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MODEL ROCKET ENGINE PERFORMANCE

BY EDWIN D. BROWN
Mgr. Estes Industries Engine Development

INTRODUCTION

A model rocket engine is a scientific device. To produce thrust it operates exactly the same way as a "big" rocket. Its solid propellant burns, producing gases, which are expelled through a scientifically designed nozzle. These gases exit at supersonic velocities, producing thrust in the opposite direction (in accordance with Newton's Third Law of Motion).

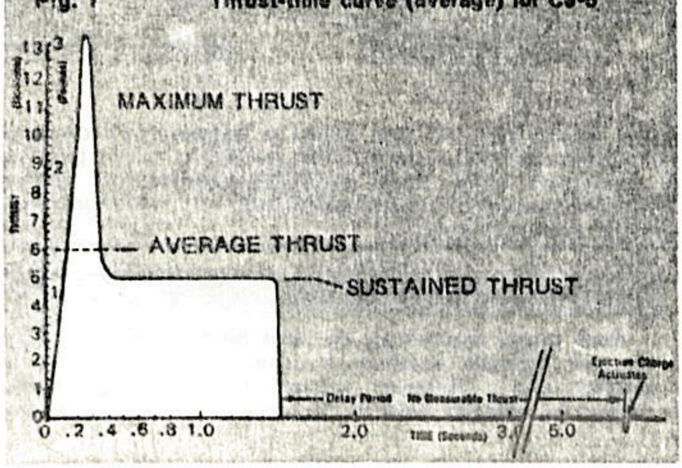
The thrust characteristics of a model rocket engine are measured in the same way, and with the same terminology, as large rocket engines. The total impulse (total power) is measured in pound-seconds or newton-seconds. Suppose a rocket engine produces an average thrust of 1.35 pounds (6 newtons) for 1.5 seconds. This would be total impulse of 2.02 lb.-sec. or 9.0 n.-sec. Looking at the information in the Estes Catalog, you'd find an engine with this much total impulse is a type "C" engine. A time-thrust curve for this type of engine is shown in Figure 1.

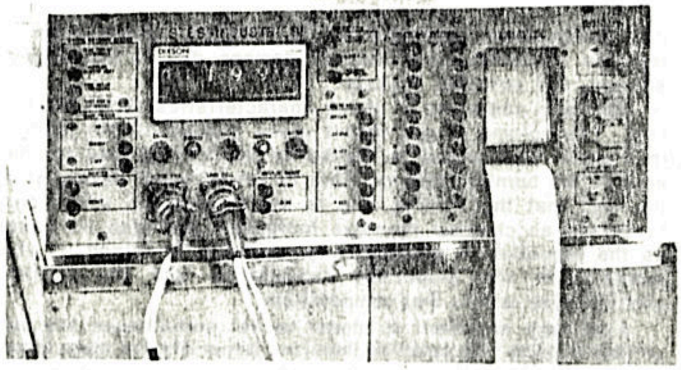
Model rocket engines are also designed to provide other supplementary functions to make the rocket perform satisfactorily. If the rocket engine just produces thrust, then ejects the parachute, you're in trouble. The rocket may be traveling as fast as two or three hundred miles per hour. The wind will literally tear off the shroud lines and/or rip the chute to shreds. Therefore, the model rocket engine is designed with a special slow-burning "delay" material which also produces a dense smoke. This allows you to see your rocket as it coasts upward while slowing down. The ejection charge is initiated after several seconds of coasting flight. This ejects a large volume of gas which rushes into the forward part of the rocket body. The pressure from this gas can be made to push out a parachute, trigger a camera, move an elevator surface, eject the spent engine casing, etc.

All of the above major functions of a model rocket engine are important. Failure of any one part can cause your rocket to not work properly. This is why all Estes engines are carefully tested throughout production. Although the production process is automatic and many tests are done as the engines are made, other supplementary evaluations are continually performed. As the engines come off the machines, three out of every 100 are static tested on an electronic device. Performance information recorded in each of these tests includes peak thrust, average thrust, thrust duration, total impulse, time delay, and the strength of the ejection charge. Tests similar to these are also performed in the field of professional rocketry. The fact that model rockets and professional rockets are so much alike makes the hobby even more exciting and valuable to those who participate.

