THE COMPARATION OF TWO LAPAN'S PROPELLANTS OF DIFFERENT COMPOSITION

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ABSTRACT

This paper compares two LAPAN's propellant compositions. The A propellant has a composition of AP/AL/Binder with ratio of 70/10/20, and propellant B has ratio 75/7.5/17.5, both with HTPB base but different curing agent. As comparator of this simulation is RX-1512.01 rocket that has flight tested. The performances which compared are specific impulse and efficiency characteristic.

Result shows that the propellant B gives higher specific impulse, however higher losses is, about 5.373%. One to be considered is that the propellant B produce combustion temperature higher than propellant A by ± 200 K, that may influence the rocket structures.

ABSTRAK

Paper ini membandingkan dua buah komposisi propelan LAPAN. Propelan A dengan komposisi AP/AL/Binder 70/10/20, sedangkan propelan B 75/7.5/17.5, keduanya dengan dasar HTPB tetapi dengan *curing agent* yang berbeda. Sebagai dasar pembanding adalah basil simulasi roket RX-1512.01. Performa yang dibandingkan adalah *impuls* spesifik yang dihasilkan serta efisiensi karakteristik.

Hasil yang diperoleh menunjukkan bahwa propelan B memberikan *impuls* spesifik yang lebih besar, tetapi justru mengalami rugi-rugi yangjuga lebih besar, yaitu 5,373%. Satu hal yang harus dipertimbangkan adalah bahwa propelan B memiliki temperatur pembakaran yang lebih besar + 200 K, yang dapat mempengaruhi struktur roket

Kata kunci: *Impuls specific. Characteristic efficiency, Propellant, Simulation*

1 INTRODUCTION

Specific impulse, *Isp,* is the thrust force per unit weight of propellant. Specific impulse is important figure of merit of the performance of a rocket propulsion system, which is similar to the concept of miles per gallon parameter used with automobiles [Sutton, 2001]. A higher specific impulse means better performance.

Beside specific impulse, characteristic velocity, C (m/s), is independent of nozzle characteristics and can be determined by experimental data of chamber pressure, throat diameter, and propellant mass flow rate. C" efficiency is also an important performance parameter as it is used to express the degree of completion of the energy release and the creation of high temperature and pressure gas in the chamber [Sutton, 2001].

Jihad et. al., 2006, has evaluated five methods to determine specific impulse, ***Isp,*** to featuring propulsion system performance. The previous research used RX-1512.01 data of LAPAN's rocket (see table 3), obtained different value between five methods used. The researcher concluded that method number four is most suitable to calculate specific impulse. Research conducted of LAPAN propellant of AP/AL/Binder composition with ratio 70/10/20, in this papers named by propellant A. specific impulse obtained is 225,273 second *(Isp_4)*, the method that suggested.

Data taken to the calculation are result of simulations using software developed by Propulsion Department of LAPAN, and GDL ProPEP software (Gas Dynamics Lab. Ver. 1.2 by James E. Lanier). This software based on PEP (Propellant Evaluation Program) written by DR. Cruise at NWC that described on NWC TP 6037 with subject Theoretical Computations of Equilibrium Compositions, Thermodynamic Properties, and Performance Characteristics of Propellant Systems".

This research, continuing previous effort, aims to compare characteristic witii another propellant composition produce by LAPAN's, named propellant B. Composition of propellant B is AP/AL/Binder with ratio 75/7,5/17,5. The five methods of determining specific impulse will be discussed again, followed by the expression for characteristic velocity, which includes C* theoretical, C* actual, and C* efficiency.

2 Isp CALCULATION METHODS

To reviewing equation will be used, below presented each method and its equation.

2.1 Impulse Specific Equations

2.1.1 Method 1

Method 1 is the direct measurement of rocket thrust and propellant mass flow rate. The specific impulse (sec) is defined as thrust force, F, divided by the sum of propellant mass flow rate, w_p, (oxidizer mass flow rate w_0, and fuel mass flow rate, Wf, for liquid rocket). Here go is the gravitational acceleration at sea level, and g_c is the dimensional conversion constant 32.174 (lb_m-ft)/(lbr sec^2) in the English system and 1.0 (dimensionless) in the SI system [Sutton, 2001; Law, 2003].

