Hybrid Propellant Motor Design

Kent Evans, Chris Ryan, Sam Heyd, Danny Cha, Johnny Wang

2/23/15

Washington State University

Recap

- Testing is vital
- Oxidizer - N2O

  Fuel - Undecided (HTPB, Parrafin, Mixture, Nylon)
- Hard to make design decisions without data!

Engine Ignition/Burn

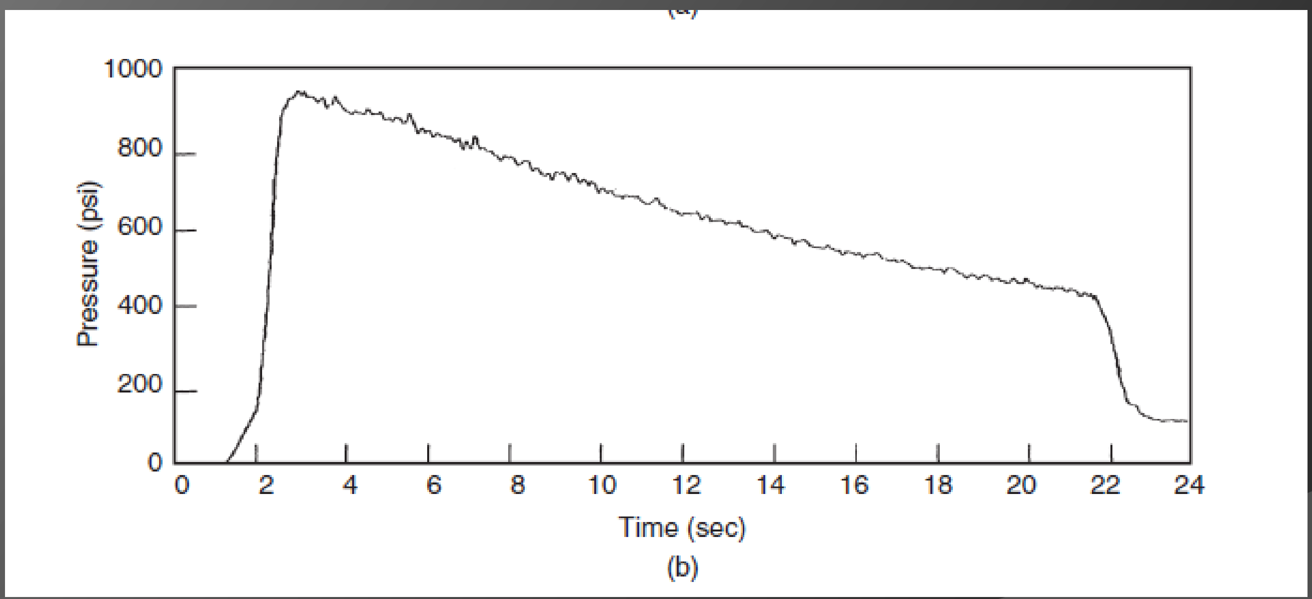


FIGURE 16–10.

Important Components/Considerations

Design Elements

- Chamber Pressure
- Chamber Temperature
- Injector
- Igniter
- Nozzle

Design Tools

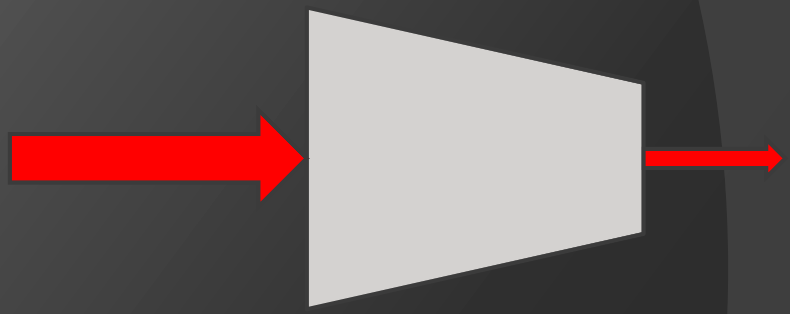
- RPA Modeling
- EES Modeling
- Test Stand/Empirical Data

Big Decisions

- Casing Size/Diameter
- Fuel Grain/Overall Length
- Oxidizer Tank Size
- Dependent on Overall Weight

Chamber Pressure

- Extremely Important
- Chamber Pressure analysis is complex
- “Hard Start” Danger
- Higher Chamber Pressure -> More Thrust (Generally)
- Nitrous Oxide self pressurizes around 730 psi
- Chamber pressure must be < 600 psi to ensure unhindered oxidizer flow and therefore stable combustion
- Use 500 psi to model at least initially



RPA Modeling

- Gas Constant
- K-Values
- Optimized Area Ratio
- Generates an Optimal Fuel Mixture Ratio for your input criteria

Thermodynamic properties (O/F=7.490)

Parameter	Injector	Nozzle inlet	Nozzle throat	Nozzle exit	Unit
Specific heat (p=const)	3.9222	3.9222	3.7800	1.5655	kJ/(kg·K)
Specific heat (V=const)	3.3517	3.3517	3.2473	1.2747	kJ/(kg·K)
Gas constant	0.3005	0.3005	0.2970	0.2858	kJ/(kg·K)
Molecular weight	27.6713	27.6713	27.9993	29.0957	
Isentropic exponent	1.1475	1.1475	1.1451	1.2278	
Density	3.4658	3.4658	2.1399	0.1528	kg/m^3
Sonic velocity	1068.3860	1068.3860	1030.2876	854.1806	m/s
Velocity	0.0000	0.0000	1030.2876	2398.0423	m/s
Mach number	0.0000	0.0000	1.0000	2.8074	
Area ratio	infinity	infinity	1.0000	6.0170	
Mass flux	0.0000	0.0000	2204.7267	366.4167	kg/(m^2·s)

- Very useful for getting accurate numbers for EES Modeling

Free download at: http://www.propulsion-analysis.com/

Mixture Ratio

- Optimal Mixture Ratio is still unknown
- Most likely between 7-8 (For highest specific impulse)
- RPA Modeling of N2O/HTPB predicts an optimal ratio of 7.490
- Approximately a 7.5 O/F ratio will be suitable for our rocket
- Keep this ratio in mind when selecting valves for the oxidizer injection
- A high oxidizer flow rate like this will be attainable due to a typically low regression rate
- Ratio will shift throughout the burn -> Hard to predict