The following semi-technical description of the model rocket engine and how it works will not only be helpful to you in model rocketry, but will also give you a basic understanding of some important characteristics of all types of solid propellant rockets.

Fig. 1 Thrust-time curve (average) for C6-5





The Estes Semiautomatic Portable Engine Test System (ESPETS) uses 75 solid-state integrated circuits and some 100 transistors. Using the ESPETS' analog and digital readouts, all the major parameters of model rocket engines can be determined and recorded in less than one minute per engine.

THE ENGINE

Choosing the C6-5 type engine as an example, we have the following facts available to us: The C means that the total impulse must be between 1.13 and 2.24 pound-seconds (5.01 and 10.0 newton-seconds); the 6 tells us that the average thrust is 6 newtons (1.35 pounds); and the 5 tells us that there is a 5 second delay after the thrust stops before the ejection charge is ignited. Looking at Figure 2 we see that this engine has the following parts: casing, nozzle, propellant, delay element, ejection charge, and retainer cap. Figure 1 shows a typical thrust-time trace for this engine.

FUNCTIONS OF A TYPICAL ENGINE

The nozzle guides the products of the chemical reaction as they are ejected from the rocket engine.

The propellant is a composite which produces the reaction products by a self-sustaining combustion process. These reaction products allow us to take advantage of Newton's Third Law, "For every action there is an equal and opposing reaction," making our rockets fly.

The delay element is a slow-burning, smoke-producing mixture which allows the rocket to reach its peak altitude before igniting the ejection charge and provides a smoke trail for tracking purposes.

The ejection charge provides a fixed amount of gas which is used to activate the recovery system, etc.

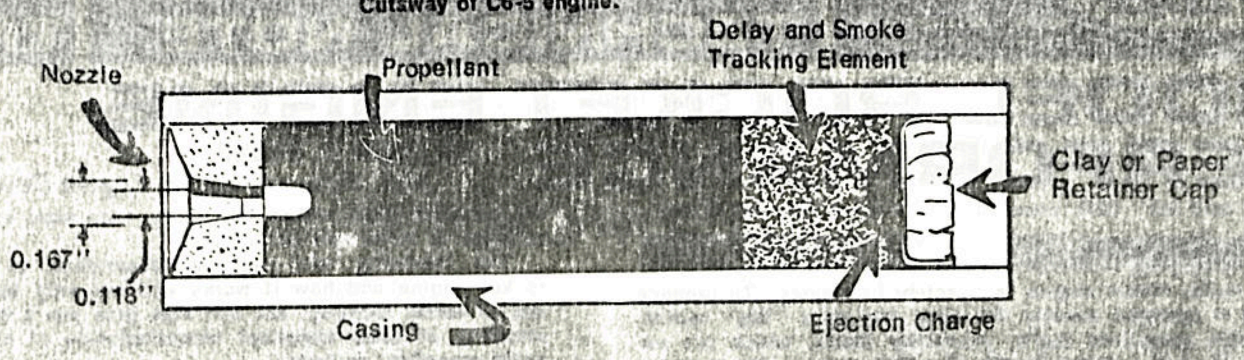
The retainer cap serves only to retain the ejection charge until it is ignited.

Since the propellant, grain configuration, and the nozzle determine the major portion of the engine's performance, we will discuss them further in the next sections.

PROPELLANT CHARACTERISTICS

The important characteristics of a propellant are: burning rate, specific impulse, density, characteristic exhaust velocity, specific heat ratio, temperature of combustion, pressure

Fig. 2 Cutaway of C6-5 engine.



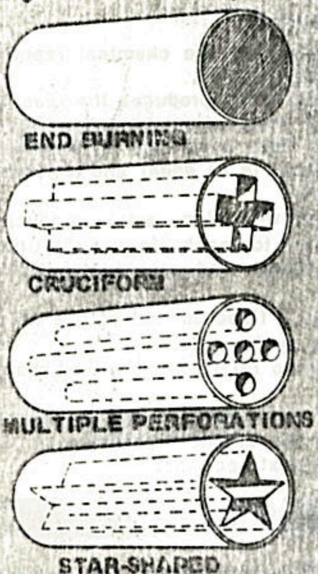
and temperature requirements for ignition, composition of reaction products, resistance to damage due to handling or storage, and possible toxicity.