For solid rocket,

$$Isp_1 = \frac{F}{w_p}\frac{g_c}{g_o} \qquad (2\text{-}1)$$

And for liquid rocket,

$$Isp_1 = \frac{F}{(w_o + w_f)}\frac{g_c}{g_o} \qquad (2\text{-}2)$$

2.1.2 Method 2

Method 2 involves the chamber pressure and area to determine thrust along with propellant mass flow rate. The specific impulse is a function of thrust coefficient, CF, stagnation chamber pressure, P_c, throat diameter, D_t, and propellant mass flow rate. The constants, go and gc, are the same for method 1 [Law, 2003].

$$Isp_2 = \frac{C_F.P_c.\pi.D_t^{\,2}}{4w_p}\frac{g_c}{g_o} \qquad (2\text{-}3)$$

The thrust coefficient can be thought of as representing the amplification of thrust due to the gas expanding in the supersonic nozzle as compared to the thrust that would be exerted if the chamber pressure acted over the throat diameter only. It is a convenient parameter for seeing the effects of chamber pressure or altitude variations in a given nozzle configuration, or to correct sea-level result for flight altitude condition. Thrust coefficient is a function of pressure ratio P_c/P_e, nozzle area ratio (expansion ratio), e = Ae/A_t, and specific heat ratio [Sutton, 2001; Davenas 1993].

$$C_F = \sqrt{\gamma}.\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{2(\gamma-1)}}\sqrt{\frac{2\gamma}{\gamma-1}\left[1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right]} - \frac{A_e}{A_t}\left(\frac{P_e}{P_c}-\frac{P_a}{P_c}\right) \qquad (2\text{-}3a)$$

2.1.3 Method 3

Method 3 involves the use of ideal rocket relationships. The method evaluates specific impulse of rocket in terms of a chamber temperature, T_c, chamber pressure, P_c, exit pressure, P_e, ambient pressure, P3, propellant mass flow rate, w_p, and throat diameter, D_t. the relation is based on the assumption that there is no change in the composition of the exhaust gas as it progresses through the nozzle. Again, the constants, g_0 and g_c, are the same as previous methods [Law, 2003].

$$Isp_3 = \sqrt{\frac{2R_c g_c}{g_o^2}\left(\frac{\gamma}{\gamma-1}\right)\left(\frac{T_c}{M}\right)\left[1-\left(\frac{P_e}{P_c}\right)^{\left(\frac{\gamma-1}{\gamma}\right)}\right]} + \frac{(P_e - P_a)\pi D_e^2 g_c}{4 w_p g_o} \quad (2\text{-}4)$$

Isentropic expansion relations in the rocket nozzle are assumed, where the maximum heat is converted to' kinetic energy of the jet. The energy loss to the wall is also neglected because the losses are difficult to measure and usually very small in nozzles. The ideal rocket relationships assume well-designed supersonic nozzles where the conversion of thermal energy into directed kinetic energy of the exhausted gases proceeds smoothly without normal shocks or discontinuity. Any increase in the chamber temperature (usually caused by increase in energy release) or any decrease of molecular mass at the propellant will help improve the performance of the rocket; thus, increase the specific impulse [Sutton, 2001].

2.1.4 Method 4

Method 3 uses one-dimensional model equation with the assumption that the exhaust gases exiting the nozzle are axially directed. In reality, the gases exiting are directed at an angle to the motor centerline depending upon the nozzle geometry curvature. This results in a loss of propulsive efficiency due to nozzle divergence effects. In order to compensate for the non-axial behavior of the exhaust gas velocity profile, a theoretical correction factor can be applied to the momentum term of the thrust equation. Method 4 considers nozzle divergence angle effect on performance [Ostlund, 2002; Sutton, 2001; Davenas, 1993; CLeary, 1992; Stitt, 1990]. The nozzle divergence factor, AN, (some authors like O'Leary write as Ageo, or just A) is defined as

$$\lambda_N = \frac{1}{2}(1+\cos(\alpha)) \quad (2\text{-}5)$$

where a represents the nozzle divergence half angle. Small nozzle divergence half angles cause most of the momentum to be axial, and thus give a high specific impulse. However, the long nozzle has a penalty in rocket propulsion system mass, vehicle mass, and also design complexity, and vice versa. Modifying the method 3 specific impulse expression to account for the nozzle divergence effect yields,

$$Isp_4 = \lambda_N\sqrt{\frac{2R_c g_c}{g_o^2}\left(\frac{\gamma}{\gamma-1}\right)\left(\frac{T_c}{M}\right)\left[1-\left(\frac{P_e}{P_c}\right)^{\left(\frac{\gamma-1}{\gamma}\right)}\right]} + \frac{(P_e - P_a)\pi D_e^2 g_c}{4 w_p g_o} \quad (2\text{-}6)$$

2.1.5 Method 5

Method 5 evaluates specific impulse assuming chemical equilibrium is maintained during the nozzle expansion process. In this case, reaction due to high temperature dissociation and recombination of the single fluid are considered. The specific impulse is based on the exit velocity assuming a negligible nozzle gas flow inlet velocity.

