spherical; d) concave spherical; e) flat.

Fig. 4 – Variants of the protective fairing partition shape

For the flap of the fairing with a flat partition, the flow field was simulated with a discrepancy of the rotation angle α equal to 1˚. It has been determined the significant changes in the flow pattern occur during depressurization when flap opens at 1˚. Further, all the flow fields are similar to each other up to the angle 22˚, which was taken as the extreme one for the transition to autonomous flap movement. So it can be taken smaller samples and interpolate intermediate values by known ones. The next step after choosing the best shape is to optimize the size and placement of the partition. In addition, the autonomous movement of the flap is described by other equations. It is also advisable to model this motion after selecting the optimal shape and size of the fairing partition and determining the engine parameters.

Research results. The results of computer simulation are projections of aerodynamic forces and torque. The air pressure distribution patterns for the most characteristic angles are illustrated in Figures 5 – 9.

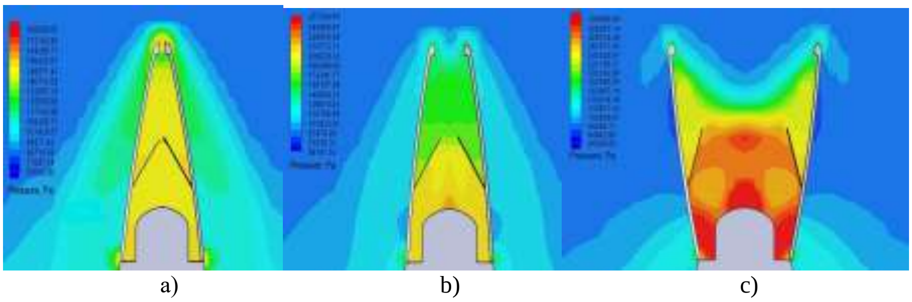
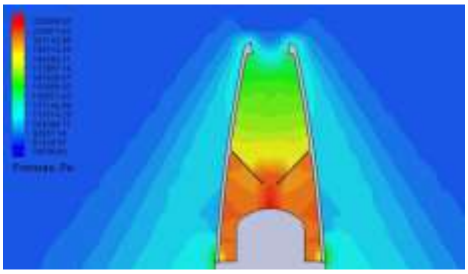


Fig. 5 – The pressure distribution in the symmetry plane at the rotation of fairing flap with a conical partition at the half-open angle: a)1˚; b) 5˚; c) 22˚

a)



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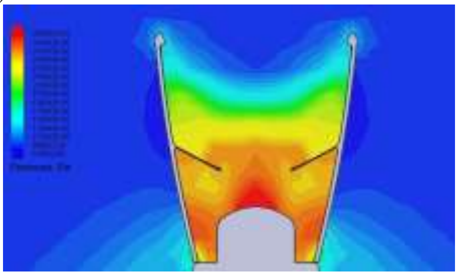
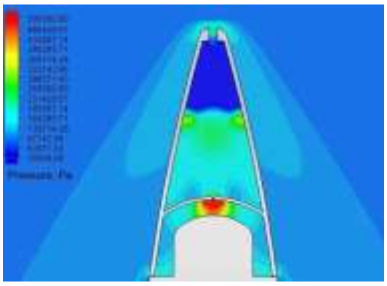
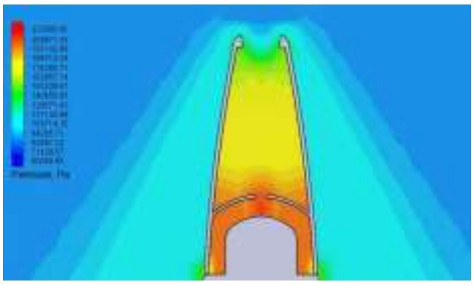


c)

Fig. 6 – The pressure distribution in the symmetry plane at the rotation of fairing flap with a conical concave partition at the half-open angle: a)1˚; b) 5˚; c) 22˚



a)



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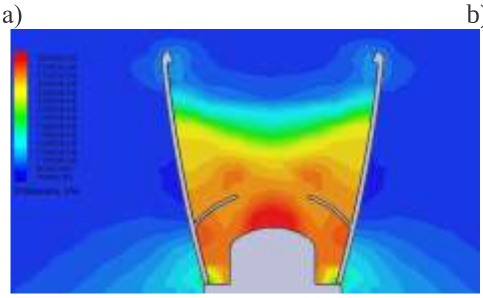
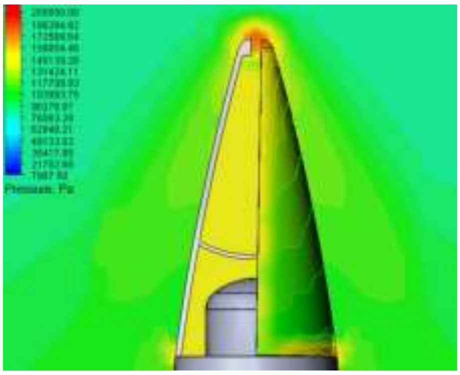
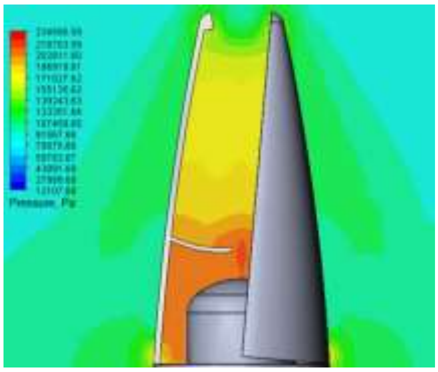
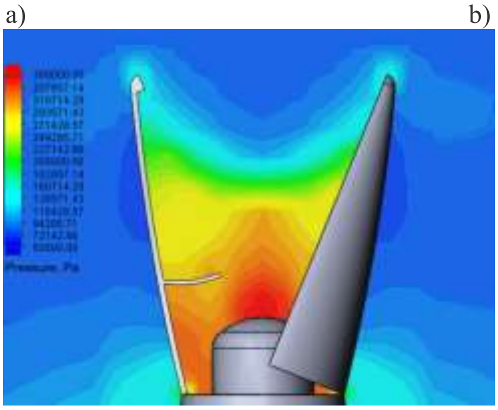


