



Prediction of Internal Ballistic Parameters of Solid Propellant Rocket Motors*

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Abstract. A modular computer program named SPPMEF has been developed which is intended for purposes of predicting internal ballistic performances of solid propellant rocket motors. The program consists of the following modules: TCPSP (Calculation of thermo-chemical properties of solid propellants), NOZZLE (Dimensioning of nozzle and estimation of losses in rocket motors), GEOM (This module consists of two parts: a part for dimensioning of the propellant grain and a part for regression of burning surface) and ROCKET (This module provides prediction of an average delivered performance, as well as mass flow, pressure, thrust and impulse as functions of burning time).

The program is verified with experimental results obtained from standard ballistic rocket test motors and experimental rocket motors. Analysis of results has shown that the established model enables high accuracy in prediction of solid propellant rocket motors features in cases where influence of combustion gases flow on burning rate is not significant.

Keywords: rocket motors, solid propellant, burning rate, internal ballistic performances prediction, losses, computer program

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1. INTRODUCTION

Initial phase of solid propellant rocket motor development is characterized by number of parametric studies undertaken in order for the rocket mission to be accomplished. During the process of assessment of possible solutions for propellant charge shape, configuration of motor and type of propellant charge, problems of production are being considered along with demands for specific motor performances and exploitations conditions. Even though these preliminary project studies are comprehensive from a practical side, it is not good practice to treat all the influencing factors parametrically. Instead, after first assessment of possible solutions, an optimal construction is chosen. It is then further subjected to detailed analysis. Using this analysis, the following is tested critically: propellant type – geometry of propellant grain – motor structure, in order to determine whether the motor will satisfy parameters necessary for solid propellant rocket motor design. One of the main objectives for designers of solid propellant rocket motors is defining of the propellant grain which will enable required change of thrust vs. time, needed for fulfillment of the rocket's mission, taking care of other specific limitations (envelope, mass, etc.).

Analysis of solid propellant rocket motors progresses in two levels, where, independent of the level, it is needed to assess following four basic steps [1, 2]:

- Assessment of several types of propellant types/configurations,
- Defining the geometry of propellant grain which satisfies conditions of internal ballistics and structural integrity,
- Approximate determination of erosive burning and potential instability of burning process,
- Determination of structural integrity of the grain during time of pressure increase during ignition.

First level or preliminary analysis of the design uses tools that have to be simple and adaptable to the user. There are usually simple computer codes, based on analytical models or diagrams that give the first simplified results.

The second level is the level of the propellant charge's final design. Tools for this task are more refined and these are handled by experts for propellant grain design. Computer codes are based on finite difference methods or finite element methods, with 1D, 2D or 3D models of physical phenomena (internal ballistics, fluid dynamics and continuum mechanics structural analysis). They allow precise calculations, or optimization for defining of final geometry.

Countries with high technological level (USA and western countries) focus their continual research on prediction of theoretical performances of solid propellant rocket motors. They base their research on development of high range ballistic guided rockets, based on composite propellant charges.

Large volumes of experimental research, conducted during the development of these rocket systems, enabled huge databases of influencing factors on dispersion of real from ideal performances of rocket motor, for every system individually.

Most of today's models for prediction of the internal ballistic performances of solid propellant rocket motors are based on one-dimensional (1D) mathematical models for solving basic equations of fluid mechanics (continuity, momentum and energy equations). One-dimensional models, which can be found in commercial programs (SPP-Solid Performance Program [3-7], SNIA-BPD, Bombrini Parodi-Delfino S.p.A., Defense and Space Division, Colleferro, Italy [8]), have the advantage of fast calculation times.

The SPP program has become the standard reference computer program throughout the United States for predicting the delivered performance of solid propellant rocket motors. The nozzle performance methodology starts with the ideal performance and addresses each of the following performance loss mechanisms: finite rate chemical kinetics, nozzle throat erosion, nozzle submergence, nozzle flow divergence, two phase flow, combustion efficiency, and the nozzle wall boundary layer. The Grain Design and Ballistics (GDB) module calculates the ideal pressure-thrust history and subsequently modifies these values based on the nozzle performance efficiencies. The SPP program is used by leading manufacturers of solid propellant rocket motors in USA and many other countries. This program enables prediction and/or analysis of performances for hundreds of different rocket motors but most of these data are unavailable for other countries [3].

