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A Quick Approach to Correct Range Prediction of A Surface to Surface Rocket Fitted with a Nonstandard Fuze

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Abstract

Equipping a nonstandard fuze to an unguided artillery rocket could affect the rocket characteristics and hence different flight trajectory. Consequently, the firing tables provided by the rocket manufacturer are no longer accurate. This paper investigates a quick and low cost approach that can mitigate this problem. The approach was applied to a case study of a 122 mm artillery rockets fitted with a fuze whose shape and mass are different from the original design. Available data from live fire tests were utilized to evaluate the accuracy of the prediction. The results suggested that the error was higher at greater quadrant elevation and the error of one sample point near the maximum range was up to 7.8%.

Keywords*:* firing tables, artillery rockets, trajectory simulation, external ballistics.

Nomenclature

A_{ref}	Reference area (m^2)
a_x, a_y, a_z	Translations in the launching axes (m/s)
ac_x, ac_y, ac_z	Acceleration due to Earth's rotation in the launching axes (m/s^2)
C_A	Axial force coefficient
C_l	Rolling moment coefficient
C_{lp}	Rolling moment coefficient derivative with roll rate (1/rad)
$C_{m\alpha}$	Pitching moment coefficient derivative with angle of attack (1/rad)
C_{mq}	Pitching moment coefficient derivative with pitch rate (1/rad)
$C_{n\beta}$	Yawing moment coefficient derivative with side slip angle (1/rad)
$C_{N\alpha}$	Normal force coefficient derivative with angle of attack (1/rad)
$C_{Y\beta}$	Side force coefficient derivative with side slip angle (1/rad)
$Drift_{Nominal}$	Drift in nominal case (m)
$Drift_{Aero\ Var}$	Drift in aerodynamic coefficient variation case (m)
F_{dx}, F_{dy}, F_{dz}	Force due to disturbance in the launching axes (N)
F_{px}, F_{py}, F_{pz}	Force due to propulsion in the launching axes (N)
F_{rx}, F_{ry}, F_{rz}	Force due to aerodynamics in the launching axes (N)
F_{rbx}, F_{rby}, F_{rbz}	Force due to aerodynamics in the rocket body axes (N)
g_x, g_y, g_z	Acceleration due to Earth's gravity in the launching axes (m/s^2)
I_{bx}, I_{by}, I_{bz}	Moments of inertia of the rocket in the rocket body axes (kg.m^2)
L_{ref}	Reference length (m)
m	Total mass of the rocket (kg)
M_{rbx}, M_{rby}, M_{rbz}	Moment due to aerodynamics in the rocket body axes (N.m)
M_{dbx}, M_{dby}, M_{dbz}	Moment due to disturbances in the rocket body axes (N.m)
p, q, r	Angular rate of rocket body in the rocket body axes (rad/s)
QE	Quadrant elevation (mil, deg)
R_{Nom}	Range in nominal case (m)
$R_{Aero\ Var}$	Range in aerodynamic coefficient variation case (m)
V	Total velocity (m/s)
X_{cg}	Center of gravity position (m, caliber)
$X_{cg,ref}$	Reference center of gravity position when calculating aerodynamics (m, caliber)
α, β	Angle of attack and side slip (rad)
ρ	Atmospheric air density (kg/m^3)
σ	Standard deviation of range (m)
Δ_{Range}	Difference in range between nominal case and modified case (m)

1. Introduction

Computing firing data for unguided artillery rockets is a classic gunnery problem. The primary objective is to determine the azimuth and quadrant elevation for delivering an effective fire on the target under given conditions. The azimuth is the angle in the horizontal plane that determines the direction of fire. The quadrant elevation is the angle in the vertical plane that determines the range of impact point. To compute these two angles, artillerymen can follow standard procedures [1] and utilize either tabular firing tables in print or a fire control programs. These tools are normally provided by the manufacturer of that rocket system. Data in the firing tables are generated from a rocket trajectory model that has been verified by several live fire tests [2]. The generated data is often put in standard tabular format such as STANAG 4119 [3] and the verified trajectory model is employed in a fire control program. Since the geometry, mass properties, thrust, etc., are different from one rocket model to another, the firing tables and the rocket trajectory model are valid for one rocket with specific configurations only.

In some occasions, it is necessary to modify a rocket from its original design due to several possible reasons. Such modification is replacing an original fuze with another one not specified by the manufacturer. This kind of modification certainly affects the aerodynamic characteristic and mass properties of the rocket and consequently affects the rocket trajectory. As a result, the original firing tables and the fire control program supplied by the manufacturer are no longer accurate. In the ideal case, the aerodynamics of the rocket should be precisely evaluated again by a reliable method such as wind tunnel testing. New aerodynamic data and other parameters that are affected by the change should be updated in the trajectory model. Then the data in the firing tables can be regenerated and verified by some live fire tests. However, both time and budget are a major constraint in most cases. Moreover, requesting updated firing tables from the manufacture can incur extra cost and lead time. Therefore, there is a need for a low cost approach to counter this problem.

This paper investigates a quick and low cost approach to correct the range prediction of an unguided artillery rocket that is fitted with a nonstandard fuze. From the following sections, section 2 describes the problem background. The proposed approach is explained in section 3. For verification, selected data from live fire tests are presented and discussed in section 4. Section 5 concludes the paper.

