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# Design and Performance Characteristics of a Rocket Using Potassium Nitrate and Sucrose as Propellants

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**Abstract:** ***The field of rocketry is one of the most complex and expensive on the planet today, where only the most developed and rich countries delve into its full applications. Rocket propulsion is a very complex field where lots of equations are involved. When designing, a lot of factors and conditions are considered because it has to deal with explosives and it can be quite expensive. Its complex thermodynamics doesn't allow much room for error as a simple miscalculation can be quite devastating. This work simulate the ideal theory equations which govern the working of the rocket using MATLAB and crating a virtual way of determining and improving it performance before it is being launched. By so doing, certain dangers and expenses can be avoided. The parameters determined through MATLAB were in close agreement with the experimental results by Nakka. The predicted operating pressure of $KNO_3$ propellant is 19 MPa, this in not beyond what stainless steel material can withstand and the thrust coefficient are in a range of 1.5 and 1.9. It was extended to predict variation of pressure with time and the thrust generated which could carry the load effectively.***

**Keywords:** Propellant, Specific Impulse, Burning time, Chamber pressure, MATLAB.

## 1. Introduction

Rocket engines are reaction engines and obtain thrust in accordance with Newton's third law [15]. It is a machine that develops thrust by the rapid expulsion of matter. This matter is a high-speed fluid that is nearly always a gas which is created by high pressure combustion of solid or liquid propellants, consisting of fuel and oxidizer components, within a combustion chamber [5, 6]. The major component of a chemical rocket assembly are rocket motor or engine propellant consisting of fuel and oxidizer, a frame to hold the components, control systems and a cargo such as satellites [8]. A rocket differs from other engines in that it carries its fuel and oxidizer internally, therefore it will burn in the vacuum of space as well as within the earth's atmosphere [9, 14, 18].

Rocket propellant is mass that is stored, usually in some form of propellant tank prior to being ejected from a rocket engine in the form of fluid jet to produce thrust [16,18]. Propellant is the chemical mixture that burned to produce thrust in rockets and consists of a fuel and an oxidizer. A fuel is a substance that burns when combined with oxygen producing gas for propulsion. An oxidizer is an agent that releases oxygen for combustion with fuel. The ratio of oxidizer to fuel is called the mixture ratio and for this case it is 65/35 according to Nakka [13]. Propellants are classified according to their state – liquid, solid or hybrid. There are two families of solid propellants: homogeneous and composite. Both types are dense, stable at ordinary temperatures and easily storable. The propellant for a chemical rocket engine is usually consists of a fuel and an oxidizer. Sometimes a catalyst is added to enhance the chemical reaction between the fuel and the oxidizer. The principal advantage of solid propellant is that it is relatively stable therefore it can be manufactured and stored for future use. Solid propellants have a high density and can burn very fast. They are relatively insensitive to shock, vibration and acceleration. No propellant pumps are required thus the rocket engines are less complicated [1, 4]. Disadvantages are that once ignited, solid propellants cannot be throttled, turned off and then restarted because they burn until all the propellant is used. The surface area of the burning propellant is critical in determining the amount of thrust being generated. Cracks in the solid propellant increase the exposed surface area, thus the propellant burns faster than planned [2]. If too many cracks develop, pressure inside the engine rises significantly and the rocket engine may explode. Manufacture of a solid propellant is an expensive precision operation.

The gauge for rating the efficiency of rocket propellants is specific impulse, stated in seconds. Specific impulse indicates how many pounds of thrust are obtained by the consumption of one pound of propellant in one second. The value of specific impulse will vary to some extent with the operating conditions and design of the rocket engine.

An igniter creates the temperature and pressure conditions that ignite the exposed inner surface of the propellant. The resulting high thermal energy created by the combustion process can then be converted into kinetic energy through the nozzle, effectively creating thrust [7,18].

Solid rocket propellant that was used in this work is potassium nitrate as oxidizer and sucrose as organic fuel. The primary objective of this effort was to develop a model to predict the performance of solid rocket motors using computer simulation software. The model is divided into two; the first being able to determine the safe operating pressure of the cylinder used as the motor casing based on certain parameters peculiar to the material used. The second model is designed as a simplistic model capable of predicting the pressure and thrust generated by a rocket motor from a given set of input parameters.

