UDK: 621.45.054:621.456
COSATI: 21-08, 16-04, 16-03

Design of Shoulder - Launched Unguided Rocket with Thermobaric Warhead

Siniša Jovančić[1)]
Danica Simić[1)]

This paper presents a calculation methodology of geometrical and mass parameters for shoulder-launched rocket with thermobaric warhead. An optimization of design rocket parameters for specified tactical and technical requirements is carried out by the criteria of the mass efficiency using two calculation methods. According to the first method, the rocket calibre for given a maximum range is determined by the function of payload mass. The second method predicts calibre value and based on that, all rocket sub-assemblies masses can be calculated, which means that total rocket mass is determined either. An optimal calibre value for rocket motor with limited operating time is common and unique solution for both of these iterative procedures. In the concrete example the application of the exposed model is shown – for specified tactical and technical requirements and known material characteristics, carried out calculation and analysis parameters of warhead and rocket motor. For selected rocket design parameters, a simulation of motor operating time and moving rocket throughout its launching tube are done for three typical usage temperatures: -30 ºC, +20 ºC and +50 ºC. The obtained values of the chamber pressure, operating time of the rocket motor and initial rocket velocity are analyzed considering the shooter's safety and realization of required ballistic characteristics.

Key words:. **rocket launcher, shoulder-launcher, unguided rocket, thermobaric warhead, rocket motor, impulse rocket motor, projection, parameters optimization, numerical simulation.**

Nomenclature

a	− Acceleration
A	− Area
b	− Coefficient of the San Robert’s burning rate law
C_f	– Friction coefficient of flow within aperture
C_F	– Thrust coefficient
C_x	− Aerodynamic axial coefficient
C^*	– Characteristic velocity
d, D	− Diameter
EL	− Elevation angle
F	− Thrust force
g	− Acceleration of gravity
h	− Height of target
I	− Impulse
J_*	− Critical Viljunov number
k_ε	− Erosive burning coefficient
K_i	− Ratio of burning surface to flow area
K_n	− Ratio of burning surface to throat area
l, L	− Length
m	− Mass
$\dot{m}$	− Mass flow rate
n	− Exponent of the San Robert’s burning rate law
n_c	− Number of tubular grains
p	− Pressure
r	− Propellant burning rate
R	− Specific gas constant
S	− Surface
t	– Time
T	− Temperature
u	− Particle velocity of gas flow
V	– Velocity
w	– Web tickness
x	– Axial distance
X_r	− Direct fire range
z_b	− Gap between the tubular grains
Z	– Propellant mass fraction
α_{td}	– Thermal diffusivity of propellant
Δ	– Deviation and nominal value ratio of the observed parameter
γ_n	– Half angle of divergent conical nozzle
η	– Correction factor of theoretical performance
δ	– Erosive burning rate factor
δ_1	– Unsteady burning rate factor
ε	– Chamber charging coefficient
ε_n	− Expansion ratio
κ	– Ratio of specific heats
λ	– Coefficient which describes how fast the static friction approaches the dynamic one
μ	– Friction coefficient
ν	– Safety coefficient

[1)] Military Technical Institute (VTI), Ratka Resanovića 1, 11132 Belgrade, SERBIA
Correspondence to: Siniša Jovančić; e-mail: jovancicsinisa@gmail.com

ρ – Density
σ_e – Yield strength
σ_m – Ultimate strength
σ_p – Temperature sensitivity of burning rate
ψ_{10}, ψ_{11} – Pressure function coefficients in expression of unsteady burning rate

Subscripts

b – Burning parameters
c – Chamber parameters
cr – Reduction chamber parameters
d – Dynamic parameters
dc – Chamber bottom parameters
e – Nozzle exit parameters
eb – Explosive burster parameters
ex – Thermobaric explosive parameters
f – Final rocket parameters (after rocket propellant is ejected)
LT – Launching tube parameters
m – Mean valaue of parameters
max – Maximum valaue of parameters
n – Nozzle parameters
o – Initial parameters
p – Propellant parameters
R – Rocket parameters
s – External (or static) parameters
sp – Specific parameters
t – Nozzle throat parameters
tot – Total parameters
ti – Thermal insulation parameters
u – Lethal parametrs
w – Warhead parameters
x – Axial direction

Introduction

ROCKET design development includes the determination of basic parameters, that defines the mass, geometrical and ballistic characteristics related to specified tactical and technical requirements by predetermined optimization criteria. The optimization problem during developing design proces can be reduced to a model that gives the minimum start rocket mass value as a function of the basic parameters that provides reliable delivery of payload to target at a given range, and the required hit probability, [1-4].

Optimization model based on the criteria of the mass efficiency, unguided rocket design comes down to the achievement of a maximum given range, which is basically conditioned by intensity of velocity and angle of velocity at the end of the active phase of flight, as aerodynamic drag force. Since the rocket mass is proportional to rocket cube diameter, and thrust force and aerodynamic drag force are proportional to rocket square diameter, ref. [1-3], the optimization problem is reduced to determining the optimal calibre values.

One of the additional requirements for the shoulder-launched unguided rocket is to increase shooter's safety. Practically, this requirement limits the rocket motor operating time, because the action time of rocket motor is less than the time of rocket's moving throughout its launching tube. If propellants with a relatively high burning rate are not available, indirectly, it can be concluded that the limit of operating time causes small web thickness, and therefore lower charging chamber coefficient, respectively lower propellant mass, which significantly affects the range limit.

Unlike the methodology given in references [1, 2, 4], the integrated aproach of resolving the problem of the rocket optimization with the impulse motor is given in this paper. Beside the optimal rocket calibre determination for a given maximum range and the warhead efficiency on the target, the problem of optimal rocket masses is included which is significant for the shoulder-launched rocket.

Conception, operating principle and design parameters of thermobaric warhead

The war conflict experiences have shown that the shoulder-launched unguided rocket should have various types of warheads. Beside the shaped charge special purpose warheads, incendiary and thermobaric warheads are in use. Thermobaric explosive charge instead of high explosive charge (eg. TNT), achieves significantly greater destructive effect per area unit, although the lower values of the overpressure on the detonation wave rim are accomplished. However, at the same time it has a significantly wider zone of chemical reactions, ref. [5]. Although there are solutions of warhead with the solid explosive charge achieving the effect similar to thermobaric, ref. [6-8], in this paper, only the warhead with the liquid explosive will be considered. One of the performed thermobaric warhead solutions for the shoulder-launched unguided rocket is shown in Fig.1.

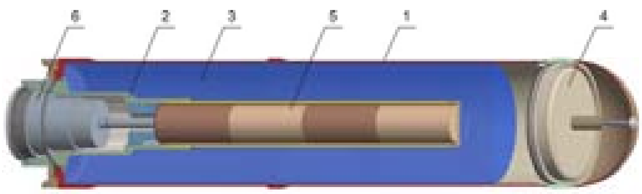

Figure 1. Thermobaric warhead: 1 – shell of warhead, 2 – shell of burster charge, 3 – thermobaric explosive mixture, 4 – spherical cap, 5 – exsplosive burster, 6 – fuze

Thermobaric explosive mixture consists of: primary explosive charge (liquid aromatic hydrocarbon) and autoignition initiator (small metal particles). Explosive burster (secondary explosive charge with high explosive), provides instant and uniformly breaking of the warhead shell, and then quickly scattering of the thermobaric mixture. Primary explosive charge, scatters into tiny dispersion particles, mixes with air and forms an aerosol cloud. At the moment when the optimal concentration of thermobaric mixture and air in aerosol cloud is reached, the autoignition is initiated by small metal particles, and instant burning process becomes a detonation, ref. [6]. A rapid temperature and pressure increasing is occured in the detonation transforming area, wherein the main part of the mechanical energy is spending on generating a shock wave, which is the blast effects carrier. With increasing the distance from the center of explosion, the pressure, the temperature and the rate of the explosion gaseous products rapidly decrease. At the moment when the explosion gaseous products cool down and water vapor condensation begins, surrounding air starts to vaccum in the direction of the center of explosion.

This method of thermobaric warhead operating provides forming aerosol cloud of high homogeneity, volumetric explosion is nearly ideal, with a high rate of chemical energy transformation into the shock wave mechanical energy. The external meteorological effects on warhead efficiency are reduced by a short time of forming the volumetric explosion.