2. Problem Description

The rocket presented in this paper is an unguided surface to surface artillery rocket that was acquired by Defence Technology Institute (DTI) of Thailand for its research and development works. The maximum range in the standard condition [4], i.e. standard temperature and pressure, sea level, etc., is above 40 km and the rocket can be equipped with a drag ring for range reduction. The rocket caliber is 122 mm and the total length is almost 3 m. The initial weight is slightly lower than 70 kg. The rocket is aerodynamically stabilized by utilizing 4 wrapped around fins. The rocket is originally designed to be used with MRV-U fuze but was replaced by the M423 fuze due to some constraints during the live fire tests.

MRV-U fuze is an impact fuze that is commonly used with several 122 mm artillery rockets. The warhead of this rocket was designed to fit smoothly with the fuze shape. The fuze has 140 mm of exposed length after assembled to the rocket. It weighs about 0.72 kg. On the other hand, M423 fuze is a fuze that is widely used with 2.75 inch rockets fired from helicopter or low speed aircrafts [5]. The fuze weighs 0.34 kg, which is about half of MRV-U fuze. An adapter was required to fit the M423 fuze to the 122 mm test rocket. It has 83 mm of exposed length. Note that these two fuzes are commonly used by armed forces in many countries so their specifications can be found in several references [6–9]. Some values in the references are be slightly different from data in presented this paper.

Fig. 1 presents both MRV-U and M423 fuze. MRV-U is twice the size of M423 and they have different screw thread size. An adapter was made to fit M423 to the rocket. Fig. 2 shows both fuzes assembled to the rocket warhead. It could be seen that the warhead with M423 fuze has more steps and the streamline looks less smooth. Fig. 3 shows the warheads when they are equipped with a drag ring. Note that all parts shown in the figures are dummy parts for demonstration only. But all dimensions are identical to the actual parts. Fig. 4 summarizes all rocket configurations: A) the original rocket with MRV-U fuze, B) the original rocket with MRV-U fuze and a drag ring, C) the rocket with M423 fuze, and D) the rocket with M423 fuze and a drag ring. Tabular firing tables are provided by the manufacturer for the rocket with MRV-U fuze in configuration A and B only.

Fig. 1 MRV-U (left) and M423 (right)

Fig. 2 Warhead with MRV-U (left) and M423 (right)

Fig. 3 Warhead equipped with a drag ring and MRV-U (left) and M423 (right)

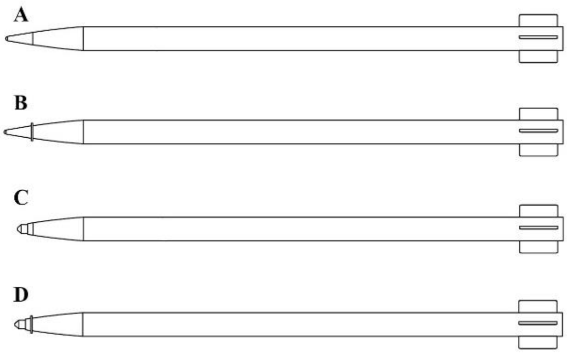

Fig. 4 Rocket configurations

It is quite expectable that changing a fuze from MRV-U to M423 would have affected the aerodynamics of the rocket because the external geometry of both fuzes are noticeably different. The mass and moment of inertia of the rocket could also be affected. Therefore, the original fire control program and firing tables provided by the manufacturer need to be updated with these new values so they can predict the impact points of the rocket accurately. However, it was impossible to update the fire control program and regenerate the firing tables because the source code of the program, thrust profile, and original aerodynamic parameters were not available.

3. Method Description

An approach described in this paper is proposed to predict the range of the modified rocket in the situation when

- the original fuze is replaced with another nonstandard one;
- firing tables of the rocket with the original fuze is provided;
- the source code of the firing control program and original aerodynamic data are not available.
- trajectory data from previous live fire tests are not available.

Overall, the working steps are similar to those in the development process of firing tables and a trajectory model for a new rocket. But the whole process requires much less time because original firing tables are already available. The data in the original firing tables are no longer valid since the rocket was modified from its original design. But they are still extremely valuable because they can be used as a benchmark. Fig. 5 summarize the main steps in the method. The first three steps are done to recreate a trajectory model, of which the results match or almost replicate those data in the original firing tables. The next two steps are done to correct parameters in the trajectory model to account for any modification on the rocket. The last step is utilizing the updated trajectory model to predict the impact point.

