PROPELLANT CHARACTERIZATION

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DETERMINING SOLID ROCKET MOTOR PROPELLANT INTERNAL BALLISTICS CHARACTERISTICS AND MOTOR DESIGN TIPS FOR MODEL, HIGH POWER AND EXPERIMENTAL / AMATUER SOLID ROCKET MOTORS

Let's assume you are doing a very small scale rocket motor development project. You have no electronic thrus equipment. What you DO have is a 50 lb. capacity "baby scale" and a video camera. Here's an approach that has for others in the past.

Build a small test stand that will transfer the motor thrust to a Baby Scale with an analog face. (So you can easily read pounds on the scale!) Select a small, manageable motor size, say a G size casing that you have purchased or built and whose estimated top thrust is a conservative fraction of the scale's top capacity. A recommended motor design for this BATES grain test motor is presented below.

A small Basic Test S can be easily constru using wood and a b scale, as this docun refers to. To check c simple solution cli **HERE**.

Within the ROGERS AEROSCIENCE ENGMOD Program, select a middle of the road propellant formulation. One that you have high confidence in, or is identical or very similar to a formulation that has proven successful in similar sized motors in the past.

Assume a ratio of specific heats in the 1.20 - 1.24 range. Choose a desired operating pressure that you have re believe the motor will tolerate. Phenolic or fiberglass motor cases typically can withstand sustained operating c pressures of 450 psi.

Run ENGMOD and design a very neutral-burning grain. To minimize core Mach number effects on the characteriz your propellant ballistic characteristics, design the motor to have a relatively large core diameter relative to it diameter. A typical BATES grain test motor design is presented here.

BATES GRAIN TEST MOTOR

3 BATES Grains
Core Diameter = 0.60 in
Case Inside Diameter = 0.95 in
Grain Lengths = 1.60 in each
Throat Diameter = 0.23714 in
Exit Diameter = 0.50 in
Divergence Half Angle = 15.0 deg
Ratio of Specific Heats = 1.19

Fired with propellant AT-STD (Standard AP-HTPB ATP-8442), this motor will operate at a maximum chamber pre 450 psi, and will have a propellant weight of 55.57 grams. Fired at sea level with a Straight Throat conic nozzle choice ST) this motor will produce an average thrust of 27.3 lbs and will burn for 0.97 sec. Most importantly this m produce an extremely flat thrust curve. For 77% of the thrust curve the thrust will be within +/-3.0% of the averag The maximum deviation from the average thrust will occur just prior to burnout, where the thrust will still be within the average thrust. Repeat the design process in ENGMOD designing a second motor with a slightly differen throat which will operate at a different burning surface area to throat area ratio, or Kn.

Physically construct both motor designs using your own propellant. Fire the first motor on a "baby scale" tes Videotape the runs. The video will give you the thrust as a function of time; having that, you can back out the c pressure using the PCROC function.

Running the motor on ENGMOD will give you the ratio of the motor burning surface area divided by the throat ar versus time, from which you can deduce the average Kn of the motor. With the average Kn, and the average c pressure from the motor firing, you will have measured one point on the Kn-chamber pressure curve.

Using a stop watch, or the slow motion time function from the video camera, measure the burntime of the motor. \ motor having a flat thrust curve, the motor will also have a flat chamber pressure time history. Thus dividing t thickness (the initial propellant thickness from the core to the inside case wall) by the burntime, you will have m one burnrate-chamber pressure data point.

The next step is to fire a second motor of the same design, but with a different throat diameter. Repeat the p procedure, and you will have determined second points on both the Kn-chamber pressure and burnrate-c pressure curves. From the two points on the Kn-chamber pressure and burnrate-chamber pressure curves the EN program will construct the entire curves.

If you were to construct a motor with the core diameter smaller than throat diameter you would create a conditio gas flow in the core will choke (reach Mach 1) at the end of the core and motor will fire nozzeless. To minimize burning from core Mach number effects, the throat diameter should be no larger than 3/4 of the core diameter.