c)

Fig. 7 – The pressure distribution in the symmetry plane at the rotation of fairing flap with a spherical partition at the half-open angle: a)1˚; b) 5˚; c) 22˚

a) b)

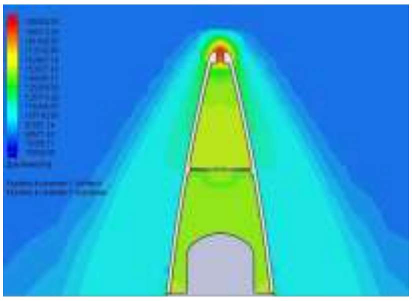
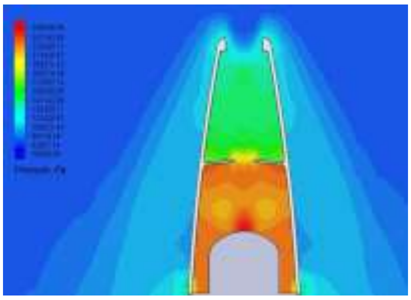


c)

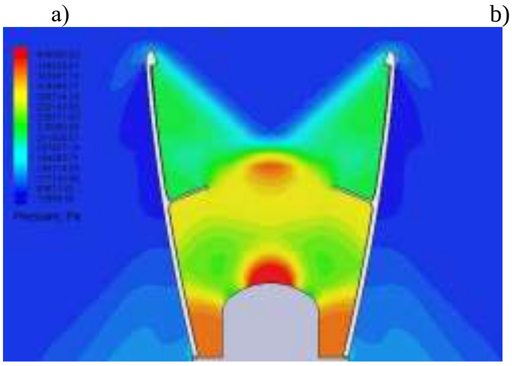
Fig. 8 – The distribution of pressure in the plane of symmetry when rotating the flap of the fairing with a spherical concave partition at the half-open angle: a) 1˚; b) 5˚; c) 22˚

The distribution of pressure in the plane of symmetry for rotating the fairing with a flat partition at the different half-open angle is shown in fig 9.

As it can be seen from the figures, when the flap (the fairing flap) is opened,

a) b)



c)

Fig. 9 – The distribution of pressure in the plane of symmetry for rotating the fairing with a flat partition at the half-open angle: a) 1˚; b) 5˚; c) 22˚

the flowing stream hits the payload and locally increases the value of the pressure at the bottom of the fairing under the protective partition. The aerodynamic forces and torque that arise at this moment counteract the decoupling process only up to a certain angle of 220, when the mass center of the flap is above the axis of flap rotation (point O). As the angle of rotation continues to increase, they change direction and facilitate the retraction process. However, there is the great likelihood of a flap strike the rocket body. The optimal angle can be found from the calculation of the dynamics of autonomous flap movement.

The work performed by aerodynamic force factors for each partition variant is shown in table 2.

Table 2 – Aerodynamic power factors for each partition

№	Form of protective partition	Work performed by external forces at the flap rotating A, J.
1	Spherical	16740
2	Concave spherical	11480
3	Cone	18470
4	Concave cone	22890
5	Flat	22560

According to table, the smallest value of the work is performed by external forces at the flow over flaps with a concave spherical partition. Therefore, it is advisable to make further calculations with this option for optimal placement and optimization of the radius of the sphere. In all the calculated variants, except for the concave cone partition, the projection value F_y is positive. That is, the air, penetrating the fairing, strikes the payload, slows down the speed and at the same time raises the pressure in the area under the partition. Its size is quite large, that leads to appearance a force trying to lift the flap up against the stream. This serves as an auxiliary factor in removing the flap from the rocket body and reduces the engine thrust used for these purposes. The minimum required thrust value of the engine in this case is 235.1 kN. Approximately this order of magnitudes was achieved in the experiments conducted at the experimental station of the Institute of Technical Mechanics of the National Academy of Sciences of Ukraine and the State Space Agency of Ukraine

Research results. A method for calculating the basic parameters of a detonating corded rocket engine for removing the fairing flap from the rocket in flight conditions in Earth dense layers of the atmosphere is developed. Five types of flaps with different protective partition are modeled: conical, concave conical, spherical, concave spherical and flat. The variant with a concave spherical protective partition was the optimum, in terms of minimum energy costs. Effects for reducing the required thrust of the engine for removing flap from the rocket body have been found. In all cases, pressure distribution was determined, which further allows to carry out calculations for strength of sructure. The minimum required thrust of the corded detonation engine for removal of the flap from the rocket is determined.

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Received on 19.04.2020,
in final form on 26.06.2020