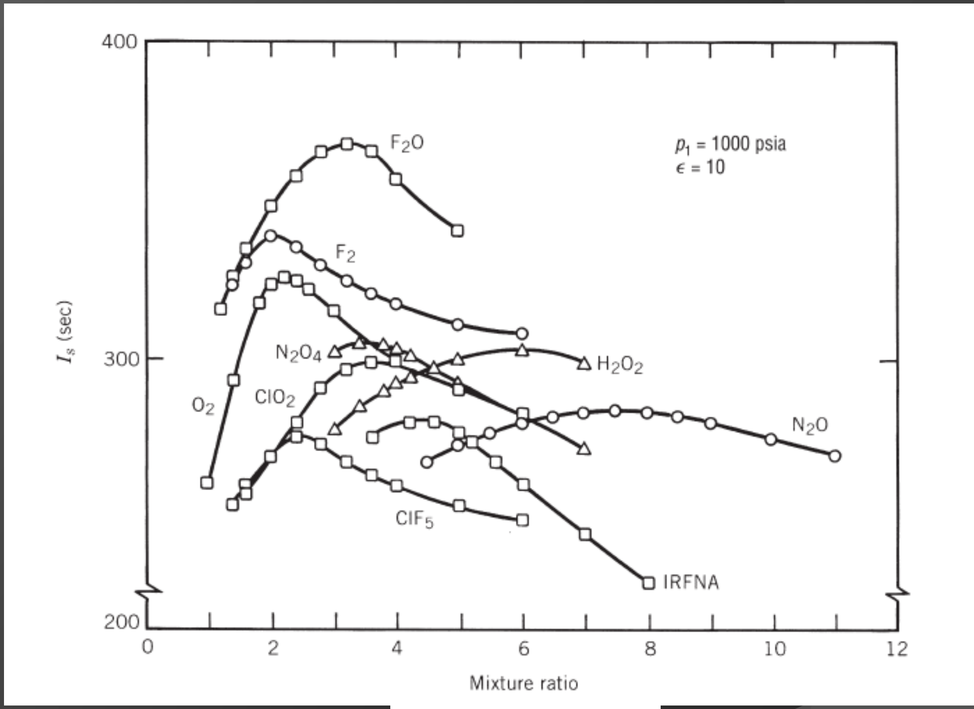


FIGURE 16–2.

EES Modeling

- Plenty of Assumptions were made
  - Assumed Burn Time (10s)
  - Assumed Initial Mass (60 lbs)
  - Constant Chamber Pressure/Temperature (500psi)
  - Ideal Compressible Flow
  - Choked Flow Throughout Burn
  - Casing Diameter (8 Inches) <- used for drag force calculation
  - No Throat Erosion/Clogging
- Initial Model predicts throat diameter of .6 inches will get us to 10500 ft with a ten second burn time after about 28 seconds

1 $total_{time}$	2 y_{ft}	3 $\dot{m}$	4 mass [kg]	5 Thrust	6 F	7 acc [m/(s2)]
25.51	10567	0.4261	22.95	1053	-229.1	-9.98
26.53	10613	0.4261	22.95	1053	-226.7	-9.875
27.55	10575	0.4261	22.95	1053	-225.3	-9.816
28.57	10621	0.4261	22.95	1053	-225.1	-9.805
29.59	10597	0.4261	22.95	1053	-224.3	-9.771
30.61	10500	0.4261	22.95	1053	-222.4	-9.689

EES Refinement

- Changing Chamber Pressure
  - Testing -> More Realistic Pressure Ranges/Change -> Changing Chamber Pressure in the model
- Incorporate Chamber Pressure into the Oxidizer flow rate model
- Integrate Regression rate (found experimentally) to model the mass flow rate into the post combustion chamber and the total mass lost by the rocket
- Develop a model for chamber pressure?

Nose Cone Considerations

- Low Half Angle limits drag
- Less drag

  ->Less Thrust Required

  ->Lighter Rocket
- Possible

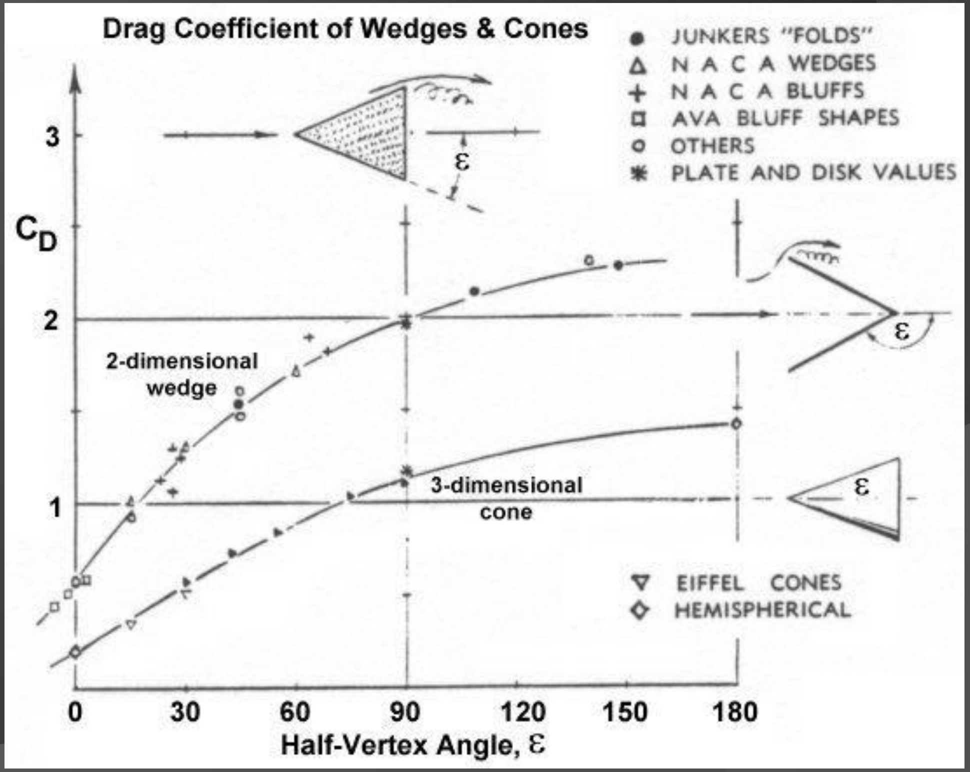
  Payload/Recovery System

  Storage
- 20° = .386, 11 inch Length

  30° = .5, 7 inch Length

$$C_D = 0.0112\,\varepsilon + 0.162$$

http://www.aerospaceweb.org/question/aerodynamics/q0231.shtml



Nozzle: Parameters

- Chamber length
- Chamber diameter
- Throat diameter
- Nozzle length
- Half angle
- Exit diameter
- Expansion ratio
- Contraction ratio