The specific impulse can be related in terms of chamber temperature, nozzle exit temperature, and pressure, and ambient conditions:

$$Isp_5 = \sqrt{\frac{2C_{p,ef} g_c}{g_o^2}\left(\frac{T_c - T_e}{M}\right)} + \frac{(P_e - P_a)\pi D_e^2 g_c}{4 w_p g_o} \quad (2\text{-}7)$$

Typically, iterative calculations are performed in determining the level of dissociation at several points in the nozzle to obtain the temperature of the exhaust gas. The method used assumes both a chamber and exit temperature can be measured in order to asses the performance [Law, 2003].

2.2 Characteristic Velocity

2.2.1 C* Theoretical

For the ideal case, the maximum value of C* is a function of gas properties such as specific heat ratio, y» chamber temperature, Ti, universal gas constant, Re, and molecular mass, M.

$$C^*_{teo} = \frac{\sqrt{\gamma \frac{R_c}{M} T_1 g_c}}{\gamma\sqrt{\left[\frac{2}{(\gamma+1)}\right]^{\frac{(\gamma+1)}{(\gamma-1)}}} \quad (2\text{-}8)$$

2.2.2 C* Actual

Characteristic velocity, C^*_{act} (m/sec), is independent of nozzle characteristic and can be determined by chamber pressure, P_c, throat area, A_t, throat diameter, D_t, and total propellant mass flow rate, $\dot{m}$.

$$C^*_{act} = \frac{P_c.A_t.g_c}{m} = \frac{P_c.\pi.D_t^{\ 2}.g_c}{4w_p} \qquad (2\text{-}9)$$

2.2.3 C* Efficiency

C^*_{eff} can be determined by the ratio of the actual value, as determined from equation (2-9) and the theoretical value, as obtained from equation (2-8),

$$\eta_{C^*} = \frac{C^*_{act}}{C^*_{teo}} \qquad (2\text{-}10)$$

C^*_{eff} is an important performance parameter that is used to express the degree of energy release and the creation of high temperature and pressure gas in the chamber.

3 CALCULATED PERFORMANCE PARAMETER

3.1 ISP

Data of propellant A as presented on table la and lb, taken from Jihad, et al., 2006. By entering the propellant B composition (weight fraction), initial propellant temperature, chamber and ambient pressure to GDL ProPEP software resulting as presented in Table 3-2a, and 3-2b. Combustion gases are known to remain in the nozzle for a period of 10^4 to $10''^3$ second; that information permits to set the solution between two extreme models [Filipovic, 2003]; a). Gaseous mixture, generated by the burning of solid propellant, stays frozen, frozen equilibrium expansion, or b). Gaseous mixture stays in chemical equilibrium, which depends on local pressure and temperature in any moment of expansion, shifting equilibrium expansion.

In this simulation, combustion process and exit gases through the nozzle assume as frozen, thus the parameters used to calculate the specific impulse taken from the tables above in row of frozen. Table 3-3, represent the thrust prediction using Grain software that developed by Propulsion department of LAPAN.

Using data represented above, the Isp theoretical calculation are:

- Method 1:
  Value of w_p obtained by divide propellant mass to burning time, here w_p = 24.466/4.6 = 5.319 Kg/s; F = 1115 Kg (obtained from static test result), thus get Ispi = 209.638 second.
- Method 2:
  CF obtained from equation (2-3a) is 1,535, P_c = 45 Kg/cm^2, throat diameter, D_t = 46 mm. Using equation (2-3) obtained Isp2 = 215,761 second.
- Method 3:
  Re = 8314,3 J/Kg-mol. K; obtained M = 24,721 g/mol; chamber temperature, Tc = 2766 K; specific heat, Y - 1.2103; exit pressure, P_e = 1.03 Kg/cm^2; pressure chamber, P_c = 45 Kg/cm^2. Substituting these values to equation (2-4), obtained Isp3 = 232.272 second.
- Method 4:
  The divergence half angle of RX-1512.01 nozzle is 13°, using equation (2-5), correction factor of divergence half angle nozzle, AN = 0,9872, thus Isp4 = 229,299 second.
- Method 5:
  Exhaust gas temperature, T_e = 1380 K, Cpeff = 42.030, obtained Isp_5 = 236,362 second.