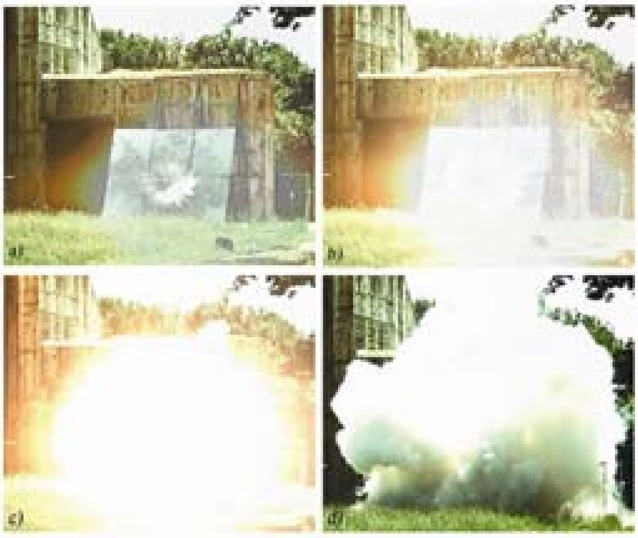

Figure 2. The mechanism of thermobaric warhead functioning: a) thermobaric mixture scattering, b) aerosol cloud forming, c) detonation, d) explosion gaseous products cooling down and surrounding air vaccuming

As shown in the reference [9], the primary injury mechanisms of thermobaric warhead are blast and heat. Secondary injury mechanisms are flying fragments created by interaction of the blast with structures (eg. flying bricks, shell and metal debris, etc.) and suffocation through the generation of toxic gases and smoke. The level of structural damage and injury caused by blast is dependent on the peak pressure, impulse of pressure positive phase duration, and the elastic–plastic strength and natural period of oscillation of the structure or body. In the human body, the blast wave interacts with many types of tissues (eg. skin, fat, muscle and bone) that differ in mechanical characteristics (density, elasticity and strength). Each tissue type, when interacting with a blast wave, is compressed, stretched, sheared or disintegrated by overload according to its material properties. Internal organs that contain air, as sinuses, ears, lungs and intestines, are particularly vulnerable to the blast. The whole body may also be thrown by a blast wind, which can result in fractures. Besides the obvious blast injuries, recent research has shown that there are neurological, biochemical and blood chemistry changes caused by the blast effects.

Thermobaric explosive mixture mass in warhead depends on the lethal overpressure value p_u at the given lethal range R_u, and per ref. [5] following the next form:

$$m_{ex} = a_o \cdot p_u^{a_1} \cdot R_u^{a_2} \tag{1}$$

Coefficients a_o, a_1 and a_2 depend on on the type of thermobaric explosive mixture, and are determined by semi-empirical methods. The lethal overpressure value differs for each type of target (infantry, vehicles, different purpose buildings, fortifications, etc.), but also depends on the desired level of target vulnerability (heavy, medium or low, etc.). These values are usually determined empirically and for some type of the targets are available in the reference [5]. Total warhead mass is equal to a sum of all element masses that are shown in Fig.1. According to the references [1-3], in the initial phase of the rocket design, the warhead mass can be determined based on the existing similar solutions using statistical data. Introducting the coefficient k_w, the warhead mass will be:

$$m_w = \frac{m_{ex}}{k_w} \tag{2}$$

According to the reference [5], the mass m_{eb} and diameter d_{eb} of the explosive burster (secondary explosive charge) can be determined by using the statistical coefficients k_{eb} and k_d, and known calibre d, by following the relations:

$$m_{eb} = \frac{m_{ex}}{k_{eb}} \tag{3}$$

$$d_{eb} = \frac{d}{k_d} \tag{4}$$

Since thermobaric mixture includes both solid and liquid phase, it is necessary to provide a free volume in warhead for vapour spreading that it could eventually occur during the exploitation. Using basic mathematical transformation, per reference [5], for known density of thermobaric explosive mixture ρ_{ex}, warhead length is as follows in the next form:

$$L_w = \frac{d}{2} + \frac{4 \cdot m_{ex}}{3 \cdot \rho_{ex} \cdot d^2 \cdot \pi} \cdot \frac{4k_d^{\,2} - 1}{k_d^{\,2} - 1} \tag{5}$$

Optimal design parameters determination of the shoulder-launched unguided rocket

The rocket design parameters optimization, which should provide the achievement of a maximum given range for known warhead mass, per [1-3] is reduced to a determination of maximum rocket velocity at a minimum start rocket mass.

According to the references [10, 11], velocity at the end of the shoulder-launched unguided rocket burning is determinated based on the given maximum value of direct fire range X_r and height of the target h. Since the thermobaric warheads are designed for destroying infantry hidden in all sorts of shelters (bunkers, fortifications, buildings, caves, etc.), often as a height of the target, the standard ceiling height of residential buildings is accepted. This is often acceptable in case of anti-terrorism act in the city areas. As the range is relatively small and the rocket flight is subsonic, all forces are neglected, except gravity. The velocity reached in the moment when the rocket leaves the launching tube, according to the ref [10, 11], it will be:

$$V_o = \sqrt{\frac{g \cdot \left(X_r^{\,2} + 16 \cdot h^2 \right)}{8 \cdot h}} \tag{6}$$

The rocket acceleration is essentially proportional to the net thrust. The usual approach in the initial phase of calculation is adopted that the thrust vs. time law is approximately of a rectangular shape, ref. [10, 12]. However, in this paper the thrust vs. time law will be adopted by an approximately trapezoidal shape. This shape better corresponds to thrust vs. time by the impulse rocket motor. Based on the above, the adopted law acceleration vs. time is presented in Fig.3. The first segment in Fig.3 – segment $k_1 \cdot t_o$, presents the start up and achievement of rocket motor operating mode, where t_o is a total operating time of motor, and k_1 is the empirical coefficient. The second segment – segment $(k_2 - k_1) \cdot t_o$ presents operating mode of rocket motor, while the segment $(1 - k_2) \cdot t_o$, presents shutdown mode and combustion residues of a propulsive charge, known as the tail off. Empirical coefficients k_1 and k_2 can be determined based on the existing similar solutions of the impulse rocket motor. Although tubular grains are adopted in this paper, which have a neutral surface burning, the combustion residues of a shutdown mode are the consequence of a variety of geometric imperfections of propellant and possible sliver existence.

Figure 3. Rocket acceleration vs. time for impulse rocket motor

Based on the method shown in [10], it is easy to derive that the total rocket moving time throughout its launching tube of length L_{LT} (motor operating time, i.e.) and adopted thrust vs. time law is:

$$t_o = \frac{L_{LT}}{V_o} \cdot \frac{3 \cdot (1 - k_1 + k_2)}{2 + k_2 (2 - k_2) + k_1 (k_1 - 3)} \tag{7}$$

In the same way, the maximum rocket acceleration in the launching tube can be derived:

$$a_{\max} = \frac{2 V_o^2}{L_{LT}} \cdot \frac{2 + k_2 (2 - k_2) + k_1 (k_1 - 3)}{3 \cdot (1 - k_1 + k_2)^2} \tag{8}$$

By adopting that the effective exhaust gas velocity is approximately equal to a specific impulse of the rocket motor I_{sp}, according to the Ciolkovski equation [1, 2, 10], mass ratio of propellant and final rocket mass can be determined:

$$Z = \frac{m_p}{m_f} = \exp\left(\frac{V_o}{I_{sp}}\right) - 1 \tag{9}$$

It should be noted that the specific impulse could be reliably estimated by some methods presented in [11] or [12]. Otherwise, specific impulse can be determined as a product of the characteristic velocity C^* which is thermochemical parameter, and the thrust coefficient C_F which in practice is in the range $1,3 \div 1,7$.

