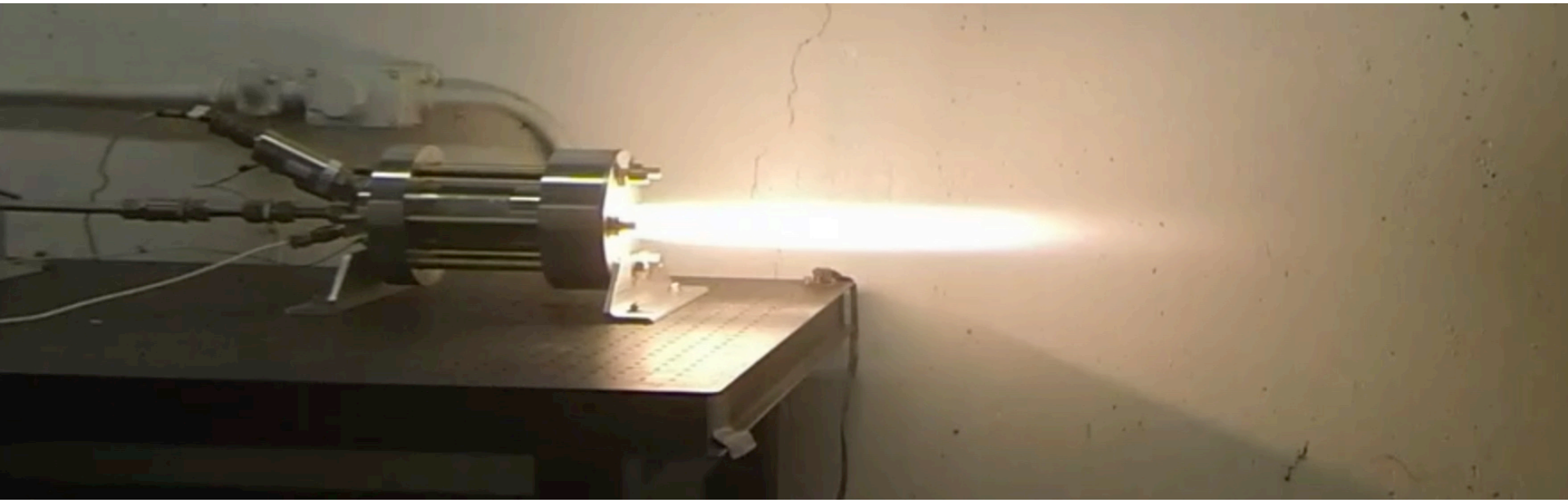
AA150R
Hybrid Rocket Propulsion



Prepared by Ashley Karp, Ph.D.