Between September 1997 to 2007, the Center for Simulation of Advanced Rocket (CSAR) at the University of Illinois at Urbana-Champaign, was developing a program for the U.S. Department of Energy predicting performances of solid propellant rocket motors based on numerical simulations [9-12]. The goal of the CSAR is the detailed, whole-system simulation of solid propellant rockets from first principles under both normal and abnormal operating conditions. The design of solid propellant rockets is a sophisticated technological problem requiring expertise in diverse sub-disciplines, including the ignition and combustion of composite energetic materials; the solid mechanics of the propellant, case, insulation, and nozzle; the fluid dynamics of the interior flow and exhaust plume; the aging and damage of components and the analysis of various potential failure modes. These problems are characterized by very high energy densities, extremely diverse length and time scales, complex interfaces, and reactive, turbulent and multiphase flows. All of these modules are verified using scaled experimental rocket motors and real rocket motors. Models enabling numerical simulation for these type of problems demand high performance computers (longer calculation times).

The Defense Technology Department at the Mechanical Engineering Faculty Sarajevo also developed their own model and program, under the name SPPMEF, for prediction of internal ballistic performances of solid propellant rocket motors, which can solve problems with high accuracy. The SPPMEF program is intended for rocket motors where influence of gas flow and mass flux on burning rate is not significant, as well as for rocket motors with a central nozzle [13].

2. MODEL FOR DIMENSIONING AND PREDICTION OF INTERNAL BALLISTIC PERFORMANCES OF SOLID PROPELLANT ROCKET MOTORS

The SPPMEF program consists of a series of modules which are integrated to provide a method for predicting the average delivered performance (Fig. 1):

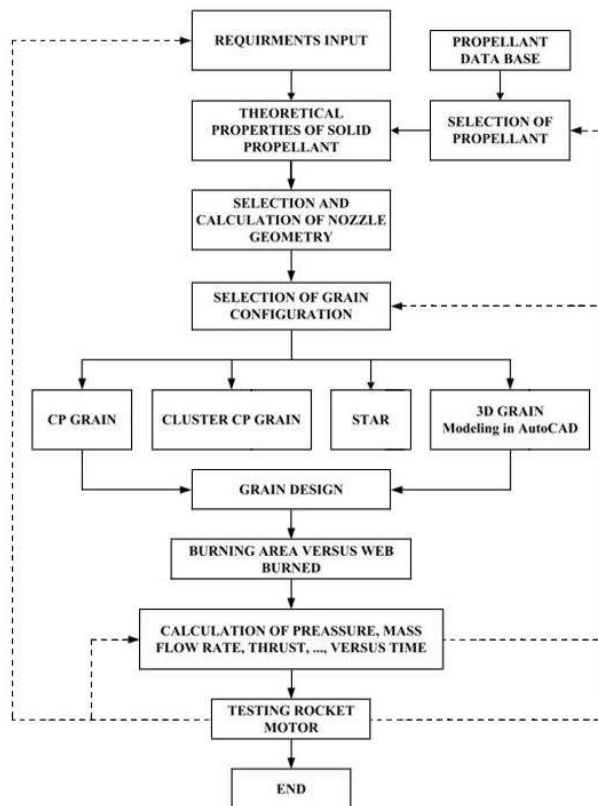


Fig. 1. SPPMEF Model

- TCPSP – calculation of the Thermo-Chemical Properties of Solid Propellants,
- NOZZLE – dimensioning of the nozzle and estimating losses in a rocket motor,
- GEOM – this module consists of two parts: a part for dimensioning the propellant grain and a part for regression of the burning surface, and
- ROCKET – this module provides prediction of average delivered performance, as well as mass flow, pressure, thrust and impulse as functions of burning time.

These modules, together with analytical or experimental expressions, are used to describe physical and chemical processes occurring in a rocket motor. Effectiveness of these models depends on assumptions and the numerical model used.

Verification of models for prediction of internal ballistic performances is only possible using experimental tests. Experimental research helps in identifying of quantities that influence dispersion of internal ballistic parameters obtained experimentally from ideal parameters.

2.1. TCPSP Module

This module enables calculation of the combustion products' composition at chemical equilibrium (Gibbs energy minimum model), transport properties of gaseous combustion products and theoretical performances of rocket motors. Calculation of theoretical performances of rocket motors is based on assumption of the Infinite-Area Combustion chamber (IAC) model. This model describes procedures for obtaining theoretical performances of rocket motors for both cases of expansion, at "frozen" equilibrium and "shifting" equilibrium conditions. Three cases are considered as follows:

- Expansion to given Mach number (condition for throat area),
- Expansion to given pressure at the nozzle exit,
- Expansion to given expansion ratio (program enables expansion calculations for 3 different expansion ratios).