DEFINITION OF THE "BURN RATE EXPONENT"

The burnrate exponent (n), and the chamber pressure exponent (l/(l-n)), describe the burnrate of the propellant chamber pressure, and the chamber pressure that results from a given motor Kn (motor burning surface area di throat area) based on the following equations:

$$\text{burnrate} = \text{constant} * (\text{chamber pressure})^n$$

$$\text{chamber pressure} = (\text{constant} * Kn)^{(l/(l-n))}$$

It can be derived, theoretically, that the exponent variable "n" in each equation is identical. In the real world the variables will have nearly the same value for most propellants, but can and will be different. If little is known particular propellant, assuming the two "n"s are identical is a good assumption. The higher the burnrate expon higher the propellant burnrate for a given chamber pressure. The higher the chamber pressure exponent (1/(1- higher the motor chamber pressure for a given Kn.

THRUST COEFFICIENT:

The thrust coefficient of the rocket motor nozzle is the thrust of the nozzle divided by the chamber pressure ti throat area;

$$\text{Thrust Coefficient} = \text{Thrust} / (P_c * A_{th})$$

The thrust coefficient is a very useful parameter as it is a function of nozzle expansion ratio, atmospheric pressure exit pressure, and the ratio of specific heats for the flow in the nozzle. The nozzle exit pressure is itself a functio expansion ratio and chamber pressure. From the thrust coefficient equation it is apparent that the chamber press a first order effect on thrust. For a given fixed nozzle expansion ratio and throat area at a constant altitude doub chamber pressure will produce very close to double the thrust.

SEA LEVEL SPECIFIC IMPULSE:

The sea level specific impulse presented in the specific impulse versus chamber pressure plot is calculated assu ideal expansion ratio ($P_e = P_a$), zero divergence losses, and zero losses in the nozzle throat and divergent sectio friction losses or other losses associated with a drilled or non-rounded, non-optimum throat geometry. Typically or high power rocket motor with a properly expanded nozzle will deliver a specific impulse equal to 95% of this va given chamber pressure. The ENGMOD program will calculate the thrust losses due to under or over exp divergence losses, and nozzle throat and friction losses to predict the actual delivered specific impulse for the mot run.

NOZZLE TYPES AND LOSSES:

The primary nozzle thrust loss is from under or over expansion. This can easily be eliminated by properly expan nozzle. The highest thrust is produced when the exit pressure (P_e) is equal to the atmospheric (P_a). The ENGM PCROC programs print out P_e/P_c (PEPC in the program output) calculated for the nozzle, the exit pressure of the is simply:

$$Pe = Pc \times (Pe/Pc)$$

This identifies to the user the relative under or over expansion of the nozzle that has been selected. Rocket operate at varying chamber pressures and hence have varying Pe, the user can run the ENGMOD program inter to determine the expansion ratio that produces the highest delivered specific impulse for the entire burn of the mot

The second largest thrust loss is due to the divergence half-angle, which is the half-angle of the nozzle in the no plane. This nozzle half-angle creates opposing normal components of thrust relative to the nozzle axis that car produce no contribution to axial thrust. As the momentum of this normal component of velocity is wasted the ener to create it is lost and the thrust of the nozzle is decreased. For a typical nozzle divergence half-angle of 15 deg the conic divergent sections of model and high power rocket nozzles the resultant thrust loss is small, only 1.7%. advantage of a bell nozzle is it's near zero divergence angle, for a rocket flying to orbit a 1.7% decrease in thr specific impulse is critical and would eliminate a substantial portion of the orbital payload for the vehicle.

Further losses in the nozzle are a function of the nozzle throat type and the shape of the nozzle divergent Typically model and high power rocket motors use standard convergent and divergent sections for a family of mot the nozzle throat drilled-out for the proper throat area for a particular motor. This straight, or drilled, throat p losses from the sharp corners at the throat, in addition to nozzle friction losses due to the boundary layer in the These nozzle losses typically result in a approximately 3.3% thrust loss for the straight or drilled throat. Combine typical nozzle divergence half-angle of 15 deg. the total thrust loss of 5% results in the typical model or high powe motor delivering 95% of the theoretical optimally expanded specific impulse for the propellant at a particular c pressure.

