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Refurbishing Project of Rocket Motor for a Rocket Artillery Projectile

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In this paper, an optimization method for a rocket-refurbishing project is presented. While considering the technical solution the level of reconstruction needs to be decided, from which refurbishment costs are dependent, as well as eventual increase of rocket performances. The idea of rocket motor refurbishment is to extend the life cycle and eventually increase performances for a minimum of an investment. The design shown in this paper is supposed to be an economic solution for the M63 "PLAMEN" 128 mm caliber artillery rocket, resulting in the increase in performance, without significant changes to the construction.

Key words: **solid propellant, rocket motor, thrust, internal-ballistic calculation, external-ballistic calculation, modernization, refurbishment, etc.**

Introduction

THE idea of replacing a propellant of a rocket motor in the refurbishing process, with eventual increasing performance of a rocket system, falls into some of the most interesting operations in the field of rocket propulsion in present times.

The idea of refurbishing expensive parts of rocket motors is interesting from an economic standpoint. In recent history, the United States are considering valuation methodologies for rocket motors from excess intercontinental ballistic missiles (ICBMs), as to use them to launch government payloads [1].

Solid rocket propellants are the most used for minor tactical weapon systems, because of their reliability, cost effectiveness, ease of use and storage, as well as the simple production technologies that are used for their manufacture [2], [3].

Today's trends are overhauling of existing outdated projectiles with expired service life, and their modernization by implementation of new intelligent subsystems [4]. A good example can be the overhaul of the 122 mm rocket projectile "Grad", conducted by few worldwide manufacturers [5]. With implementation of the overhaul package called the "Electro-Optical Precision Integration Kit" [6] any standard soviet 122 mm "Grad" artillery rocket can be turned into a precision guided projectile, capable of hitting stationary or moving targets.

One of the problems during any refurbishing process is to choose an optimal level of rocket motor redesign, having in mind the whole line of usual problems throughout the exploitation of a rocket system. For the choice of an optimal variant of the redesign process, different criteria can be established. Usually there are several possibilities as refurbishing concepts vary, from an approach of maximization of performance of the new projectile, to the approach of minimizing operation costs and extensiveness of mechanical revamps [4].

The approach of performance maximization is based on a complete redesign of rocket motors, which implies application of the newest technological solutions, materials and propellants with maximum energy [7]. Oppositely, the approach of minimizing operation costs and extensiveness of mechanical revamps leads to solutions with lesser performances. On the other hand, with minimization we get overhauled ordnance with full resource and minimal cost. Eventually, this solution can also yield some increase in performance.

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Nomenclature

A_b	Burning area
A_{be}	Burning area at the end of the process
A_t	Critical nozzle area
C_d	Discharge coefficient
d_t	Throat diameter
d_{t0}	Throat diameter, starting value
D	Diameter
I_{tot}	Total impulse
K_i	Pobedonoscov coefficient
L	Length
M	Molar mass

$\dot{m}$	Mass flow
p_0	Chamber pressure at the start of the process
r	Burning rate
r_{30}	Burning rate on 30°C
r_{50}	Burning rate on 50°C
T_0	Combustion temperature
V_0	Free combustion chamber volume
X_c	Resulting range
δ	Combustion chamber wall thickness

Analysis of solutions

The 128 mm caliber rocket designated - M63 "PLAMEN" is a gyro-stabilized rocket artillery projectile; it is an old design and a mass-produced piece of ordnance. During its production, old technologies were used which resulted in low production costs, average material quality, but high construction reliability. The result of this concept is a cheap, reliable rocket with less than average performance. The construction of the rocket and the basic components of the rocket motor are displayed in Figure 1.

To choose the level of redesign of the rocket motor during the overhaul, a performance analysis was conducted for several variants. Those variants are design models, each with a different level of performance. Every higher level of performance leads to an increase in complexity and cost of overhaul operations. By increasing propellant mass, using a propellant with a higher specific impulse, as well as by replacing rocket motor parts with ones made of a modern lightweight material, an increase of performance can be achieved [8].

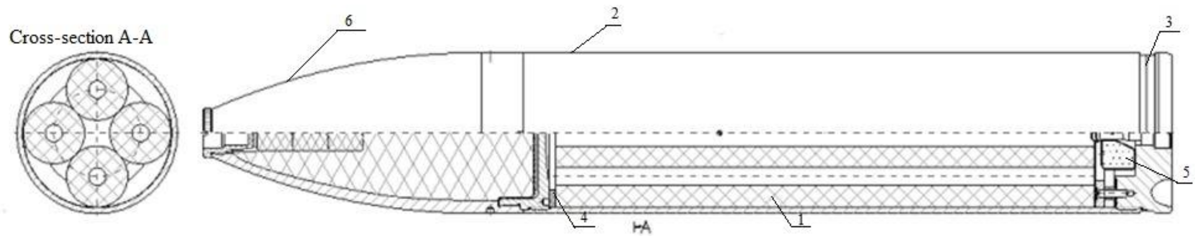

Figure 1. Cross-section of the 128 mm M63 PLAMEN

1 - Double base propellant, free standing grains, four uninhibited tubes. 2- Case with thick cylindrical wall, made from steel without thermal insulation. 3 - Multi-nozzle block - steel aft closure with eight nozzles. 4 - Forward closure from steel. 5 - Igniter. 6 - Warhead with fuse.