The most important of these characteristics is its burning rate. The volume of gas that a given propellant can produce in a given time period is limited by the burning rate and the area of the burning surface. This is complicated somewhat by the fact that the burning rate is not a constant. It not only increases as chamber pressure increases, but also increases as the propellant's preignition temperature is raised. It also varies with the propellant composition and the oxidizer particle size within that composition.

Also very important to model rocket performance are propellant density and specific impulse. Generally the more dense (heavy) the propellant is, the less space a given weight of propellant will occupy. Most model rocket propellants are made of a dense material, thus increasing overall efficiency.

Specific impulse is a measurement of propellant efficiency. It is expressed in seconds and is determined by dividing the total impulse of the engine by the weight of the propellant. For example, a C6-5 engine which has a total impulse of 2.25 lb.-sec. and contains 0.028 lb. of propellant will have a specific impulse of 80.36 seconds. Most model rocket engines have specific impulses between 50 and 100 seconds. In professional solid propellant rocketry, where the chamber pressures are higher and more exotic fuels are used, specific impulses of 180 to 250 seconds are common. However, most of these fuels are less dense and require relatively heavy motor casings and rocket frames. Thus, part of the performance increase obtained with the higher energy fuels is lost.

Fig. 3 Typical Grain Shapes



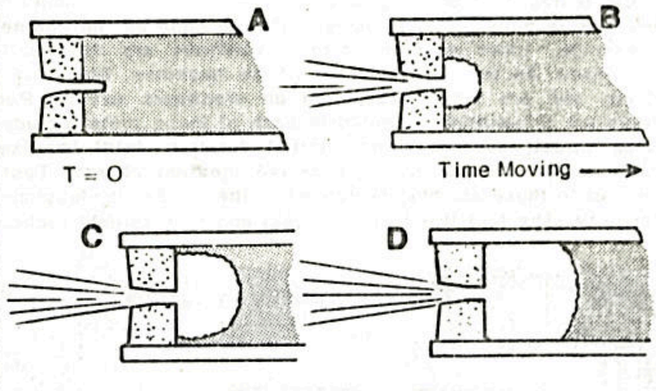
The propellant charge in a solid propellant rocket engine is called the grain. By shaping the grain into different configurations the burning is controlled to obtain desired results. For example, the popular star shape is designed for a constant thrust during the entire burn. Fast burning propellants like those used in model rocketry frequently use a modified end burning grain which is illustrated in Figure 4.

We will not cover the other propellant characteristics in this report because of limited space. However, the serious student may gain more knowledge in these areas by referring to the publications listed at the end of the report.

PROPELLANT GRAIN DESIGN

The primary purpose of varying propellant grain design (grain geometry) is to provide the burning area necessary to produce the desired chamber pressure. The most common grain design found in model rocket engines is a combination of core burning and end burning as shown in Figures 2 and 4. Core burning is also known as progressive burning since the burning area increases with time. End burning is sometimes called neutral burning since the burning area remains constant. The purpose of combining the two types in model rocket engines is to provide a high initial thrust to accelerate the rocket to a high enough speed to stabilize it while it is still being guided by the launch rod and to bring the model up to its maximum speed more or less gradually to minimize drag buildup. (Drag is proportional to the square of the velocity.) Figure 4 illustrates the burning of the propellant in a typical model rocket engine.

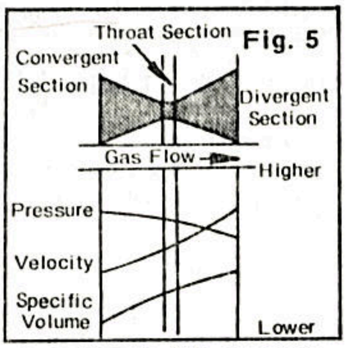
Fig. 4 Transition from core to end burning



Note that in step "C" the engine is approaching a condition of maximum burning surface. As shown in Fig. 1, this produces a peak thrust which occurs about 0.24 seconds after ignition. When the burning has progressed to step "D", the thrust drops to the sustained level.

