THE STUDY OF THE INFLUENCE OF PROPELLANT PERFORMANCES CHANGES ON THE BALLISTIC CHARACTERISTICS OF ANTI-HAIL ROCKET

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Abstract: *The paper proposes to analyse the influence of utilising an alternant propellant with similar characteristics on ballistic performances of anti-hail rocket. For that purpose a comparative analyse of usual performances of rockets used so far will be made, giving a special attention to the safety aspects of product operation. Following these results was obtained several relevant recommendations helpful in the future development of this kind of weather modification means, and in anti-hail working procedures also.*
Keywords: *rocket, hail, propellant, trajectory*

1. INTRODUCTION

A national anti-hail system was established in our country in 1999 based on the cloud's seeding method using silver iodide. The method of seeding the clouds with ground-launched rockets was chosen according to the national strategy [1], [2].

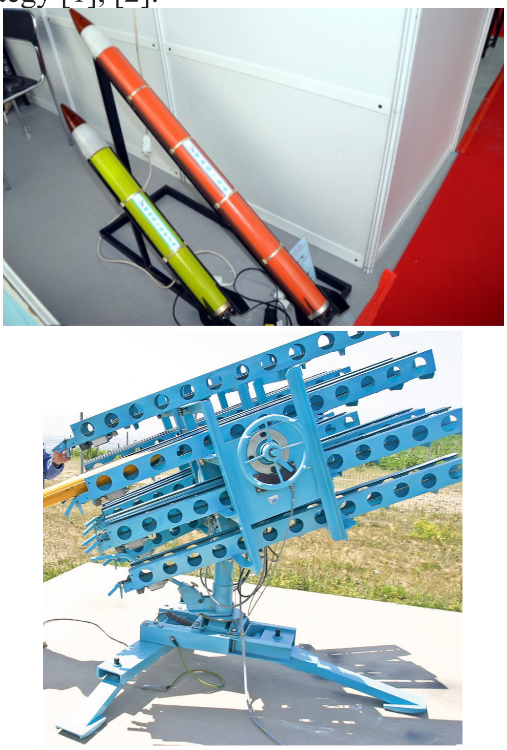

Fig. 1 Anti-hail rocket and launch platform

The system is composed of several fighting hail units. A hail Combat Unit includes a weather station equipped with weather radar with a Doppler effect, a central control point and several local units of anti-hail rocket launch (Fig. 1).

The functioning of anti-hail rocket ensures the transport of reactive agents at the necessary altitude where the substances are spreading according to rocket's sequences presented in figure 2. For secure reasons the self-destruction must occurred at a pre-set altitude.

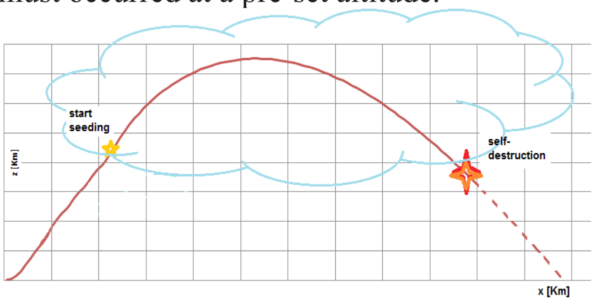

Fig. 2 Rocket's sequences

Unlikely the rocket terminology we define the active zone of the rocket's trajectory as the zone between the start seeding point and self-destruction point. The active zone of the trajectory determines some constraints in positioning of launching units. The launching units' disposal the must be designed according to radial active area resulted, so as to obtain a better coverage of the desired protected area.

The component that ensures the required propulsion power is the rocket engine, propellant playing a crucial role.

Since there are various suppliers in the market results differences in propellant's performance. The aim of this paper is to quantify the deviations due to these differences in the functional characteristics of the product and to set acceptable limits for the propellant's performance.

2. MODEL DESCRIPTION

In order to predict the trajectory of an unguided rocket, six degrees of freedom (6-DOF) mathematical model is used according to [4,8]. All aerodynamic forces and moments coefficients of the configuration are previously calculated according to [3], and they are considered as input data.

The mass properties (mass, mass centre, moments of inertia) are calculated considering the change of the rocket mass during propellant burning till the propellant burn-out (active part), then the rocket will fly the rest of its trajectory as a projectile of fixed mass (passive part) until self-destruction. The 6-DOF model assumed the rocket is ideal, where the axis of symmetry of the exterior surface coincides with the longitudinal principal axis of inertia, and the two lateral principal moments of inertia are identical.The input data for calculations and for obtaining the necessary data for simulation of rocket trajectory are (table 1): mass and geometric data, experimentally data for thrust obtained on firing bench (Fig. 3) for different temperatures and the input parameters of launching conditions: launch angle, wind speed etc.