Considering these criteria, the next variant models, are considered:

1. Variant of minimal overhaul, achieved by replacing the propellant with a new one with higher mass and energy. The new propellant would be made with a better usage of combustion chamber space, with inhibited or case bounded propellant grain, to achieve the motor walls thermal isolation. A double based propellant has a lower price per unit of mass than a composite propellant, so this variant would be the most economic. It is also possible to use composite propellants with a lower specific impulse, to avoid temperature overload for the motor elements;
2. Variant of medium level overhaul, achieved by using a composite propellant with a high specific impulse, as well as by decreasing the mass of all rocket motor elements, where it is feasible. Some motor elements must be modified with additional thermal insulation padding, i.e. the nozzle block, parts of the combustion chamber and front aft. An increase in performance is expected because of the higher specific impulse and propellant mass. The price of the overhaul is increased because of additional work and materials;
3. Variant of total overhaul, achieved by using a composite propellant with a high specific impulse, and with a decreased mass for all rocket motor elements, where it is feasible. Like in the previous variant, metal elements must be thermally insulated, and the heaviest motor elements should be redesigned to decrease mass. For example, combustion chamber, nozzle block, aft closure, etc. Performance of this model would be significantly increased compared to previous variants, but the price would be increased as well, drastically.

The first variant was chosen for an experimental realization of our pilot project, as well as for verification of the optimization methodology, and for appraisal of manufacturing costs.

When we have opted for the overhaul variant, the first step is to decide on the type of propellant. In the analysis, the composite rocket propellant and the double base propellant are compared, while following arguments are taken into consideration:

1. Composite rocket propellants can have both a very high specific impulse and combustion temperature, but also lower values, close to the ones of double base propellants.
2. Composite propellants are more expensive, compared to double base propellants, with an about 30% higher price.
3. Composite propellants have some disadvantages regarding combustion product properties, for example corrosivity towards metal parts of the launcher, and high smoke production, with the former resulting in a shortened launcher lifespan, and the latter in a de-masking effect as well as obstructing the optical guidance system [9].

Considering that the propellant mass of a several kilograms is relatively small, the difference in price of the propellants would be a few tens of euros. Besides, another contribution to the decision to realize the first variant with a composite propellant charge, are the production capacities which are at disposal. Finally, in a modern tactical approach, the negative aspects of composite propellant can be acceptable having in mind their higher performance.

Maximization of the rocket motor performance

The maximization of motor performance in the chosen variant, which should provide the best rocket range, imposes the following propellant grain requirements:

1. A maximum propellant mass, which can be achieved with the highest propellant density and maximum grain volume, regarding the combustion chamber volume.

2. Maximum acceptable propellant specific impulse, regarding the thermal loads.

3. Optimal thrust-time curve, for achieving the most suitable external ballistics and highest range.

To summarize, the motor's total impulse is one of the key parameters which influence rocket projectile performance. The maximum total impulse value can be achieved by increasing the composite propellants mass and specific impulse.

The propellant mass that can be fitted inside the combustion chamber depends on the chosen propellant grain configuration and combustion chamber dimensions [10]. The choice fell for a composite propellant, which has a higher density than a two-base propellant. The propellant will most likely have higher combustion product temperature, so the most suited choice is a propellant configuration with outer inhibition, so thermal protection of the walls of the combustion chamber would be assured. From an aspect of specific impulse levels, the ideal would be a near neutral combustion, that is, a constant value of chamber pressure during the rocket motor's effective operating time [11]. Propellant grain configurations that offer high volume usage, which can be applied in this case, are rod, tube or slotted tube. Despite having a smaller volume usage, two-dimensional configurations are also acceptable [10]. The production process would be simplified and supporting elements i.e. holders, thermal insulators, grates, etc. could be avoided. Also, it is important to bear in mind that with the increase of grain volume of a propellant's grains, it becomes susceptible to the occurrence of erosive burning. Based on previous experiences, the limit of controllable erosive burning is approximately at a Pobedonoscev coefficient level of around Ki=120 [12].

The composite propellant's specific impulse depends on the level of heat potential and is in correlation with the combustion processes temperature. The negative side of increasing the combustion process' temperature is an increased thermal load on the motor's elements. Because of this, it is necessary to optimize the specific impulse's level, to preserve reliability and functionality of the rocket motor.

The design process of a new propellant grain for the rocket motor consists of modeling in several scientific technical fields: thermo-chemical calculations, internal ballistics, geometry analysis, fluid dynamics, heat transfer and rocket dynamics. With the use of these complex design processes, three variants are analyzed for the chosen propellant grain geometry, with different composite propellants: CP1, CP2 and CP3, the results are shown in Table 1. For comparison purposes the results of thermo-chemical calculations [13] for a two-base propellant – DB, are also shown.

Table 1. Parameters of analyzed propellants: double based (DB) and composite (CP).

Parameters	**Units**	**DB**	**CP1**	**CP2**	**CP3**
Propellant mass	kg	6.17	6.29	6.36	6.55
Combustion temperature	K	2422	2377	2800	3108
Specific impulse	Ns/kg	2073	2124	2190	2272
Average thrust	N	10660	11130	11600	16530
Total impulse	Ns	12793	13357	13930	14883
Range	m	10186	10413	10696	11091

According to the external ballistic calculation, based on Euler equations [14], the variant CP3 has the best performance. With a total impulse of 14883 Ns, it yields the greatest gain in range for the rocket. The range of the rocket increased to 11091 meters, which is about 2600 meters more than the original rocket. Based on the internal ballistic, gas-dynamic and heat transfer calculations, a heat overload of the nozzle block is expected [15]. The variant CP1 is thermally the least demanding, regarding combustion temperature (Table 1), and it is calculated that it still leads to a significant increase of performance. Because of the chosen refurbishing variant, a composite propellant with a low specific impulse, along the lines of 2100 Ns/kg, must be used. During the rest of the assignment, the composite propellant CP2 will be considered as the optimal solution.