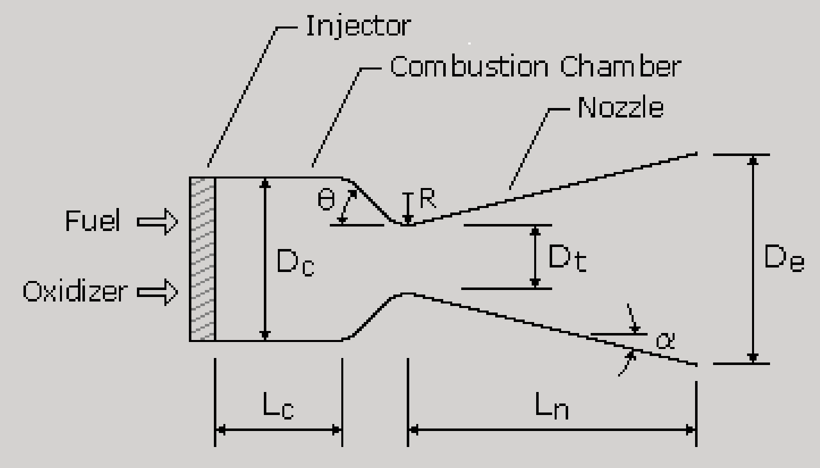


Figure 1.4

http://www.braeunig.us/space/pics/fig1-04.gif

Nozzle: Loss Sources

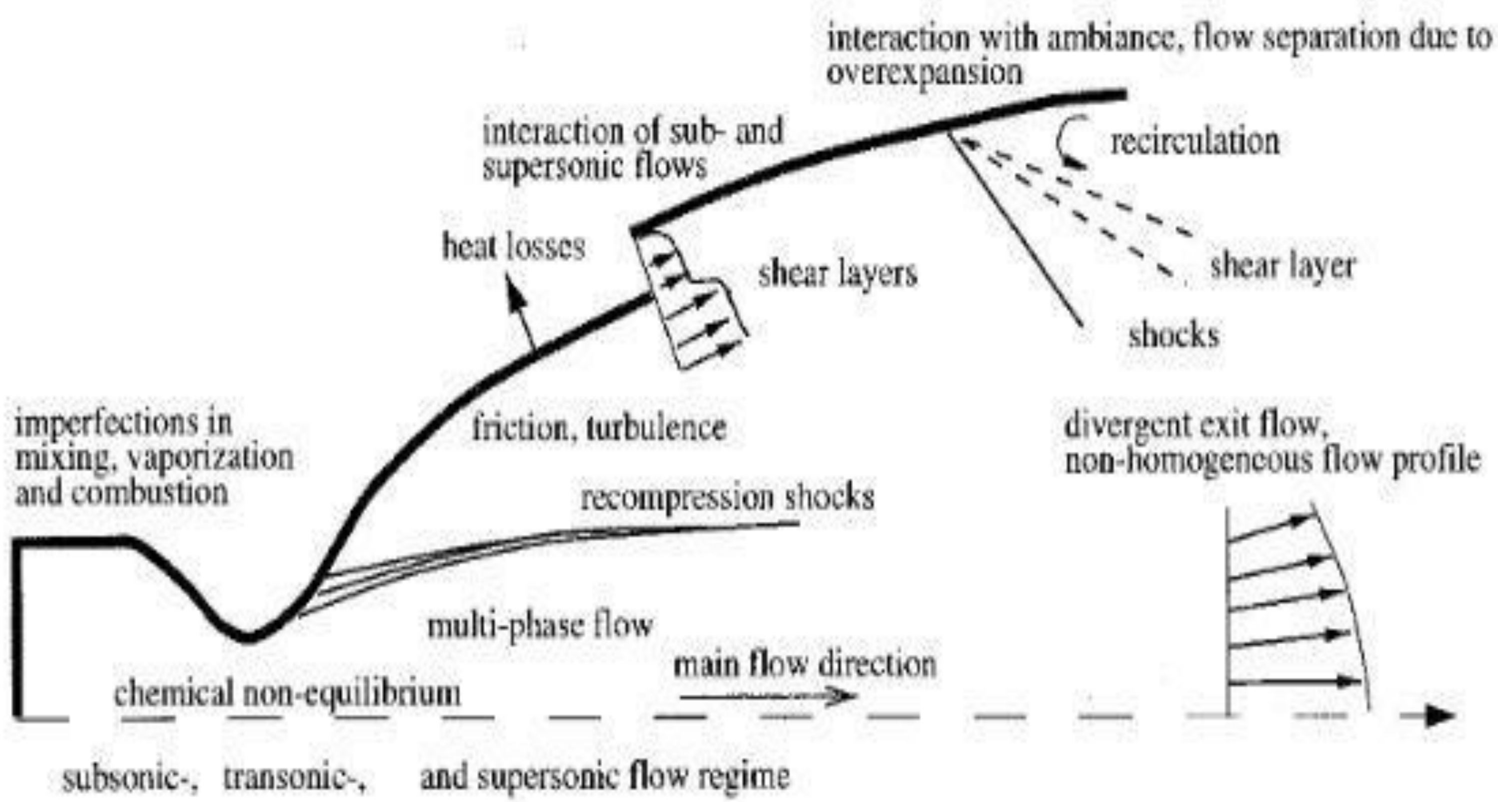


Fig. 1 Flow phenomena and loss sources in rocket nozzles.

http://www.space-propulsion.info/resources/articles/Advanced_nozzles.pdf

Nozzle: Important Loss Sources

Table 1 Performance losses in conventional rocket nozzles[a]

Losses	Vulcain 1, %	SSME, %
Chemical nonequilibrium	0.2	0.1
Friction	1.1	0.6
Divergence, nonuniformity of exit flow	1.2	1.0
Imperfections in mixing and combustion	1.0	0.5
Nonadapted nozzle flow	0–15	0–15

[a]Other loss sources also shown in Fig. 1.

http://www.space-propulsion.info/resources/articles/Advanced_nozzles.pdf

Nozzle: Important Loss Sources

- Viscous effects because of turbulent boundary layers
- Nonuniformity of the flow in the exit area
- Non adaptation of the exhaust flow to varying ambient pressures
  - Induces a significant negative thrust contribution
- Ambient pressures that are higher than nozzle wall exit pressures increase the danger of flow separation inside the nozzle
  - Results in possible generation of side loads!

Nozzle: Altitude Adaptation

- Nozzles can be complex, and be adaptive for higher performance with changing altitude.
- Simple nozzles are easy, but come with significant performance losses during off-design operation
- Many different nozzle types exist and are suitable for a variety of scenarios

Nozzle: Off-Design Examples

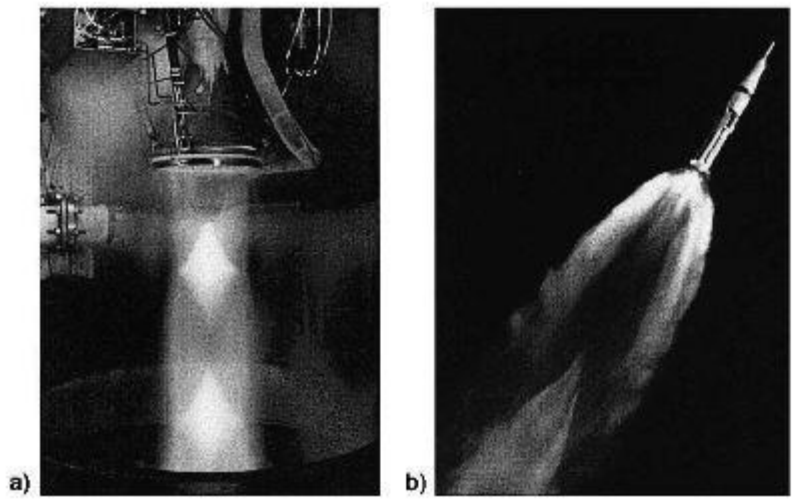


Fig. 2 Rocket nozzle flowfields during off-design operation: a) overexpanded flow RL10A-5 engine and b) underexpanded flow Saturn-1B, Apollo-7 (Photographs, United Technologies Pratt & Whitney, NASA).