A rounded throat used with a conic or bell nozzle will typically reduce these nozzle throat losses to approximatel this remaining thrust loss being primarily due to friction losses in the nozzle. As mentioned previously, the advantage of the bell nozzle over the conic nozzle is the reduction to near zero of the nozzle divergence half-ar ideally expanded bell nozzle with a zero divergence half-angle utilizing a rounded throat can have thrust losses f theoretical ideal performance of 1.0% or less.

The ENGMOD and PCROC programs also allow the user to run the motor assuming theoretical ideal performance given expansion ratio of the motor nozzle. Divergence losses can be added to this theoretical ideal performance.

PROPELLANT CHARACTERIZATION:

In order to run the ENGMOD program you need to have the ballistic characteristics, Which are the Kn vs. c pressure curve, and the burn rate vs. chamber pressure curve. These curves are plotted on log-log plots; plots th log scales for both the abscissa and the ordinates.

Classically, propellants whose characteristics are plotted in this fashion will show the data points falling on a stra on the log-log plot. So the technique that is typically used to characterize propellant is to perform at least two te sometimes three tests, for each of the plots where the data points can be plotted, and then a straight line is drawn plots. Professionally this data is acquired by doing "bomb tests", where propellant is put in an enclosed sphe cylindrical device and then is burned under pressure. This is known as a "strand-burner test", because the pr sample is a strand; a long rectangular section. The rate at which this propellant burns down the strand is meas determine the propellant burn rate.

What has been found, for model, high power and experimental/amateur rocketeers, is that a better technique because it requires less sophisticated equipment and it actually gives you a better answer because it gives propellant actually installed in the motor, is to build a BATES grain motor to characterize the propellant. Again, w do a strand-burning test in a "bomb", as it's called, you do very accurately measure the burn rate and the Kn vs. c pressure of the propellant; but the propellant will act a little differently in the motor; it doesn't burn exactly the sa because of the physical characteristics of the chamber, which is the motor. Therefore characterizing the propellan motor is a good technique to use to get the actual INSTALLED ballistic characteristics of the propellant.

A BATES grain motor is a central cylindrical core-burning motor, where the cylindrical core burns from the insid outside; except instead of having one long grain, the grain is sliced into multiple segments. In a typical core-burnin the surface area is progressive with time. The cylindrical core grows in radius until it reaches the radius of the pr grain outside diameter. And as that radius grows, the contribution to the burning surface area from the core (whi pi r * length) increases. You don't want a motor to be extremely progressive; certainly not for a test where characterizing the propellant. So the technique used with the BATES grain is to take a single long grain that wou core-burner and to slice it into three or four grains. As the ends of these smaller grains burn they help to sho length of each of the grains; therefore keeping the surface area nearly constant.

Using a motor of this design, you can design a motor that fires at essentially a constant surface area; and theref constant Kn. In doing this you can produce a motor that, for example, fires at a Kn of 200 and maintains approximately 200 for most of the motor burn. You fire this motor on a thrust stand, and if the thrust is essentially c it can be a rather simple thrust stand like a spring or even a scale. The thrust of the motor can be used with the program to determine what chamber pressure the motor was operating at. As an example, you could fire the mot 100 lbs of thrust, and given the throat diameter, the exit diameter, the divergence angle of the motor and an a ratio of specific heats you can, using the PCROC program, determine what chamber pressure in that motor produ thrust.

You can fire two or three of these motors with two or three different Kn's, and you'll get two or three different thru two or three different chamber pressures. Each one of these data points is one point on the Kn - chamber pressur You plot all of the two or three points, draw a line through them, and you have characterized the propellant fo chamber pressure.

To characterize the propellant burn rate vs. chamber pressure the best technique to use is again to create a BATE type motor and note the web thickness of the motor. (Web thickness: the distance from the cylindrical core to the diameter of the propellant; i.e. the distance the propellant burns back from the face of the core to the inside case; course the motor runs out of propellant and the firing is over.) If the motor operates at essentially a constant Kn th operate at essentially a constant chamber pressure, and therefore will operate at an essentially constant burn rate with these two or three different tests with two or three different Kn's, thus two or three different chamber pressure get two or three different burn times ... and for each of those burn times you will have a chamber pressure to go Thus you will have now characterized at what burn rate the propellant burns at a given chamber pressure. Using tl rate vs. chamber pressure graph, you plot the three points, draw a line through the points and you have character propellant.