The optimal pressure-time curve is obtained by analyzing the effect on range of several different shapes, using external ballistic calculations [14]. In the active flight phase of the rocket, the velocity changes in accordance with the thrust curve and aerodynamic drag characteristic. Depending on the Mach number, the drag changes significantly, the value is determined by semi-empirical equations [16]. During the optimization process it was predicted that the maximum range of the rocket could be achieved with a suitable thrust curve.

Keeping in mind the following constraints and estimates, the optimization process was carried out:

- The effective operating time of the rocket motor (effective time) t_e is in the range of 0.5 to 2 s, limited by drastic increase of thermal loads, which happen more in long-running rocket motors;
- The optimal thrust curve is neutral, taking into consideration the motor casing's structural strength and internal ballistic performance.
- The tolerance for the thrust curve neutrality is average, in order of magnitude of 30%.
- A higher level of thrust is desirable at motor ignition, to increase muzzle velocity of the rocket.

The optimization process was conducted through varying of two major functions that have the highest influence on range: effective time and thrust shape:

- First, the operating time of the rocket motor is being varied for the neutral thrust curve, for a total impulse of 14000 Ns and an elevation angle of 45º.
- Second, all three thrust curves (neutral, progressive and degressive) are being tested for the determined operating time.

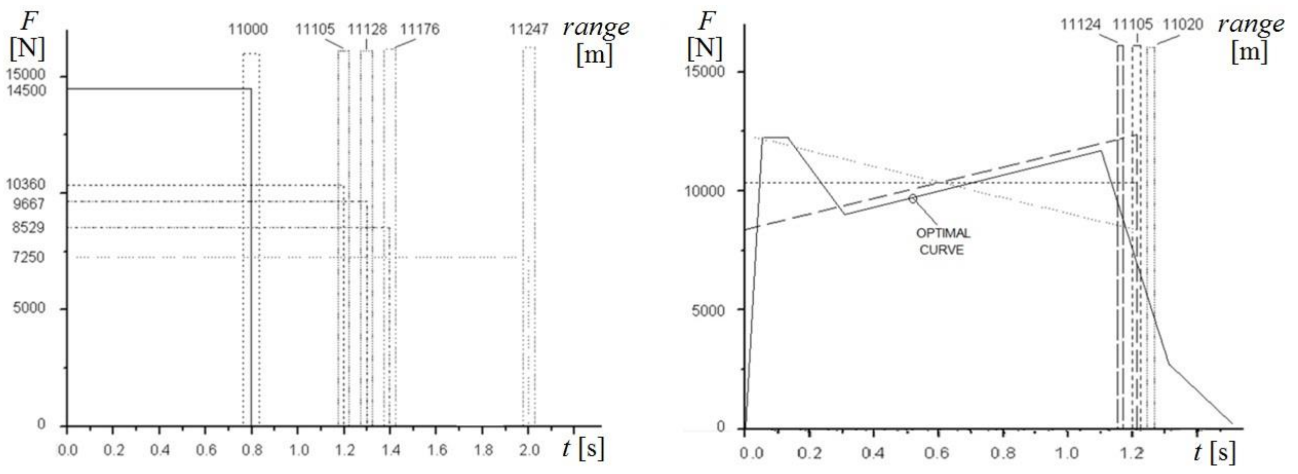

Figure 2. Influence of thrust-time curve on the rocket range

By analyzing the left diagram in Figure 2, it is concluded that with the increase of effective time for 0.1 seconds on the interval of 1 to 2 seconds, the range is increased at an average of 0.215%. Prolongation of the effective time has relatively small contribution to the range, and it also has negative implications, from an aspect of thermal loads. A compromised operating time of 1.2 seconds is chosen, and for this time span a second optimization step is conducted (Figure 2, right).

The best increase in range is achieved with a progressive curve, a smaller one with a neutral, and the least increase in range with a degressive curve. The progressive thrust curve with an increasing level of around 40% leads to an increase in range of 0.17%, in comparison to the neutral curve. A digressive curve with the same negative gradient gives a smaller range, for about 0.765%, in comparison to the neutral curve.

The introduction of a higher starting peak for thrust should generally be taken into consideration, in order to achieve higher muzzle velocity. The optimal realistic thrust – time curve (Figure 2) can be achieved by following these guidelines:

- The requested full thrust should be achieved at 0.07 seconds from ignition;
- A short time starting peak is desirable;
- After the starting peak the thrust curve should have a progressive shape.

Internal ballistic design

Design requirements

Having in mind the defined task and the existing rocket motor construction, the design demands can be drawn from the following characteristics: chamber pressure, burning area, burning rate, etc.

Originally the rocket motor and combustion chamber were designed for an uninhibited propellant without thermo-isolation. The propellant originally used has a low combustion temperature, short combustion time, which is under one second. The concept is simple, with a low construction cost, from a time when inhibitor technologies and materials for inhibitors and thermo-isolation were still in development. The integrity of the combustion chamber is achieved with a steel wall which is 4.5 mm thick. It is designed to absorb the generated amount of heat during the work cycle, as well as simultaneously withstand mechanical loads in accordance with the predefined safety factor [7]. The case is designed to carry the sum of the chamber pressure and equivalent of inertial and thermal loads. In this variant of overhaul, an inhibited propellant grain is used to provide thermal-insulation so an optimal chamber pressure, along the lines of values of up to 150 bar can be achieved, and the combustion chamber can be used to its fullest potential [19]. With the chamber pressure predetermined, the main internal ballistic characteristics can be calculated, such as mass flow rate, burning area and burning rate of the propellant.