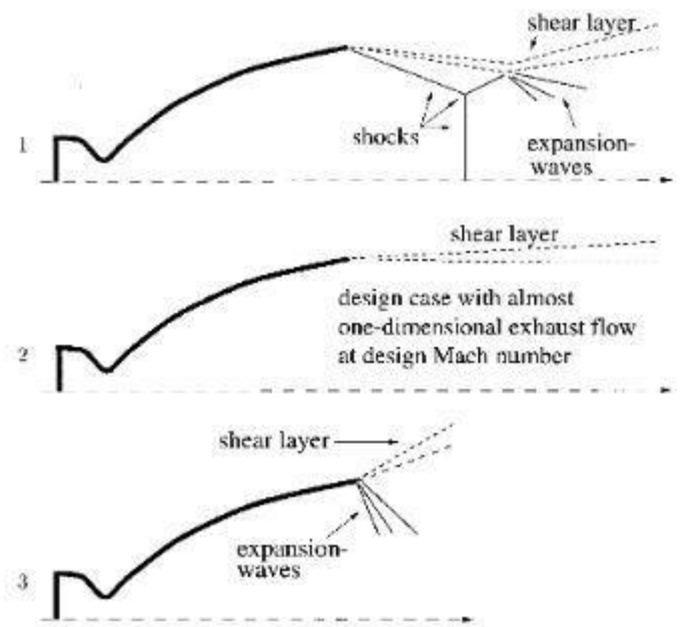


Fig. 4 Flow phenomena for a conventional rocket nozzle.

http://www.space-propulsion.info/resources/articles/Advanced_nozzles.pdf

Nozzle Types

- Nozzles with inserts for controlled flow separation
- Two-position nozzles
- Dual-bell nozzles
- Dual expander/ dual throat nozzles
- Expansion - deflection nozzles
- Plug nozzles

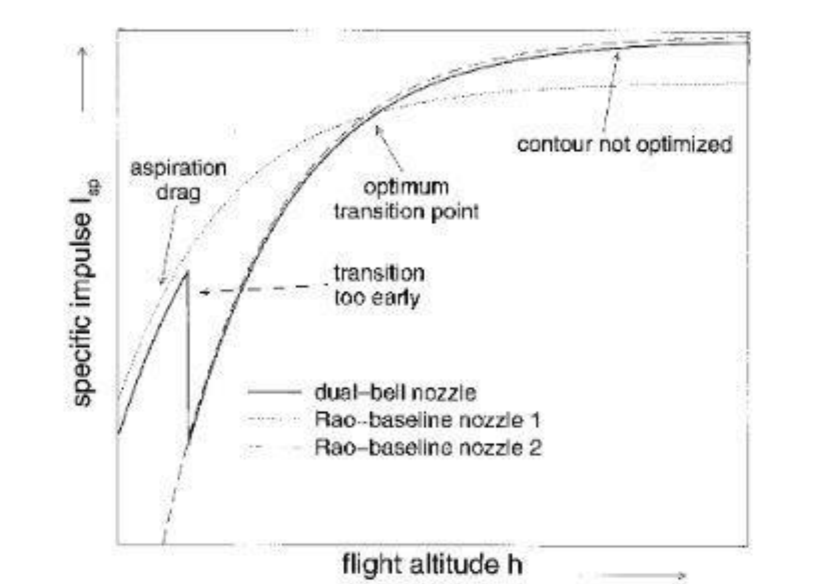


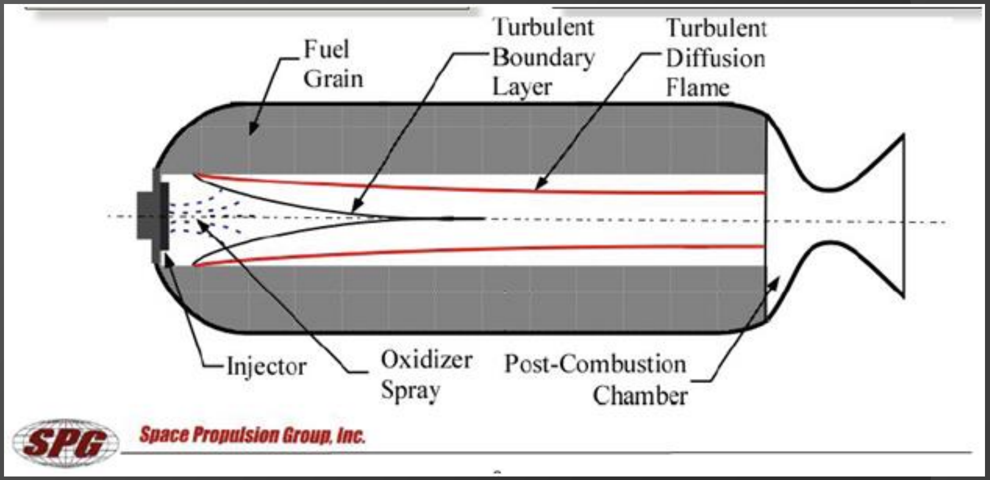
Fig. 11 Performance characteristics of a dual-bell nozzle. Performance is compared with two baseline bell-type nozzles as function of flight altitude (baseline nozzle 1: same area ratio as dual-bell base nozzle; baseline nozzle 2: same area ratio as nozzle extension).

Nozzle Selection

- Our design - 10,000ft.
- Only a small area expansion ratio is required
- We can neglect “aspiration” drag at such low altitudes and small areas, its effects will be minor
- Optimizing at one altitude will get the job done
- Options
    - Conical
    - Single bell contour

Injector

- Consists of holes, orifices, and passageways
  - Cross flow area
  - Flow rate
- Self-pressurizing oxidizer
- Design considerations
  - Two phase flow
  - Combustion Stability
  - Hole diameter
  - Hole inlet geometry
- Designs
  - Stanford 3” 9,600ft
  - Fintels 5” 15,000ft



http://aa.stanford.edu/events/50thAnniversary/media/Karabeyoglu.pdf

Injector - Self Pressurization Oxidizer

- N_2O - high vapor pressure
  - no pumps or pressurization system
  - = 730 psi @ room temp
- Problems
  - Dynamic and thermodynamic properties difficult to predict
- Flow rate affected by
  - vapor volume
  - heat and mass transfer