3.2 Characteristic Velocity

By entering the values of each variable obtained from previous calculation, the characteristic velocity are,

$$C^*_{teo} = \frac{\sqrt{1{,}2103\ \frac{8314{,}3}{24{,}721}.2766\ .9{,}8}}{1{,}2103\ \sqrt{\left[\frac{2}{(1{,}2103\ +1)}\right]^{\frac{(1{,}2103+1)}{(1{,}2103-1)}}}} = 1{,}486\ .10^3$$

Its actual value is

$$C^*_{act} = \frac{45.\pi.(4{,}6)^2.9{,}8}{4.(5{,}319)} = 1{,}406.10^3$$

These two values give, η_C = 94.627% that indicates any losses about 5,373%.

Table 3-1a: SIMULATION RESULT AT P_c = 70.31 Kg/cm²; P_a = 1.034 Kg/cm² (PROPELLANT A)

Condition	T (K)	P (atm)	Combustion Enthalpy	Entropy	C_p/C_v (γ)	Gas	RT/V	BM	ρ (gr/cm³)
Chamber	2576	68.02	-52.45	238.35	1.2159	4.234	16.031	22.538	1.6325
Exit	1201	1	-118.96	238.35	1.2635	4.236	0.225	22.617	-

Performa:

Equi	Isp_{id}	Is Ex	T*	P*	C*	ε_{opt}	D-Isp	T_e
Frozen	236.7	1.2329	2307	37.97	4751.0	8.63	386.3	1150
Shifting	240.6	1.2048	2336	38.29	4821.3	8.72	392.8	1201

Table 3-1b: SIMULATION RESULT AT P_c = 45 Kg/cm²; P_a = 1.034 Kg/cm² (PROPELLANT A)

Condition	T (K)	P (atm)	Combustion Enthalpy	Entropy	C_p/C_v (γ)	Gas	RT/V	BM	ρ (gr/cm³)
Chamber	2571	45	-52.51	242.09	1.2160	4.245	10.257	22.574	1.6325
Exit	1312	1	-114.27	242.09	1.2589	4.236	0.225	22.613	-

Performance:

Equi	Isp_{id}	Is Ex	T*	P*	C*	ε_{opt}	D-Isp	T_e
Frozen	228.0	1.2294	2306	24.33	4760.7	6.27	372.1	1260
Shifting	231.9	1.1372	2408	25.12	4958.4	6.15	378.5	1312

Table 3-2a: SIMULATION RESULT AT P_c = 70.31 Kg/cm²; P_a = 1.034 Kg/cm² (PROPELLANT B)

Condition	T (K)	P (atm)	Combustion Enthalpy	Entropy	C_p/C_v (γ)	Gas	RT/V	BM	ρ (gr/cm³)
Chamber	2775	68	-53.87	238.48	1.2103	4.060	16753	23.818	1.6642
Exit	1335	1	-122.80	238.48	1.2510	4.045	0.235	23.902	1.6642

Performance:

Equi	Isp_{id}	Is Ex	T*	P*	C*	ε_{opt}	D-Isp	T_e
Frozen	241.1	1.2251	2494	38.07	4839.3	8.76	401.3	1266
Shifting	244.9	1.1640	2568	38.88	4987.6	8.79	407.6	1335

Table 3-2b: SIMULATION RESULT AT P_c = 45 Kg/cm²; P_a = 1.034 Kg/cm² (PROPELLANT B)

Condition	T (K)	P (atm)	Combustion Enthalpy	Entropy	C_p/C_v (γ)	Gas	RT/V	BM	ρ (gr/cm³)
Chamber	2766	45	-53.87	242.08	1.2106	4.064	10.715	23.798	1.6642
Exit	1454	1	-117.78	242.08	1.2466	4.045	0.235	23.901	1.6642

Performa:

Equi	Isp_{id}	Is Ex	T*	P*	C*	ε_{opt}	D-Isp	T_e
Frozen	232.1	1.2225	2489	24.39	4844.9	6.34	386.2	1380
Shifting	235.8	1.1621	2563	24.90	4985.0	6.36	392.5	1454