To circumvent additional work, the use of the existing nozzle block without modification is predicted. Equivalent nozzle critical area - At can be calculated from the nozzle's geometry. Combustion products mass flow through the nozzles is conditioned by their gas-dynamic characteristics, which can be expressed by the theoretical discharge coefficient (1):

$$C_d = \sqrt{\frac{\kappa}{R \cdot T_0} \cdot \left(\frac{2}{\kappa+1}\right)^{\frac{\kappa+1}{\kappa-1}}} = 6.84 \cdot 10^{-4} \quad (1)$$

where the next combustion product parameters, are determined by thermo-chemical calculations [13]:

specific heat ratio: κ = 1.215;

molar mass: M = 25 kg/kmol;

combustion temperature: T0= 2569 K.

Using the given values, the required mass flow rate of combustion products can be calculated as (2):

$$\dot{m} = A_t \cdot C_d \cdot p_0 = 7.66 \ \frac{\text{kg}}{\text{s}} \quad (2)$$

The required mass flow rate can be achieved by adjusting two parameters: the propellant's burning area of the grain -Ab and burning rate -r. To achieve an approximately neutral working regime for the motor, the initial burning area should be close to the value of inhibited surface of the grain. The initial burning area is approximately equal to the area at the end of combustion process -Abe, which can be calculated from the grain geometry, where the diameter of the propellant is - D and its length– L (3):

$$A_{be} = D \cdot \pi \cdot L = 0.155 \ m^2 \quad (3)$$

The required burning rate in that case, can be calculated from the relation of burned mass, using the propellant's density value – ρ (4):

$$r = \frac{\dot{m}}{A_b \cdot \rho} \approx 27 \ \frac{\text{mm}}{\text{s}} \ \ldots\ldots \ (4)$$

Based on all previous requirements and taking into account the adjustments to the propellant, a burning rate - r of 27 mm/s is achieved, at a chamber pressure p0 of 150 bar. The burning rate law is determined experimentally for a wide range of combustion chamber pressure levels, for extreme operating temperatures of service conditions. The results are given as functions in Figure 3.

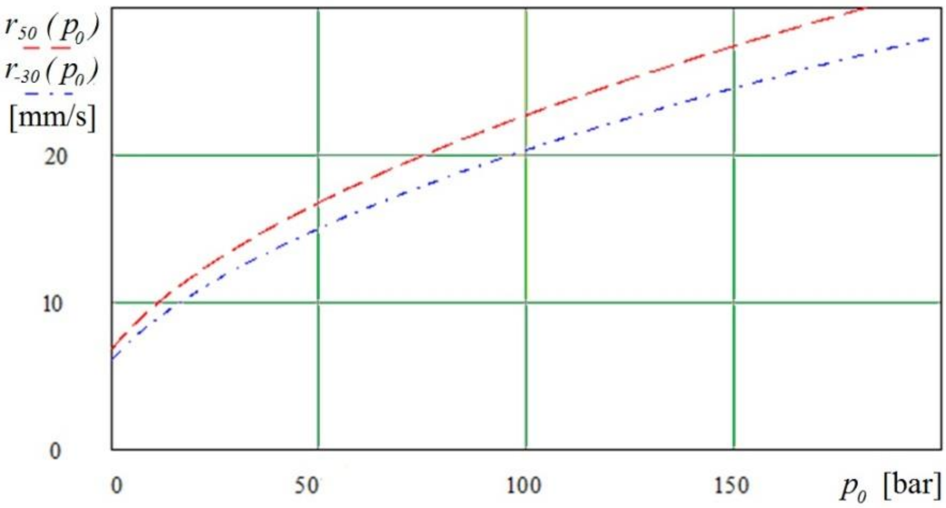

Figure 3. Experimentally determined burning propellant rate law on temperature range +50°C to -30°C

Propellant grain geometry

The propellant grain was designed based on the requirements for maximum propellant mass, that can be placed within the available volume of the combustion chamber. This characteristic can be defined by the volumetric loading fraction, which is calculated by dividing the grain volume with the available chamber volume. Based on all other proclaimed characteristics, the next properties of the propellant grain's geometry, were defined:

1. An outer inhibited surface or a case bonded propellant;
2. Maximized propellant mass, which can be expressed as the maximum value of the volumetric loading fraction;
3. Approximately neutral combustion.

The outer diameter of the propellant grain is defined by the existing case caliber and wall thickness, also taking into account inhibitor/liner thickness (adopted from semi empirical calculations).

Several different propellant grain geometry configurations can be used to fulfill stated design demands. Among them, the star configuration with a corresponding web thickness - w and high-volume ratio was chosen (Figure 4.) [17], [18].

