PREDICTION OF INTERNAL BALLISTIC PARAMETERS OF SOLID PROPELLANT ROCKET MOTORS

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Abstract. A modular computer program for prediction of internal ballistic performances of solid propellant rocket motors SPPMEF has been developed. The program consists of following modules: TCPSP (Calculation of thermo-chemical properties of solid propellants), NOZZLE (Dimensioning of nozzle and estimation of losses in rocket motor), GEOM (This module consists of two parts: a part for dimensioning 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).
Program is verified with experimental results obtained from standard ballistic rocket test motors and experimental rocket motors. Analysis of results has shown that established model enables has 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.

1. INTRODUCTION

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

Analysis of solid propellant rocket motors is progressing in two levels, where, independent of 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 design uses tools that have to be simple and adaptable to user. There are usually simple computer codes, based on analytical models or diagrams that give simple first results.

Second level is level of final design of propellant charge. 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, continuum mechanics structural analysis). They allow precise calculations, or optimization up to defining final geometry.

Countries with high technological level (USA and western countries) focus their continual research on prediction of theoretical performances of solid propellant rocket motor. They base their research on development of high range ballistic guided rockets, based on composite propellant charges. Large number of experimental research, conducted during the development of these rocket systems, enabled huge database 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.

Program SPP 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. Program SPP 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 different rocket motors, but most of these data are unavailable for other countries [3].

From september 1997. to 2007., Center for Simulation of Advanced Rocket (CSAR), the University of Illinois at Urbana-Champaign, for needs of U.S. Department of Energy, was developing program for prediction of the performances of solid propellant rocket motors based on numerical simulation [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).

Defense Technology Department, at 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, but for rocket motors where influence of gas flow and mass flux on burning rate is not significant, as well s for rocket motors with central nozzle [13].

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

The program SPPMEF consists of a series of modules that are integrated to provide a method to predict the average delivered performance (figure 1):

- TCPSP – Calculation the Thermo-chemical properties of solid propellants,
- NOZZLE – Dimensioning of nozzle and estimating losses in rocket motor,
- GEOM – This module is consisted of two parts: a part for dimensioning the propellant grain and a part for regression of 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.

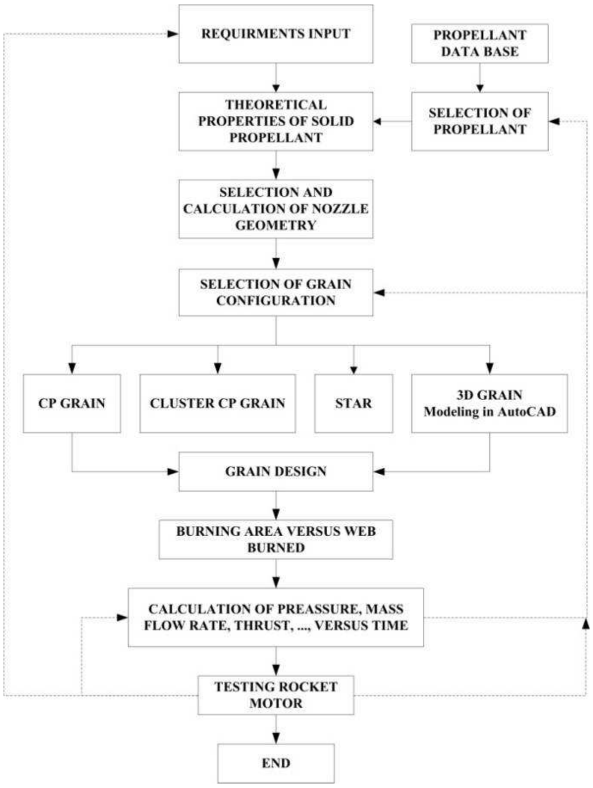

Figure 1. Model SPPMEF

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

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

2.1 Module TCPSP

This module enables calculation of the combustion products composition at chemical equilibrium (model Minimum of Gibbs energy), 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 calculation expansion for 3 different expansion ratio)

Module TCPSP 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 to predict 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.