The TCPSP Module enables calculation of theoretical performances of rocket motors with propellants consisting of the following chemical elements: Al, C, Ca, H, K, Mg, N, Na, O, P, S, Si, Ti, F, Fe, Cl, Pb. The database consisting of propellant ingredients based on available data published by MARTIN MARIETA [16] and STANAG 4400 [15] has been established. This program is capable of predicting properties of combustion products mixture with 156 gaseous and 39 phase-condensed ingredients. The database, which consists of propellant ingredients and combustion products, can be upgraded with new ingredients.

A very good agreement of calculated theoretical performances of rocket motors is obtained by the TCPSP module, with the Ophelie and CEA referent programs (Table 1) [14].

Table 1. Comparative analysis of some properties in the combustion chamber for solid propellant AP/CHOS-Binder/Al/MgO/H₂O (wt. %: 72.06/18.58/9/0.2/0.16)

P [MPa]	3.447			1.724		
	TCPSP	CEA ^[20]	DEVIATION	TCPSP	CEA ^[20]	DEVIATION
T [K]	2716.80	2724.46	-0.28%	2700.20	2708.02	-0.29%
C _p [J/gK]	2.4185	2.4079	0.44%	2.5407	2.5317	0.35%
γ	1.1969	1.1945	0.20%	1.1926	1.1890	0.30%
s [J/gK]	10.529	10.575	-0.44%	10.788	10.824	-0.34%
h [J/g]	-2028.30	-2028.24	0.00%	-2028.30	-2028.24	0.003%
ρ [g/m ³]	3527.0	3520.9	0.17%	1772.0	1768.1	0.22%
M (1/n)	23.112	23.136	-0.10%	23.071	23.096	-0.11%
M _w [g/mol]	22.262	22.282	-0.09%	22.225	22.246	-0.09%
a [m/s]	1080.1	1081.4	-0.12%	1075.4	1076.6	-0.11%
(dvt) _p	1.0457	1.0518	-0.58%	1.0690	1.0686	0.04%
(dvp) _t	-1.00260	-1.00263	-0.003%	-1.00350	-1.00342	0.01%

2.2. NOZZLE Module

This module enables dimensioning of the nozzle, estimating losses in a rocket motor and prediction of specific impulse delivered.

Process of dimensioning of the nozzle demands that following is known:

- Average values of thrust F_{aver} (determined in external-ballistics analysis of the missile mission).
- Combustion pressure in a rocket motor chamber (determined during the process of choosing the type of propellant).
- Theoretical values of thermo-chemical parameters of the propellant, for case of equilibrium and “frozen” state of combustion products, for adopted working combustion pressure and ratio of exit and throat area section of nozzle (from module TCPSP: Mole fraction condensed phase, Specific impulse for equilibrium and frozen expansion, Thrust coefficient), and
- Losses in a rocket motor nozzle (for assessment of losses we need to know the following parameters: material of nozzle, nozzle half angle, burning time, radial erosion rate of the throat and submergence length).

Prediction of real value of specific impulse of rocket motor is a complex task which encompasses theoretical values of specific impulse of propellant, combustion process coefficient of efficiency η_{C^*} and thrust coefficient of efficiency η_{CF} :

$$I_{sp} = I_{sp_{teo}} \cdot \eta_{C^*} \cdot \eta_{CF} \quad (1)$$

For prediction of real specific impulse, empirical formulas are used in assessment of losses, recommended from AGARD-a in Propulsion and Energetic Panel Working Group 17 or similar method, used in the SPP program [7, 18, 19]. The program currently handles the following losses: divergence (ϵ_{DIV}), Two Phase Flow (ϵ_{TP}), Boundary Layer (ϵ_{BL}), Kinetics (ϵ_{KIN}), Submergence (ϵ_{SUB}) and Throat Erosion (ϵ_{EROS}).

Thrust coefficient efficiency is

$$\eta_{CF} = 1 - 0,01 \cdot (\epsilon_{DIV} + \epsilon_{TP} + \epsilon_{BL} + \epsilon_{KIN} + \epsilon_{SUB} + \epsilon_{EROS}) \quad (2)$$

Experimental values of specific impulse are determined using the formula:

$$I_{sp_{exp}} = \frac{\int F dt}{m_p} \quad (3)$$

Results of comparative analysis of assessed specific impulse and experimentally determined specific impulse for 4 types of rocket motors (first 3 with double base propellant and last one with composite propellant type TP-H-3062 [20]) are shown in Table 2.