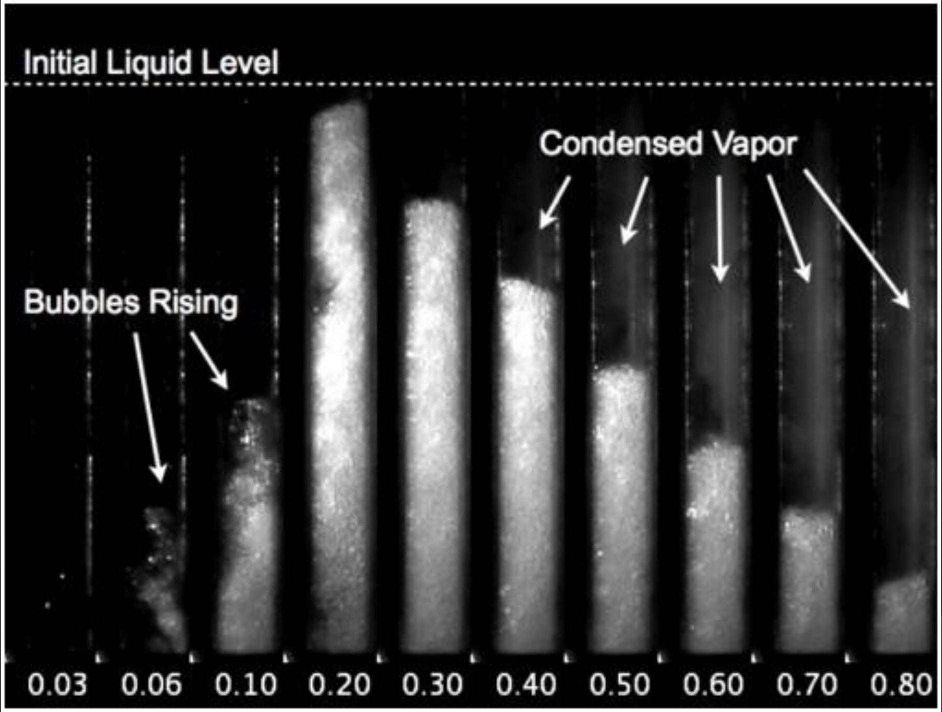


Figure 3: Still images from a video of a cold flow test. The white numbers at the bottom indicate the normalized time for each image.

http://spase.stanford.edu/Self-Pressurizing_Propellant_Dynamics.html

Injector - Two Phase Flow

- Static pressures within injector reach values below vapor pressure
  - Cavitation occurs
  - Flash vaporization
  - Decreases bulk fluid density
- Homogeneous Equilibrium Model (HEM)

$$\dot{m}_{HEM} = A\rho_2\sqrt{2(h_1 - h_2)}$$

  - Assumptions
    - liquid and vapor phases in thermal equilibrium
    - no velocity difference between phases
    - flow is isentropic in injector

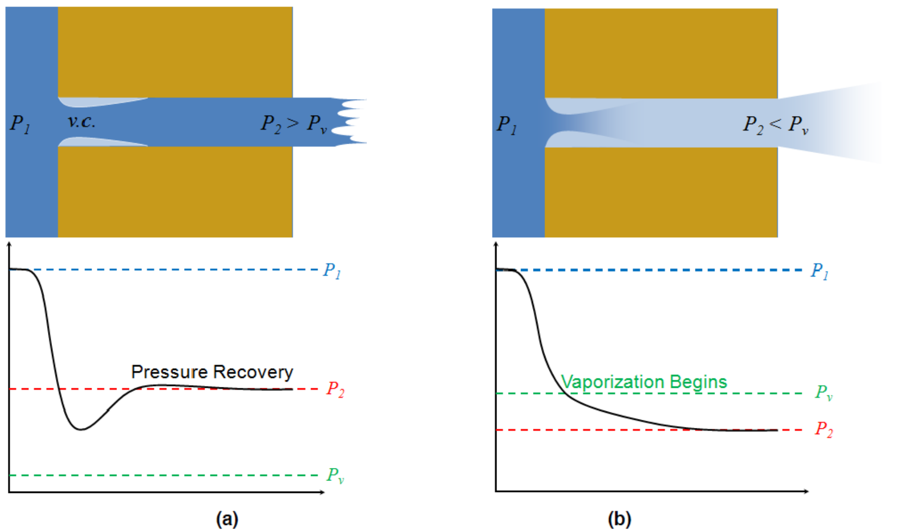


Figure 3: Conceptual injector pressure history as adapted from work by Dyer et al.[11] showing a low vapor pressure propellant (left) and a high vapor pressure propellant (right) with flow from left to right.

http://spase.stanford.edu/Self-Pressurizing_Propellant_Dynamics_files/Waxman%20et%20al.%20-%202013%20-%20Mass%20Flow%20Rate%20Characterization%20of%20Injectors%20for%20Use%20with%20Self-Pressurizing%20Oxidizers%20in%20Hybrid%20Rockets.pdf

Injector - Combustion Stability

- Chamber pressure lower than 80% of vapor
  - Decrease chamber pressure
  - Increase vapor pressure
- Increase temperature of oxidizer

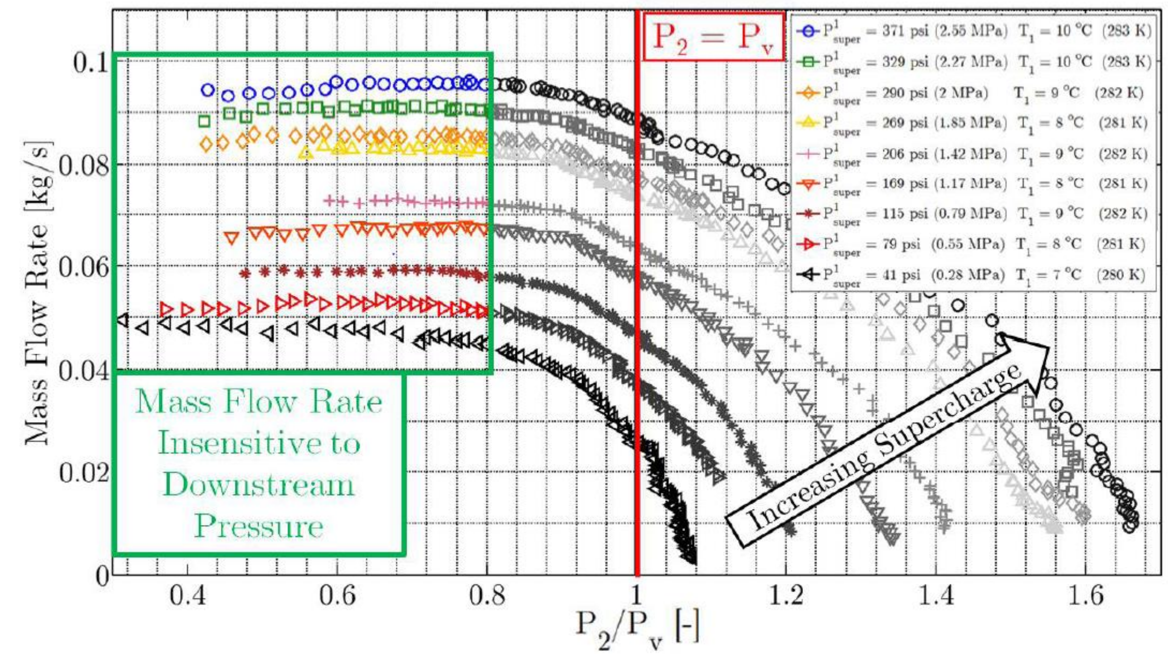


Figure 12. Mass flow rate vs. the ratio of chamber pressure to oxidizer saturation pressure (P_2/P_v). Data is presented for the flow of nitrous oxide through a sample injector over a range of supercharge values. Details of the injector design and the cold flow test apparatus used to make these measurements can be found in Ref. 5.