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

Table 1. Comparative analysis of some properties in the combustion chamber for the 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]	2716.8	2724.46	-0.28%	2700.2	2708.02	-0.29%
Cp [J/gK]	2.4185	2.40789	0.44%	2.5407	2.531738	0.35%
γ	1.1969	1.1945	0.20%	1.1926	1.189	0.30%
s [J/gK]	10.529	10.57506	-0.44%	10.788	10.82443	-0.34%
h [J/g]	-2028.3	-2028.24	0.00%	-2028.3	-2028.24	0.00%
ρ [g/m^3]	3527	3520.9	0.17%	1772	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%	1.069	1.0686	0.04%
(dvp)t	-1.0026	-1.00263	0.00%	-1.0035	-1.00342	0.01%

2.2 Module NOZZLE

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

Process of dimensioning of nozzle demands that following is known:

- Average values of thrust F_{aver} (determined in external-ballistics analysis of missile mission).
- Combustion pressure in rocket motor chamber (determined during the process of choosing the type of propellant).
- Theoretical values of thermo-chemical parameters of propellant, for case of equilibrium and “frozen” state of combustion products, for adopted working combustion pressure and ration 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 rocket motor nozzle (for assessment of losses we need to know 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 complex task, which encompasses theoretical values of specific impulse of propellant, combustion process coefficient of efficiency η_{C^*} and thrust coefficient of efficiency η_{C_F} :

$$I_{sp} = I_{sp_{teo}} \cdot \eta_{C^*} \cdot \eta_{C_F} \tag{1}$$

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 program SPP [7,18,19]. The program currently treats 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

$$\eta_{C_F} = 1 - 0{,}01 \cdot \left(\varepsilon_{DIV} + \varepsilon_{TP} + \varepsilon_{BL} + \varepsilon_{KIN} + \varepsilon_{SUB} + \varepsilon_{EROS}\right) \tag{2}$$

Experimental values specific impulse are determined using:

$$I_{sp_{exp}} = \frac{\int F\, dt}{m_p}. \tag{3}$$

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_{C_F}	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 losses of performances of rocket motor is in very good agreement with results obtained experimentally. Maximal deviations of specific impulse is up to 0,5%.

2.3 Module GEOM

Grain dimensioning modules 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 modelling 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 channels for gas flow. 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 computer program OPTIM [22], which insure to choose optimal geometry of star grain by variation of seven independent geometric variables (figure 2) of propellant with assumptive intervals of volumetric loading, relative rest of propellant which is not burned (sliver - σ) and degree of neutral burning area of propellant ($\Gamma_{min}=S_{max}/S_{aver}$), has been developed.

Comparative analyses of results from OPTIM computer code with referring code SPP (Solid Performance Program) [23] have been carried out and very good agreement has been obtained (figure 3).

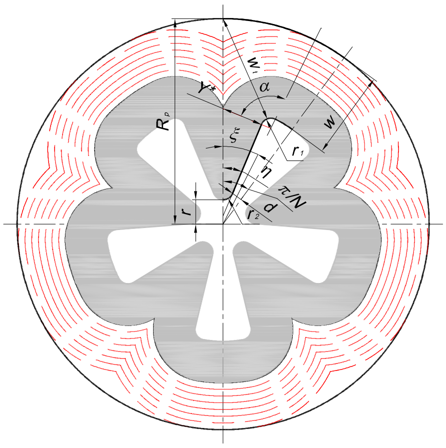

Figure 2. Geometric definition of star grain and regression of burning surface

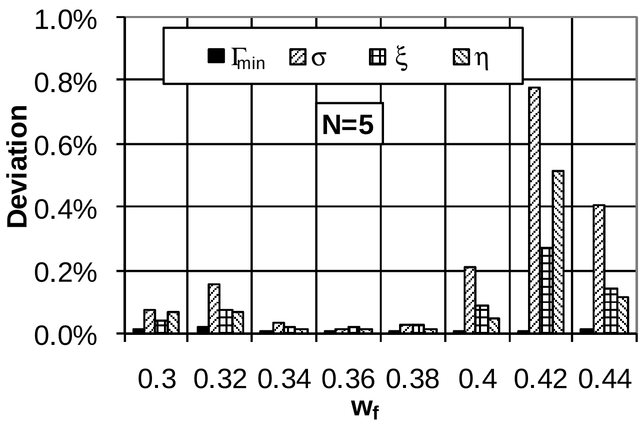

Figure 3. Comparative analyses of results from OPTIM computer code with reference [23] for optimization of star grain with 5 sides (V_l=0.85 and $r_1/R_p=r_2/R_p$=0.05)

2.4 Module ROCKET

Mathematical model which describes flow filed in rocket motor is based on 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 Vielle's) law within a limited pressure range:

$$r_0 = a \cdot p^n \tag{4}$$

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 test of ballistic evaluation motors [24-27,13]. Influence of initial temperature of propellant charge on burning rate and combustion pressure can be expressed as:

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

where: a_0 – temperature constant for temperature T_0 = 20°C, T_p - propellant temperature and

$$\sigma_p = \pi_K (1 - n). \tag{6}$$

- 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 component of burning rate in normal burning (no erosive burning) r_0 and component which is result of erosive burning r_e [3,7,12]:

$$r_b = r_0 + r_e \tag{7}$$

The LR model defines the erosive burning contribution as:

$$r_e = \alpha \cdot G^{0,8} \cdot \exp(-\beta \cdot r_b \cdot \rho_s / G) / L^{0,2} \tag{8}$$

$$\alpha = \frac{0.0288 \cdot c_{p_g} \cdot \mu_g^{0,2} \cdot \Pr_g^{-2/3}}{\rho_s \cdot c_s} \cdot \left(\frac{T_c - T_s}{T_s - T_0} \right) \tag{9}$$

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 equation:

$$r_e = \alpha \cdot G^{0,8} \cdot \exp(-\beta \cdot r_b \cdot \rho_s / G) / f(D_h) \tag{10}$$

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]:

$$C^* = C^*_{p_{cnom}} \eta_{C^*} \tag{11}$$

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 rocket motor as a function of time is based on continuity equation - mass of gas made by combustion of propellant charge $\dot{m}_g$ is equal to sum of mass of combustion products accumulated in rocket motor *dM/dt* and mass of combustion products through nozzle $\dot{m}_n$, (figure 4):

$$\dot{m}_g = \frac{dM}{dt} + \dot{m}_n . \tag{12}$$

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

$$\dot{m}_g = \rho_s \cdot A_b \cdot r_b \tag{13}$$

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

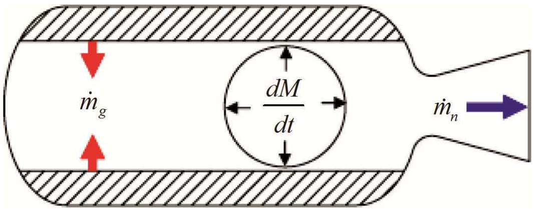

Figure 4. Balance of gas mass by combustion of propellant charge in rocket motor

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

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

where: $\rho_g = p_c / (R_g \cdot T_c)$ – density of combustion gas products in 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

$$\frac{dM}{dt} = \rho_g \frac{dV}{dt} + \frac{V}{R_g \cdot T_c} \cdot \frac{dp_c}{dt} . \tag{15}$$

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

$$\dot{m}_n = \frac{p_c \cdot A_{th}}{C*} \tag{16}$$

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):

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

Change of thrust is calculated using:

$$F_i = C_{F_i} \cdot p_{c_i} \cdot A_{th_i} \tag{18}$$

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

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

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 of 57 mm (RM-1), rocket motor with 128 mm diameter with Cluster CP Grain (RM-1), rocket motors with 128 mm diameter with star grain with a central nozzle (RM-3) and with multiple perforated nozzle (RM-4) and rocket motor with 204.7 mm with CP grain (STAR-8) [20,28].

Table 3. Data on tested real rocket motors

Rocket motor	Propellant type	Grain	$r=a(P_c[MPa])^n$ [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 “plato” effect (first law of burning applies to 14 MPa, and second law - above)

3.1 Rocket motor 57 mm – RM-1

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

In fig. 6 comparative results of change pressure and thrust vs time were shown for prediction model and 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 (β =120). Also, influence of 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 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 represent good agreement with experimental research.

3.2 Rocket motor of 128 mm – RM-2

In chamber of rocket motor RM-2 there are four CP grain with internal-external burning, without restriction of burning surface. Rocket motor have central nozzle without erosion of throat nozzle section. Fig. 7 shows change pressure and thrust vs time for rocket motor RM-2 obtained with program SPPMEF and experimentally.

Also here there is excellent agreement in our prediction of thrust change vs. time, with experimental data. Deviation of total impulse value is 1,2%, integral of pressure up to 0.6%.

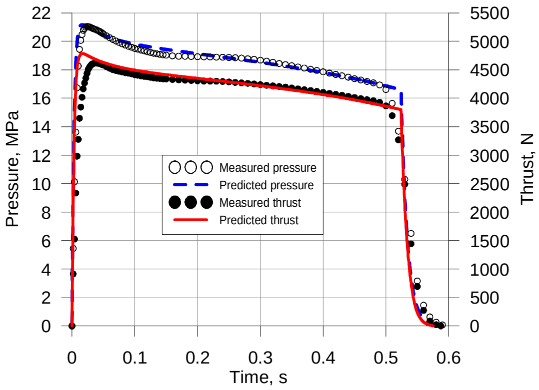

Figure 6. Pressure vs time and thrust vs time for rocket motor RM-1

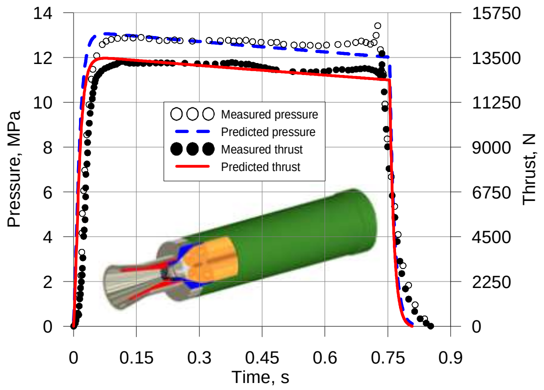

Figure 7. Pressure vs time and thrust vs time for rocket motor RM-2

3.3 Rocket motor of 128 mm – RM-3

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

In fig. 8 comparative results of change pressure and thrust vs time were shown for prediction model and experimental test (average values of pressure). In simulation, 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 (β =120). Influence of HUPM effect was analyzed. The prediction has shown good agreement with test results.