Jet Propulsion Laboratory, Caltech

May 2016

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Common Types of Rockets

Liquid Rockets

Advantages:
- High performance
- Throttle/restart

Disadvantages:
- Low density propellants
- Complex

Image Credit: NASA

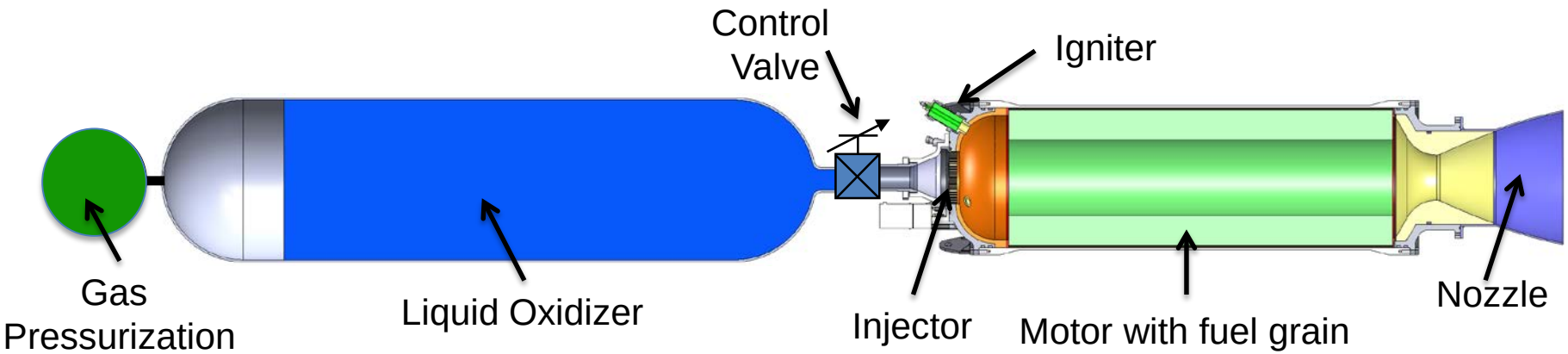
Solid Rockets

Advantages:
- High density propellants
- Low cost
- Simple

Disadvantages:
- Low performance
- Safety
- Can’t throttle
- Thrust is propellant temperature dependent

What is a Hybrid Rocket?

- Hybrid rockets use a solid fuel and liquid or gaseous oxidizer



Benefits of Hybrid Propulsion

- ***Safety:*** Hybrid propulsion systems are inherently more safe than liquid or solid systems. The fuel and oxidizer are separated by both phase and physical location.
- ***Low Cost:*** Inert propellants lead to increased safety, ease of handling and decreased operational costs. Short development times of large scale systems have been demonstrated, for example AMROC's DM-01 motor was designed and tested in just over one year [4].
- ***Tolerance to Debonding and Cracks:*** Burning down cracks is unlikely because oxidizer is not present at the fuel surface.
- ***Decreased Complexity***: Hybrids only require half the propellant delivery system compared to a liquid rocket.
- ***Throttling:*** The solid regression rate only depends on the oxidizer flow rate, therefore, they can be throttled by opening or closing one valve.
- ***Performance:*** Hybrids have increased performance compared to solids and generally enjoy a specific impulse more nearly compared to liquid systems. The solid fuel grain makes it easy to add performance enhancing materials (e.g. aluminum).
- ***Packaging Flexibility:*** The oxidizer and pressurant (if required) can be stored in any shaped pressure vessel or even split between multiple vessels and the solid fuel can be stored within the combustion chamber.
- ***Temperature Tolerance:*** The regression rate (and therefore thrust) of hybrid fuels is not pressure dependent as it is in solids. It is possible to launch hybrids in a greater range of temperatures and with minimal to no thermal conditioning for most applications.
- ***Chamber Pressure Tolerance:*** The solid fuel regression rate is only weakly dependent on chamber pressure over most of the range of useful oxidizer mass fluxes.
- ***Environmental Safety:*** Hybrids burn relatively cleanly because of the high hydrogen to carbon ratios of the fuels. They do not use perchlorates or nitrates nor do they produce hydrochloric acid.

Challenges of Hybrid Propulsion

- ***Low Regression Rate:*** Classical hybrid rockets suffered from low regression rate fuels and diffusion limited mixing. Thrust levels required by most applications led to the requirement for increased burning area through the use of multiple ports.
- ***O/F Shift:*** The port area in the combustion chamber increases as the solid fuel burns. This causes the oxidizer to fuel ratio (O/F) to shift over the course of the burn. This can lead to operation at less than optimal mixture ratios. Careful design can minimize the impact on performance (peak Isp) to less than 1%
- ***Transient Times:*** Hybrid rockets have multiple transient events that are necessary for nominal operation (ignition and shut down), some that are imposed by mission requirements (throttling) and several that are undesirable (instabilities). Ongoing research has been focused on their mitigation.
- ***Low Technological Readiness Level:*** Hybrids are at a comparatively low TRL and have not been fight proven for most practical applications. The exception is SpaceShipOne (an HTPB/N2O hybrid), which won the Ansari X Prize.
  - Their TRL is steadily rising due to renewed interest in hybrids driven by paraffin fuels and the development of SpaceShipTwo for space tourism. Hybrids are also being considered for CubeSat propulsion systems and a potential Mars Ascent Vehicle.