http://spase.stanford.edu/Self-Pressurizing_Propellant_Dynamics_files/Waxman%20et%20al.%20-%202014%20-%20Effects%20of%20Injector%20Design%20on%20Combustion%20Stability%20in%20Hybrid%20Rockets%20Using%20Self-Pressurizing%20Oxidizers%282%29.pdf

Injector - Hole Diameter

- Discharge Coefficient decreases as hole diameter increase

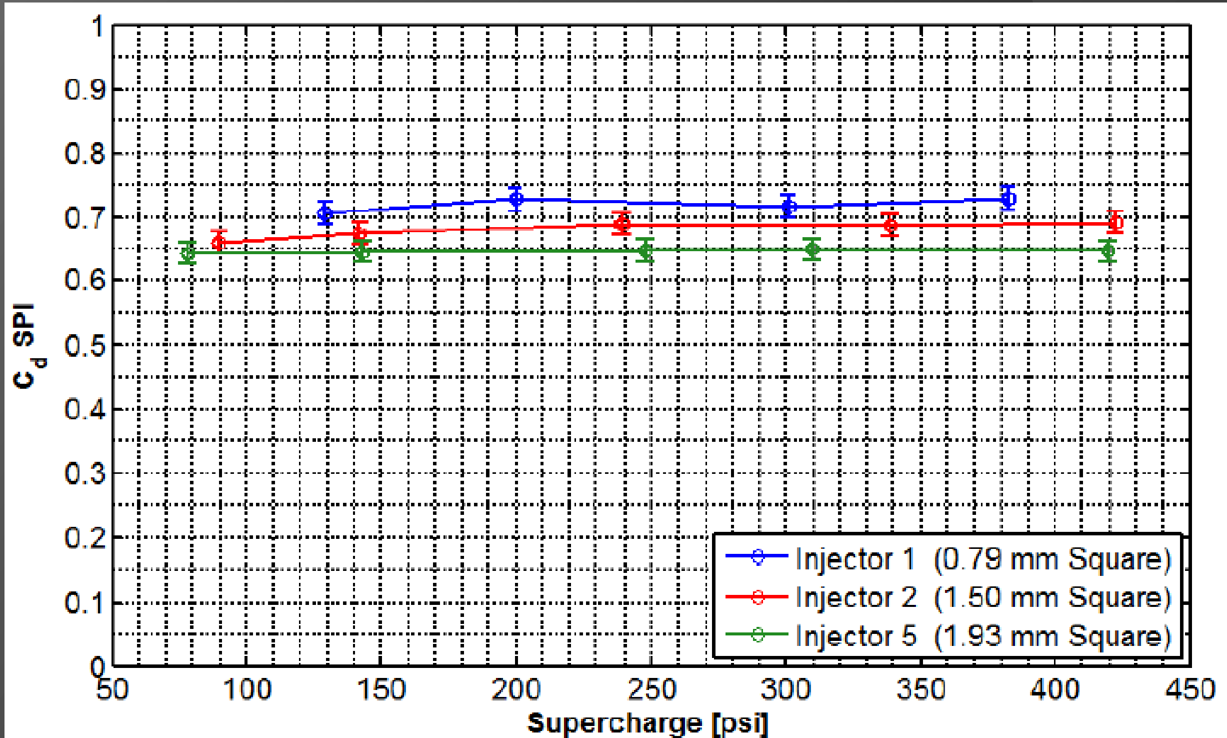


Figure 15. C_d in the single-phase region vs. supercharge for injectors number 1, 2 and 5 with nitrous oxide.

http://spase.stanford.edu/Self-Pressurizing_Propellant_Dynamics_files/Waxman%20et%20al.%20-%202013%20-%20Mass%20Flow%20Rate%20Characterization%20of%20Injectors%20for%20Use%20with%20Self-Pressurizing%20Oxidizers%20in%20Hybrid%20Rockets.pdf

Injector - Hole Inlet Geometry

- Square is significantly lower
- Rounded slightly higher than chamfered
- Extra effort to round edges is not worth the slight improvement

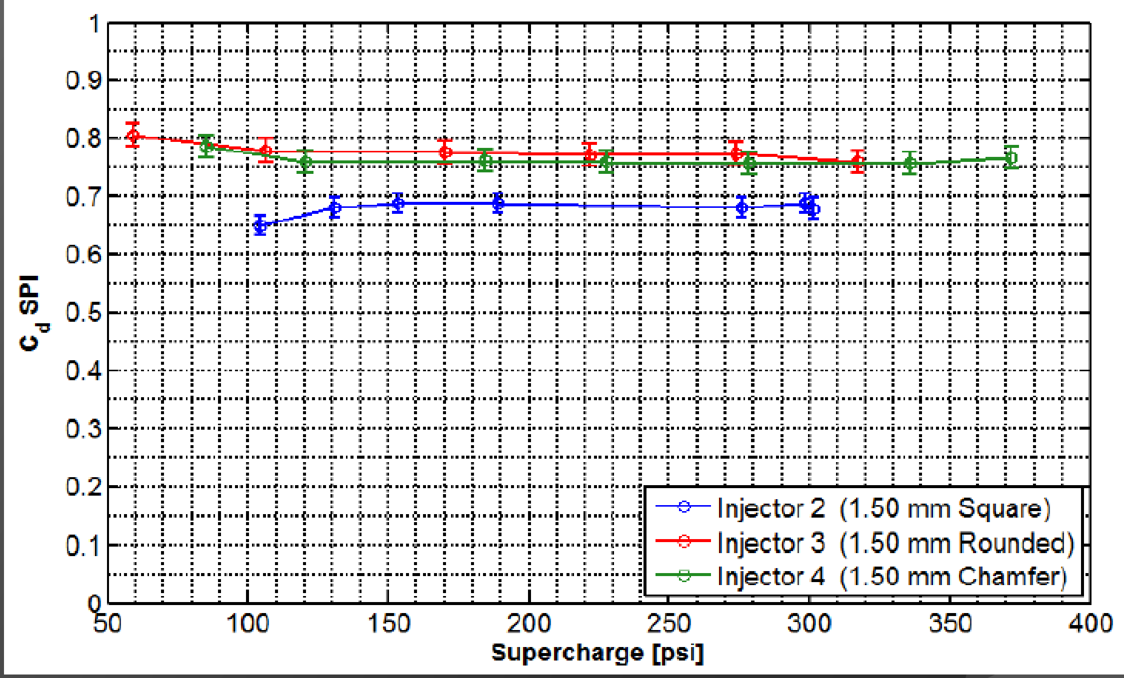
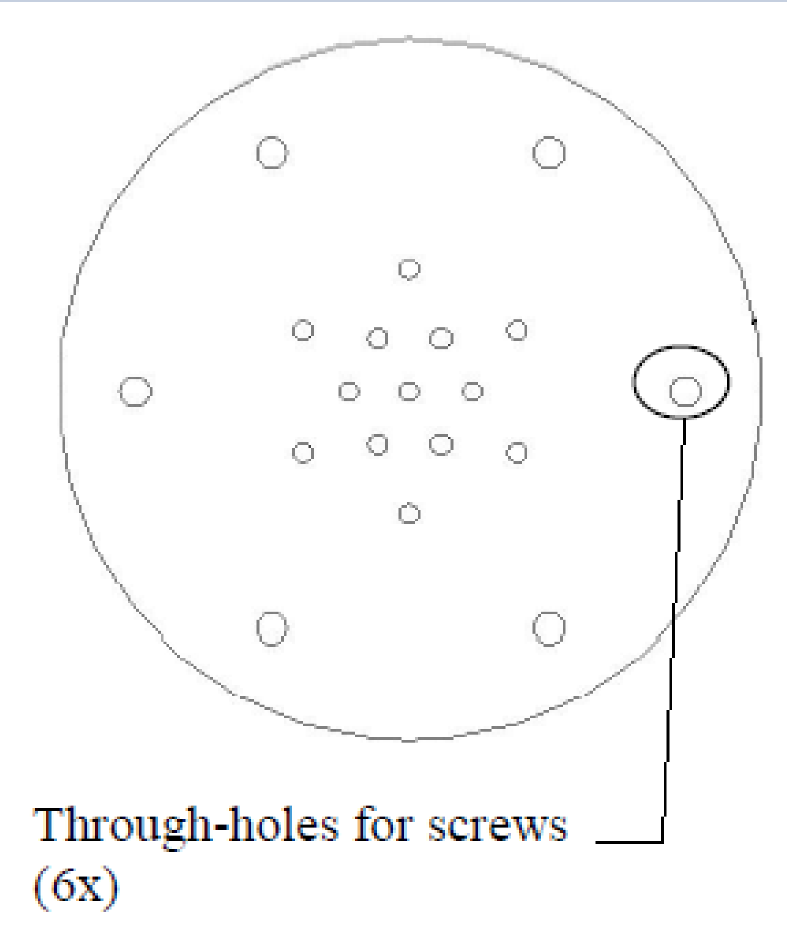