Table 3-3: SIMULATION DATA OF RX-1512.01

Propellant grain configuration	Star,7
Propellant weight	24.46 kg
Thrust prediction: Average Maximum Minimum Thrust (from static test)	 1050 Kg 1208 kg 975 kg 1115 Kg
Pressure: Average Maximum Minimum	 45 Kg/cm2 52 Kg/cm2 42 Kg/cm^2
Burning time	4,6 second
Nozzle: Throat diameter Exit diameter Inlet diameter Divergence half angle	 46 mm 120 mm 136.47 mm 13 degree

[Source: Tim Rekayasa Roket Detekgan, 2003]

4 DISCUSSION

Before compares Isp calculation result between two type of propellant, i.e. propellant A and propellant B, it is better to discuss the differences of simulation result first (see Table 3-la, 3-lb, 3-2a, and 3-2b). To obtain comparable data between propellants, the test must be performed with identical rocket motors. The propellant grain geometries used are well suited to obtain the desired precise data.

Simulation shows that the density of propellant B higher than A, which gives higher volumetric loading. With the same propellant grain and volume, so propellant B will give higher mass flow, of course will give higher thrust also. But, because the burning direction is radial, same configuration and burning rate, assumed these propellants have same burning time value, 4.6 second that obtained from simulation, and F = 1050 Kg. The static test gives difference value, *F* ** 1115 kg. We can conclude that increasing density will increase volumetric loading, so, pressure exponential at burning time has to be corrected for getting real burning rate. Logically, higher volumetric loading will decrease burning rate due to burning time increase. Isp calculation is presented in Table 4-1

In method 1, the differences in result caused by higher thrust value used to calculate Ispi, like stated above, the value obtained from static test, compare with propellant A, where value of thrust obtained from software simulation. Therefore, Ispi for propellant B higher than propellant A. While, Isp2, for propellant B lower than A, this is caused by higher of propellant mass flow of propellant B, although with same throat. Software that developed by Propulsion Department obviously cannot give an accurate result about density to burning time differences. Whereas, Isp2 equation, for both propellant which all variable keep constant and the difference just specific heat ratio, Y> and propellant mass flow, *m.*

Opposite result gained uses Isp3 equation, propellant B give higher specific impulse than propellant A. This equation is an ideal specific impulse. Could be understood that the increasing of Isp caused by increasing of chamber temperature (see table 3-la through 3-2b, where propellant B has higher chamber temperature). The increasing of chamber temperature absolutely influence by composition and ingredient of propellant. Method 4, with same divergence half angle of RX-1512.01 nozzle, of course propellant B gives higher specific impulse. Obtained the specific impulse for propellant B is, Isp_4 = 229.299 second.

Table 4-1:RESULT OF ISP CALCULATION BETWEEN PROPELLANT A AND B

Propellant	Isp Method (second)					ηc(%)
	Isp_1	Isp_2	Isp_3	Isp_4	Isp_5	
A	201.25	219.6	228.65	225.723	232.987	96.20
B	209.638	215.761	232.272	229.299	236.362	94.627

Significant difference shown when calculates using Isp.s equation. Considering equation (2-7), the dominant factors are temperature differentiation and effective molecular weight combustion gas, M. Theoretically, when chamber temperature increase, the molecular mass of gas as combustion product decrease, and higher specific impulse will be obtained. Table 3-la through 3-2b shows this case.

Table 4-1 shown that propellant B has higher loss than propellant A. Couldn't determined yet what kind of losses as dominant factors. Further more, see O'Leary, 1992 and Gokhale, 1989. However, because *r\c* is independent and specific to nozzle characteristic, therefore an effort to redesign *nozzle* that suite with propellant B is needed. As seen on table 4, that propellant B gives higher Isp, thus, propellant B is better than A. One to be considered that propellant B has higher temperature than A, this is an important parameter to design the structures of rocket motor including the nozzle.

5 SUMMARY

Five methods that used for calculate Isp express the difference result for two propellant considering. Propellant B gives higher Isp than propellant A, but with one concern that propellant B has higher temperature that might influences whole of the structures of rocket motors particularly nozzle structure.

In conclusion, the great caution is necessary when discussing specific impulse. Indeed, a rigorous performance comparison between various propellants requires; identical rocket motors (shape, mass, insulation, and contour and material of the nozzle, etc.); operating points corresponding to standard condition; test condition and equipment sufficient to secure a good level of precision.

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