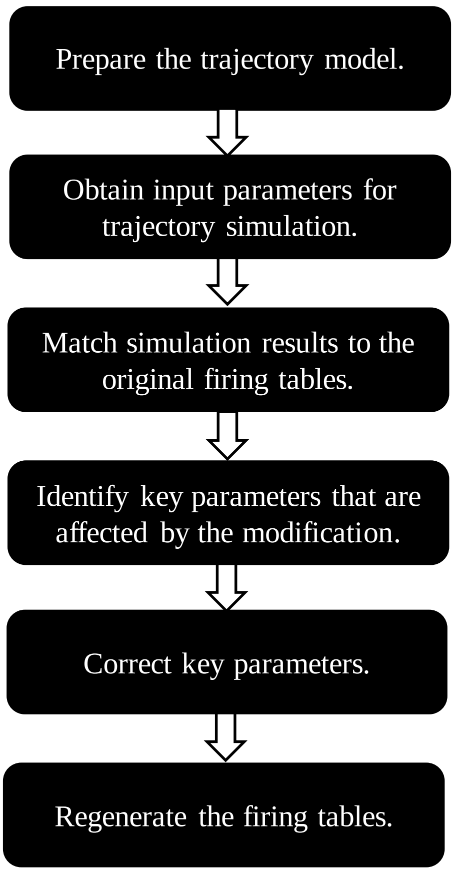

Fig. 5 Overall working steps

3.1 Preparing the Trajectory Model

The first step is to choose a trajectory model that is suitable for the problem. Several models for rigid body motion have been described in many references [10–15]. These references provide a very useful guideline to develop a trajectory simulation computer program. However, one can also utilize existing trajectory simulation program if it is already available. This is frequently the case for those who are working in this exterior ballistic area. Utilizing previously verified computer codes of other similar rockets can be a very convenient short cut and save a lot of time.

This paper utilizes a computer program that consists of a 6 degrees of freedom (6DOF) trajectory model and firing solution algorithms in our previous work [16], in which the trajectory model was formulated based on the literatures mentioned above. The motion was calculated in two axis systems. They are the rocket body axis and the launching axis system, as illustrated in Fig. 6. In the rocket body axis system, the origin is located at the initial center of gravity and it is moved together with the rocket. The X axis is pointing from the origin to the nose of the rocket and is coincident with the longitudinal axis. The Y axis is perpendicular to the X axis and pointing to the right side of the rocket body. The Z axis is pointing downward to satisfy the right hand axis system. In the launching axis system, the origin is fixed to the Earth and located at the initial position of the rocket center of gravity before launching. The X axis is pointing in the same direction of the launching azimuth. The Z axis is point downward vertically and is perpendicular to the ground surface. The Y axis is point to the right to complete the right hand axis system. Quaternions are utilized to formulate the transformation matrix and calculate attitude angles relative to the earth.

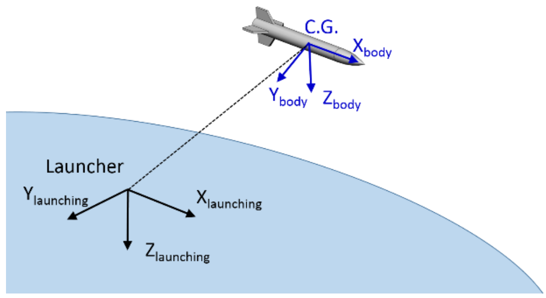

Fig. 6 An example of a thrust profile

Three translations are calculated in the launching axis by Eqs. (1) to (3). Three rotations are calculated in the rocket body axis by Eqs. (4) to (6) for convenience of calculating moment inertia properties. The disturbance forces and moments were neglected so F_{dx}, F_{dy}, F_{dz}, M_{dbx}, M_{dby}, M_{dbz} were set to zero. There is no thrust vectoring and thrust misalignment was neglected so F_{py}, F_{pz} were zero. In addition, WGS84 ellipsoid was utilized to represent Earth's shape. The calculation is performed in time marching scheme until the rocket hit the ground. The fourth order Runge-Kutta method was used for integration with the time step of 0.01 s.

$$a_x = \frac{1}{m}\left(F_{px} + F_{rx} + F_{dx}\right) + g_x + ac_x \tag{1}$$

$$a_y = \frac{1}{m}\left(F_{py} + F_{ry} + F_{dy}\right) + g_y + ac_y \tag{2}$$

$$a_z = \frac{1}{m}\left(F_{pz} + F_{rz} + F_{dz}\right) + g_z + ac_z \tag{3}$$

$$\dot{p} = \frac{1}{I_{bx}}\left(M_{rbx} + M_{dbx}\right) \tag{4}$$

$$\dot{q} = \frac{1}{I_{by}}\left(M_{rby} + M_{dby}\right) + \left(I_{bz} - I_{bx}\right)pr \qquad (5)$$

$$\dot{r} = \frac{1}{I_{bz}}\left(M_{rby} + M_{dby}\right) + \left(I_{bx} - I_{by}\right)pq \qquad (6)$$

Components of aerodynamics force and moment in the rocket body axis were estimated by Eqs. (7) to (12). The Magnus effect [17] and yawing moment derivative with pitch rate were neglected. From axisymmetry of the rocket shape, it was assumed that the magnitude of $C_{N\alpha}$ and $C_{Y\beta}$, $C_{m\alpha}$ and $C_{n\beta}$, C_{mq} and C_{nr} are equal [10]. Then the aerodynamic force components in the rocket body axis are transformed to the launching axis using a transformation matrix. Atmospheric pressure, density, and temperature were obtained by first order interpolation of meteorological data. The wind was taken into consideration when calculating the relative velocity of rocket body to the surrounding air.

$$F_{rbx} = \frac{1}{2}\rho V^2 A_{ref}\left(-C_A\right) \qquad (7)$$

$$F_{rby} = \frac{1}{2}\rho V^2 A_{ref}\left(-C_{Y\beta}\beta\right) \qquad (8)$$

$$F_{rbz} = \frac{1}{2}\rho V^2 A_{ref}\left(-C_{N\alpha}\alpha\right) \qquad (9)$$

$$M_{rbx} = \frac{1}{2}\rho V^2 L_{ref}\left(C_l + C_{lp}\,p\frac{L_{ref}}{2V}\right) \qquad (10)$$

$$M_{rby} = \frac{1}{2}\rho V^2 L_{ref}\left(C_{m\alpha}\alpha + C_{mq}\,q\frac{L_{ref}}{2V}\right) \qquad (11)$$

$$M_{rbz} = \frac{1}{2}\rho V^2 L_{ref}\left(C_{n\beta}\beta + C_{nr}\,r\frac{L_{ref}}{2V}\right) \qquad (12)$$