Table 2. Comparative analysis of assessed and experimentally determined specific impulses

Rocket motor	P _{aver} [MPa]	Nozzle				η_{C^*}	η_{CF}	Specific impulse [Ns/kg]			Diff. [%]
		Type	d _t [mm]	α [°]	ϵ			Isp ⁰	Isp _{ex}	Isp _{pred}	
RM-1	17.342	Conical	14,0	7,5	6.250	0.999	0.909	2210.3	2011.5	2006.9	-0.229
RM-2	12.437	Conical	29,4	13	14.050	0.982	0.921	2320.9	2098.5	2099.6	0.051
RM-3	12.262	Conical	29,4	11	12.867	0.974	0.928	2250.7	2026.9	2033.7	0.335
Star-8 ^[23]	9.827	Conical	22,4	15	27.1	0.955	0.916	3069.4	2677.1	2685.9	0.329

Model for prediction of performance losses of rocket motors is in very good agreement with results obtained experimentally. Maximal deviations of specific impulse reach 0.5%.

2.3. GEOM Module

The Grain dimensioning module contains three standard grain design shapes: CP Grain (cylinder with internal burning surface, cylinder with internal-external burning surfaces), Cluster CP Grain (multiple cylinders with internal-external burning surfaces), Star Grain. For 3D grain we prefer using database-obtained modeling of the grain in AutoCAD.

This module is consisted of two parts: a part for dimensioning the propellant charge and a part for regression of burning surface. For predicting the grain regression, analytical methods are used [13, 22].

Based on parameters determined in preliminary analysis, choice of general configuration of grain in this model is based on following parameters: character of thrust change, relative thickness of combustion (w_f), volumetric loading and ratio L/D of propellant grain. Determination of propellant grain dimensions for first two types of configurations depends on volumetric loading (directly related to relative web) and conditions of flow inside the gas flow channels. That is why it is possible to establish faster assessment and define geometry of these types of grain by using simple expressions and tables.

Star Grain charge gives possibility of different geometry that satisfy conditions from preliminary analysis. Procedure of optimization of star grain charge is based on assumptions defined in references [21, 22]. The OPTIM computer program [22] has been developed to ensure choice of optimal geometry of a star grain by variation of seven independent geometric variables (Fig. 2) of propellant with assumptive intervals of volumetric loading, relative rest of unburned propellant (sliver – σ) and degree of propellant's neutral burning area ($\Gamma_{\min} = S_{\max}/S_{\text{aver}}$).

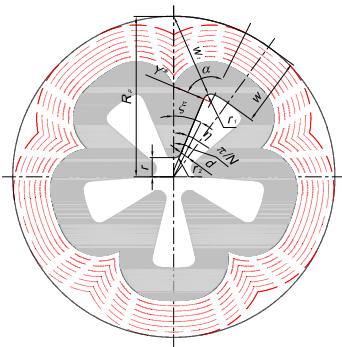


Fig. 2. Geometric definition of star grain and regression of burning surface

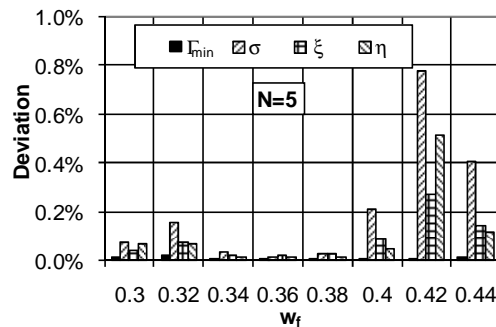


Fig. 3. Comparative analyses of results from OPTIM computer code with reference [23] for optimization of star grain with 5 sides ($V_1 = 0.85$ and $r_1/R_p = r_2/R_p = 0.05$)

Comparative analyses of results from OPTIM computer code with referring SPP code (Solid Performance Program) [23] have been carried out resulting in a very high agreement values (Fig. 3).