It would be helpful to use a reduced value of some parameters (overlined parameters), ref. [1, 2]. The reduced value is defined as a ratio of the considered parameter and unknown rocket calibre value. According to the references [1, 2], the reduced chamber diameter is determined for a given maximum chamber pressure p_c and yield strength of chamber σ_{ec}. The temperature field in the wall of the chamber should be taken into account, as well. Now, the reduced chamber diameter is as follows:

$$\overline{d_c} = \frac{\sigma_{ec}}{\sigma_{ec} + p_c} \tag{10}$$

The grain web thickness is equal to the product of the average burning rate for the expected chamber pressure r_m and the motor operating time t_o:

$$w = r_m \cdot t_o \tag{11}$$

The maximum rocket velocity corresponds to the optimal value of propellant mass fraction, that for a given payload mass can be achieved only for certain rocket calibre value. In general, the equation which expresses the conditions of the maximum range for a given payload, according to the references [1, 2], is:

$$\frac{\partial l_p}{\partial \varepsilon} \varepsilon + l_p + \frac{\pi}{4} \cdot k_L \cdot \frac{d^2 \cdot \left(1 - \overline{d_c}^2\right) \cdot \rho_{cr}}{\overline{m}_{kt}} \cdot l_p^2 = 0 \tag{12}$$

Thereby, the coefficient which takes into account the increase in the length of the cylindrical part of rocket motor in relation to the length of the propellant charge, is usually adopted from the range $k_L = 1,02 \div 1,05$. Further, ρ_{cr} is a reduced density of the chamber case and thermal insulation, $\overline{m}_{kt}$ is a mass of virtual payload. To solve the equation (12) it is necessary to know the maximum propellant charge length in function of the chamber charging coefficient ε. A function of maximum propellant charge length which meets the requirements of combusting without erosion by the Pobedonoscev criteria, in the rocket motor with the limited operating time, is as follows:

$$l_{p\max} = \frac{1 - \varepsilon}{\varepsilon} \cdot K_i \cdot w \tag{13}$$

By differentiating previous equation with respect to the chamber charging coefficient ε, and substituting into (12), the equation is obtained that gives a value of chamber charging coefficient which corresponds to a maximum range in function of the rocket calibre:

$$\left(\frac{\varepsilon}{\varepsilon - 1}\right)^2 = \frac{\pi}{4} \cdot k_L \cdot \frac{d^2 \cdot \left(1 - \overline{d_c}^2\right) \cdot \rho_{cr}}{\overline{m}_{kt}} \cdot K_i \cdot w \tag{14}$$

To solve the optimal rocket calibre value for motor with the limited operating time, it is necessary to introduce the project parameter N, ref. [1, 2]. This parameter actually represents a solution of the equation (14), so as to obtain a maximum value of the chamber charging coefficient $\varepsilon_{\max}$. According to the references [1, 3], the design parameter is as follows in the next form:

$$N = \frac{\sqrt{Z}}{\sqrt{\frac{\rho_p}{\rho_{cr}} \cdot \frac{\overline{d}_c^{\,2}}{1-\overline{d}_c^{\,2}} \cdot \frac{1}{k_L} - \sqrt{Z}}} \tag{15}$$

where ρ_p is density of the propellant.

It is also necessary to define virtual payload mass which is equal to a sum of warhead mass and masses of the rocket parts that are not explicit function of the propellant charge length, such as: grain carriers, bottom of chamber, membranes, seals, spacers, screws, etc. If it is assumed that the mass of these rocket elements is directly proportional to warhead mass through the coefficient k_{em}, the mass of the virtual payload is as follows:

$$\overline{m}_{kt} = m_w \cdot (1 + k_{em}) \tag{16}$$

The exact value of the coefficient k_{em} is unknown, and it can be assumed in the range $k_{em} \approx 0 \div 0,4$, ref. [1-3], that is based on the statistical parameters implemented in similar rocket construction. Thus, for each adopted value of coefficient k_{em}, the virtual payload mass can be calculated. Then, for the calculated mass $\overline{m}_{kt}$, per ref. [1, 2], the optimal rocket calibre value for rocket motor with limited operating time is defined by the following relation:

$$d = \frac{2N}{\sqrt{\frac{\pi \cdot \rho_{cr} \cdot \left(1-\overline{d}_c^{\,2}\right) \cdot k_L \cdot K_i \cdot w}{\overline{m}_{kt}}}} \tag{17}$$

Now, based on the known rocket calibre value, the propellant mass, which provides a maximum rocket velocity and at the same time meets the requirements of the combustion without erosion in the rocket motor with the limited operating time by the Pobedonoscev criteria, is calculated by the equation:

$$m_p = \rho_p \cdot \frac{K_i \cdot w}{N+1} \cdot \frac{d^2 \cdot \overline{d}_c^{\,2}}{4} \cdot \pi \tag{18}$$

According to the known value of the propellant mass fraction Z, final mass m_f, and start rocket mass m_R, are:

$$m_f = \frac{1}{Z} \cdot m_p \tag{19}$$

$$m_R = \frac{1+Z}{Z} \cdot m_p \tag{20}$$

Rocket caliber and mass rocket characteristics selection

As shown in the previous chapter, for the assumed coefficient k_{em}, calibre value is determined by the equation (17). In that manner, the series of different optimal calibre values are determined so as to obtain maximum value of velocity, that corresponds to the requirement for each start rocket mass. In this approach, for each value of coefficient k_{em}, the calibre in function of start rocket mass can be defined by some discrete form:

$$d = f_1(m_R) \tag{21}$$

On the other hand, for the known calibre value and concretely chosen technical decision of designed rocket sub-assemblies, it is possible to calculate very reliable mass characteristics for all rocket sub-assemblies, respectively total rocket mass. It follows that for a predicted calibre value, it is possible to determine some new discrete function, that after mathematical transformation can be expressed in a form:

$$d = f_2(m_R) \tag{22}$$

Solution of the discrete system equations (21) and (22), for the designed rocket with concretely chosen technical decision of its sub-assemblies, is unique and optimal by means of given maximum range value.

On the basis of the previous consideration, a function described by the equation (22) is needed to be determinated. That means that for predicted value caliber, masses of all rocket sub-assemblies (cylindrical case mass, chamber bottom mass, nozzle mass, etc.) are determinated, respectively total rocket mass is determinated.

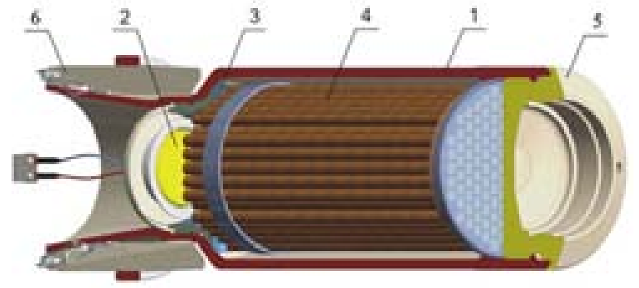

Figure 4. Impulse rocket motor: 1 – chamber case, 2 – igniter, 3 – nozzle, 4 – propelling charge, 5 – chamber bottom, 6 – stabilizing fin

According to the references [1] and [2], the mass of the rocket motor cylindrical case is:

$$m_c = \rho_{cr} \cdot l_{p\max} \cdot k_L \cdot d^2 \cdot \left(1-\overline{d}_c^{\,2}\right) \cdot \frac{\pi}{4} \tag{23}$$

The chamber bottom is usually made as a part of ellipsoid, a sphere or a torus, which is generally favorable from the mass reduction point of view, but can take a lot of space. Therefore, the impulse motor design is often a matter of compromise. At the expense of insignificant mass increase, but due to reduction of the space, the approximate flat circular panel shape is adopted. If it is assumed that in a certain case the chamber bottom is flat circular panel, ref. [15], the chamber bottom mass is as follows:

$$m_{dc} = 0.34 \cdot d^3 \cdot \sqrt{\frac{p_c \cdot \nu_{dc}}{\sigma_{mdc}^{radial}}} \tag{24}$$

where σ_{mdc}^{radial} is the ultimate strength of chamber bottom in radial direction, and $\nu_{dc} = 1,20 \div 1,30$ is the safety factor of chamber bottom.

According to the reference [15], divergent nozzle part mass is essentially proportional to the total impulse, wich is the product of the propelling charge mass m_p and the specific impulse I_{sp}:

$$m_n = k_n \cdot m_p \cdot I_{sp} \tag{25}$$

Based on the statistical data, the coefficient of proportionality in the previous equation is approximately:

$k_n \approx (2,4 \div 2,6) \cdot 10^{-4}$. According to [16], the thermal insulation mass can be shown in the next form:

$$m_{ti} = k_{ti} \cdot m_p \cdot I_{sp} \quad (26)$$

Statistical coefficient in the previous equation takes values in the range: $k_{ti} \approx (1,0 \div 2,5) \cdot 10^{-5}$, where the lower coefficient value refer to the rocket motor that has a shorter oparating time. The statistical coefficient value for the rocket motor whose not using the thermal insulation (the most common for impulse rocket motor), $k_{ti} = 0$ is adopted.

For the predicted calibre value, the total rocket mass is equal to the sum of all sub-assembly rocket masses, such as: warhead, propelling charge, cylindrical case of rocket motor, chamber bottom, nozzle and thermal insulation. According to the analysis which is exposed in ref. [17], it is necessary to make a correction of this calculated rocket mass, because it does not take into estimate the mass of some sub-assemblies such as: stabilizing fin mass, igniter mass, grain carriers, screws, hermetic mass, etc. Now, the equation of total rocket mass is as follows:

$$m_R = \left[m_w + m_p + m_c + 2m_{dc} + m_n + m_{ti} \right] \cdot (1 + k_R) \quad (27)$$

Based on the statistical data, ref. [17], the coefficient of increasing rocket mass is typically: $k_R \approx 0,01 \div 0,03$. In that manner, the discrete equation (22) is completely defined by (27), and along with the equation (17) creates the system of equations that provides the optimum calibre rocket value.