The change in center of gravity position during the flight due to burning propellant mass is taken into account by Eqs. (13) and (14).

$$C_{m\alpha} = C_{m\alpha,ref}\left(X_{cg} - X_{cg,ref}\right)/L_{ref} \qquad (13)$$

$$C_{n\beta} = C_{n\beta,ref}\left(X_{cg} - X_{cg,ref}\right)/L_{ref} \qquad (14)$$

3.2 Obtaining the Input Parameters for Trajectory Simulation

Parameters of the original rocket must be determined then input into the trajectory simulation program. These parameters include mass, center of gravity, moment of inertia, thrust data, and aerodynamic coefficients. To obtain accurate value of some parameters can be a challenging task and requires a lot of works. So some parameters that are difficult to measure are roughly estimated first. But they will be adjusted in the next step using data in the original firing tables as a benchmark.

Quality control reports or inspection sheets of each rocket can provide data on mass, center of gravity, and moment of inertia. These documents may be provided by the manufacturer at free of charge or minimum cost. If these documents are not available, weight scales can be used to measure mass and the center of gravity position. The moment of inertia requires more special tools but it can alternatively be estimated by computer aided design (CAD) software.

A thrust profile can be measured by performing a static test on the rocket motor. To conduct such a test, preparation works, experienced staffs, and a dedicated test facility are required. In an ideal situation, static tests should be repeated to see variation between each round. In addition, static tests should be performed at different propellant temperature because the burn time is affected by the propellant temperature [18]. Alternatively, the thrust data can be estimated using an internal ballistic software [19,20]. Fig. 7 shows an example of a thrust profile.

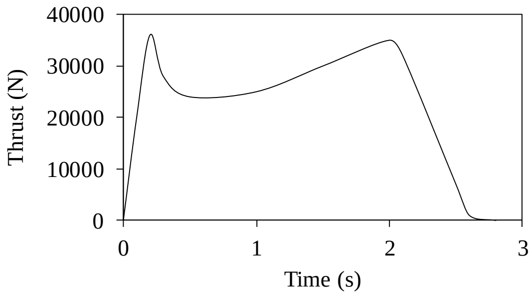

Fig. 7 An example of a thrust profile

To determine aerodynamic coefficients, one may conduct experiments in a wind tunnel or an aero ballistic range. Alternatively, computational fluid dynamic (CFD) simulation can be employed. Several advanced CFD techniques, such as CFD simulation in couple with rigid body dynamics [21], have been investigated and they could provide considerably accurate estimation. Furthermore, there are several aerodynamic prediction software that can quickly estimate the aerodynamic coefficients. These software are Missile DATCOM [22], PRODAS [23], MISL3 [24], and Aeroprediction [25], etc. These software require much less time and cost than experimental methods or CFD simulation. But they are suitable for some rocket configurations only. It was recommended that their results may not be accurate for all projectile configurations, Mach number, or angle of attack range [26–28].

In this paper, the mass properties of the rocket were obtained from the rocket data sheets. Most parameters were measured again by the quality control staffs. Digital scales with a resolution better than 10 g and the moment of inertia measurement tools with a resolution better than 0.005 kg.m^2 were employed to measure the mass, moment of inertia, and center of gravity position. Aerodynamic coefficients were simply determined using Missile DATCOM. In addition, a static test on the rocket motor was carried out to measure the thrust data. Unfortunately, only one static test was allowed. The test was conducted at the room temperature, which would not much differ from the temperature during the fire tests.

3.3 Matching the Simulation Results to the Original Firing Tables

Once the input parameters are obtained, the trajectory simulation code is ready to generate outputs. At first, the results may be very different from the data in the original firing tables because some input parameters are estimated inaccurately. In this step, some input parameters are adjusted until the simulation results at most quadrant elevations match the data in the original firing tables or have least error. This step is trial and error and can be tiresome because it involves many input parameters.

For simplification and shorten the process, the parameter adjustment should focus on the input parameters that are expected to be inaccurate but have a large effect on the range. To identify these parameters, a quick study on the effect of input parameters on the range of the impact point was carried out using a trajectory simulation program from the previous step. Previous research works [29,30] suggested that aerodynamic coefficients, especially the axial force coefficient, and some other input parameters had great effect on the range of impact point. So a brief study on the effects of aerodynamic coefficients on the range was carried out to confirm critical coefficients for our trajectory model. The aerodynamic coefficients included in the study were C_A, $C_{N\alpha}$, C_l, C_{lp}, $C_{m\alpha}$, and C_{mq}. For the axial force coefficient C_A, it is defined as $C_{A,Pon}$ for the period when the rocket motor is still burning and $C_{A,Poff}$ for the period after the period after the rocket motor has burnt out. Besides aerodynamic coefficients, the study were carried out on other input parameters including m, I_{bx}, I_{by}, I_{bz}, X_{cg}, T_x and error in launching QE in the test. This addition analysis provided better understanding of the sensitivity of the range to these input parameters.