2.4. ROCKET Module

The mathematical model which describes flow filed in a rocket motor is based on the continuity equation of mass, moment and energy in one-dimensional form. Basic assumptions for this model are:

- Products of combustion are considered ideal gasses,
- Propellant burning rate is mostly influenced by the combustion chamber pressure and is expressed by Saint Robert's (or Vieille's) law within a limited pressure range:

$$r_0 = a \cdot p^n \quad (4)$$

The pressure exponent n and the burn rate coefficient a are dependent on chemical composition of a solid propellant and initial temperature of the propellant charge. These coefficients are usually determined by means of firing tests of ballistic evaluation motors [24-27, 13]. Influence of initial temperature of propellant charge on burning rate and combustion pressure can be expressed as:

$$a = a_0 e^{\sigma_p (T_p - T_0)} \quad (5)$$

where: a_0 – temperature constant for temperature $T_0 = 20^\circ\text{C}$, T_p – propellant temperature and

$$\sigma_p = \pi_K (1 - n) \quad (6)$$

- Influence of mass flux or erosive burning on burning rate in rocket motor chamber is considered using modified formula of *Lenoir and Robillard* (LR). In this model total burning rate contains a component of burning rate in normal burning (no erosive burning) r_0 and component which is a result of erosive burning r_e [3, 7, 12]:

$$r_b = r_0 + r_e \quad (7)$$

The LR model defines the erosive burning contribution as:

$$r_e = \alpha \cdot G^{0.8} \cdot \exp(-\beta \cdot r_b \cdot \rho_s / G) / L^{0.2} \quad (8)$$

$$\alpha = \frac{0.0288 \cdot c_{p_g} \cdot \mu_g^{0.2} \cdot \text{Pr}_g^{-2/3}}{\rho_s \cdot c_s} \cdot \left(\frac{T_c - T_s}{T_s - T_0} \right) \quad (9)$$

where G – the mass flux of the combustion gasses, ρ_s – density of propellant [kg/m^3], L – characteristic length [m], c_{pg} – constant pressure specific heat of gasses [J/kgK], Pr – Prandtl number, T_c , T_s , T_0 – temperature of combustion products, burning surface and initial condition of propellant [K], c_s – constant pressure specific heat of propellant [J/kgK].

Using equations 8 and 9, the erosive burning contribution can be calculated using only one empirical value (β), which is essentially independent of propellant composition and approximately 53 [3, 7, 12]. The value of in equation 9 can also be assigned from empirical data rather than calculated with transport properties. A further improvement to the LR model is presented by the authors of the solid propellant rocket motor performance computer program (SPP) [3,12] using the equation:

$$r_e = \alpha \cdot G^{0.8} \cdot \exp(-\beta \cdot r_b \cdot \rho_s / G) / f(D_h) \quad (10)$$

where $f(D_h) = 0.90 + 0.189 \cdot D_h \cdot [1 + 0.043 \cdot D_h \cdot (1 + 0.023 \cdot D_h)]$, D_h – the hydraulic diameter (calculated using the wetted perimeter, not burning perimeter, and port area).

- Characteristic velocity is not a function of combustion pressure but propellant type and it is determined using [13]:

$$C^* = C_{p_{cnom}}^* \eta_{C^*} \quad (11)$$

where: $C_{p_{cnom}}^*$ – characteristic velocity obtained based on theoretical calculation of rocket motor performances under nominal value of combustion pressure for case of equilibrium expansion; η_{C^*} – coefficient of combustion efficiency.

Calculation of pressure inside a rocket motor as a function of time is based on the continuity equation – mass of gas generated by combustion of propellant charge \dot{m}_g is equal to the sum of the mass of combustion products accumulated in a rocket motor dM/dt and mass of combustion products through nozzle \dot{m}_n , (Fig. 4):

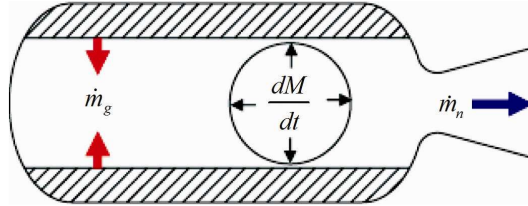


Fig. 4. Balance of gas mass by combustion of propellant charge in a rocket motor

$$\dot{m}_g = \frac{dM}{dt} + \dot{m}_n \quad (12)$$

Mass of gas generated by combustion of propellant charge \dot{m}_g is given as,

$$\dot{m}_g = \rho_s \cdot A_b \cdot r_b \quad (13)$$

where A_b – area of combustion of propellant charge [m²];