The ENGMOD program utilizes this data by having the user input just two points from the Kn vs. chamber pressu and two points from the burn-rate vs. chamber pressure curve. Using those two points the ENGMOD program co straight lines on the log-log plots and solves for the burn rate exponent (n)and the chamber pressure exponent (These are the exponents that are in the equations for chamber pressure as a function of Kn, and for burn ra function of chamber pressure. Once these exponents have been determined the equations for the chamber press the burn rate can be used to put the entire ballistic characteristics into the program. Finally, you also need to w propellant; to know how much went into the motor to be able to determine the propellant density.

... (Kn vs Pc; Rb vs Pc; Isp vs Isp from an old Aerotech catalog) ATP 8441-

For the chamber pressure ranges that high power rocket motors, and experimental/amateur rocket motors, operate at (250-1000 psi) the assumption that the propellant ballistic data will be a straight line on a log-log plot ho well. As you get to chamber pressures below 250 psi it tends to fall off a little. As you get to chamber pr approaching 3000 psi it can begin to fall off a little and some of the linearity on the log-log plot can be lost. But these high and low chamber pressures the linearity is an excellent assumption.

EROSIVE BURNING - What causes it? How to prevent it.
Erosivity is the process where the hot, high temperature gases flowing at a high velocity inside the core of th flowing over the burning surface of the core speed up the burn rate of the propellant The burn rate of the pr speeds up because of the scrubbing action; hence the term "erosivity", as this hot gas flows over the burni surface.

The Mach number in the core of a solid propellant rocket motor starts at zero at the head of the motor; builds subsonic value at the end of the core before it makes it to the plenum behind the core and then the nozzle, then o reaches Mach 1 at the throat of the nozzle. While the Mach numbers in the core are subsonic and low, the temper the gas in the core is extremely high, 5000 F. At this temperature the speed of sound of the gas in the core is at 3 per second. Therefore even though the core Mach number is subsonic the velocity can be quite high. One fundamental things that is understood about erosivity is that a propellant that is already exhibiting a low burn rate w its burn rate increased by a much higher percentage than a propellant that's already burning at a high burn rat propellants or motors operating at low chamber pressures or with low burn rates are more susceptible to erosivity predicting erosivity gets very empirical because the increase in burn rate is a function of many things; the mo elasticity of the propellant (the rate at which the propellant can bend and flex), and the size of the ammonium per crystals in the propellant (larger crystals tend to be rougher in terms of the propellant surface so as the gas scru the surface the burn rate will increase at a proportionally higher rate).

A general design challenge for solid propellant rocket motors is that you want to pack as much propellant into th as possible (a higher volumetric loading). An immediate way to get a higher volumetric loading is to decrease t diameter. In fact you can decrease the core diameter until the core diameter is identical to the throat diameter. approach that condition, you're getting more propellant in the motor but you're also increasing the core Mach num therefore the erosivity. If the core of the motor has the same diameter as the throat, the Mach number at the en core will be Mach 1, and in fact the throat could be removed and you would have a nozzle less rocket motor ... erosivity would be very, very high.

A general rule of thumb to use when designing solid propellant rocket motors or running the ENGMOD program is the diameter of the throat be approximately 3/4 the diameter of the core. The Mach numbers in the core will not high. It's very conservative but it will give good operability of the motor and you'll have a good, safe, conservative The diameter of the core can be decreased, but as it decreases towards the diameter of the throat the cor numbers will rise and the erosivity will get higher and higher.

Failure of motors from erosive burning effects is a very common occurrence. As the erosive burning causes the b to increase and raises the chamber pressure, this increase in the chamber pressure in turn raises the burn ra further.

Finally the erosive burning becomes so severe that as the scrubbing of the hot gas over the propellant continues, of propellant can be ripped from the face of the grain, greatly increasing the surface area. A motor failing from burning will typically explode right at the beginning of the burn as the core Mach number and the erosive burning i highest at the beginning of the burn. In fact a design trade can be made where erosive burning can provide a little spike at the beginning of the motor thrust; then as the core diameter grows the core Mach number falls, and the is reduced.