## 2. Materials and Methods

### 2.1 Design Features

Rocket has three major parts which are nose cone, combustion chamber and nozzle. The nose cone is the forward-most part of the rocket. The purpose of the nose is to reduce the aerodynamic drag on the model [19]. Combustion chamber is where the propellant burns. One important parameter that is needed for the design of the motor casing is the combustion chamber pressure ($P_c$) and temperature – it must be able to withstand the pressure and temperature developed. Nozzle is the exhaust duct of a rocket engine's combustion chamber. Gases from propellant combustion are accelerated to higher velocities in the nozzle [10].

### 2.2 Design Considerations

The small size rocket was design based on experimental results by Richard Nakka for potassium Nitrate and sucrose propellants. The size that was used is a small scale of 60 mm outside diameter, thickness 2 mm and length of the combustion chamber is 830 mm. The stainless steel was chosen as the casing and its properties are presented in table 1. The following parameters were considered in the design of a rocket: specific impulse, characteristic exhaust velocity, density, combustion temperature and specific heat ratio. The parameters calculated using Matlab are: specific impulse and mass flow rate, characteristic length and thrust coefficient, Nozzle area ratio, throat and nozzle diameter.

**Table 1:** Mechanical properties of Stainless steel [17]

| Properties | Values |
|---|---|
| Yield Strength | 517MPa |
| Ultimate Strength | 862Mpa |
| Modulus of Elasticity | 186.3GPa. |
| Poisson ratio | 0.27 |

### 2.3 Design Calculations

To carry out the design calculations on rocket design and characteristics, the values of some parameters got by Nakka as presented in table 2 was used and an ideal rocket unit assumption presented below [19] was used:

1. The working substance (or chemical reaction products) is homogeneous.
2. The working substance obeys the perfect gas law.
3. There is no heat transfer across the rocket walls; therefore, the flow is adiabatic.
4. There is no appreciable friction and all boundary layer effects are neglected.
5. The gas velocity, pressure, temperature, and density are all uniform across any section normal to the nozzle axis.

**Table 2:** Values taken from Nakka's experiment to be used in the MATLAB script [13]

| Parameters | Unit | Value |
|---|---|---|
| Burn rate pressure exponent | Nil | 0.319 |
| Burn rate constant | mm/sec | 8.62 |

### 2.4 Design of component parts of the Rocket

#### 2.4.1 Nozzle Design

The nozzle throat is responsible for the acceleration of the gases. It must not be too small or large. It may be computed if the total propellant flow rate is known and the propellants and operating conditions have been chosen. Assuming perfect gas law theory as stated by [19]:

$$\boldsymbol{A_t} = \frac{w_t}{P_t}\sqrt{\frac{RT_t}{kg}} \quad (1)$$

Where R = gas constant, given by

$$\boldsymbol{R} = \frac{\bar{R}}{M} \quad (2)$$

$\bar{R}$ is the universal gas constant
M is the molecular weight of the gas.
$k$ is the ratio of gas specific heats and is a thermodynamic variable.
g is a constant relating to the earth's gravitation.
$T_t$ is the temperature at the throat of the nozzle given by the equation below

$$\boldsymbol{T_t} = \boldsymbol{T_c}\left[\frac{1}{1+\frac{k-1}{2}}\right] \quad (3)$$

$T_c$ is the combustion chamber temperature
Pt is the gas pressure at the nozzle throat. Therefore, it can be calculated as follows:

$$\boldsymbol{P_t} = \boldsymbol{P_c}\left[1 + \frac{k-1}{2}\right]^{-\frac{k}{k-1}} \quad (4)$$

The nozzle exit area corresponding to the exit Mach number resulting from the choice of chamber pressure is given by

$$\boldsymbol{A_e} = \frac{A_t}{M_e}\left[\frac{1+\frac{k-1}{2}M_e^2}{\frac{k+1}{2}}\right]^{\frac{k+1}{2(k-1)}} \quad (5)$$

The Mach number at the nozzle exit is given by a perfect gas expansion expression

$$\boldsymbol{M_e^2} = \frac{2}{k-1}\left[\left(\frac{P_c}{P_{atm}}\right)^{\frac{k-1}{k}} - 1\right] \quad (6)$$