History of Hybrid Rockets

- 1933: GIRD-9, Soviets
- 1960s: Research across the US,
  - Marxman, Wooldrige, Muzzy, Altman, etc. develops classical theory
  - Motor testing up to 40,000 lbf
- 1964-1984: Flight System Development
  - Sandpiper, HAST, and Firebolt target drones (USA, Chemical Division of UTC)
  - LEX Sounding Rocket (ONERA)
  - FLGMOTOR Sounding Rocket (Sweeden)
- 1973-1982 Indian Institute of Science
  - Hypergolic solid fuel with RFNA
  - 96% c* efficiency

GIRD-09 Rocket. Cantwell, Karabeyoglu, and Altman. “Recent Advances in Hybrid Rockets.” International Journal of Energetic Materials and Chemical Propulsion 9(4), 305–326 (2010)

History of Hybrid Rockets

- Early History (1933-1960)
  - First flight of a hybrid rocket was on 17 August 1933

Date	Rocket	Who built it?	Notes
1932-1933	GIRD-9: LOx/Gellified Gasoline suspended on a metal mesh	Tikhonravov and Korolev	500 N, 7 in diameter
1937	Coal/Gaseous Nitrous Oxide	I. G. Farben, Germany	10kN
1938-1939	LOx/Graphite	Prof. Hermann Oberth, Germany	
1938-1941	Coal/GOx	California Rocket Society	
1947	Douglas Fir/LOx (XDF), Wax/LOx, and Rubber/LOx	Pacific Rocket Society	Doug Fir Nozzle (bad idea)
1951-1956	Hydrogen Peroxide (90%)/Polyethylene	GE	>300 tests

Chiaverini and Kuo, “Fundamentals of Hybrid Rocket Combustion and Propulsion”

History of Hybrid Rockets (cont.)

Enlightened Period (1960-1980's)

- High-Energy Combinations
  - NASA sponsored technology development for space motors
  - Tried propellants like Lithium, Hydrogen and Fluorine and FLOX
- Target Drones
  - Sandpiper: PMMA/Mg/MON-25 (UTC/Beech Aircraft), 1968. Advances: rapid development in 18 months. 8:1 throttle, Mach 4, 100,000 ft, flight time > 300 s.
  - High Altitude Supersonic Target (HAST): PB/PMMA/IRFNA, pressurized with a ram air turbine, 10:1 throttle, cruciform port with 4 injectors
  - 48 second generation HAST motors were delivered to military specs by CSD and Teledyne Ryan Aircraft in the mid 1980's
- High Altitude Sounding Rockets
  - ONERA/SNECMA (1964-1967): hypergolic using RFNA and amine fuel (metatolunere diamine/nylon), record altitude of > 100 km
  - Sweeds (Volvo Flygmotor): hypergolic using RFNA and Tagaform and later Sagaform

High Energy upper stage Li/LiH/PBAN with F_2/O_2
(Fig 3 from Chiaverini & Kuo)

History of Hybrid Rockets (cont.)