3.4 Rocket motor of 128 mm – RM-4

Rocket motor RM-4 has the same propellant and charge configuration as rocket motor RM-3. 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 rocket motor RM-3. In fig. 9 comparative results of pressure and thrust change vs. time were shown for prediction model (SPPMEF) and experimental test.

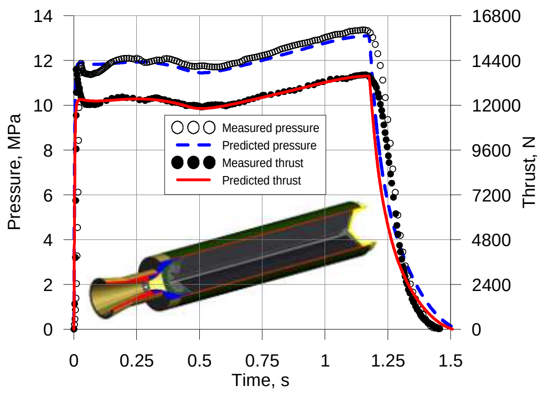

Figure 8. Pressure vs time and thrust vs time for rocket motor RM-3

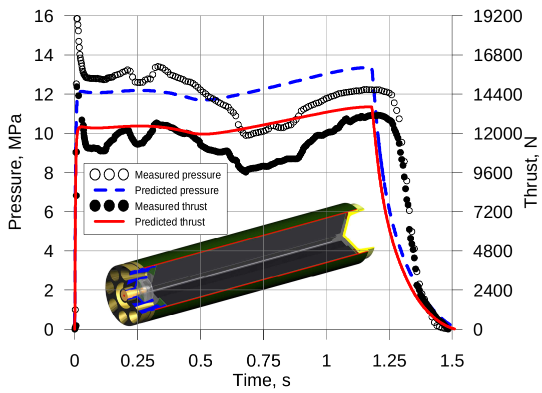

Figure 9. Pressure vs time and thrust vs time for rocket motor RM-4

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 internal cavity of propellant grain they don't immediately enter the nozzles (multiple perforated nozzle), but gas flow is curled and it is forming 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 the changes in development of pressure in rocket motor and changes of internal-ballistic parameters (i.e. total and specific impulse of rocket motor) [27].

At this moment Defense Technologies Department is conducting the research pointed to the expansion of model where complexity of gas flow between propellant charge and nozzle, by means of numerical simulation, is taken into account.

3.5 Rocket motor STAR 8

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 wit composite propellant TP-H-3062 and used 6AI-4V titanium case, pirogen igniter, and centred nozzle.

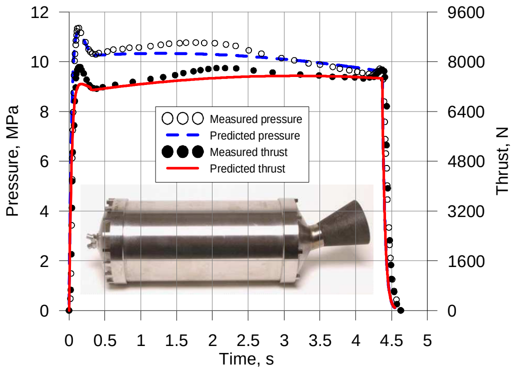

Figure 10. Pressure vs time and thrust vs time for rocket motor STAR 8 (-30°C, vacuum)

In fig. 10. comparative results of pressure and thrust change vs. time were shown for prediction model and experimental test.

In simulation 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 HUMP effect these agreement would be even better.

4. CONCLUSIONS

A modular computer program SPPMEF is developed for 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 program SPPMEF 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 now significant influence of gas flow and mass flow on burning rate, as well as rocket motor with central nozzle.
- For rocket motors that have stable work it is possible to determine average values of pressure and thrust, as well as their integrals with accuracy up to 2%, and for rocket motor with significant instability in combustion, maximal error in prediction is up to 5%.
- Understanding of complexity of gas flow, in case of rocket motor with multiple perforated nozzles, is only possible using methods of numerical simulation.

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