$P_c$ is the pressure in the combustion chamber and $P_{atm}$ is the atmospheric pressure,

#### 2.4.2 Rocket Motor Casing

One important parameter that is needed for the design of the motor casing is the combustion chamber pressure ($P_c$). It is directly proportional to the hoop stress of the material used for the casing. This is given as

$$\boldsymbol{P_c} = \frac{2\sigma_{hoop}t}{D} \quad (7)$$

where D is the casing diameter and (t) is the wall thickness. The chamber pressure is also related to the thrust, F, develop by motor. This is given by

$$\boldsymbol{F} = \boldsymbol{C_f\, A_t\, P_c} \quad (8)$$

where $C_f$ is the thrust coefficient
$A_t$ is the nozzle throat area (as stated in Eqn. 1)
The combustion chamber cross-sectional area is given by

$$\boldsymbol{A_c} = \frac{\pi D^2}{4} \quad (9)$$

The combustion chamber volume is given by

$$\boldsymbol{V_c} = \boldsymbol{A_c L_c} \quad (10)$$

### 2.5 Thermodynamic Relations

#### 2.5.1 Thrust

Thrust is generated by the flow of exhaust gases from the combustion of the propellants in the combustion chamber. It can be expressed mathe matically as

$$\boldsymbol{F} = \int \boldsymbol{PdA} = \dot{m}\boldsymbol{v_e} + \boldsymbol{A_e}(\boldsymbol{P_e} - \boldsymbol{P_{atm}}) \quad (11)$$

where ṁ is the mass flow rate of the exhaust products,
Pa is the atmospheric pressure
Ae is the cross-sectional area at the exit plane of the nozzle.

(Pe − Patm) is also-called *pressure thrust*. The maximum thrust is generated when the nozzle is optimized so that the exit pressure is equal to the ambient pressure or Pe = Patm.

$$\dot{m} = {}^{m_p}/_{t_b}, \tag{12}$$

where total usable propellant mass is $m_p$. total duration of burning is $t_b$

Hence an optimized equation gives

$$F = \dot{m}v_e \tag{13}$$

**2.5.2 Thrust coefficient**

This is the measure of the efficiency of the extracted energy from the gases flowing from the combustion chamber. It is related to the chamber pressure and the throat area as given below

$$C_f = \frac{F}{P_c A_t} \tag{14}$$

**2.5.3 Exhaust Velocity**

It is the velocity at which the combustion gases produced as a result of burning of the propellant leave the nozzle. It is mathematically represented by

$$V_e = \sqrt{\frac{2kRT_c}{k-1}\left[1 - \left(\frac{P_e}{P_o}\right)^{k-1/k}\right]} \tag{15}$$

where $T_c$ is the combustion temperature as stated in Eqn. (3)
R is gas constant as stated in Eqn. (2)
By substituting (12) in (10), we have

$$F = \dot{m}\sqrt{\frac{2kRT_c}{k-1}\left[1 - \left(\frac{P_e}{P_o}\right)^{k-1/k}\right]} \tag{16}$$

By substituting (3.13) in (3.11)

$$C_f = \frac{1}{P_c A_t}\,\dot{m}\sqrt{\frac{2kRT_c}{k-1}\left[1 - \left(\frac{P_e}{P_o}\right)^{k-1/k}\right]} \tag{17}$$

**2.5.4 Total and Specific Impulse**

Total impulse is defined as the integral of the thrust over the operating duration of the motor

$$I_t = \int F dt \tag{18}$$

Specific impulse can be expressed as the efficiency of a propellant per unit weight

$$I_{sp} = \frac{I_t}{w_p} \tag{19}$$

The script code was written in the MATLAB programming language.

**2.6 Mathematical Rocket Design**

In order to design a prototype rocket, values for outer diameter, $D_o$ and casing wall thickness, t, were assumed. The value for the length was also assumed. The availability and cost of the material informed this assumption. As stated in table 2, the experimental results of burn rate pressure exponent (n), burn rate constant and burn rate (r) were input together with assumed values into the numerical code written in MATLAB to solve the design problem. The effectiveness and accuracy of the design parameters were determined.