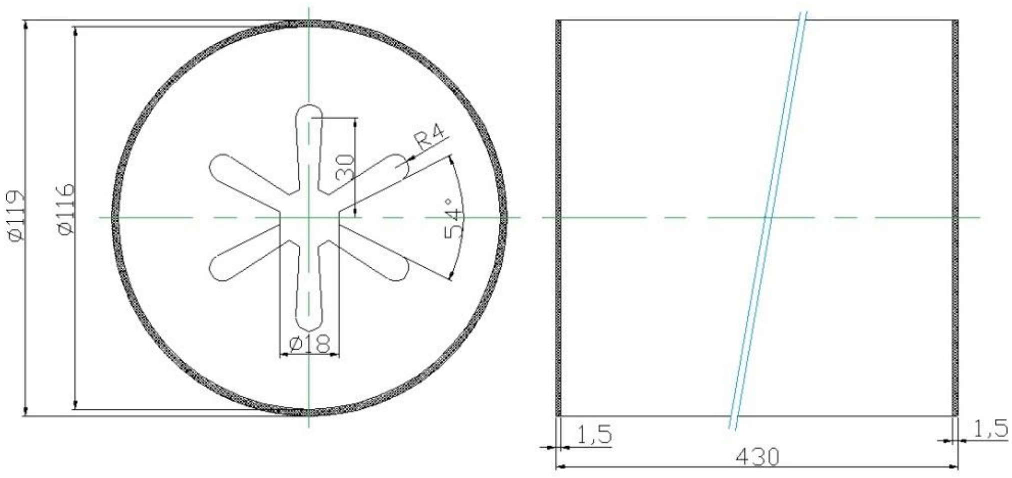

Figure 4. Geometry of the propellant grain

With the chosen geometry, the following propellant characteristics are gained:

–A mass of 6.655 kg;

–A volumetric loading fraction value of 87,93 %;

–A sliver size of 10.55 %.

Another necessary input function for internal ballistic calculations is the function of burning area regression. It represents the change of combustion surface area of the grain versus the burned web thickness. The calculation was conducted using the numerical program SVOD [10], and the resulting curve is shown in Figure 5.

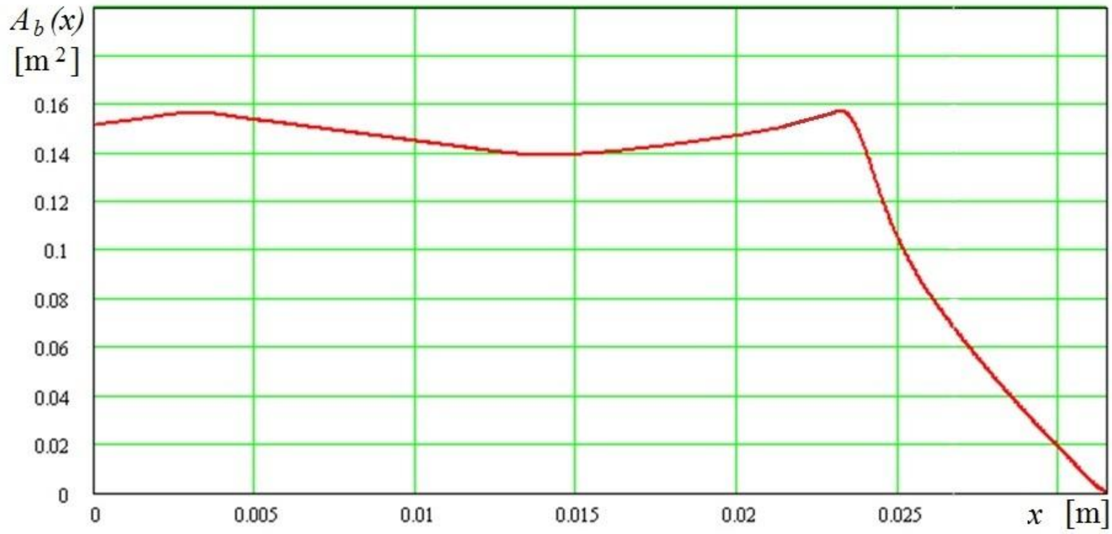

Figure 5. Burning area dependence from burned web, numerically calculated by SVOD software.

The chosen grain geometry mostly satisfies the preset requirements. A nearly neutral combustion is achieved with a deviation in the value of 10%. The radius at the root of the star is optimized to decrease the concentration of stress and to lower the size of the remaining sliver. On the other hand, a very high-volume ration is achieved by using an reverse valley angle of 54º, which is smaller than the angular fraction of the star which is 60º (360º/6) (Figure 4).

Internal ballistic calculation

The design of a new propellant grain for an existing construction of a rocket motor can be considered as a reverse internal ballistic calculation. The results of these calculations are pressure – time and finally thrust – time curves. Those resulting curves should be as similar as possible to the requested optimal curve (Figure 2).

The internal ballistic calculation is based on the mass, momentum and energy conservation equations, etc. One of the methods of conducting this calculation is described in literature [10], by solving the system of differential and algebraic internal ballistic equations, without mathematical simplifications. In addition to the characteristics mentioned above, it is also required to geometrically calculate the combustion chambers free volume: V0 = 7×10-4m3.

Using the Runge-Kutta numerical calculation [10] pressure - time functions for two extreme operating temperatures are obtained, and they are shown in Figure 6.