Determining design parameters of the impulse rocket motor

On the basis of the selected calibre rocket value, mass and web thickness, it is needed to determine the other rocket motor parameters. The impulse rocket motor can use different forms of propelling charges (tubular, strip, cord, shaped strip, etc.), but the most common tubular uninhibited grains are applied, and it is accepted in this paper. The internal chamber diameter of the rocket motor is given by relation, ref. [1, 2]:

$$D_c = d \cdot \overline{d_c} \quad (28)$$

Optimal tubular grain geometry needs to meet several criteria: ultimate strength, combustion without erosive burning on outer and inner tubular surfaces, and maximum value of chamber charging coefficient. Considering that the impulse rocket motor inertial force is dominant due to a relatively high rocket acceleration in the launching tubes, for dimensioning tubular length, we use the ultimate strength criteria of grain. According to the reference [18], propellant length limit value must meet the following requirement:

$$l_p \le \frac{\sigma_{mp}}{a_{\max} \cdot \nu_p \cdot \rho_p} \quad (29)$$

where σ_{mp} is the ultimate strength of propellant grain, and $\nu_p = 1,15 \div 1,30$ is the safety factor of tubular propellant grain.

For the selected value of propellant grain length, and from the flow condition in internal diameter of tube, which meets the requirements of combusting without erosion by the Pobedonoscev criteria, the internal diameter of tube is calculated as follows:

$$d_u = \frac{4 \cdot l_p}{K_i^{\,u}} \quad (30)$$

Although, the flow conditions in this apertures are not the same, in this paper to simplify, we assume that the value of the Pobedonoscev number for the external and internal apertures are the same: $K_i = K_i^{\,s} = K_i^{\,u}$. For the known web thickness value, previously determined by equation (11), the external diameter of tube is:

$$d_s = d_u + 4 \cdot w \quad (31)$$

On the basis of the tube dimensions calculated and the propellant mass that fulfill specified project requirement, the number of tubular grains is calculated by equation:

$$n_c = \frac{4 \cdot m_p}{\pi \cdot \rho_p \cdot l_p \cdot \left(d_s^{\,2} - d_u^{\,2} \right)} \quad (32)$$

For geometrical parameters defined by relations (28 ÷ 31) it should be checked if the number of tubular grains n_c can be placed into the rocket motor chamber. Analysis of the maximum possible tubular grains of known dimensions that could be placed in the chamber, are exposed in details in ref. [12, 18]. As noted in [12], the maximum possible number of tubular grains into the chamber can be provided using the chess layout. The chess layout of tubular grains is defined as follows: one tube is located in the center of the chamber, while the other tube centers are located in the vertices of equilateral triangles, with the side $d_s + z_b$, as shown in Fig.5.

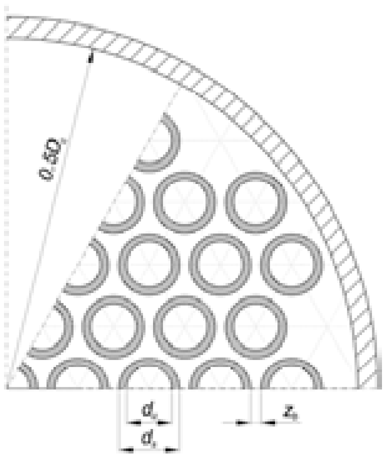

Figure 5. Sketch of tubular grains in chess layout

The total number of tubular grains that can be placed in the rocket chamber is:

$$n_{c\max} = 6 \cdot \left(n_{c2} + \sum_{i=1}^{n_{c1}-1} \left(n_{c1} - i \right) \right) + 1 \quad (33)$$

where n_{c1} is the first lower integer number calculated by equation:

$$n_{c1} = \frac{D_c - (d_s + z_b)}{2(d_s + z_b)} + 1 \quad (34)$$

The number of tubular grains that can be placed at the largest radius regarding to the chamber axis is n_{c2}. As shown in [12], number n_{c2} is determined by numerical method. Essentially, it determines whether the coordinates of the tubular grains centers in the last row from the chamber center fulfill the following requirement:

$$s_c \leq \frac{D_c - (d_s + z_b)}{2} \tag{35}$$

At the same time the distance s_c is defined by expression:

$$s_c = (d_s + z_b) \cdot \sqrt{(n_{c1} + 0.5 \cdot i - 1)^2 + 0.75 \cdot i^2} \tag{36}$$

According to the equation of mass conservation, ref. [18], nozzle troat area is determined. The complete mass flow rate of gases caused by combusting into chamber, should be ejected through a nozzle, and according to [18] follows:

$$A_t = \rho_p \cdot S_{bo} \cdot \sqrt{\frac{R \cdot T_c}{\kappa}} \cdot \sqrt[n]{\frac{b}{r_m^{(1-n)}}} \cdot \left(\frac{2}{\kappa+1}\right)^{\frac{\kappa+1}{2\cdot(1-\kappa)}} \tag{37}$$

where S_{bo} is the initial propellant burning surface, T_c is the absolute chamber temperature, R is a specific gas constant of the combustion gases, κ is a specific heats ratio of the combustion gases, b is the coefficient of the San Robert's burning rate law, n is exponent of the San Robert's burning rate law.

During the design and development, it is necessary to determine rocket length, based on the results of the previously presented calculation methods. The length of the rocket with thermobaric warhead for the shoulder-launched, can be estimated using the following relation, ref. [16]:

$$L_R \approx L_w + 0,67 \cdot d + k_L \cdot l_p + \frac{d_t \cdot \left(\sqrt{\varepsilon_n} - 1\right)}{2 \cdot tg\gamma_n} \tag{38}$$

At the same time, according to [12], the expansion nozzle ratio for the impulse rocket motor usually takes values in the range: $\varepsilon_n = 2 \div 4$, and the half angle of divergent conical nozzle usually takes values in the range: $\gamma_n = 8^o \div 16^o$.

Rocket ballistic characteristics analysis

In order to perceive the selected rocket design parameters according to the previously exposed model, it is necessary to compare ballistic rocket characteristics with expected values for given tactical and technical requirements. It means that the chamber pressure, the rocket motor's action time and initial rocket velocity for selected geometric and mass rocket parameters should be calculated.

Mathematical model which describes the process in the impulse rocket motor chamber includes the equations of unsteady burning propellant rate and burning generation gases flow through internal and around external surfaces of the propellant charge. In general, the burning rate of solid propellant depends on: combustion chamber pressure, initial temperature of the propellant, mass velocity of combustion gas trough stream apertures (erosive burning, i.e.), pressure gradient, axial and radial rocket acceleration, and induced grain stress, ref. [18, 19]. Withouth the loss of accuracy for motor operating in short period of time (impulse rocket motor), the influence of propellant strain may be neglected. The influence of axial rocket acceleration can be neglected too, because its vector is perpendicular to the normal vector of burning surfaces (except insignificant effect on increasing head-on burning rate of propellant part close to the nozzle). Spin rocket acceleration for higher order values is lower than the axial acceleration, so its influence for the impulse rocket motor is also neglected. According to the references [18, 19], a differential equation that describes the unsteady burning rate is as follows:

$$\frac{\partial w}{\partial t} = b \cdot p_c^{\,n} \cdot e^{\sigma_p (T_p - T_{oo})} \cdot \left[1 + \delta \cdot k_\varepsilon \cdot \left(\frac{\rho u \cdot \sqrt{C_f}}{\rho_p \cdot b \cdot p_c^{\,n}} - J_*\right)\right] \cdot \left[1 + \delta_1 \cdot \left(\psi_{10} + \psi_{11} \cdot \left(\frac{p}{p_o}\right)\right) \cdot \frac{\alpha_{td}}{b^2 \cdot p_c^{\,2n+1}} \cdot \frac{\partial p_c}{\partial t}\right] \tag{39}$$

where σ_p is temperature sensitivity of burning rate, T_p is the propellant temperature, T_{oo} is the reference temperature, δ is erosive burning rate factor, k_ε is erosive burning coefficient, ρu is mass velocity of the combustion gases per unit area, C_f is friction coefficient of the flow trough aperture J_* is the critical Viljunov number, δ_1 is unsteady burning rate factor, ψ_{10} and ψ_{11} are pressure function coefficients in expression of unsteady burning rate, p_o is ignition pressure, α_{td} is thermal diffusivity of propellant.