COMBUSTION INSTABILITY:

Typically when testing motors, especially as an experimental/amateur rocketeer, one may not have much thru equipment. You will create a new propellant and you really don't understand much about its ballistic charact Different propellants at a Kn of, as an example 200, may exhibit chamber pressures anywhere from 100 psi to 80 would depend on the propellant, the size of the Ammonium Perchlorate (AP) crystals in the propellant, the differe rate additives that were added ... it is a very empirical phenomenon determining just what the chamber pressure So a typical design technique used by experimental and amateur rocketeers is to construct the motor and fire it.

If the motor fires with a phenomenon known as "chuffing", where the motor spurts and fires, goes out, spurts ag goes out...this is typical of a motor where the chamber pressure has gotten too low. Chuffing is a phenomenon ca unstable pressure waves inside the motor. Not having a convergent section of the nozzle that goes all the way to of the motor, just having a small or no convergent section, all of these things can produce chuffing.

Typically chuffing can be avoided by having the motor operate at a higher chamber pressure. In fact chuffing is ex rare above 250 psi chamber pressure. In high power rocket motors for years chuffing was a problem but it has been completely eliminated by just operating the motors at chamber pressures above 250 psi. If the motor ch have to raise the chamber pressure, and the way to do that for the same design (same surface area of the gra reduce the diameter of the throat and therefore raise the Kn of the motor.

Also if you fire the motor and it blows up, typically that is because you operated at too high of a chamber p Therefore to get the motor to operate properly for the same grain design and surface area you would drill out the t reduce the Kn, reducing the chamber pressure of the motor.

In its crudest form in amateur rocketry you can have a new propellant that you create so you build a series of m they blow up - drill out the throats; if they chuff - make the throats smaller; and if they work you're in an acc though possibly sub optimal, operating range.

CHAMBER PRESSURE: typical values

In general the best chamber pressure to operate high-power rocket motors is typically between 450-500 psi c pressure. As an example, looking at the ATP-8441 propellant, at 250 psi chamber pressure it has a *theoretical impulse of 207 seconds. If the chamber pressure is raised to 500 psi the chamber pressure theoretical specific will increase to 227 seconds. As you can see from those two figures there's a big performance advantage to be going from 250 psi to 500 psi chamber pressure. When you raise the chamber pressure again to 750 psi the impulse only increases to 236 sec. So the big increase in the specific impulse is obtained by raising the chamber p from 250 psi to 500 psi. There's little benefit to be gained from raising the chamber pressure from 500 psi to 75 fact, as you go past 750 psi you typically cannot use fiberglass or phenolic cases. You end up having to go to case or to a carbon fiber-type case and these can get very expensive.

You'll see some interesting things by running the ENGMOD program, and then using the motors you design in th flight simulation program. In many cases by backing off on the chamber pressure you will lose specific impu lengthen the motor burn time. This produces a motor that, while having a lower delivered specific impulse, has a lower-thrust thrust curve that can propel the rocket to a higher altitude.

Typically for the cases that are used for high power rocket motors; phenolic and filament-wound fiberglass, if y operating at 600-700 psi you are getting very close to the approximate 800 psi where many of these cases and/c bulkheads will fail. In fact you will be so close to the limit that the normal production variances between mo produce an unacceptable level of bad motors. By operating at 450-500 psi you will have very high factors of safety ultimate strength capability of the phenolic and filament-wound fiberglass cases. Most high power rocket motors in the 450-500 psi chamber pressure range. An exception is the Aerotech reloadable motors as they use aluminum that can take higher pressures; many of these motors operate at chamber pressures up to 700-800 psi.

* note:
(Theoretical Isp = the ideal specific impulse delivered with a perfect nozzle. With a real-world nozzle and m delivered specific impulse is typically 95% of the theoretical specific impulse. The ENGMOD program determi actual delivered specific impulse.)