Table 1 summarizes the variation used in the study. Variation of the aerodynamic coefficients was conservatively chosen based on the prediction method. The other parameters were chosen conservatively based on the measurement tools or expected production tolerance.

Table. 1 Variation of input parameters

Input Parameters	Variation
C_A , $C_{N\alpha}$, C_l , C_{lp} , $C_{m\alpha}$, C_{mq}	±20%
m	±0.1 kg
T_x	±0.5%
X_{cg}	±0.5 caliber
I_{bx}	±0.01 kg.m^2
I_{by} , I_{bz}	±0.1 kg.m^2
QE	±2 mil

The trajectory simulation was performed at quadrant elevation (QE) 533 mil (30°) and 800 mil (45°), which are around the firing angle that were initially expected to be used during the tests. The nominal trajectory was simulated then the nominal range was record. Next the input parameters were varied one parameter at a time and a trajectory simulation was performed again. The range of the impact point was recorded. The nominal range for QE 533 mil and 800 mil is about 30 km and 40 km respectively. Change Δ_{Range} in range relative to the nominal range is calculated by Eq. 15.

$$\frac{\Delta_{Range}}{R_{Nom}} = \frac{|R_{Aero\,Var} - R_{Nom}|}{R_{Nom}} \times 100\% \tag{15}$$

The results for the aerodynamic coefficients are presented in Figs. 8 and 9. It could be seen that C_A had large effect on range and $C_{A,Poff}$ had greater effect than $C_{A,Pon}$. The reason was obvious since the burn time of this rocket is very short or less than 3 s. In addition, Figs. 10 and 11 show that the effect of I_{bx}, I_{by}, I_{bz}, and X_{cg} on the range was quite small. This trend of results agrees with previous research works [29,30].

Following the study above, C_A was our main focus when trying to adjust input parameters to match the predicted range to the data in the original firing tables provided by the manufacturer. The value of m, I_{bx}, I_{by}, I_{bz}, X_{cg}, T_x were not adjusted.

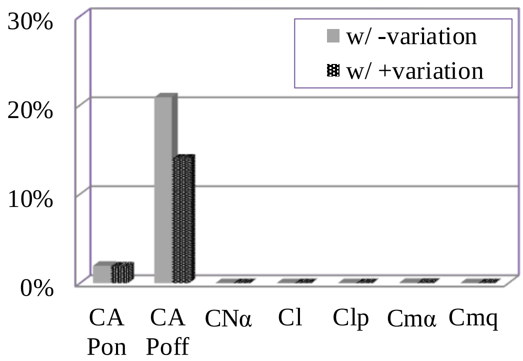

Fig. 8 Δ_{Range}/R_{Nom} due to aerodynamics at QE 30°

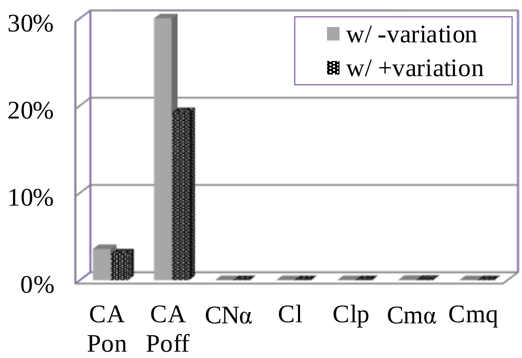

Fig. 9 Δ_{Range}/R_{Nom} due to aerodynamics at QE 45°

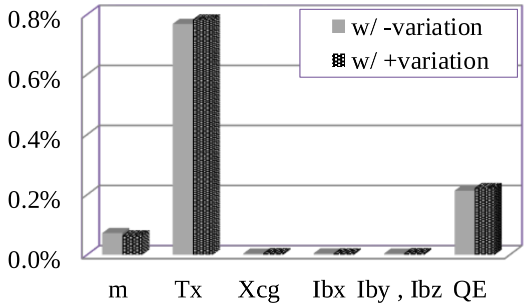

Fig. 10 Δ_{Range}/R_{Nom} due to other parameters at QE 30°

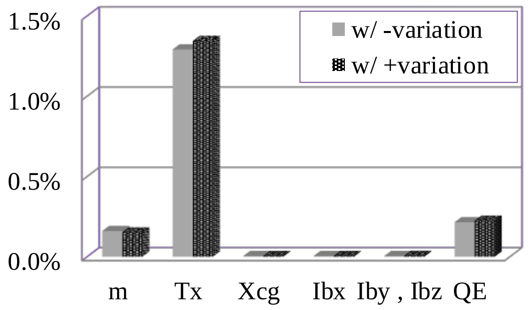

Fig. 11 Δ_{Range}/R_{Nom} due to other parameters at QE 45°

3.4 Identifying Key Input Parameters Affected by the Modification

All previous steps the focus on the rocket in the original configuration only. Their primary objective is to create a trajectory simulation program, of which the results are as close as possible to the data in the original firing tables. From this step, the input parameters will be updated again to account for changing fuze. But first it is necessary to identify which input parameters are affected by the modification.

Changing from MRV-U to M423 fuze would certainly change the aerodynamics because external geometry and dimensions of both fuzes were noticeably different. The PRODAS® software were employed to estimate the aerodynamic coefficients of the rocket fitted with both fuzes and make comparison. It was found that only the axial force coefficient was noticeably changed especially in the supersonic region. Fig. 12 compares the axial force coefficient of the rocket fitted with the original MRV-U fuze and the new M423 fuze.