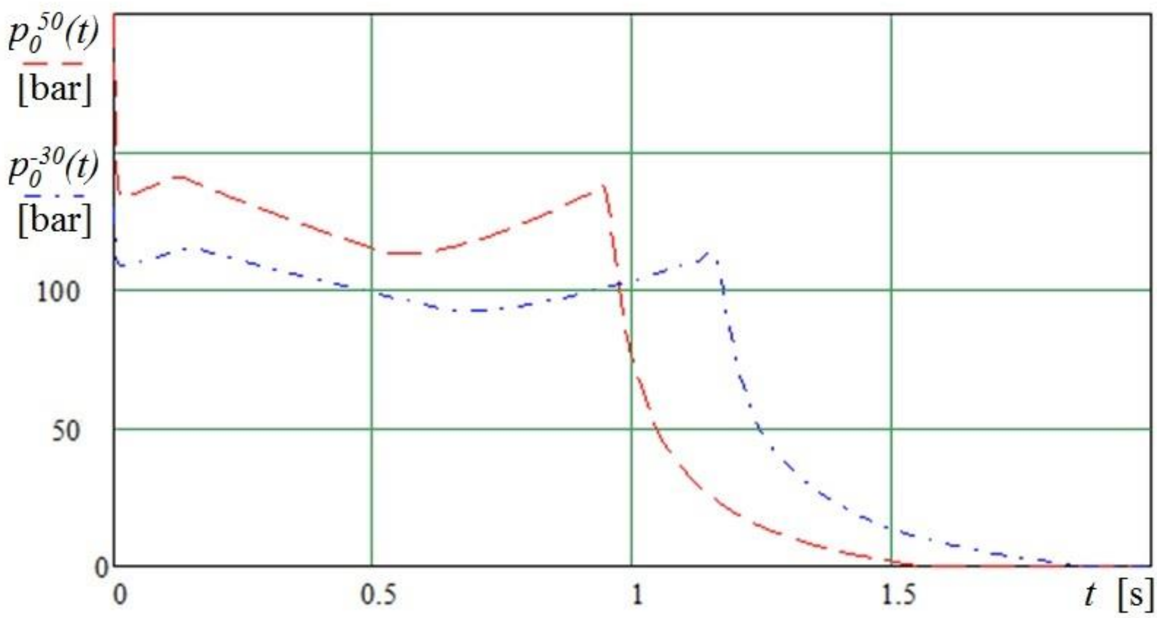

Figure 6. Calculated rocket motor chamber pressure curve during work time on extreme temperatures -30ºC and 50ºC

Considering prior development and static tests of earlier versions of the rocket motor, some infrequent processes in solid rocket propulsion can be expected, which are caused by certain design specifics:

1.Influence of the rocket rotation on the burning rate [7],
2.Intense nozzle throat erosion [10].

The effect of the burning rate increasing is triggered by centrifugal force, generated under high-speed rotation, which is used for the gyroscopic stabilization of the rocket. This influence is considered, based on research [7], by the semi empiric formulas 5 and 6:

$$r_{50}(p_0) = \begin{cases} r_{50}(p_0); & if\ t < 0.3s \\ r_{50}(p_0)\left(1 + 0.3\frac{t-0.3}{t_e}\right); & if\ t \geq 0.3s \end{cases} \quad (5)$$

$$r_{-30}(p_0) = \begin{cases} r_{-30}(p_0); & if\ t < 0.32s \\ r_{-30}(p_0)\left(1 + 0.4\frac{t-0.32}{t_e}\right); & if\ t \geq 0.32s \end{cases} \quad (6)$$

The nozzle throats erosion is simulated by introducing a time function for the diameter increase (formula 7), it is assumed the function is linear during the entire operating time of the rocket motor, starting from the diameter value of d_{t0}:

$$d_t(t) = d_{t0} + 0.0025\frac{t}{t_e} \quad (7)$$

The phenomena of nozzle erosion, consequently, produces the difference in the shape of the measured thrust and pressure curves. The average increase of the nozzle effective throat area is 34%, the total increase of the effective throat diameter is statistically obtained from experiments.

With these two effects considered, an improved internal ballistic calculation gives much more realistic results of pressure and thrust - time curves (Figure 7).

Based on obtained pressure-time curves it can be concluded that the chosen propellant configuration complies to the initial demands: nearly neutral working regime and about 1.2 s effective work time.