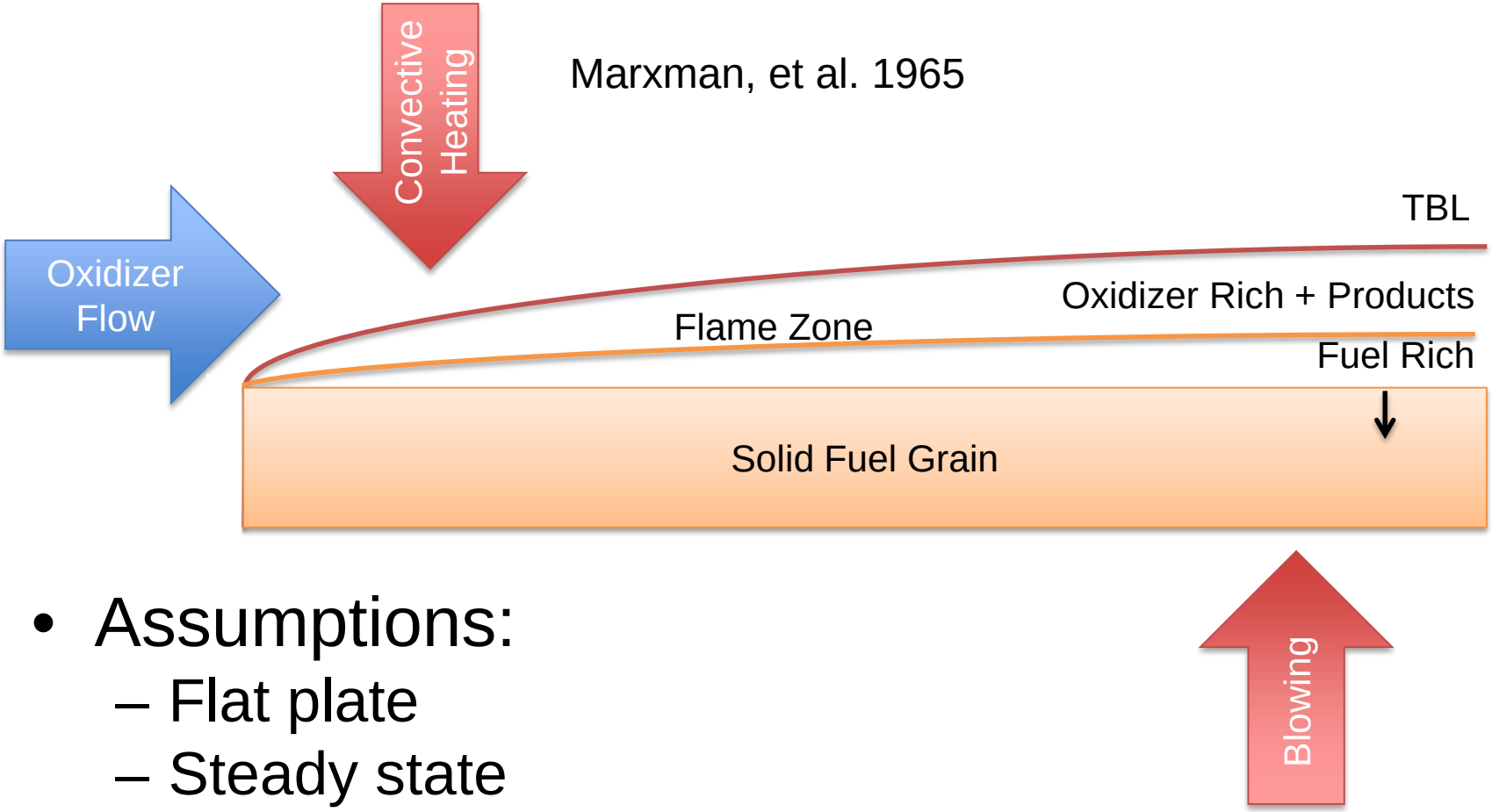
- Recent History
- **1981-1985**: STARSTRUCK company developed and sea launched the Dolphin sounding rocket LOx/PB (35,000 lbf of thrust, 42 in diameter)
- **1985-1995**: AMROC (New name for STARSTRUCK)
  - Tested 35,000-250,000 lbf thrust \motors. LOX/HTPB motor.
- **1990's**: Hybrid Propulsion Development Program (HPDP).
  - Successfully launched a small sounding rocket.
  - Developed and tested 250,000 lbf thrust LOX/HTPB motors.
- **1998**: First firing of paraffin-based hybrid at Stanford University.
- **2002**: Lockheed developed and flight tested a 24-inch LOX/HTPB sounding rocket called HYSR (60 klbf. thrust).
- **2003**: Scaled Composites and SpaceDev developed 20klbf. Nitrous oxide/HTPB hybrid motor for the sub-orbital manned vehicle SpaceShipOne.
- **2013**: First powered flight of SpaceShipTwo
- **2016/2017**: Peregrine Sounding Rocket, NASA Ames