The first square bracket term on the right side of differential equation (43) represents the influence of erosive burning effect per the theory of Viljunov. The second square bracket term represents the influence of pressure gradient on the burning rate. In this paper, the friction coefficient of the flow trough aperture C_f is determined using the Nikuradze relation, ref. [18, 19], because it gives the most unfavorable value of the Viljunov number. As well, it is necessary to consider the other friction laws, such as the laws of Blazius, ref. [18, 19] or Colburne – Ishihara, ref. [20], in a detailed analysis of the rocket chamber flow. Differentiating with respect to time, an equation that describes a change of the propellant burning surface of tubular grains charge in function of web, ref. [18, 19], is given by equation:

$$\frac{\partial S_b}{\partial t} = -2 \cdot \pi \cdot n_c \cdot (d_u + d_s) \cdot \frac{\partial w}{\partial t} \tag{40}$$

Differential equation that describes the pressure change of rocket motor chamber can be derived from the equation of mass conservation of gases caused by the combusting rocket motor's grain, ref. [15]:

$$\frac{\partial p_c}{\partial t} = \frac{4R \cdot T_c}{\pi\left[D_c^{\,2} - n_c\left(d_s^{\,2} - d_u^{\,2}\right)\right] + 4S_b w} \cdot \left(\rho_p - \frac{p_c}{R \cdot T_c}\right) \cdot \left[w\frac{\partial S_b}{\partial t} + S_b\frac{\partial w}{\partial t}\right] - \frac{p_c \cdot A_t}{\eta_{C^*}} \cdot \sqrt{\frac{\kappa}{R \cdot T_c} \cdot \left(\frac{2}{\kappa+1}\right)^{\frac{\kappa+1}{\kappa-1}}} \tag{41}$$

The thrust of a rocket may be expressed as a mesure of performance of the expansion process in the nozzle (C_F), ref. [15, 18]:

$$F = p_c \cdot A_t \cdot \eta_{C_F} \cdot C_F \tag{42}$$

Velocity and trajectory of the rocket are determinated by solving differential equation of motion particles of variable mass. The lock force value in the initial motion moment is neglected, and it is supposed that the geometrical axis of rocket symmetry and direction of thrust force are overlapped. It is also supposed that the axis of launching tube takes the known elevation angle *EL* to the horizont. In the direction of the rocket's motion throught the launching tube, the product of the mass and the acceleration has to equal the sum of all forces, namely the propulsive, aerodynamic, gravitational and friction forces:

$$m(t)\cdot\frac{\partial V_x}{\partial t}=F(t)-\frac{1}{2}\rho A\cdot C_x(M)\cdot V_x^{\,2}-\\-m(t)\cdot g\cdot\left[\sin(EL)+\mu\cdot cos(EL)\right]\tag{43}$$

where ρ is ambient density, A is rocket reference area, and $C_x(M)$ is aerodynamic axial coefficient.

The second term on the right side of differential equation (43), which describes force of aerodynamic drag, may be neglected for rocket's moving throughout its launching tube, since its influence is insignificant and withouth loss of acquired results accuracy. The change of rocket mass in any interval of time is determined by the following equation:

$$m(t)=m_R-\frac{A_t}{\eta_{C^*}}\cdot\sqrt{\frac{\kappa}{R\cdot T_c}\cdot\left(\frac{2}{\kappa+1}\right)^{\frac{\kappa+1}{\kappa-1}}}\cdot\int_0^t p_c(t)\,dt\tag{44}$$

In this paper the most prominent friction coefficient model in the system dynamics literature is applied given by a simple velocity dependence, that assumed to be exponentially weakening with increasing velocity, and by [21] as follows:

$$\mu=\mu_d+(\mu_s-\mu_d)\cdot e^{-\lambda\cdot V_x}\tag{45}$$

where there are three independent experimental friction parameters: μ_s - static friction coefficient, μ_d - dynamic friction coefficient, λ - coefficient which describes how fast the static friction approaches the dynamic one.

Numerical example and analysis of results

The calculation of the shoulder-launched unguided rocket with thermobaric warhead design parameters was executed for the values that are presented in the following Table 1 and 2.

Table 1. Input data for warhead calculation

Parameter:	Notation:	Value:
Lethal range	$R_u\,[m]$	7,0
Lethal overpressure	$p_u\,[bar]$	0,3
Thermobaric explosive density	$\rho_{ex}\,[kg/m^3]$	1278
Experimental coefficients of thermobaric warhead	a_o	0,0245
	a_1	1,1547
	a_2	3,0023
	k_w	0,495
	k_{eb}	9,410
	k_d	2,250

Table 2. Input data for rocket motor calculation

Parameter:	Notation:	Value:
Direct fire range	$X_r\,[m]$	250
Height of target	$h\,[m]$	2,6
Specific impulse	$I_{sp}\,[Ns/kg]$	2130
Launcher tube length	$L_{LT}\,[m]$	0,960
Acceleration vs. time law coefficients	k_1	0,113
	k_2	0,600
Maximum chamber pressure	$p_c\,[bar]$	450
Pobedonoscev number	K_i	170
Mechanical characteristics of propellant	$\rho_p\,[kg/m^3]$	1620
	$\sigma_{mp}\,[Pa]$	10,8·10^6
Usage material mechanical characteristics of rocket motor	$\rho_{cr}\,[kg/m^3]$	3100
	$\sigma_{ec}\,[Pa]$	550·10^6
	$\sigma_{md}^{radial}\,[Pa]$	550·10^6
Gas dynamic flow parameters in rocket motor	$p_o\,[bar]$	25
	κ	1,231
	$R\,[J/kgK]$	322
	$T_c\,[K]$	2620,5
Reference temperature	$T_{oo}\,[K]$	293,15
Expansion ratio	ε_n	3,815
Burning rate parameters	$r_m\,[m/s]$	35,97·10^-3
	$b\,[m/s]$	4,518·10^-3
	n	0,118498
	$\sigma_p\,[1/K]$	3,542·10^-3
	J_*	15,775
	ψ_{10}	1,55
	ψ_{11}	1,45
Experimental friction coefficients during rocket's moving throughout its launching tube	μ_s	0,20
	μ_d	0,14
	λ	5,0

Design parameter values necessary for the rocket calibre optimization of motor with the limited operating time are determined according to equations (1÷15) and initial parameters (Table 1 and 2). The calculation results are given in Table 3.

Table 3. The rocket parameters required for caliber determination

Parameter:	Notation:	Value:
Thermobaric mixture mass	$m_{ex}\,[g]$	2100
Warhead mass	$m_w\,[g]$	4243
Initial rocket velocity	$V_o\,[m/s]$	171,8
Total rocket moving time throughout its launching tube	$t_o\,[ms]$	9,90
Maximum rocket acceleration	$a_{\max}\,[m/s^2]$	23304
Ciolkovski number	Z	0,0840
Reduced chamber diameter	$\overline{d_c}$	0,9244
Web thickness	$w\,[mm]$	0,35
Design parameter	N	0,2006
Maximum chamber charging coefficient	$\varepsilon_{\max}$	0,1671

On the basis of relations (17÷27), for the shoulder-launched rocket calibre with thermobaric warhead optimization can be applied. The optimal rocket calibre value is in the cross section of the two curves obtained by calculation using methods f_1 and f_2, according to the equations (21, 22), respectively. The results of the rocket calibre optimization are presented in Fig.6.

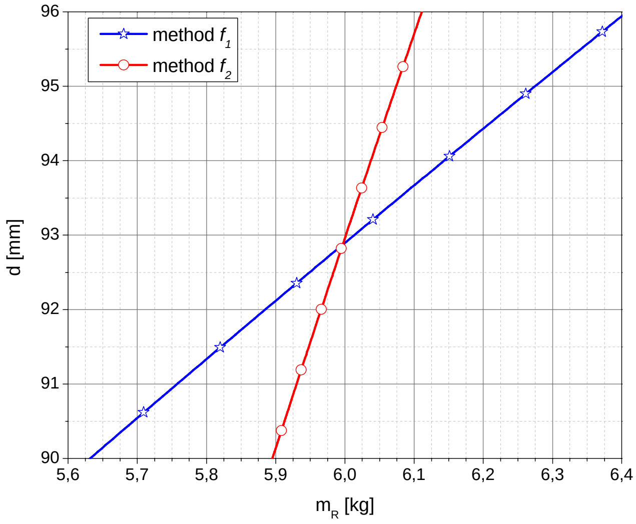

Figure 6. Optimal rocket calibre selection

Design requirements are usually contradictory because besides the minimum values of mass and dimensions, the maximum range and efficacy on the target are also required (Table 1 and 2). If it is assumed that in any phase of the design some technical requirements should be changed, but not the materials which are used for the rocket production, their individual influence on the possible change of calibre and start rocket mass should be examined, too. A partial influence of the direct fire range, target height, maximum value of chamber pressure, lethal range and launcher tube length at changing the input requirements for ±30% are presented in Figures 7 and 8. All curves have cross section in a point which presents the optimal selected rocket calibre value for the originally given requirements.