THE NOZZLE

Most rocket engines use de Lavâl nozzles. These consist of three separate sections: a convergent section, a throat section, and a divergent section. The convergent section causes the reaction products to increase in velocity in order to pass through the throat section in much the same way that water speeds up when flowing through a narrow part of its channel. (Note: The model rocket engine does not have a true convergent taper in the convergent section.) In the divergent section things become slightly more complicated. The velocity continues to increase because we are exhausting to a lower pressure region and the gaseous reaction products are expanding to this pressure. Figure 5 illustrates what happens to the velocity, pressure, and specific volume (volume occupied by a unit of mass) of gaseous reaction products in a de Laval nozzle.



Once grain design and propellant composition are fixed, then the nozzle and its design become the controlling factors in model rocket engine performance. By varying its design and size, we can vary chamber pressure, specific impulse, thrust level, engine efficiency, etc. The following equations and illustrations show how this happens.

$F = C_F P_c A_t$ eq. 1

$c = I_{sp} g$ eq. 2

$c^* = c/C_F$ eq. 3

$I_{sp} = F/\dot{W}$ eq. 4

F = Thrust (pounds)

C_F = Thrust Coefficient (a dimensionless, relative measure of nozzle efficiency)

P_c = Chamber pressure (pounds per square inch absolute)

A_t = Nozzle throat area (square inches)

c = Effective exhaust velocity (feet per second)

I_{sp} = Specific Impulse (seconds). A measure of propellant efficiency

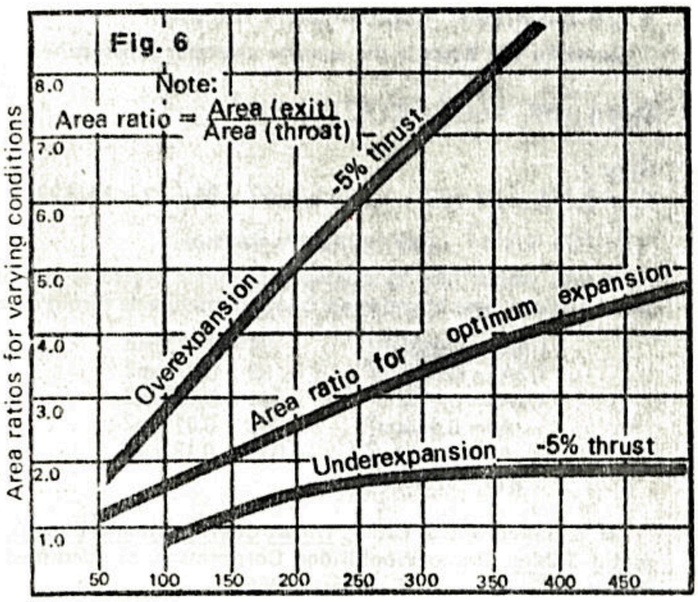
g = Acceleration due to gravity (32.17 feet per second^2)

c^* = Characteristic exhaust velocity (feet per second)

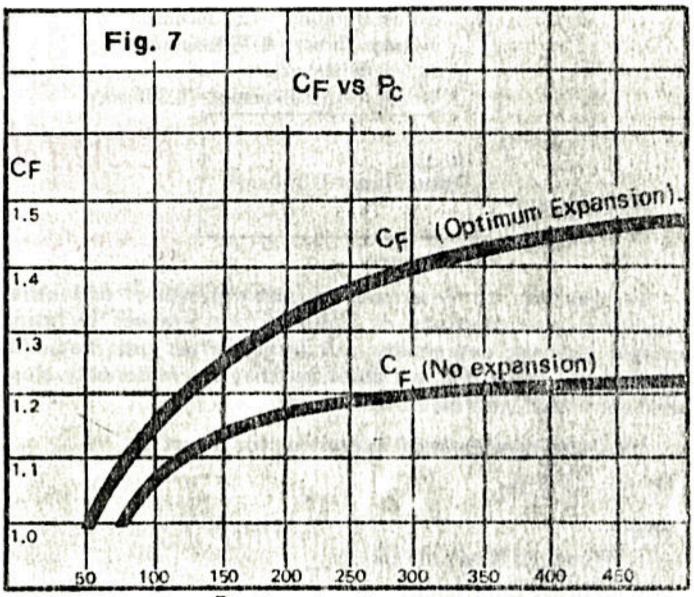
$\dot{W}$ = Weight flow rate (pounds per second)

Obviously these equations can be rearranged into many different forms to find the value of various terms.