Besides the aerodynamics, the mass of M423 was about half of MRV-U fuze. So these parameters needed to be updated too.

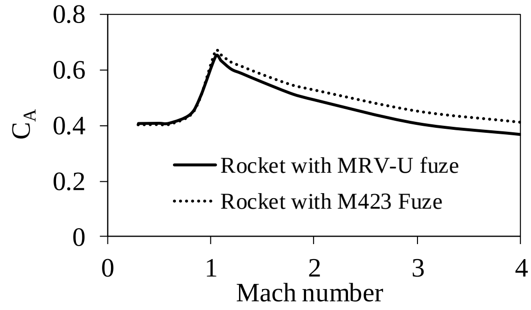

Fig. 12 Axial force coefficient of the rocket

3.5 Correct Key Parameters

There are two approaches to correct he input parameters in this step. The first one is to simply overwrite the old value with a new one. This approach is suitable when the input parameters can be measured accurately so we adopted it to update the mass properties.

The second approach is to relatively increase or decrease the old value by some ratios. This ratio can be calculated from the estimated new value to the estimated old value. This approach is suitable for the input parameters that are difficult to estimate or measure accurately. For example, the aerodynamic coefficients was estimated using a semi-empirical software instead of testing in a wind tunnel. So we used this approach to update the aerodynamic coefficients. The ratio of the new axial force coefficient ($C_{A,new}$) to the old one ($C_{A,old}$) was calculated, as shown in Fig. 13. Then it was applied for updating the old value. In addition, the ratio for the rocket with a drag ring were also determined and found that their trend was similar to Fig. 13. With higher axial force coefficient, it was expected that the range of the rocket fitted with M423 fuze would be decreased.

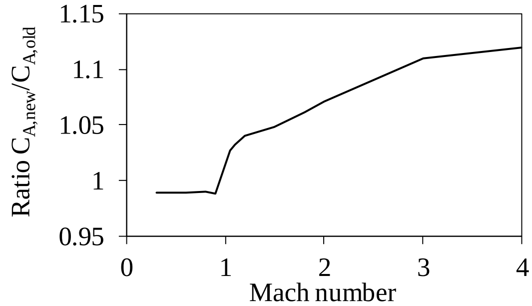

Fig. 13 Ratio of $C_{A,new}$ to $C_{A,old}$

3.6 Regenerating the Firing Table Data

After key input parameters that were affected by changing fuze had been updated, the trajectory simulation program should be rechecked for any possible error. The simulation results should be carefully examined whether they are reasonable. Then the updated simulation program is ready to be used to predict the impact points of the modified rocket and regenerate tabular firing tables. In the next section, the accuracy of the approach are evaluated by comparing its prediction to data that obtained from live fire tests.

4. Verification

4.1 Live Fire Tests

Available data from two live fire tests that were conducted by a joint effort between DTI, Royal Thai Navy, and Royal Thai Army were utilized for verification of the proposed approach. Both tests were conducted at the same location at sea level in a coastal test range in Thailand. It is important to note that these live fire tests were conducted for other research objectives so the data available for this study was limited and the quadrant elevation was not selected systematically. Yet these fire tests provided very useful data. Totally, impact point data of 9 rounds of this rocket are available for use in this paper. The rockets were fired from a multiple launched rocket system (MLRS) platform in different launch tubes. The dimensions and alignment of all tubes were inspected by the quality control staffs prior to the test to ensure that there was no significant variation between all tubes. Furthermore, the launch tubes were also inspected after firing to check whether any tube was damaged.

The impact points were detected using shipborne surveillance X-band radar systems, Raytheon Anschütz NSC-25 and Selex RAN-30X/I. The radar detected the water splash caused by the impact instead of detecting a high velocity rocket at the moment of impact. From environment conditions and equipment specification, it was conservatively estimated by the operators that the actual impact points were located within 90 m radius from the reported impact locations. The information on the velocity, flight path along the trajectory could not be measured.

Standard artillery computer meteorological message (METCM) [31] was obtained using the radiotheodolite VAISALA® RT20A system. Due transportation constraints, radiosondes were released near the launcher instead of an ideal location at the middle of the trajectory. This data was utilized in the trajectory simulation by using first order interpolation to obtain the atmospheric data at the desired altitude. Figs. A1 to A4 in the appendix presented the head wind and cross wind components and density that were determined from METCM. The head wind and cross wind were calculated relatively to launching azimuth. Atmospheric density was calculated using ideal gas law. Most Met Data sets contains data up to about 19 km altitude with the exception of Met Data 3 where the radiosonde unexpectedly stopped transmitting the data at 16.5 km altitude. But the data was sufficient because that the maximum ordinate of all rounds were estimated to be below 16.5 km.

Table 2 summarizes the test data. Round 1, 2, and 6 to 9 were obtained the first test while Round 3 to 5 were obtained from the second test. Round 1 to 3 were aimed for the range near the maximum range, which was expected to be 37 to 40 km. Round 4 and 5 were aimed at 28 km. Four rounds equipped with a drag ring were aimed at about 20 km.