Table 1

Input data	model (product) standard	model (product) modified	Difference %
initial mass [kg]	8.5	8.2	3.53
final mass [kg]	4.87	4.87	0.
initial mass center [mm]	750	640	1.33
final mass center [mm]	686	686	0.

For the present paper two double base propellants with little geometry differences (both existing on market) were utilized as components in propulsion system of the rocket.

This will induce differences of input data for thrust value, burning time, mass variation in time, and others that will conduct inevitable to other results for trajectory of the rocket [5,7].

The aerodynamic configurations are the same for both cases. All the deviations in rocket functioning must be evaluated. In figure 3 there are represented the thrust diagram for the standard and modified models.

The thrust data are obtained experimentally by testing the rocket engine on the firing bench based on methodology described in [6].

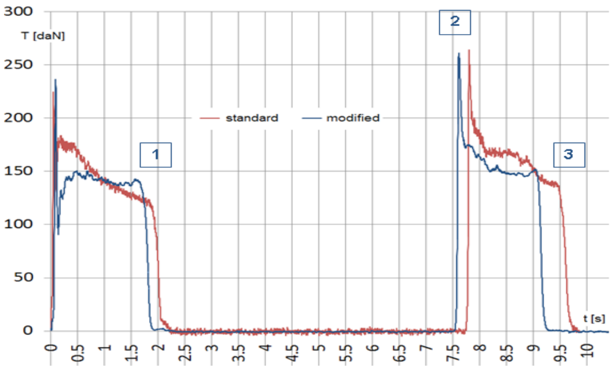

Fig. 3 Thrust diagram
(experimental bench data)

The propulsion occurs in 2 steps (corresponding to the two stages - see Fig. 3), separated by a delay time. The measurements performed for the force vs time provide necessary information to estimate the average value of thrust and specific impulse for each propellant.

Moment 1 corresponds to the final of stage1, also marking the burning time for stage1, moment 2 corresponds to the beginning of stage2 - burning and moment 3 corresponds to the final of stage 2 (Fig. 3). Knowing the experimental values *T(t)* from the thrust diagram (Fig. 3) for both rocket engines, the average value of thrust for each stage was calculated as the average value of a function:

$$T_m = \frac{\int_{t_i}^{t_j} T(t)dt}{\Delta t} \quad (1)$$

All these average values of thrust are presented in Table 2 for both models.

Table 2

Parameter	standard	modified	Difference %
Propellant mass [kg]	3.1	2.8	9.68
Average value of thrust for stage1 [daN]	132.68	132.5	0.14
Average value of thrust for stage2 [daN]	146.93	146.17	0.52
moment 1 [s]	2.15	1.85	11.63
moment 2 [s]	7.75	7.6	1.94
moment 3 [s]	9.75	9.25	5.13
self-destruction [s]	43.5	42.45	2.41

By definition, the specific impulse for each stage is:

$$I_{sp} = \frac{T_m}{\dot{m}.g_0} \quad [s] \qquad (2)$$

Where:

T_m is the average value of thrust obtained from the engine [N];

$\dot{m}$ is the mass flow rate [Kg/s];

g_0 is the acceleration at the Earth's surface [m/s^2].

Although the difference of thrust between the two propellants is not significant, however the difference of specific impulse is 4,86% for the first stage, while 5,83% is the difference for the second stage.

Because the geometric configuration is the same all the aerodynamic coefficients are unchanged.

The static stability margin is defined as the distance between the center-of-pressure *(CP)* and center-of-gravity *(CG)* locations normalized with respect to the body length *l*:

$$SM = \frac{X_{CP} - X_{CG}}{l} \qquad (3)$$

The influence of mass variation is reflected in variation of centre mass position. Since CP position is the same because the geometry is unchanged only the differences of CG position will have an influence. Analysing the variation of CG position we conclude that are less than 2% (Table 1), so the differences are too small to endanger the static stability of the rocket.

3. SIMULATION RESULTS

The results obtained refer to a number of parameters (coordinates, speed, angles, etc. vs. time) that describe the performance of the rocket during flight.

For this type of rocket will have to pay attention to:

- maximum range position and time
- maximum altitude position and time
- position and time for start-seeding
- position and time for self-destruction
- distance during seeding and duration

Analysis was performed for all the necessary launching angles, but in the present paper only results for 45^0 and 50^0 will be referred. Because it will make a comparative analysis, these trajectory diagrams will be represented compared to the standard case, as in figure 4.

For a better interpretation the time moments corresponding to the associate position in space are also represented on the diagram.