IGNITOR CONSIDERATIONS:

When firing any motor it is very important to place the igniter at the head end of the grain, not down at the end of tl by the nozzle. It's recommended to use an igniter with more pyrogen (the combustible material at the end of the ig in some cases taping thermalite to the end of the igniter it to increase its effectiveness. If the igniter is inadvertently at the nozzle end of the grain what will happen is only a small part of the propellant will get lit by the igniter If the i placed at the head of the motor the igniter will initially light the propellant located at the head end of the motor. O small amount of propellant ignites, the flame and gases from that propellant traveling down the core will ignite the the motor. If the igniter, though, is placed by the propellant down by the nozzle end, when the nozzle end prop ignited a lot of the gases become trapped in the core making it more difficult for the hot gas to travel UP the core the propellant at the head end, and in fact a large amount of the hot gas will travel out of the nozzle without ligh additional propellant in the motor. This can have the effect of igniting some of the propellant but not having it reac enough pressure to choke the nozzle and therefore achieve significant thrust, or enough propellant can ignite to ch nozzle and produce thrust, but at a very low chamber pressure.

This can be a very dangerous situation as the rocket may have enough thrust to clear the pad but not enough thru safely. In fact the rocket can leave the pad, flop around on the ground, have the entire surface area of the pr eventually ignite; or as is often seen the rocket sits on the pad with the motor burning without enough thrust to and this burns up the motor and eventually the rocket on the pad. In general if unsure whether the igniter has ma the way to the head end of the grain, remove the igniter, check for buckling in the wires and attempt to re-install th If the igniter is only 2/3 or 3/4 of the way up the core its not nearly as bad of a situation as if it is down by the nozzl general you should try to get the igniter as far up the core as possible.

CHOKING THE NOZZLE

Rocket motors use a nozzle known as the DeLaval nozzle. Once the chamber pressure in the plenum beh nozzle ...in this case after the core of the motor, is twice the ambient pressure outside the motor, the flow will ac to sonic velocity at the throat of the motor. Therefore for a given chamber pressure in the motor the nozzle is kn being "choked". Changing the chamber pressure inside the motor will change the mass flow through the nozzle interesting to note that once the nozzle is choked with the chamber pressure constant there is no variation in ma Since the flow is supersonic in the divergent section of the nozzle the pressure in that area does not feed forward throat or into the motor.

RATIO OF SPECIFIC HEATS

The ratio of specific heats is the Cp/Cv for the given propellant. Cp is the specific heat of the gas in the core at c pressure. Cv is the specific heat of the gas in the core at constant volume. The ratio of the specific heats is kr gamma (r) and is an important thermodynamic parameter for the gas flow. Gamma plays an important par equations for thrust of a rocket motor. Typically its values vary very little, between 1.20 and 1.30 at the temperatures in the core and nozzle throat region of a rocket motor. As an example, air at ambient conditions ha of specific heats, gamma, of 1.40. But the ratio of specific heats, gamma, for the gases in the core and in the n typically 1.20 to 1.30. The best way to determine gamma is to look up a typical value for a propellant that is very s the one that you're using. Many of the more advanced rocketeers building high power rocket moto experimental/amateur rocket motors can get access to runs of the JANAF or NASA Lewis thermochemical equ codes, where they look at the actual constituents of the gas, the species and the chemical products and detern actual gamma of the flow.

For ammonium perchlorate/hydroxyl-terminated polybutadiene (HTPB) propellant typical gammas are between 1 1.24. With higher percentages of binder (HTPB) in the propellant, which produces more black exhaust during because of unburned hydrocarbons, the gamma falls typically between 1.24 and 1.20. Generally, though, a good use for AP/HTPB propellants is 1.24.

MACH DIAMONDS

Another phenomenon seen in firing high power rocket motors is the phenomenon known as Mach diamonds diamonds are caused by either under or over expansion in the flow of the nozzle. If the nozzle flow is perfectly ex to ambient conditions there will be few if any Mach diamonds in the exhaust. When the pressure in the exit are nozzle equals the ambient pressure then the nozzle is known to be perfectly expanded. If the nozzle is too large; a has too high of an expansion ratio; (exit area / throat area = expansion ratio) the pressure in the exit area of the will be less than ambient pressure.

More typical of high power rocket motors is that they are a little underexpanded, which means the nozzle does no high enough expansion ratio and therefore the pressure exiting the nozzle is higher than the ambient pressure. Na attempt to correct for this by putting shocks into the plume so the flow can go through the required turns to ex ambient conditions. There will also be expansion fans. These shocks and expansion fans will reflect off the boundaries, which in essence form a pipe that the flow and the exhaust is flowing through. When these sh expansion fans cross in the center of the plume they cause a flow interaction with high localized heating that cre bright spots that are known as Mach diamonds.