Estes model rocket engines use an area ratio of 2.0 in their nozzles. By area ratio we mean the nozzle exit area divided by the nozzle throat area. If you look at figure 6 you will see why this area ratio was chosen. At peak thrust we have a chamber pressure of about 225 pounds per square inch, (abbreviated psia). This drops to about 100 pounds per square inch during sustained thrust. With an area ratio of 2.0 we will not lose 5% of potential thrust until chamber pressure drops to around 60 psia. This gives us a good thrust coefficient at both our peak chamber pressure and at our sustained chamber pressure. With an area ratio of 4.0 there would be a loss of more than 5% below a chamber pressure of 150 psia.



P_c Chamber Pressure (psia)



P_c Chamber Pressure (psia)

The thrust coefficient (C_F) is a function of nozzle efficiency and chamber pressure. Figure 7 illustrates this relationship. As shown, a typical model rocket engine with a peak chamber pressure of 225 psia and a two to one (optimum) expansion ratio, the thrust coefficient will be approximately 1.33. Thus, using equation 1, the peak thrust of this engine can be determined as follows:

$$F = C_F P_c A_t$$

$$= \frac{1.33}{1} \times \frac{225 \text{ lbs.}}{\text{in.}^2} \times \frac{0.011 \text{ in.}^2}{1}$$

$$= 3.29 \text{ pounds}$$

$A_t \simeq 7.1$ m

$D_t \simeq 3.0$ m

Referring to Figure 1 this is approximately the peak thrust shown for the C6-5 Engine.

When the chamber pressure is 225 psia, the nozzle efficiency will be essentially the same for an area ratio of 2.0 or 4.0. However, when we assume a sustained chamber pressure of 100 psia, we get a C_F of about 1.15 for an area ratio of 2.0, which will give us a sustained thrust level of 1.27 pounds. As Figure 6 shows, an area ratio of 4.0 is now over-expanding enough to give us a rather substantial loss of thrust (over 5%). Thus, it is clear an expansion ratio of two to one is ideal for most model rocket engines.

Using equations 2, 3, 4, other useful characteristics of model engines can be derived.

NOTE TO READER

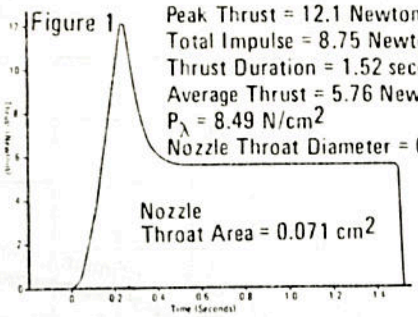
If you have further questions, we suggest that you consult the publications listed below or similar references.

Shapiro, Ascher H. The Dynamics and Thermodynamics of Compressible Fluid Flow. Volumes 1 and 2. New York: Ronald Press, 1953

Sutton, G.P. Rocket Propulsion Elements. Third Edition New York: John Wiley and Sons, Inc., 1963

Williams, F.A., Barrere, M., and Huang, N.C. Fundamental Aspects of Solid Propellant Rockets. Slough, England: Technivision Services, 1969

A Method For Calculating Chamber Pressures Of Estes Model Rocket Engines



$P_c^{max} \approx 12{,}7$ atm

$\bar{P}_c \approx 6{,}6$ atm

It's possible to determine chamber pressures and other operating characteristics of model rocket engines by using simple algebraic expressions and thrust versus time data. Although the results are not absolute, they are reasonably close and consistent with measured values.

A simple mathematical expression for thrust is:

(1) $F = C_f P_c A_t$

where

F = Thrust in Newtons (N)

C_f = Thrust Coefficient

P_c = Chamber Pressure in Newtons per square centimeter (N/cm^2)

A_t = Nozzle Throat Area in square centimeters (cm^2)

Thrust can be measured or taken from manufacturer's data. Nozzle throat area can be calculated if throat diameter is measured or provided by manufacturer. Now, if we knew Cf's value, we could easily calculate P_c.