Table. 2 Summary of prediction and test data

Round#	Test	Met Data	QE (mil)	Predicted Max Ordinate (km)	Predicted Range (km)	Test Results (km)	Error (km)	Error %
1	1st	Met 1	870	15.2	37.2	36.3	-0.9	-2.4%
2	1st	Met 1	900	16.2	37.9	36.3	-1.6	-4.4%
3	2nd	Met 2	900	16.1	37.8	35.1	-2.7	-7.8%
4	2nd	Met 3	565	7.0	28.0	28.4	0.4	1.5%
5	2nd	Met 3	565	7.0	28.0	28.3	0.3	1.1%
6	1st	Met 4	404	3.3	20.0	19.6	-0.4	-2.0%
7	1st	Met 4	404	3.3	20.0	19.7	-0.3	-1.4%
8	1st	Met 4	404	3.3	20.0	19.8	-0.2	-1.0%
9	1st	Met 4	404	3.3	20.0	19.8	-0.2	-1.0%

4.2 Measurement Accuracy and Dispersion of the Impact Points

Prior to comparing the test results to the predicted values, related factors that could possibly cause error in the test results should be discussed. So the limitation on the comparison can be clarified. First, the accuracy of the impact point detection was about 90 m as mentioned previously. So any reported range from the observer could be 90 m more or less.

Second, the dispersion due to the deviation between each round should be examined. It could be seen in the previous section that variation of rocket mass, thrust, and aerodynamics could cause deviation of the range. But the actual tolerance of the production process were normally kept within the manufacturers. Moreover, it was impossible to obtain reliable data of deviation between each round from limited number of available samples. However, the Circular Error Probable (CEP) is normally provided by the manufacturer. For this kind of unguided artillery rocket systems, including this one, the CEP is normally specified as better than 1.25% of range. So the CEP is better than 250, 350, and 450 m at the range of 20, 28, and 36 km. Note that CEP is the circle where half of rounds will theoretically fall within. The relation between CEP and standard deviation (σ) is described as CEP = 1.1774σ [14,32].

Third, the ±2 mil error in QE in our launcher could be estimated. From the simulation, increasing QE 1 mil can increase range 29, 29, and 26 m at QE equal to 404, 565, and 900 mil. Normally, the effect of the launcher is already included in the CEP of a rocket system. But we will conservatively add the launcher error into our consideration.

In summary, the detection error, launching error, and dispersion from the rocket were considered as primary factors that affect the accuracy of the test results presented in this paper. A method to calculate the total dispersion of a rocket system [14] is adapted to calculate the standard deviation of the test results in this paper. The standard deviation of the test results (σ_{result}) is estimated from a square root of the sum of all squared standard deviation from detection ($\sigma_{detection}$), launching ($\sigma_{launching}$), and rocket dispersion (σ_{rocket}), as described in Eq. 15.

$$\sigma_{result} = \sqrt{\sigma^2_{detection} + \sigma^2_{launching} + \sigma^2_{rocket}} \quad (15)$$

Let assume that the detection error and launch angle error discussed above are ±1.96σ for 95% confident level. Then σ_{result} is be calculated by Eq.15 and presented in Table 3.

This 1.96σ_{result} quantity was used a criteria when distinguishing the prediction error from the distribution of the test results. If the predicted range differed from the test result less than 1.96σ_{result}, it would be unclear whether the difference was merely caused by the prediction error. In other words, the prediction error smaller than this quantity could not be clearly identified. But there were exceptions such as bias when the error is always negative or positive. On the contrary, one may recommend that the prediction practically doesn't need to be more accurate than the dispersion of the rocket although we should try to make the prediction as accurate as possible.

Table. 3 Total standard deviation of the test results

Nominal Range (km)	QE (mil)	1.96 σ_{result} (m)	1.96 σ_{result} (of range)
20	404	272	1.4%
28	565	366	1.3%
36	900	462	1.3%

4.3 Suggested QE Before and After Correction

Using the predicted range in Table 2 as aiming range, the QE to achieve this range suggested by the original firing tables is presented in Table 4. Although it is obvious, they are compared to the suggested QE after correction to confirm that the original firing tables could be invalid when the fuze is replaced with another one not in the original design. It could be easily seen in Table 4 that the suggested QE before and after the correction are quite different.

Table. 4 QE suggested by before versus after correction

Round #	Aimed Range (km)	QE Before Correction (mil)	QE After Correction (mil)
1	37.2	727	870
2	37.9	751	900
3	37.8	733	900
4	28.0	462	565
5	28.0	462	565
6	20.0	340	404
7	20.0	340	404
8	20.0	340	404
9	20.0	340	404

4.4 Result Discussion

For Round 1 to 3, which were aimed at near the maximum range, the error was -2.4, -4.4, -7.8%. The error was more than 1.96σ_{total}. So it was suggested that the predicted range had some error at the nominal range near the maximum range. In addition, all rounds fell short so there could be some bias in the prediction.

It was noticed that Round 1 and 2 were tested almost under the same conditions. They were fired in the first test with about 30 minutes apart. So the same meteorological data was utilized for both rounds. Round 2 was fired at QE 30 mil higher than Round 1. So it was predicted that Round 2 would impact at 0.7 km further. However, both rounds impacted at almost the same range. At first, it was suspected that there might be problems during the launching. A high speed camera showed no anomaly during the first 15 m of the trajectory. Many observers at the launch site also reported no signs of any anomaly. One possibility was that the rocket might had reached its maximum range already so increasing QE would not result in increasing range. But the simulation suggested that maximum range would be achieved at much higher QE. Another possibility was that the wind condition might had changed during the first round and the second round.