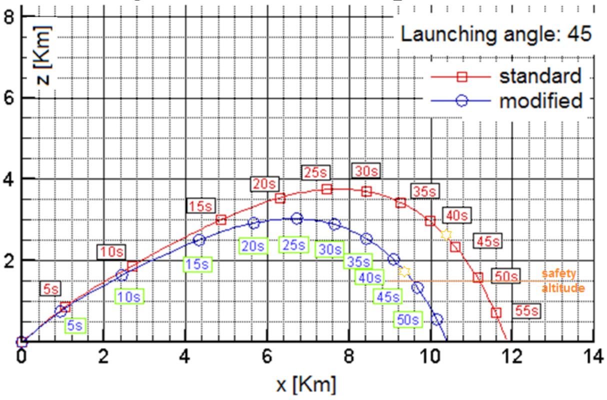

Fig. 4 The trajectories

For this case a difference in maximum altitude reached is about 25% and the maximum range decreased by about 8%. The trajectory being shorter and lower determines the total flight time to be decreased by 12%. Since the time of self-destruction must be achieved at a safe altitude of more than 1500 m, the rocket position in time must be carefully examined during flight. From figure 4 we can observe that the self-destruction (marked with yellow on the trajectories diagram) occurs lower than in standard case, so the limit of secure altitude became narrower.

CONCLUSIONS

Regarding the influence of these changes in the organization of the national system, there are two possible approaches:

i. Starting from the acceptable limits of positioning trajectories according to existing launching units and constraints of the evolution (time for seeding, time and position for self-destruction) will result limitations imposed to the propellant performances.We conclude that for the studied case a decreasing of 6% in specific impulse is the accepted lower limit for the performances of propellant. Anyway a modification of propellant, even in the accepted limits, must be studied from the point of view of all the constraints of the product functioning (trajectory via seeding time and launching angle, time and position for self-destruction, etc.)

ii. Starting from deviations that may have the propellants in the same range, just results trajectory deviations that have to be declared as acceptable and to be taken into account in future launching unit's disposal.

One particular zone of trajectory is very important. This is the active zone where the rocket performs cloud's seeding and it is important to be on the top side of trajectory on an area extended as much as possible to increase the rocket's efficiency. The area of significance (where the presence of seeding is necessary), as it is presented in figure 5, is traversed by the trajectory of standard rocket for the launching angle of 45 degrees. The length of active zone of trajectory inside the area of significance is a measure of anti-hail rocket's efficiency. We can observe that for the modified rocket the trajectory for the same launching angle is lower and the active zone inside the area of significance is shorter, therefore the efficiency of seeding is lower. A better matching for the area of significance is offered by the trajectory of 50 degrees launching angle, even though there is a noticeable horizontal shortening of the intervention.

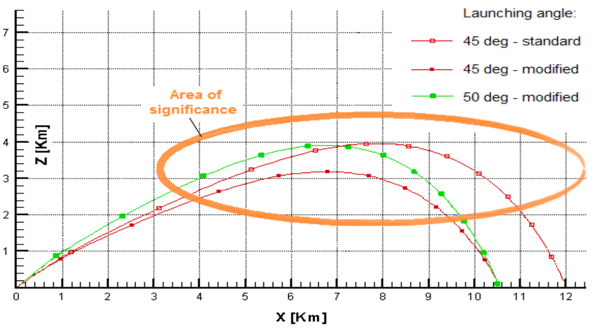

Fig 5 The seeding area

As a conclusion, the area of intervention is shorter than in standard case, and the launching angle must be increased to approach the performance from standard model. The problem have a real involvement in the operational procedures. Around the launching points, until the maximum effective range of the rocket, two zones can be distinguished (Fig. 6):

- the protection zone (near) - is the area protected 75% - 90%
- the processing zone (far) – is the area where is made the active intervention against hail (with some probability of hail falling)

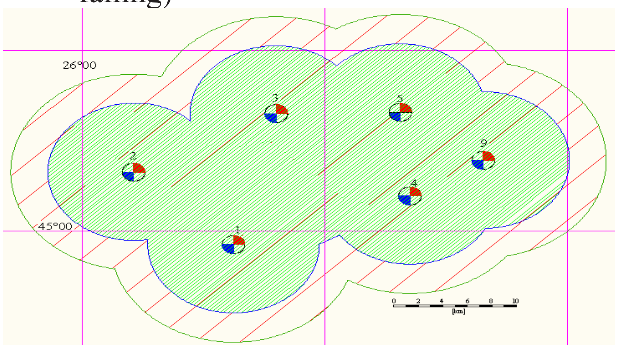

Fig. 6 The protection and processing zones

When a factor imposes to have a shorter seeding line, the processing zone became bigger and the protection zone decreases, that reduce the system efficiency.

In the case of predefined launching units inside the system will result the influence of these changes: for disposal of launching units is good to take into account such possible changes, resulting in an over coating that will allow trajectories to be within the effective area.

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