Using the values calculated by the MATLAB script, a basis was formed on which a rocket could be designed. The theoretical combustion reaction equation for potassium nitrate and sucrose propellant with an O/F ratio of 65/35 is given as [12, 13]:

$$\mathbf{C_{12}H_{22}O_{11} + 6.288KNO_3 \rightarrow 3.796CO_2 + 5.205CO + 7.794H_2O + 3.065H_2 + 3.143N_2 + 2.998K_2CO_3 + 0.274KOH}$$

## 3. Discussion of Results

The result of five important parameters that was determined numerically together with the experimental results of Nakka is presented in table 3. All the results were in close agreement with each other except specific impulse and combustion temperature that are a little bit different. This may be as a result of either some error in experimental reading of some assumptions in theoretical calculation; however, it can be used to determine variations of other important properties of the rocket with time.

**Table 3:** Comparison between Nakka's values [11] and the MATLAB's script

| Parameters | Unit | RN's value | Calculated value |
|---|---|---|---|
| Specific Impulse | sec. | 130 | 164.39 |
| Characteristic exhaust velocity | m/s | 946.7 | 913.4 |
| Density | g/cm$^3$ | 1.80 | 1.888 |
| Combustion temp. | $^o$K | 1627 | 1720 |
| Specific heat ratio | | 1.044 | 1.0437 |

**3.1 Graphical analysis of thermodynamic properties with respect to time**

Figures 1 to 9 show the variations of the different parameters which affect the flight of the rocket against the time, for complete combustion of the propellants. This is needed to understand how the parameters are interrelated in real life scenarios.

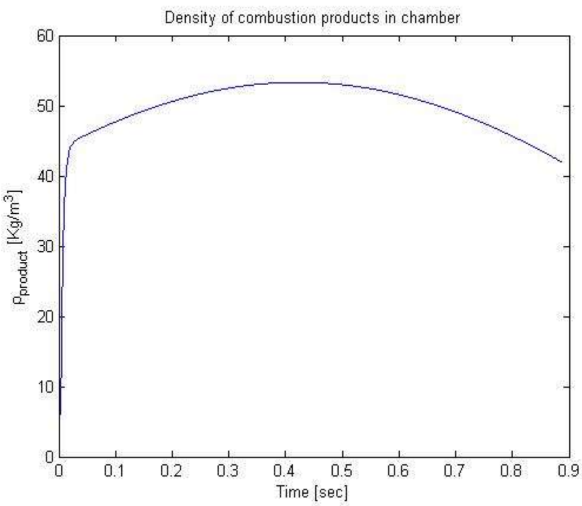


**Figure 1:** A graph of density of the combustion products against the burning time

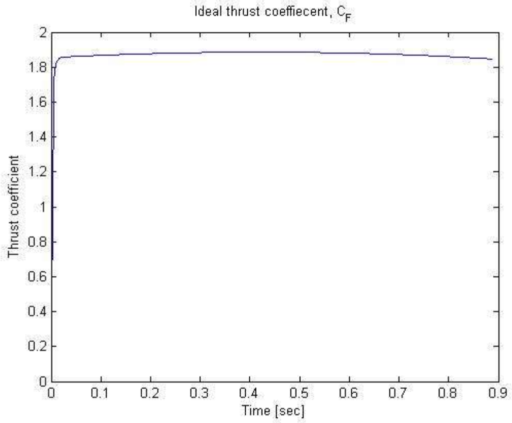


**Figure 2:** A graph of Ideal thrust coefficient against the burning time

The thrust coefficient represents the performance of a nozzle. It is the measure of the efficiency of the extracted energy from the hot gases in the combustion chamber. For most solid propellant rockets, the value of thrust coefficient ranges between 1.5 and 2.0 [3]. From the figure 2 above, it is between 1.8 and 1.9

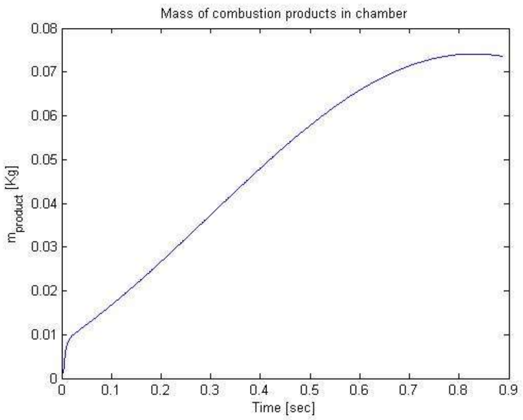


**Figure 3:** A graph of mass of the combustion products against the burning time

Figure 3 above shows the mass of products of combustion with respect to time. The curve is upwardly sloped as the mass of the burnt gases increases within the chamber as burning takes place. It reaches its peak just at the end of burning when it starts tending downwards again. Therefore, it can sustain the flight.