In Round 4 and 5, which were aimed at 28 km, the error was 1.1 and 1.5%, which is about 1.96σ_{total}. So we could not distinguish between the error from the test results and the prediction. Both rounds fell further than the predicted range but there were just two samples at this range.

In Round 6 to 9, which were aimed at 20 km, the error is 1 to 2%, which is greater than 1.96σ_{total}. Furthermore, all four rounds fell shorter than the predicted range. So the results indicated that there was prediction error, which contains some bias.

Overall, the proposed method produced up to 7.8% error. The prediction error seemed to be higher at greater QE or range. This trend was reasonable because C_A is the primary input parameter that was adjusted through the process. It was shown in Figs. 8 and 9 that the effect of C_A on range is greater at higher QE or range. So if C_A was not adjusted properly, the error would have become obvious at higher QE or range. To improve the accuracy, better tools, such as CFD simulation, could be employed to analyze the aerodynamics before and after the new fuze is installed. So the aerodynamic coefficients can be adjusted more precisely to account for the fuze change.

4.5 Limitations

The limitations of this work are addressed. First, more rounds should be included and the quadrant elevation to be fired at should cover the whole range of firing tables. However, such a dedicated test would requires a lot of resources, i.e. money, man power, etc.

Second, the accuracy of the impact point measurement should be improved. More sensors should be employed to record position, velocity, attitude, etc., during the flight. These data will allow us to adjust the trajectory model better.

Third, the radiosondes were released near the launch site and METCM does not account for variation of in horizontal dimensions. So the obtained meteorological data was unable to represent the actual weather conditions along the flight path. This problem can be mitigated if a gridded meteorological data (METGM) [33] is utilized.

Finally, the test results should be used to adjust the trajectory model again so it would be able to predict the impact point of this rocket when it is equipped with M423 fuze.

5. Conclusion

An approach to correct the range prediction of an unguided surface to surface rocket was investigated in this paper. The accuracy of the predicted range after correction was evaluated by comparing the prediction to the data obtained from other live fire tests. Due to limitations of number of available samples and the dispersion nature of the impact points, the accuracy of the approach could not be precisely determined. However, errors of the prediction results are presented. The results suggested that error was higher at greater QE or range and the prediction error at one sample was up to 7.8%.

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Appendix

Figs. A1 to A4 presents the meteorological data that were measured approximately one hour before the test. Head wind is positive when the winds blows against the rocket. Cross wind is positive when the wind blows from right to left (looking forward from the rocket rear end).

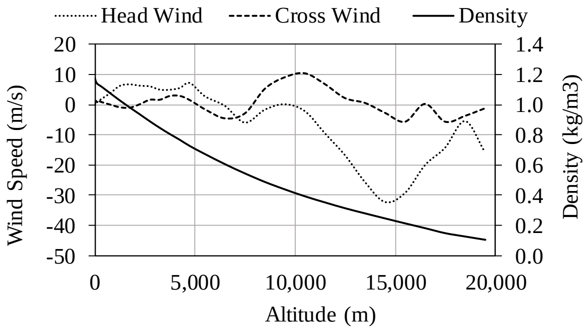

Fig. A1 Met Data 1 (used for Round 1 and 2)

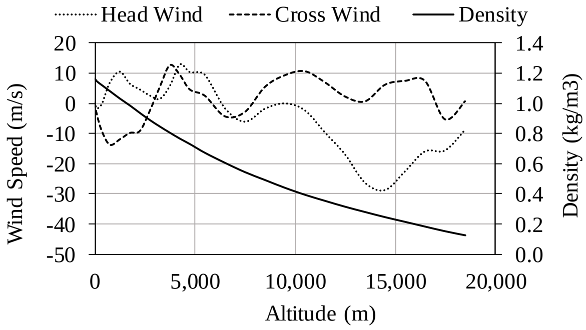

Fig. A2 Met Data 2 (used for Round 3)

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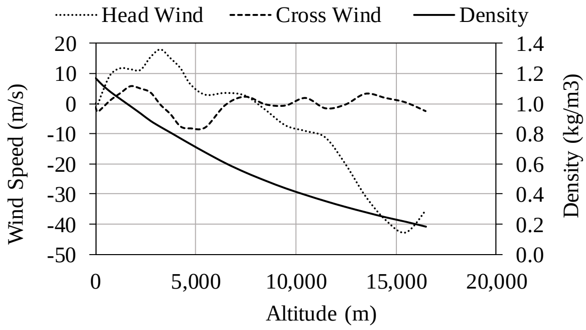

Fig. A3 Met Data 3 (used for Round 4 and 5)

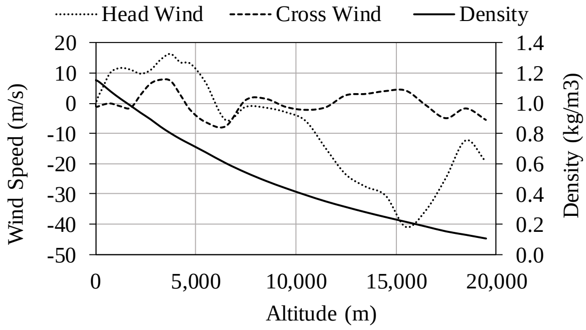

Fig. A4 Met Data 4 (used for Round 6 to 9)