Analysing Figures 7 and 8 it can be concluded that an eventual requirement for the direct fire range increasing leads to rapid rocket calibre and mass increase (and vice versa). This occurrence comes from the fact that for a higher range, it is necessary to provide a higher initial rocket velocity, as well as the mass increase of the propellant charge or apply the propulsive system with a higher specific impulse (which in this paper is not considered).

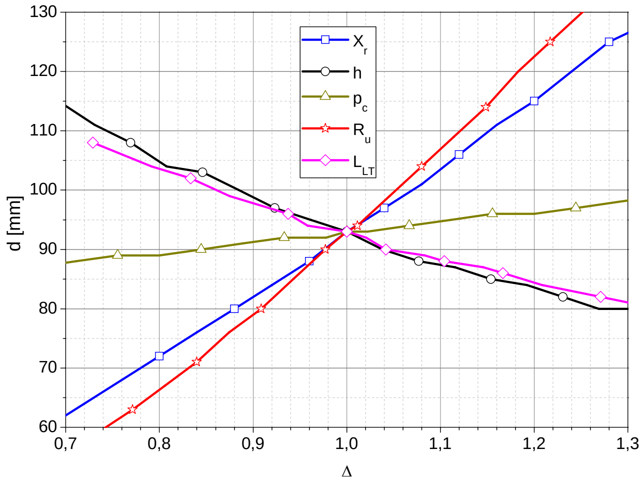

Figure 7. Partial influence of input data to select optimal calibre for rocket with thermobaric warhead

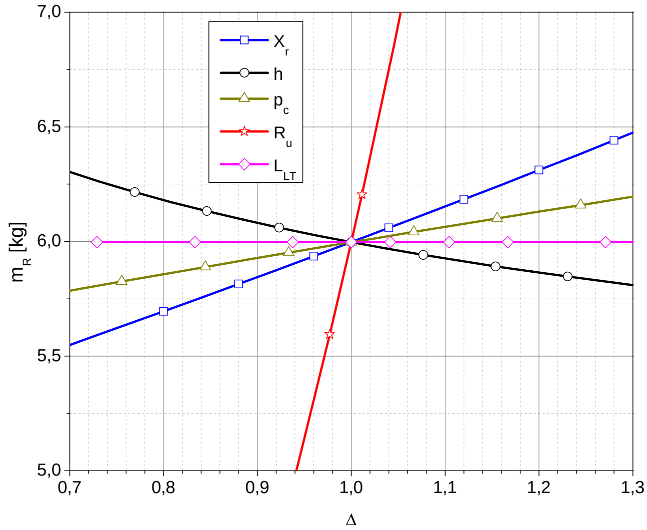

Figure 8. Partial influence of input data to rocket mass with select optimal calibre for rocket with thermobaric warhead

The target height increase has a positive effect on the rocket calibre and mass decrease, but the change is slightly lower than the direct fire range value. This occurrence can also be explained with the requirements for lower, respectively higher initial rocket velocity, in function of increasing or decreasing the target height.

The change of predefined chamber pressure causes small deviations of the rocket calibre, but some more intensive change of the rocket mass. It can be seen that the change of chamber pressure value is approximately similar to a value as for the change of target height, but the opposite sign. The consequence of mass increasing along with the chamber pressure increase occurs with the necessity for a thicker rocket motor case.

A slight change in lethal range results in a considerable change of the rocket mass and optimal rocket calibre, because for a given chemical composition of thermobaric mixture increasing lethal range requires larger and heavier warhead. Increasing of the launcher length favorably leads to a decrease of optimal rocket calibre for a long operating time of the rocket motor and has no influence on the start rocket mass, but only on the increase launcher mass. Based on the above statements, selecting the initial project requirements must be considered, as well as possible changing during the design development.

For the selected rocket calibre value from Fig.6, the calculation of geometrical and mass rocket parameters can be executed, as shown in Table 4.

Table 4. The rocket parameters calculated for selected optimal calibre

Parameter:	Notation:	Value:
Rocket calibre	d [mm]	93
Warhead length	L_w [mm]	429
Rocket motor chamber diameter	D_c [mm]	85,9
Maximum propellant length	l_{pmax} [mm]	296
Needed propellant mass	m_p [g]	465
Propellant length	l_p [mm]	238
Internal diameter of tube	d_u [mm]	5,6
External diameter of tube	d_s [mm]	7,0
Needed number of tubular grains	n_c	87

In determining the propellant length it is needed to compare the requirements from relations (13, 29), wherein the lower design parameter value of these two should be selected. Then, for selected propellant length value regarding the required propellant mass is calucalated by equation (18), and necessary number of tubular grains by equation (32). As already mentioned, for the accepted geometrical parameters of propellant charge and rocket motor, it is necessary to analyze the capability to place a maximum number of tubes in the motor’s chamber, as shown in Fig.9.

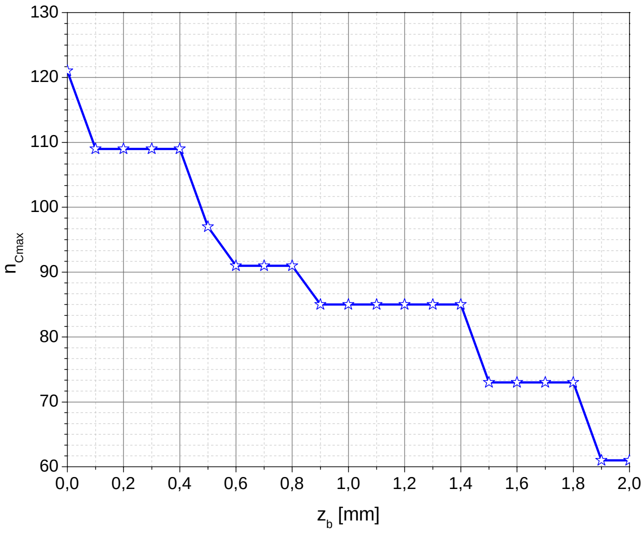

Figure 9. Maximum possible number of tubes in the rocket chamber related to grains gap

Fig.9 shows a feature of tube number discontinuality in a given circular section, and may be useful in determination of grain carriers gap tolerance. Analyzing required propellant mass and its achieved value for the adopted geometric parameters it is noticed that the propellant mass can be either higher or lower than required. An additional request in the optimization of propellant charge is flow conditions check in the rocket chamber, respectively erosive burning parameter. In this paper the Pobedonoscev number is used as an erosive burning parameter, and that is shown in Table 5. Whereas the Pobedonoscev numbers for external and internal apertures are constant, for a specified number of tubes and for any gap value of suitable range it practically means that the gap value has no direct influence on the flow character, but only on the tubular grains number.

Table 5. Maximum possible number of tubes in the rocket motor chamber in dependence of tubular grains gap

z_b [mm]	$n_{c\,max}$	$m_p\,[g]$	K_i^u	K_i^s
0,6 ÷ 0,8	91	486	170,0	207,7
0,9 ÷ 1,4	85	454	170,0	176,3
1,5	73	408	170,0	125,8

For the selected tubular grains number and previously determined geometric parameters of the propellant charge, according to the equations (20, 37, 38), the elementary geometric and mass rocket characteristics can be determined, as shown in Table 6.

Parameter simulations for the rocket motor design are carried out by equations (39-45) for three temperature usages. This system of differential equations is solved by the usage of the fourth order Runge – Kutta method. The results of the chamber pressure are presented in Fig.10, and the results of the thrust force are presented in Fig.11.