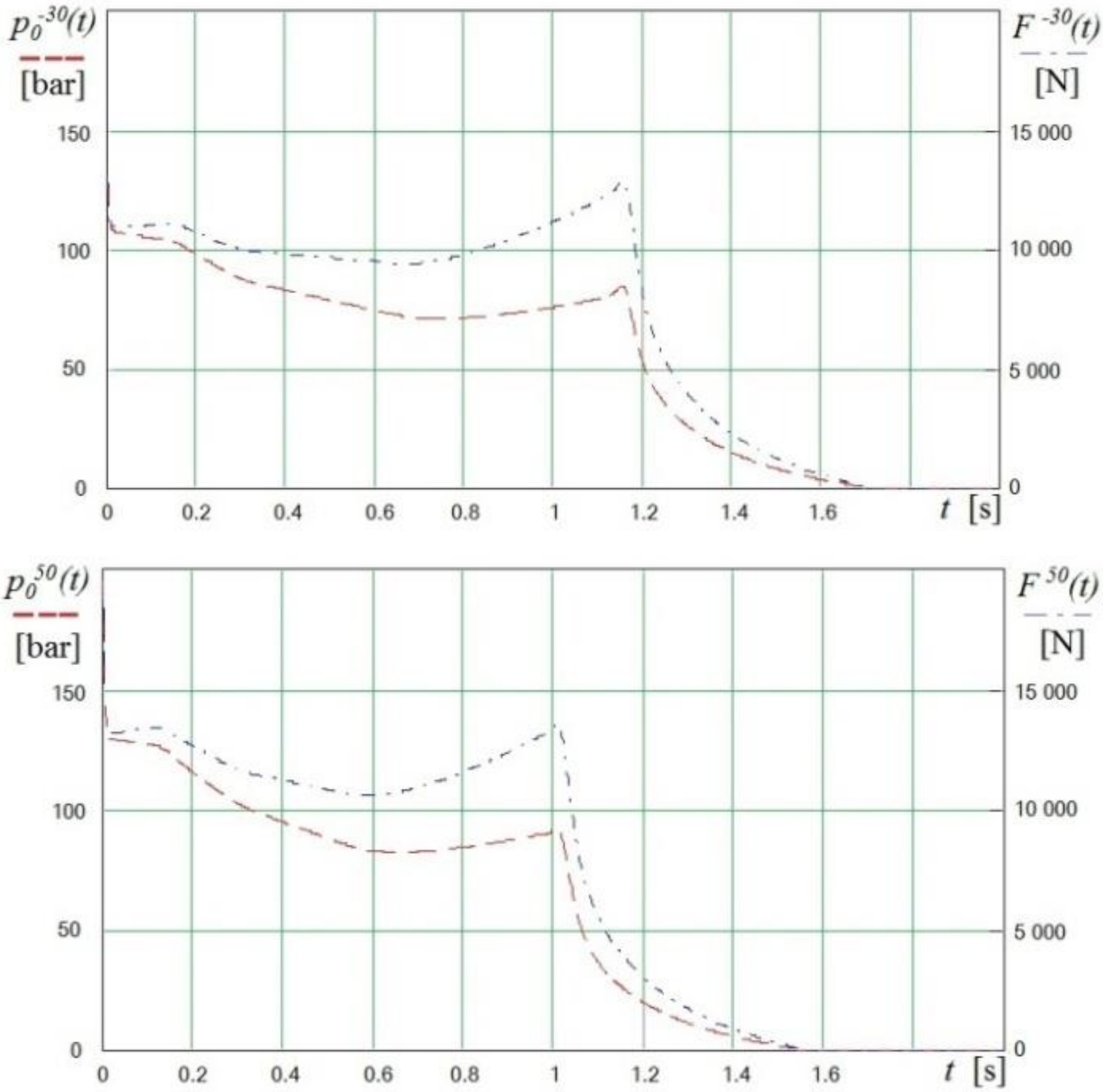

Figure 7. Calculated chamber pressure and thrust curves during the working time, on extreme ambient temperatures -30°C (above) and 50°C (below)

Static testing

The final stage of this optimized redesign project is building a prototype and experimentally validating the designed product. The prototype series of rocket motors is produced with the described construction and technology, after which they are prepared for static tests on extreme temperatures of 50°C or 323.15K and -30°C or 243,15K by conditioning in environmental chambers.

The tests were conducted on a special rotational test stand, which allows for free rotation of the rocket motor. The test stand head which holds the rocket motor, rotates together with it. The head has an axial moment of inertia similar to one of a warhead, in order to simulate rotation in flying conditions. In these static tests the chamber pressure and thrust were measured. A series of tests was conducted, and typical results are shown in Figure 8.

Comparing test results with calculated diagrams (Figure 7), a satisfying agreement of pressure and thrust curves can be seen. This agreement is proof that the applied internal ballistics model is accurate enough to predict the rocket motor's working regime, even with some complex and unusual phenomena.

For all tests the original igniters were used regularly, so it is confirmed that an igniter overhaul can be avoided in this research phase. Eventual igniter replacement is not too demanding, but would require additional activities, testing and production costs.