Figure 18: Cd in the single phase region vs. supercharge for injectors number 1, 2 and 5 with nitrous oxide.

http://spase.stanford.edu/Self-Pressurizing_Propellant_Dynamics_files/Waxman%20et%20al.%20-%202013%20-%20Mass%20Flow%20Rate%20Characterization%20of%20Injectors%20for%20Use%20with%20Self-Pressurizing%20Oxidizers%20in%20Hybrid%20Rockets.pdf

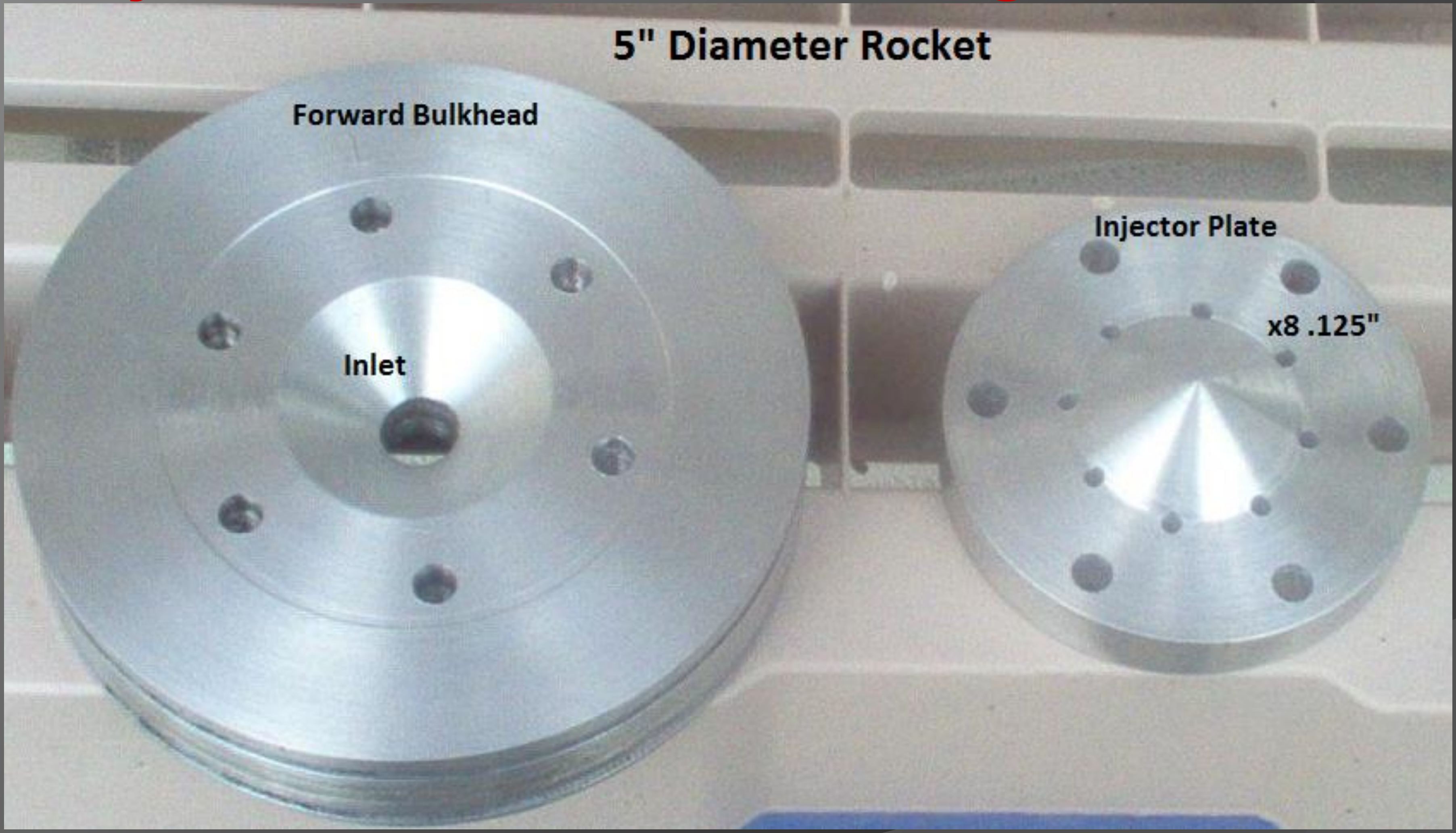
Injector - Stanford Design

- Showerhead-style 1/16” thick copper injector plate
  - Minimal weight
  - Copper is oxidizer-safe
  - Easily optimized flow
- 13-hole pattern
  - At .5” and 1” circles
- 0.067” hole diameter
  - compromise between atomization and avoidance of cavitation



http://web.stanford.edu/~hultgren/HybridRocket.pdf

Injector - Fintels Design



http://www.thefintels.com/aer/hr5.htm

Injector - Fintels

- 3 pounds of Nitrous Oxide
- [Click Here]

http://www.thefintels.com/aer/hr5.htm

Igniters

Priority of requirements:

- Specified Performance
- Specified Reliability
- Lowest Possible Cost

NASA- *Solid Rocket Motor Igniters* March 1971 pg. 45
http://electronicpackaging.asmedigitalcollection.asme.org/article.aspx?articleid=1408779
WSU Solid Team

Igniters

3D Printed Rocket Restart Test

https://www.youtube.com/watch?v=W_idSgO0jlQ

Stop and Start Igniter

- Competition only requires one flight
- Stop and start igniter would be too complicated and unnecessary

~~Stop Start Igniter~~

Igniters

Ematches

Jacobs' Rocketry, http://www.jacobsrocketry.com/experimental_rocketry.htm

Igniters

Pyrogen

- Nitrocellulose (NC) Lacquer
  - Acetone
  - Ping-Pong balls
  - 2:3 ratio of Acetone to Ping-Pong
- Black Powder

Jacobs' Rocketry, http://www.jacobsrocketry.com/experimental_rocketry.htm

Igniters

Penn Hybrid Rocket, https://sites.google.com/site/pennhybridrocket/design-1/igniter-and-pyro-grain

Igniters

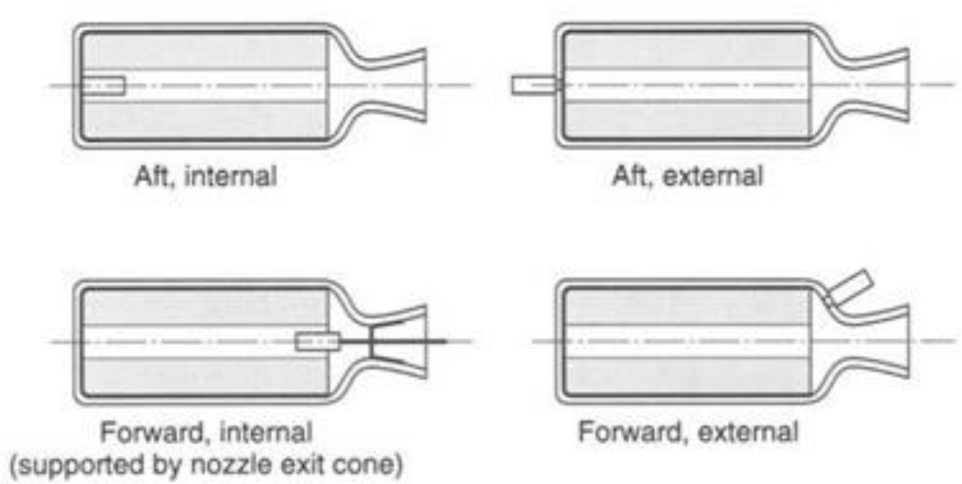


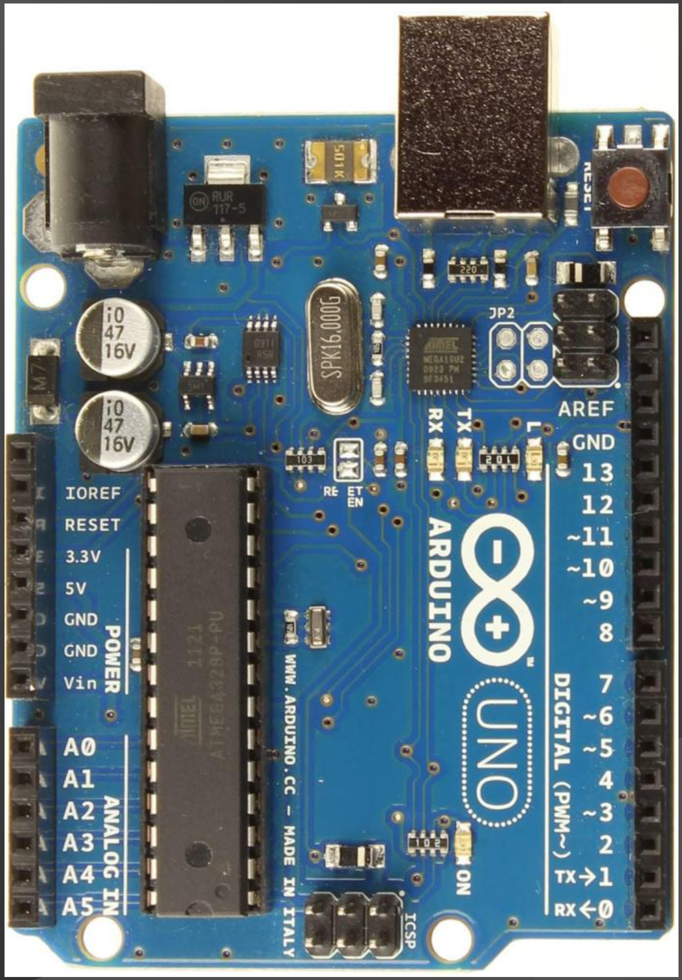
FIGURE 14–13. Simple diagrams of mounting options for igniters. Grain configurations are not shown.

Rocket Propulsion Elements, 8th Edition. Sutton, George P. and Biblarz, Oscar.

Arduino/MicroController

- Flight Controls
  - Altimeter
  - GPS
  - Oxidizer Valve Control
  - Ignition Power
- Develop Models to know when to cut the oxidizer (Prevent Overshooting 10,000)

One Unit



Arduino Choice

UNO-Atmega328

16MHz

Standard board

More Prebuilt libraries

Built-in power jack

7-12 volt

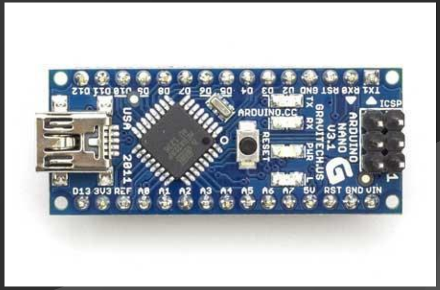
NANO-ATmega328

16MHz

Half the size

Light weight

7-9 volt



Problem

- Idea voltage 3-5 volt(DC)
  - Electric Valve 24-120 volt

Solution

- Use a transistor
- Use a Voltage divider

Testing

Testing Data:

- Average Regression Rate
- Burn Time
- Pressures on the Casing
- Oxidizer Amount/Flow rate Needed for Burn
- Temperatures for chamber and Throat

Equipment Needed:

- Thermocouples
- Pressure Transducers
- Data Acquisition System
- Flow Rate Sensors
- Load Cells
- Test Stand

Summary

- Stuck on many decisions without experimental data
- Modeling should help lighten the design load after testing
- One time ignition only -> Ematches
- Arduino board should be enough for avionics
- Injector -> Valve/“Showerplate”