Table 6. The rocket parameters calculated for the selected number of tubular grains

Parameter:	Notation:	Value:
Number of tubular grains	n_c	85
Propellant mass	$m_p\,[g]$	454
Start rocket mass	$m_R\,[g]$	5859
Nozzle throat diameter	$d_t\,[mm]$	45,9
Nozzle exit diameter	$d_e\,[mm]$	89,7
Clemmung number	K_n	485,1
Rocket length	$L_R\,[mm]$	816

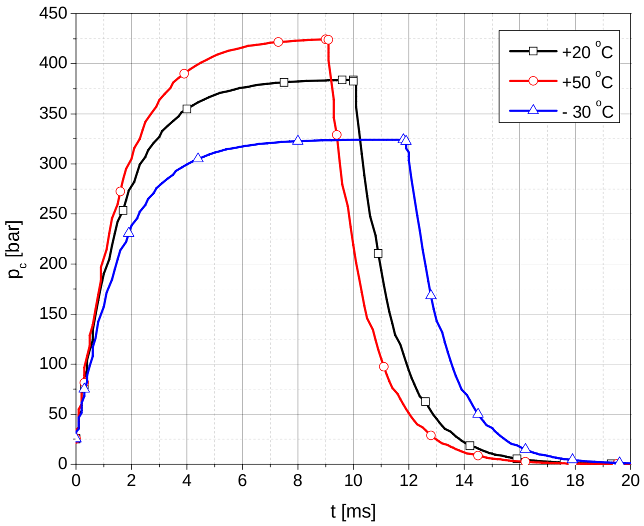

Figure 10. Chamber pressure vs. time

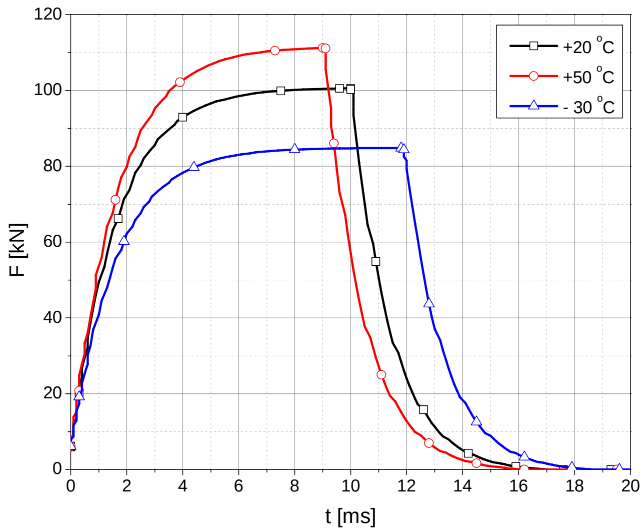

Figure 11. Thrust force vs. time

Primary ballistic parameters of the designed rocket for three usage temperatures are shown in Table 7. Maximum chamber pressure value meets the project requirements.

Total operating time t_{tot} is higer than the time of rocket's moving throughout its launching tube t_o which may adversely increase shooter's safety. Furthermore, it is noted that burning time t_b is less than the time of the rocket's moving throughout its launching tube, which means that the shooter will be located in a flow of residual gases from the chamber. Maximum nozzle exit pressure value at the end of burning is close to the atmosphere (in this example it is lower than the 2.4 *bar* for all three usage temperatures), and the shutdown process lasts several milliseconds. From this example follows that the time in which burnt products affect the shooter is very short and with negative gradient of pressure and temperature. So, the safety requirements should be checked in the real exploitation conditions. In practice, usually the other criteria are used for evaluation of the critical motor's operating time considering the shooter's safety, as the time for 50% or 10% of the maximum chamber pressure, but not total operating time of motor.

Table 7. Primary ballistic parameters of the designed rocket in dependence on the propellant temperature

Parameter notation:	Value:		
	$T_p = +20^o C$	$T_p = +50^o C$	$T_p = -30^o C$
$p_{c\max}\,[bar]$	384,0	424,4	324,2
$t_b\,[ms]$	10,20	9,06	11,89
$t_{tot}\,[ms]$	16,38	15,38	18,21
$F_{\max}\,[kN]$	100,55	111,21	84,79
$I_{tot}\,[Ns]$	990,64	995,52	987,53
$I_{sp}\,[Ns/kg]$	2182,0	2192,8	2175,2
$t_o\,[ms]$	11,87	11,41	12,76
$V_o\,[m/s]$	171,12	172,31	166,62
$V_f\,[m/s]$	174,04	177,04	173,96
$X_f\,[m]$	1,763	1,728	1,883

It should be noted that the total burning time and total impulse in real impulse rocket motor with tubular grains are less than the calculated by the previous theory. These phenomena are described by a process which appears immediately before the end of burning. Namely, because of the propellant geometric imperfections, it is reduced to ultra thin cilindric webs which break up before the end of a combustion, as well as in ejection through the nozzle due to the gas flow existence. Portion of the ultra thin cilindric webs, which was kept on the grain carriers burns, and a small portion of the gas flow exits through the throat nozzle. Unsteady burning rate influences the total time of burnout decrease, as well. It is therefore necessary to experimentally confirm that the results in the real rocket motor with values are obtained by a simulation.

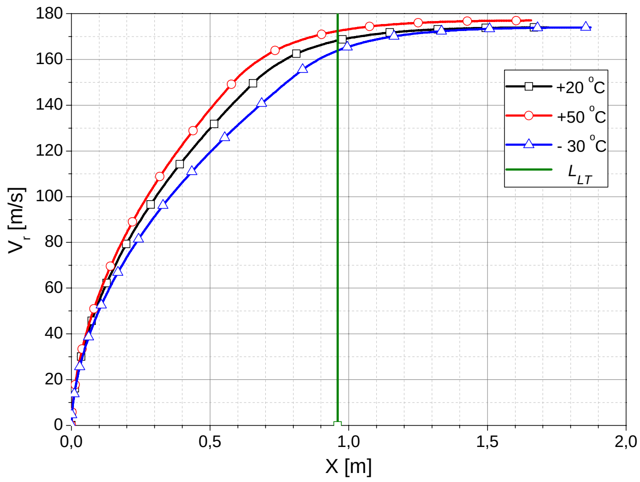

Figure 12. Rocket velocity vs. distance travelled

In the example, the rocket velocity at the end of an active phase is insignificantly higher than the necessary value which meets the technical requirements. As already shown, the real value of total impulse is most likely to be lower, and therefore it is the rocket velocity. As shown in Fig.12, although the rocket motor operates out of the launching tube, essentially there is no rocket acceleration. The references [10, 11] imply that there is practically no lateral wind effect on the rocket accuracy.

Conclusion

In this paper, the mathematical model for optimal calibre determintion of the shoulder-launched unguided rocket with thermobaric warhead is set by a predetermined tactical and technical requirements and adopted materials used for the rocket production. Since the design requirements are usually contradictory, it is necessary to consider and analyze carefully the individual contribution of each of them in the final dimension and in the rocket mass, especially in terms of existing materials usage.

Based on the selected rocket calibre, calculation of primary geometrical and mass parameters of warhead and rocket motor are presented. In respect of such selected parameters, a theoretical model of ballistic characteristics check is presented, with a special review on the unsteady burning rate by the impulse rocket motor. The estimation of expected values of basic ballistic parameters is given. In general, this theoretical model is based on the idealized stream process, which should be checked and upgraded through the development in real tests.

Based on the results of real motor tests it is necessary to prescribe safety measures and conditions for the appropriate usage of the shooter's protective equipment. That implies the criteria by which the rocket motor in some cases is allowed to operate outside the launching tube at the end of the burning process (eject phase of residual gases from the chamber).

Previously presented model of the rocket with thermobaric warhead can also be used for designing rockets with various types of warheads (shaped charge, incendiary, fragmentation, smoke, etc.) and with the motor of limited operating time.

References

[1] ШАПИРО,Я.М., МАЗИНГ,Г.Ю., ПРУДНИКОВ,Н.Е.: *Основы проектирования ракет на твердом топливе*, Военное издательство министерства обороны СССР, Москва, 1968.

[2] VUČUROVIĆ,O.: *Osnovi projektovanja raketa*, Mašinski fakultet, Beograd, 2003.

[3] БЕЛОВ,Г.В., ЗОНШАЙН,С.И., ОСКЕРКО,А.П.: *Основы проектирования ракет*, Машиностроение, Москва, 1974.

[4] ДРАКИН,И.И.: *Основы проектирования беспилотных летательных аппаратов с учетом экономической эффективности*, Машиностроение, Москва, 1973.

[5] NENADOVIĆ,LJ., KRALJEVIĆ,S.: *Ubojna sredstva na bazi prostorne eksplozije - naučnotehničke informacije*, Vojnotehnički Institut, Beograd, ISSN 1820-3418, ISBN 978-86-81123-20-1, 2006, No.2.