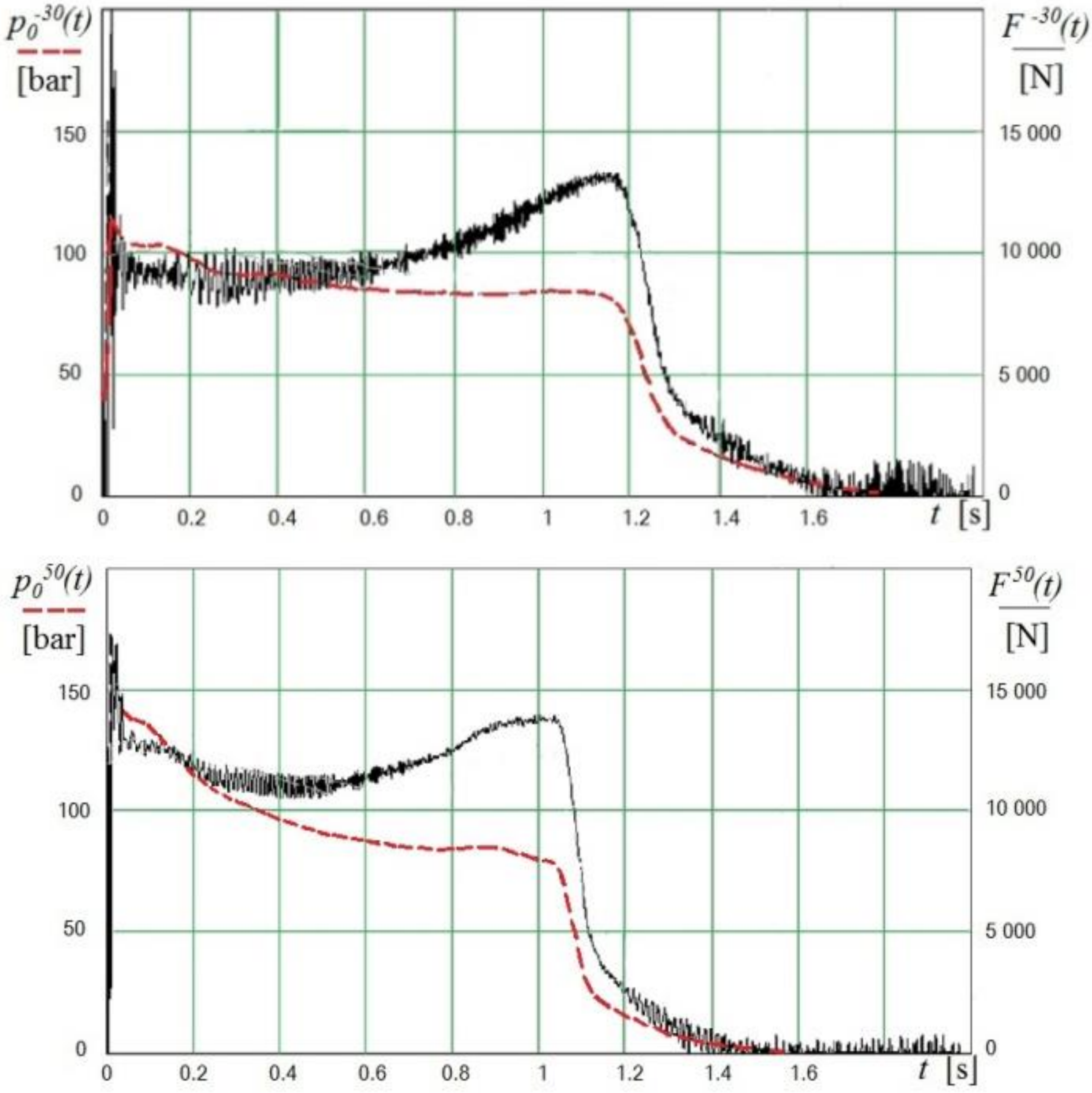

Figure 8. Results of static rocket motor tests,pressure and thrust - time curves on -30^{0}C (above) and 50^{0}C (below)

As thermal stress is one of the major issues, those effects are analyzed in the thermally most affected zones, such as the nozzle block. In Figure 9, significant material structural change due to thermal stress can be seen, as well as a significant increase of nozzle throat diameter, as mentioned before. It is concluded that nozzle erosion is more intense at tests performed at 50°C, than at -30°C. The erosion of the nozzle can increase thrust vector unalignment, which leads to a decrease in precision.

On the unprotected part of the motor case, between the propellant rear holder and the nozzle block, effects of significant thermal stress can be seen (Figure 9). A case of deformation and diameter change was not registered, which means that the stress on that area is not critical.

Measured total impulse and predicted rocket ranges are:

• Testing on 50° C: Itot = 14040 Ns, which corresponds to the range Xc = 10750 m;

• Testing on -30° C: Itot = 13730 Ns, which correspondsto the range Xc = 10600 m.

Figure 9. Nozzle block before and after static test (left), and combustion chamber thermal stress effects (right).

These values were expected and correspond to the calculation results. The resulting total impulse yields an increase in range for over 2000 m, in comparison to the original rocket, which can be considered acceptable.

Conclusion

The optimization method presented in this paper is comprised of advanced internal ballistic calculation methods, external ballistic calculations and some productive-economical considerations, which can be successfully used to accurately predict a new project solution for a rocket motor overhaul or modernization process.

In this paper, a cheaper modernization project variant was presented for the 128 mm “PLAMEN” M63 artillery rocket projectile, during which, in accordance with our resources and technological capacities, it was decided to opt for the first overhaul variant, that is the variant of minimal overhaul. A minimal number of technological operations was conducted on the rocket motor, as it was not necessary to conduct a reconstruction of the rocket motor’s elements. A change was made in the propellant, as the double-base one was swapped out for a new composite propellant, which offered a higher specific impulse along the lines of 2190 Ns/Kg.

During the design process for the new propellant, the concept deviated from the initial propellant as the old propellant was comprised of four cylinders, which laid inside the combustion chamber of the rocket motor. A new cylindrical inhibited propellant charge was made, which resulted in a better usage of the combustion chamber volume, at a value of 87,93%.

With this solution, the effects of erosive burning and nozzle erosion are within acceptable values, so they do not influence the internal ballistic process and function of the motor. The new propellant has a star shaped canal, which allows for an almost neutral combustion and an effective work time of 1,2 seconds, which is in accordance with our initial demands. The thermal stress of the rocket case is limited to an area above the nozzle block, but considering that there were no cases of deformation and diameter change, it means that the stress in that area is not critical. During the testing it was also proven that the new propellant can be reliably ignited with the original ignition, which can be used in initial research/modernization phases.

Based on the static tests conducted, after treating the motors in environmental chambers at 50° C and -30° C, the following total impulses were gained 14040 Ns at 50° C and 13730 Ns at -30° C. With the total impulse value and Euler ballistic model, the ranges of the rocket were calculated. In the first case the range matched the one of 10750 meters, while in the second case the range matched the one of 10600 meters. The end result is an increase in range of around 2000 meters in comparison to the base value, that is an increase in range of about 20%.

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Projekat remonta raketnog motora raketnog artiljerijskog projektila

U ovom naučnom radu prikazana je metoda optimizacije projekta remonta rakete. Uzimajući u obzir tehnološka rešenja, neophodno je odlučiti se za nivo rekonstrukcije od kog zavise dalji troškovi projekta remonta, kao i eventualna povećanja performansi. Ideja remonta raketnog motora je produžetak životnog ciklusa i eventualno poboljšanje performansi sa minimalnim ulaganjima. Projekat prikazan u radu treba da predstavi ekonomično rešenje za raketni motor rakete M63 „PLAMEN“ u kalibru 128 mm, koji rezultira povećanjem performansi, bez znatnih promena na samoj konstrukciji rakete.

Ključne reči: **čvrsto gorivo, raketni motor, potisak, unutrašnji balistički proračun, spoljno balistički proračun, modernizacija, remont, itd.**