Mach diamonds are typically best seen in clean burning propellants and/or propellants that do not have metal a the fuel to produce an afterburning effect. Propellants that have metal in the fuel that afterburns outside the motor a lot of unburned or partially burned hydrocarbon byproducts will tend to obscure the Mach diamonds by hav afterburning in the plume or the products from combustion basically obscuring the Mach diamonds that are almost present in the plume.

Even when perfectly expanded, some Mach diamonds are typically present in the exhaust. If the motor designe like to create Mach diamonds on purpose for a visual effect, typically reducing the expansion ratio by 10% to 2 good way to produce them.

EXPANSION RATIO

Expansion ratio is the exit area of the nozzle divided by the throat area. The total surface area of the motor burr burning surface area, divided by the throat area of the nozzle determines the Kn of the motor and therefore th chamber pressure. Given the chamber pressure that the motor is operating at, as the gases flow through the no expansion ratio determines the exit pressure of the motor, the pressure at which the flow exits the exit area of the Generally one wants to design so that the exit pressure of the motor is equal to the ambient pressure that the r intended to be operated at. For typical launch vehicles, especially upper stages of launch vehicles, very high ex ratios are used, because the pressure that the motor is attempting to match is nearly a vacuum and therefore es zero pressure. So large nozzles are put on these motors to get a very low exit pressure and therefore higher thr higher specific impulse. The highest thrust at any given altitude is achieved when the exit pressure equals the pressure. Since high power and experimental/amateur rocket motors typically operate at much lower altitudes; s to 10,000 feet; it becomes important to not either under-expand or over-expand the motor. A good value to use wh motor is operating between 450 and 500 psi chamber pressure is to use an expansion ratio of 4. This of cours expansion ratio that will be achieved when the diameter of the exit of the motor is twice the throat diameter.

HOW MANY GRAINS?

When designing a BATES grain, or coreburner, motor a question that arises is how many grains to use. Th number of grains to use is 1! In fact one of the big improvements made in high power rocket motors from the 70 mid 80's was the advent of the BATES grain motor design and its move from professional use to high power roc compare to the old coreburner designs. For example Enerjet motors, and Composite Dynamics motors, v coreburners. The problem with a coreburner is that because of the surface area progression from the initial cor final core the motors were very progressive, and progressively has several bad attributes. First, the chamber pre low at the initial part of the burn, which can induce chuffing. Furthermore with the chamber pressure low at the be of the burn the thrust is low, and that's when you want high thrust. When your rocket is leaving the launch pad you good, safe, stable launch and that's a period of flight where you would prefer a high thrust. Because a coreburne progressive it will have high thrust at the end of the burn, and this caused a lot of problems with a lot of pa plywood high power rockets because at the high thrust end of the burn the rockets would shred because of high c pressures. So you pretty much got the worst of both worlds with a coreburner; you had low thrust at the beginnin you wanted high thrust, and you got high thrust at the end and it shredded the rocket!

This is why rocketeers adopted the Bates grain motor design. Many coreburning motors can be 200% progressiv as much thrust at the end of the burn as at the beginning. For the same total grain length, by using vertical slice grain you raise the thrust at the beginning and lower the thrust at the end, resulting in a more neutral thrust trace are various advantages to various thrust traces for various Mach number regimes that a rocket may be operating i general a neutral thrust trace is one of the best. The reason why is that the motor can be operated at a constant c pressure right up against whatever limit that the designer has set and therefore achieve the highest Isp. For exa coreburner might be operating at a very high chamber pressure at the end of the burn and have a high specific i but it will have a very low specific impulse because of the low chamber pressure at the beginning of the burn, av out to a lower overall specific impulse. A better design technique is to decide what chamber pressure the motor c wants to tolerate and then operate the motor at that chamber pressure over the entire burn.

There are different ratios for the core diameter and the diameter of the grain itself, relative to the length of the gra will achieve essentially a flat thrust curve for a BATES grain motor. If fewer grains are used the motor will b progressive; if a larger number of grains are used the motor will be more regressive. By varying the number of grains in the ENGMOD program different thrust curves can be achieved.

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