Mass of combustion products accumulated in a rocket motor dM/dt is:

$$\frac{dM}{dt} = \frac{d}{dt}(\rho_g V) = \rho_g \frac{dV}{dt} + V \frac{d\rho_g}{dt} \quad (14)$$

where: $\rho_g = p_c / (R_g \cdot T_c)$ – density of combustion gas products in a rocket motor [kg/m³], V – free volume for gas flow [m³], p_c – combustion pressure [Pa], and $\frac{d\rho_g}{dt} \approx \frac{1}{R_g \cdot T_c} \cdot \frac{dp_c}{dt}$, change of density of combustion gas products,

or

$$\frac{dM}{dt} = \rho_g \frac{dV}{dt} + \frac{V}{R_g \cdot T_c} \cdot \frac{dp_c}{dt} \quad (15)$$

Mass of combustion products through nozzle \dot{m}_n is given by:

$$\dot{m}_n = \frac{p_c \cdot A_{th}}{C^*} \quad (16)$$

where: A_{th} – area of critical nozzle section [m²], C^* – characteristic velocity of gaseous combustion products.

Change of combustion pressure in rocket motor is determined using numerical integration of expression (from expression 12, after substitution of 13, 15 and 16):

$$\frac{dp_c}{dt} = \frac{1}{V_{c_i}} \cdot \left[R_g \cdot T_c \cdot \left(\sum_{j=1}^{L/\Delta x} \rho_s \cdot A_{b_{i_j}} \cdot r_{b_{i_j}} - \frac{p_{c_i} \cdot A_{th_i}}{C^*} \right) - p_{c_i} \cdot \frac{dV_{c_i}}{dt} \right] \quad (17)$$

Change of thrust is calculated using:

$$F_i = C_{F_i} \cdot p_{c_i} \cdot A_{th_i} \quad (18)$$

Coefficient of thrust is determined using a constant coefficient of ratio for specific heat of combustion products (model enables correction of the thrust coefficient and in the case of significant change of surrounding (environmental) pressure – influence of separation of gaseous flow):

$$C_{F_i} = \eta_{C_{F_i}} \cdot \sqrt{2 \cdot \frac{\gamma^2}{\gamma-1} \cdot \left(\frac{2}{\gamma+1}\right)^{(\gamma+1)/(\gamma-1)} \cdot \left[1 - \left(\frac{p_{e_i}}{p_{c_i}}\right)^{(\gamma-1)/\gamma}\right]} + \frac{p_{e_i} - p_a}{p_{c_i}} \cdot \left(\frac{A_e}{A_{th_i}}\right) \quad (19)$$

This module provides prediction of an average delivered performance, as well as mass flow, pressure, thrust and impulse as functions of time.

3. RESULTS

Results of verification for previous modules have shown very good agreement with results obtained in referent computer programs and with experimental tests. Accuracy of model for prediction of internal ballistic performances of solid propellant rocket motors can be determined by comparing results of our prediction and known experimental results for following rocket motors (Table 3): rocket motor with a 57 mm diameter (RM-1), rocket motor with 128 mm diameter with Cluster CP Grain (RM-1), rocket motors with 128 mm diameter with star grain and a central nozzle (RM-3) and with multiple perforated nozzles (RM-4) and a rocket motor with a 204.7 mm diameter using (STAR-8) CP grain [20,28].

Table 3. Data on real, tested rocket motors

Rocket motor	Propellant type	Grain	$r=a(P_c[\text{MPa}])^n$ [m/s]		$L^*=V_c/A_{th}$ [m]
			a	n	
RM-1	NGR-C (NC12% N/NG – 56.73/27.5%)	CP	0.00731	0.273	1.17
RM-2	NGR-B (NC12% N/NG – 55.7/30%)	Cluster CP	0.00276	0.5734	2.55
RM-3(4)*	NGR-A (NC12% N/NG – 55.24/ 33.84%)	STAR	0.013072 0.021616	0.2276 0.0369	1.12
STAR-8	TP-H-3062 (AP/CTPB/Al – 70/14/16%)	CP	0.004202	0.31	12.4

Note: Propellant with “plateau” effect (first law of burning applies to 14 MPa, and second law – above)

3.1. 57 mm – RM-1 Rocket motor

The RM-1 Rocket motor uses CP grain with internal-external burning (without restriction of burning surface) with a central nozzle without erosion of the throat nozzle section. During the experiment, change of thrust vs. time was measured for a group of 63 rocket motors. Standard deviation of total impulse, pressure integral and specific impulse is under 1%.

In Figure 5 comparative results of change pressure and thrust vs. time were shown for a prediction model and an experimental test. In simulation, basic burning rate is corrected with erosive burning influence (coefficients $J = A_{th}/A_p = 0.385$, $K = A_b/A_{th} = 512$, and $r_b \approx 19.5$ mm/s) by using equation 8 ($\beta = 120$). Also, influence of the HUMP effect is analyzed (obtained in analysis of burning rate based on methodology given in reference [29]).