[6] ZHANG,F. (Ed.),: *Shock Wave Science an Tehnology Reference Library*, Vol. 4, Springer – Verlag Berlin Heidelberg, 2009.

[7] SIMIĆ,D., POPOVIĆ,M., SIROVATKA,R., ANĐELIĆ,U.: *Influence of Cast Composite Thermobaric Explosive Compositions on Air Shock Wave Parameters*, Scientific Technical Review, ISSN 1820-0206, 2013, Vol.63, No. 2, pp.63-69.

[8] SIMIĆ,D., PETKOVIĆ,J., MILOJKOVIĆ,A., BRZIĆ,S.: Influence of Composition on the Processability of Thermobaric Explosives, Scientific Technical Review, ISSN 1820-0206, 2013, Vol.63, No.3, pp.3-8.

[9] WILDEGGER-GAISSMAIER,A.E.: *Aspects of thermobaric weaponry*, ADF Health, April 2003, Vol.4.

[10] ГАНТМАХЕР,Ф.Р., ЛЕВИН,Л.М.: *Теория полета неуправляемых ракет*, Государственное издательство физико-математической литературы, Москва, 1959.

[11] ĆUK,D., ĆURČIN,M., STOJKOVIĆ,S.: *O preciznosti i izboru optimalnih raketodinamičkih parametara neupravljivih protivoklopnih projektila*, Naučnotehnički pregled, ISSN 0350-0667, 1988, Vol.38, No.1, pp.3-10.

[12] SAVKOVIĆ,M.: *Analiza unutrašnje balističkih parametara impulsnog raketnog motora OSA-2*, Elaborat R-25.534, Vojnotehnički Institut, 1978.

[13] DEKKER,A.O.: *Rapid Estimation of Specific Impulse of Solid Propellants*, Journal of Jet Propulsion, 1956, Vol.26, No.7, pp.572-575.

[14] BHAT,V.K., KULKARNI,A.R., SINGH,H.: *Rapid Estimation of Specific Impulse*, Defence Science Journal, 1988, Vol.38, No.1, pp.59-67.

[15] ФАХРУТДИНОВ,И.Х., КОТЕЛЬНИКОВ,А.В.: *Конструкция и проектирование ракетных двигателей твердого топлива*, Машиностроение, Москва, 1987.

[16] СИНЮКОВ,А.М., ВОЛКОВ,Л И., ЛЬВОВ,А.И., ШИШКЕВИЧ,А.М.: *Баллистическая ракета на твердом топливе, Военное издательство министерства обороны СССР*, Москва, 1972

[17] МОСКАЛЕНКО,Г.М.: *Инженерные методы проектирования в ракетодинамике*, Машиностроение, Москва, 1974.

[18] GLIGORIJEVIĆ,N., DIVAC,S., LEMIĆ,D., PETRIĆ,A.: *Raketni motori sa čvrstom pogonskom materijom* (izabrana poglavlja), Vojnotehnički institut, ISBN 978-86-81123-63-8, Medija centar „Odbrana", ISBN 978-86-335-0391-4, Beograd, 2013.

[19] JOVANOVIĆ,V.: *Određivanje osnovnih parametara nestacionarnih procesa u komorama impulsnih raketnih motora sa čvrstom pogonskom materijom*, Naučnotehnički pregled, ISSN 0350-0667, 1990, Vol.40, No.8-9, pp. 3-11.

[20] ISHIHARA,A.: *The effect of cross-flow temperature burning rate in a solid propellant*, 24th Symposium on Combustion/The Combustion Institute, 1992, pp. 1931-1938.

[21] BERGER,E.: *Friction modeling for dynamic system simulation*, Applied Mechanics Reviews, 2002, Vol. 55, No. 6, pp. 535-577.

Received: 20.10.2015.
Accepted: 22.12.2015.

Projektovanje rakete sa termobaričnom bojevom glavom za ručni bacač raketa

Ovaj rad prikazuje metodologiju proračuna masenih i geometrijskih parametara rakete sa termobaričnom bojevom glavom za ručni bacač raketa. Optimizacija projektnih parametara rakete za zadate taktičko tehničke zahteve izvršena je prema kriterijumu masene efikasnosti pomoću dve metode proračuna. Vrednost kalibra rakete po prvoj metodi određuje se na osnovu maksimalno zadatog dometa u funkciji mase korisnog tereta. Druga metoda pretpostavlja da je vrednost kalibra poznata pa se na osnovu toga računa masa svih podsklopova, odnosno masa cele rakete. Optimalne vrednosti kalibra za raketu sa ograničenim vremenom rada raketnog motora nalaze se kao zajedničko jedinstveno rešenje oba navedena iterativna postupka. Na konkretnom primeru prikazana je primena izloženog modela – za zadate taktičko tehničke zahteve i poznate karakteristike materijala izvršen je proračun i analiza parametara bojeve glave i raketnog motora. Za izabrane projektne parametre raketnog projektila izvršena je simulacija rada raketnog motora i kretanja rakete u lansirnoj cevi za tri karakteristične temperature primene: -30ºC, +20ºC i +50ºC. Dobijene vrednosti pritiska u komori raketnog motora, vremena rada motora i početne brzine rakete analizirane su sa aspekta bezbednosti strelca i ostvarivanja zadatih balističkih karakteristika.

Ključne reči: **raketni bacač, ručni bacač, nevođena raketa, termobarična bojna glava, raketni motor, impulsni motor, projektovanje, optimizacija parametara, numerička simulacija.**

Проектирование систем с термобарической боеголовкой для ракетной установки

В данной статье представлена методика определения массы и геометрических параметров ракеты с термобарической боеголовкой для ракетной пусковой установки. Оптимизация проектных параметров ракет для заданных тактическо-технических требований была проведена в соответствии с критериями массовой эффективности, используя два метода расчёта. Значение калибра ракеты по первому методу определяется максимально заданной дальностью полёта ракеты в зависимости от массы полезной нагрузки. Второй метод предполагает, что значение калибра известно, однако, на основании того и делает расчёт массы всех субузлов, или массы всей ракеты. Оптимальные значения калибра для ракеты с ограниченным временем работы ракетного двигателя являются общим уникальным решением для обеих упомянутых итерационных процедур. В конкретном случае показано применение указанной выше модели – для заданных тактическо-технических требований, а также и для известных характеристик материала был проведён расчёт и анализ параметров боеголовки и ракетного двигателя. Для выбранных конструктивных параметров ракетного снаряда проведено моделирование работы ракетного двигателя и движения ракеты в пусковой трубе в течение трёх характерных температур -30°C, +20°C до +50°C. Полученные значения давления в камере ракетного двигателя, времени работы двигателя и начальной скорости ракеты были проанализированы с точки зрения безопасности стрелка и достижения заданных баллистических характеристик.

Ключевые слова: **реактивная установка, ракетная пусковая установка, неуправляемая ракета, термобарическая боевая головка, ракетный двигатель, импульсный двигатель, проектирование, оптимизация параметров, численное моделирование.**

Conception d'ogive thermobarique pour le lance-grenades à main

Ce papier présente la méthodologie du calcul des paramètres de masse et géométriques de fusée à ogive thermobarique pour le lance grenades à main. L'optimisation des paramètres de conception de fusée pour les exigences tactiques et techniques données a été faite selon les critères de l'efficacité de masse à l'aide des deux méthodes de calcul. Selon la première méthode la valeur du calibre de fusée se détermine à la base de portée maximale donnée en fonction de la masse du poids utile. La seconde méthode suppose la valeur connue du calibre et en partant de cela se calcule la masse de tous les sous ensembles c'est-à-dire la masse totale de fusée. Les valeurs optimales du calibre pour la fusée à temps limité de travail de moteur sont la solution unique pour les deux procédures itératives citées. A partir de l'exemple concret on a présenté l'application du modèle exposé – pour les exigences tactiques et techniques données des matériaux on a fait le calcul et l'analyse des paramètres de l'ogive et du moteur de fusée. Pour les paramètres choisis de conception du projectile on a effectué la simulation du travail de moteur de fusée et du mouvement du projectile dans le tube de lancement pour trois typiques températures d'usage: -30ºC, +20ºC et +50ºC . Les valeurs obtenues pour la pression dans la chambre du moteur à fusée , le temps de travail de moteur et la vitesse initiale de fusée ont été analysés sous l'aspect de sécurité du tireur et la réalisation des caractéristiques balistiques exigées.

Mots clés: **lance-roquettes, lance grenades à main, fusée non guidée, ogive thermobarique, moteur à fusée, moteur à impulsion, conception, optimisation des paramètres, simulation numérique.**