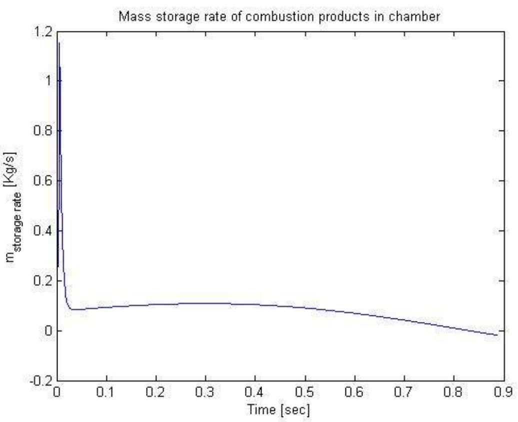


**Figure 4:** A graph of mass storage of combustion products against the burning time

It can be seen from figure 4 that the mass of the combustion product in the combustion chamber spikes immediately burning starts, but as the burning continues, the products escapes through the nozzle. The flow of the gases is what propels the rocket forward. If the mass had remained the same, there could be a build-up of pressure which causes the casing to erupt.

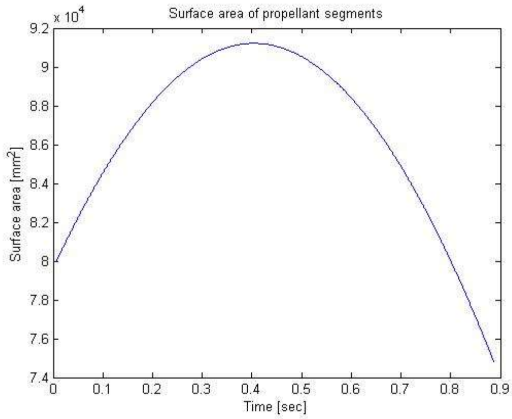


**Figure 5:** A graph of surface area of the propellant against the burning time

Figure 5 above shows the surface areas of the propellants as the burning is taking place. It is seen that the surface area peaks at 0.4 seconds which is nearly half the total burn duration, and steadily falls as burning continues.

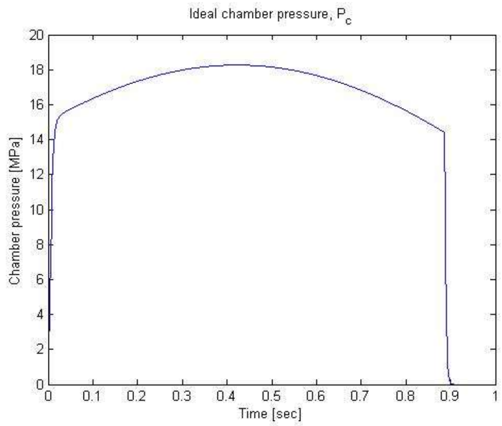


**Figure 6:** A graph of pressure of the combustion chamber against the burning time

Figure 6 above shows that the pressure in the combustion chamber can reach as high as a little below 19MPa, so it can be related to the maximum operating pressure that the material selected can withstand: this was determined for stainless steel. This can help to find the best combination for both the outer casing diameter and the thickness.

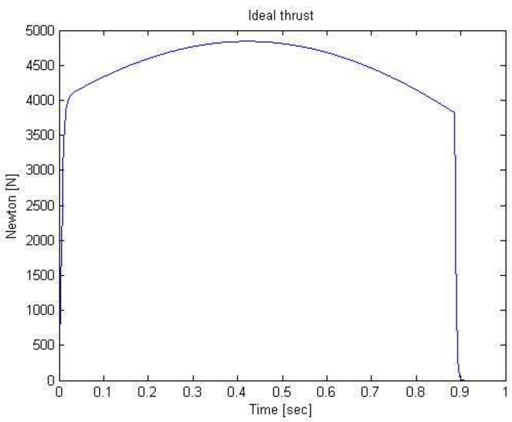


**Figure 7:** A graph of density of ideal thrust generated against the burning time

The ideal thrust in figure 7 gives the value of force produced. This is quite important for the rocket to leave the launch pad. If the thrust is smaller than that needed to propel the rocket (usually determined by the total mass of the rocket), the rocket would not leave the launch pad.