Very good agreement is achieved in prediction of thrust change vs. time, with experimental data. Agreement is especially notable in phase of quasi-stationary burning, while higher deviations are present in the exhausting phase (model doesn't consider structural integrity of the charge in final phase of burning and eventual sliver). Deviation of total impulse value is 0.3% and integral of pressure up to 0.45% which represents a good agreement with experimental research.

3.2. 128 mm – RM-2 Rocket motor

The chamber of the RM-2 rocket motor contains four CP grains with internal-external burning, without restriction of burning surface. The rocket motor has a central nozzle without erosion of throat nozzle section. Figure 6 shows change pressure and thrust vs. time for an RM-2 rocket motor obtained with the SPPMEF program and experimentally.

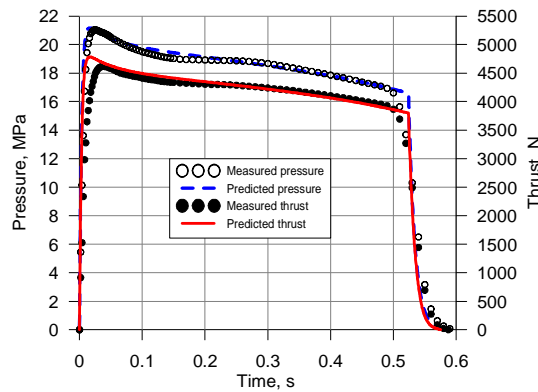


Fig. 5. Pressure vs. time and thrust vs. time for rocket motor RM-1

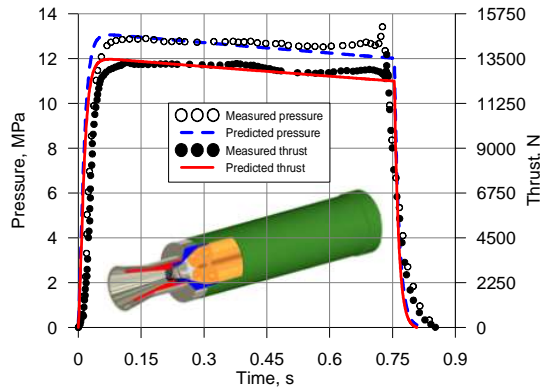


Fig. 6. Pressure vs. time and thrust vs. time for rocket motor RM-2

There is also here excellent agreement in our prediction of thrust change vs. time, with experimental data. Deviation of the total impulse value is 1.2%, integral of pressure up to 0.6%.

3.3. 128 mm – RM-3 Rocket motor

The RM-3 rocket motor contained a star grain with double base propellant and used a central nozzle without erosion of throat nozzle section. During testing of the RM-3 rocket motors combustion chamber pressures were measured at both ends of the combustion chamber. Also, thrust change vs. time was measured. Difference between pressures at both ends of the combustion chamber was around 8%.

In Fig. 7 comparative results of pressure change and thrust vs. time were shown for prediction model and experimental test (average values of pressure). In simulation, the basic burning rate is corrected with erosive burning influence (coefficients $J = A_{th}/A_p = 0.448$, $K = A_b/A_{th} = 242$, and $r_b \approx 22$ mm/s) using equation 8 ($\beta = 120$). Influence of the HUPM effect was analyzed. The prediction has shown good agreement with test results.

3.4. 128 mm – RM-4 Rocket motor

The RM-4 rocket motor has the same propellant and charge configuration as the RM-3 rocket motor. This rocket motor uses multiple perforated nozzles (8 nozzles without tangential eccentricity of nozzle) without erosion of throat nozzle section whose total surface is equal to the RM-3 rocket motor. In Figure 8 comparative results of pressure and thrust change vs. time were shown for a prediction model (SPPMEF) and experimental test.