(2) $$P_c = \frac{F}{C_f A_t}$$

The thrust coefficient (Cf) can be defined (neglecting corrections for divergence of nozzle exit cone, variation of specific heat ratio, flow separation, friction losses, and assuming one dimensional flow with isentropic expansion) by:

(3) $$C_f = \sqrt{\left(\frac{2K^2}{K-1}\right)\left(\frac{2}{K+1}\right)^{\frac{K+1}{K-1}}\left[1-\left(\frac{P_e}{P_c}\right)^{\frac{K-1}{K}}\right]} + \frac{(P_e-P_a)\ E}{P_c}$$

where

K = Ratio of specific heats (approximately 1.3 for Estes engines)

P_e = Static pressure at nozzle exit (N/cm^2)

P_a = Ambient pressure (N/cm^2)

P_c = Chamber pressure (N/cm^2)

E = Ratio of nozzle exit area to throat area (nominally 2.0 for all Estes engines).

Equation 3 looks tricky, but can be simplified. If we fired our engines in a vacuum, P_a would be zero. The vacuum thrust coefficient would then be:

(4) $$C_{f_{vac}} = \sqrt{\left(\frac{2K^2}{K-1}\right)\left(\frac{2}{K+1}\right)^{\frac{K+1}{K-1}}\left[1-\left(\frac{P_e}{P_c}\right)^{\frac{K-1}{K}}\right]} + \frac{P_e}{P_c} E$$

Now we can see that thrust coefficient is also:

(5) $$C_f = C_{f_{vac}} - \frac{P_a}{P_c} E$$

We can now obtain $C_{f_{vac}}$ for a given E and K from thrust coefficient tables. For Estes engines $C_{f_{vac}} = 1.4609$. And also for all Estes Engines:

(6) $$C_f = 1.4609 - \frac{2\ P_a}{P_c}$$

At first glance, we haven't accomplished much. We can calculate Cf if we know P_c and P_c if we know Cf. Rather than give up at this point, we'll multiply both sides of equation 6 by P_c:

(7) $$C_f P_c = 1.4609\ P_c - 2\ P_a$$

This rings a bell. We can determine $C_f P_c$ by:

(8) $$C_f P_c = \frac{F}{A_t}$$

Since we wish to find P_c, we will rearrange equation 7.

(7) $$C_f P_c = 1.4609\ P_c - 2\ P_a$$

$$1.4609\ P_c = C_f P_c + 2\ P_a$$

$$P_c = \frac{C_f P_c + 2\ P_a}{1.4609}$$

(9) $$P_c = 0.685\ C_f P_c + 1.369\ P_a$$

Equation 9 defines P_c in terms of $C_f P_c$ and we can now calculate P_c if we have values for thrust, ambient pressure, and nozzle throat area.

Step 1. Determine $C_f P_c$ using equation 8.

(8) $$C_f P_c = \frac{F}{A_t}$$

Step 2. Calculate P_c using equation 9.

(9) $$P_c = 0.685\ C_f P_c + 1.369\ P_a$$

Let's try a few calculations. Figure 1 is a reproduction of a thrust curve of a C6 engine tested at an ambient pressure of 8.49 N/cm^2.

Example #1. What is the peak chamber pressure?

Step 1. $C_f P_c = \dfrac{12.1}{.071} = 170.423\ N/cm^2$

Step 2.

$P_c = 0.685 \times 170.423 + 1.369 \times 8.49 = 116.740 + 11.623$

$= 128.363\ N/cm^2$ (approximately 186 psia)*

Example #2. What is the average chamber pressure?

Step 1. $C_f P_c = \dfrac{5.76}{.071} = 81.127\ N/cm^2$

Step 2.

$P_c = 0.685 \times 81.127 + 1.369 \times 8.49 = 55.572 + 11.623$

$= 67.195\ N/cm^2$ (approximately 97psia)*

If you would like to calculate chamber pressures for our other engines, here are nominal (maximum) nozzle throat areas:

1/4A3 = 0.063cm2	B4 = 0.123cm2
1/2A3 = 0.063cm2	B6 = 0.071cm2
1/2A6 = 0.123cm2	B14 = 0.138cm2
A3 = 0.063cm2	C6 = 0.071cm2
A8 = 0.123cm2	D12 = 0.183cm2
A10 = 0.100cm2	

H. S. Siefert and J. Crum, Thrust Coefficient and Expansion Ratio Tables, Ramo Wooldridge Corporation, Guided Missile Research Division, 29 February 1956.

G. P. Sutton, Rocket Propulsion Elements, John Wiley and Sons, Inc. 1963.

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*pounds per square inch absolute