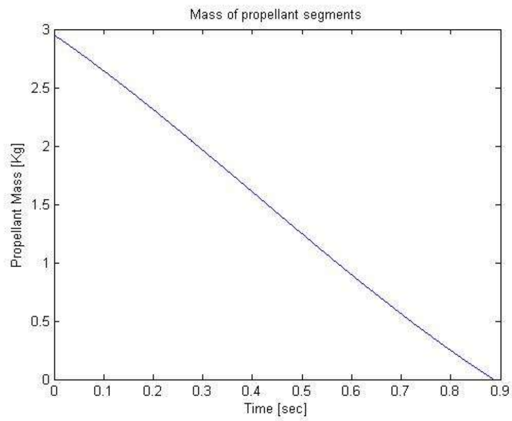


**Figure 8:** A graph of mass of propellant against the burning time

Figure 8 shows the rate at which the mass of the propellant reduces as burning takes place. It is a straight line graph and the mass reduction is proportional to the burning time.

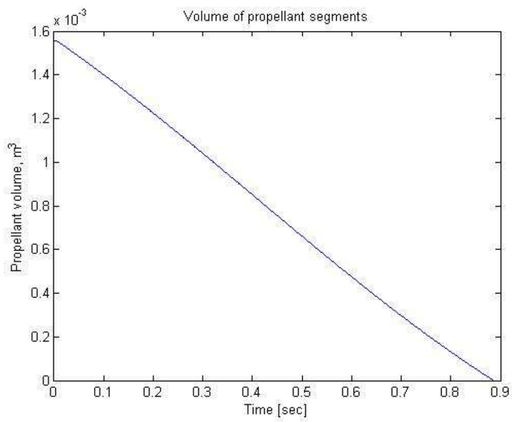


**Figure 9:** A graph of volume of the propellant against the burning time

Figure 9 shows the rate at which the volume of the propellant reduces as burning takes place. Just like the graph for the mass of propellants (Fig 8), it is a straight line graph and the volume reduction is proportional to the burning time.

## 4. Discussions

From the diagrams above, the density of products from combustion of the propellants is seen to be between 50 and 60 kg/m$^3$; the ideal thrust coefficient has a range value of between 1.8 and 1.9. This shows that the nozzle has a high efficiency.

The mass of the exhaust product in the chamber steadily rises and reaches a peak of 0.075kg just before the end of burning. This shows that the burning of the propellants is uniform and the nozzle allows for a build-up of the gases to enable propulsion. The surface area of the propellant segments as shown on the graph steadily rises for 0.4 seconds until it descends for the remaining of the burning time.

The projected pressure developed in the combustion chamber sharply rises to about 15MPa, then slopes upward to a little above 18MPa. By 0.4 seconds it reaches its peak and slopes downwards to a little below 15Mpa before falling abruptly. This information is important when selecting the material for the chamber pressure; this is to prevent explosion of the cylinder during combustion of the propellants.
The ideal thrust generated is also related to the pressure developed within the combustion chamber. The thrust is also dependent on the diameter of the nozzle throat. The thrust developed is the main parameter that determines if the rocket would fly or not. For flight to be achieved, the thrust developed should be far greater than the mass of the entire rocket assembly.

## 5. Conclusions

From the above discussions the following can be concluded:

1) The important parameters were determined which were in close agreement with the experiment of Nakka. However, further work can be done that will include erosion burning of the propellant.
2) The model allows for variation of different parameters and the pressure developed which is about 19MPa was not beyond the capacity of the pressure wall material.
3) The trust developed by the combinations of different parameters falls within the acceptable limit of 1.5 and 2.0 according to Bose and Pandey [3].
4) Based on the analysis for the rocket design using the materials and dimensions, the rocket is expected to produce a thrust of over 4000Newtons. This is enough to propel the rocket and the distance would depend on the total mass of the rocket.

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