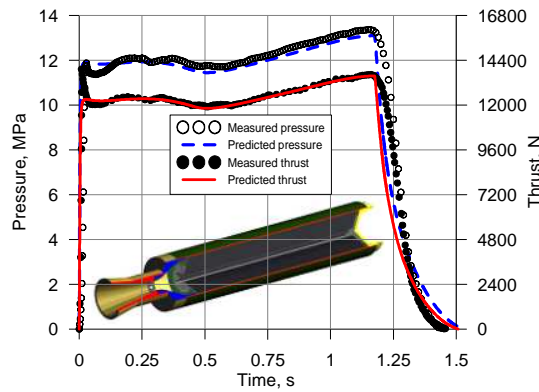


Fig. 7. Pressure vs. time and thrust vs. time for the RM-3 rocket motor

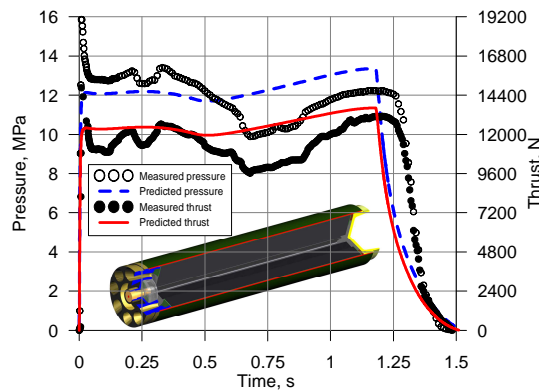


Fig. 8. Pressure vs. time and thrust vs. time for the RM-4 rocket motor

There is significant deviation in prediction of thrust and pressure change when compared to experimental tests. This is due to the fact that when products of combustion leave the internal cavity of propellant grain they don't immediately enter the nozzles (multiple perforated nozzle), but gas flow is curled and forms turbulent flow at the front of nozzle block. Only after it enters into convergent-divergent conical nozzles. During this process there is significant change of gas flow velocity vector and redistribution of gas flow pressure in this region, which is influencing changes in development of pressure in the rocket motor and changes of internal-ballistic parameters (i.e. total and specific impulse of the rocket motor) [27].

Currently, the Defense Technologies Department is conducting research aimed at the expansion of a model where complexity of gas flow between propellant charge and nozzle, by means of numerical simulation, is taken into account.

3.5. STAR 8 Rocket motor

The STAR 8 was developed and qualified (2002) as the rocket assisted deceleration (RAD) motor for the Mars Exploration Rover (MER) program for the Jet Propulsion Laboratory (JPL) in Pasadena, CA. The motor contained CP propellant grain with composite propellant TP-H-3062 and used a 6AI-4V titanium case, pyrogenic igniter, and centered nozzle.

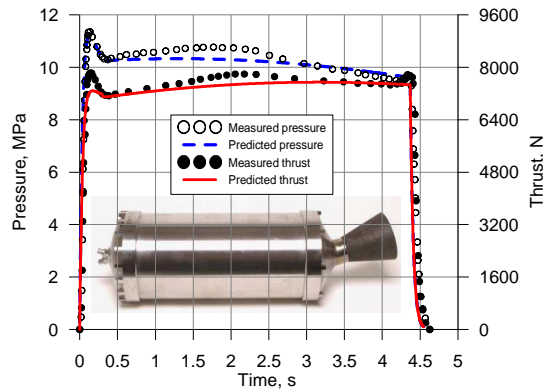


Fig. 9. Pressure vs. time and thrust vs. time for a STAR 8 rocket motor (-30°C, vacuum)

In Figure 9 comparative results of pressure and thrust change vs. time were shown for prediction model and experimental test.

In simulation, the initial surface of throat section is corrected due to the eccentricity of nozzle, based on methodology in reference [27]. Also, erosion of throat nozzle section was considered based on value of radial erosion degree, given in reference [28]. The prediction has shown a good agreement with test results. By taking into account the HUMP effect, these agreements would be even better.

4. CONCLUSIONS

A modular computer program named SPPMEF has been developed facilitating prediction of internal ballistic performances of solid propellant rocket motors, which enables:

- Calculation of theoretical performances of propellant, ideal rocket performance and prediction of losses of performances in rocket motor nozzle.
- Dimensioning and regression of burning surface of propellant grains.
- Prediction of average performances such as mass flux, pressure, thrust and specific impulse vs. time.

- Modularity of its structure enables further development of the software for improvement of particular modules in future work.

Comparative analysis of results of the SPPMEF program with results of referent program versions and experimental tests has shown following:

- Very good agreement was obtained in prediction of pressure/thrust change vs. time, when compared to experimental data where there is significant influence of gas flow and mass flow on burning rate, as well as for rocket motors with a central nozzle.
- For rocket motors which operate in a stable manner, it is possible to determine average values of pressure and thrust, as well as their integrals with accuracy up to 2% and for rocket motors with significant instability in combustion, maximal error in prediction is up to 5%.
- Understanding of complexity of gas flow in case of rocket motors with multiple perforated nozzles, is only possible using methods of numerical simulation.

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