Source: Scaled composites

Source: Virgin Galactic

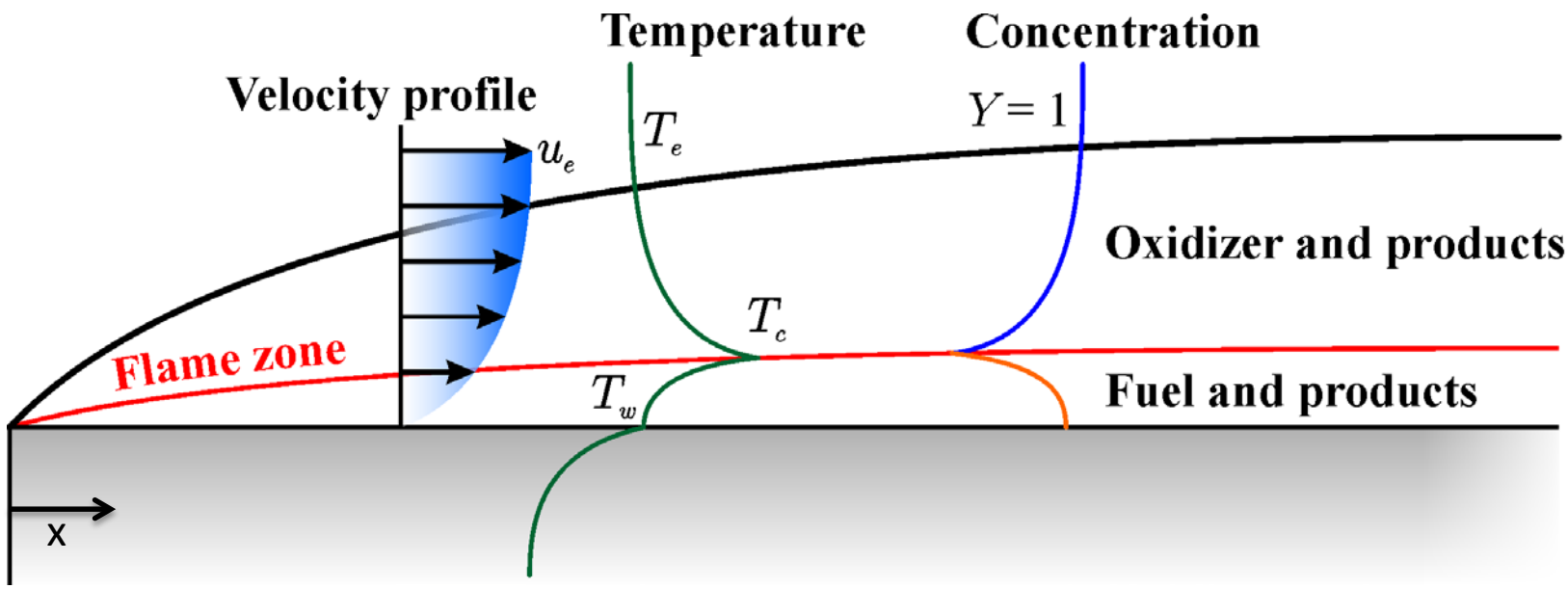
Source: ntrs.nasa.gov

Classical Hybrid Combustion



- Assumptions:
  - Flat plate
  - Steady state
  - Boundary layer is turbulent over the entire length
    - Re ~ 10^4, but mass addition destabilizes the flow

Marxman continued, Diffusion Limited Combustion



- Regression rate law:
  - 0.4≤ n ≤ 0.7
  - m is very small
  - a is an empirical constant

$$\dot{r} = \frac{a}{x^m} G^n \qquad G = \frac{\dot{m}}{\pi r^2}$$

$$\dot{m} = \dot{m}_{ox} + \dot{m}_f$$

Marxman continued, Diffusion Limited Combustion

Oxidizer mass flux simplification

$$\dot{r} = aG_{ox}^{n} \quad \text{, where} \quad G_{ox} = \frac{\dot{m}_{ox}}{\pi r^2}$$

Substitute the oxidizer mass flux into the regression rate equation, separate variables and integrate both sides:

$$\frac{\pi^n}{a} r^{2n} dr = \left(\dot{m}_{ox}(t)\right)^n dt$$

$$r(t) = \left(r(0)^{2n+1} + \frac{a(2n+1)}{\pi^n} \int_0^t \left(\dot{m}_{ox}(t')\right)^n dt' \right)^{\frac{1}{2n+1}}$$

This approximation assumes that the port radius is constant with length. Also a in the approximation case is slightly different from the original a.

Regression Rate Constant

- The regression rate constant a has units

mm/s ← $\dot{r} = \frac{a}{x^m} G^n$ → g/cm^{2}s

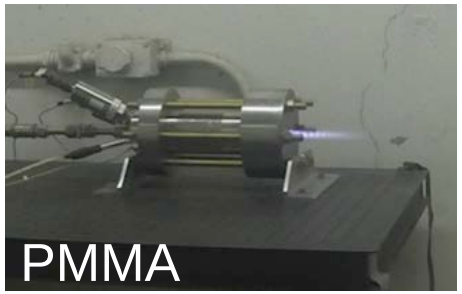
x^m → mm

- Always check the units for regression rate and G_{ox}.
- If you have to convert a from mm/s to m/s, there is a factor of n.

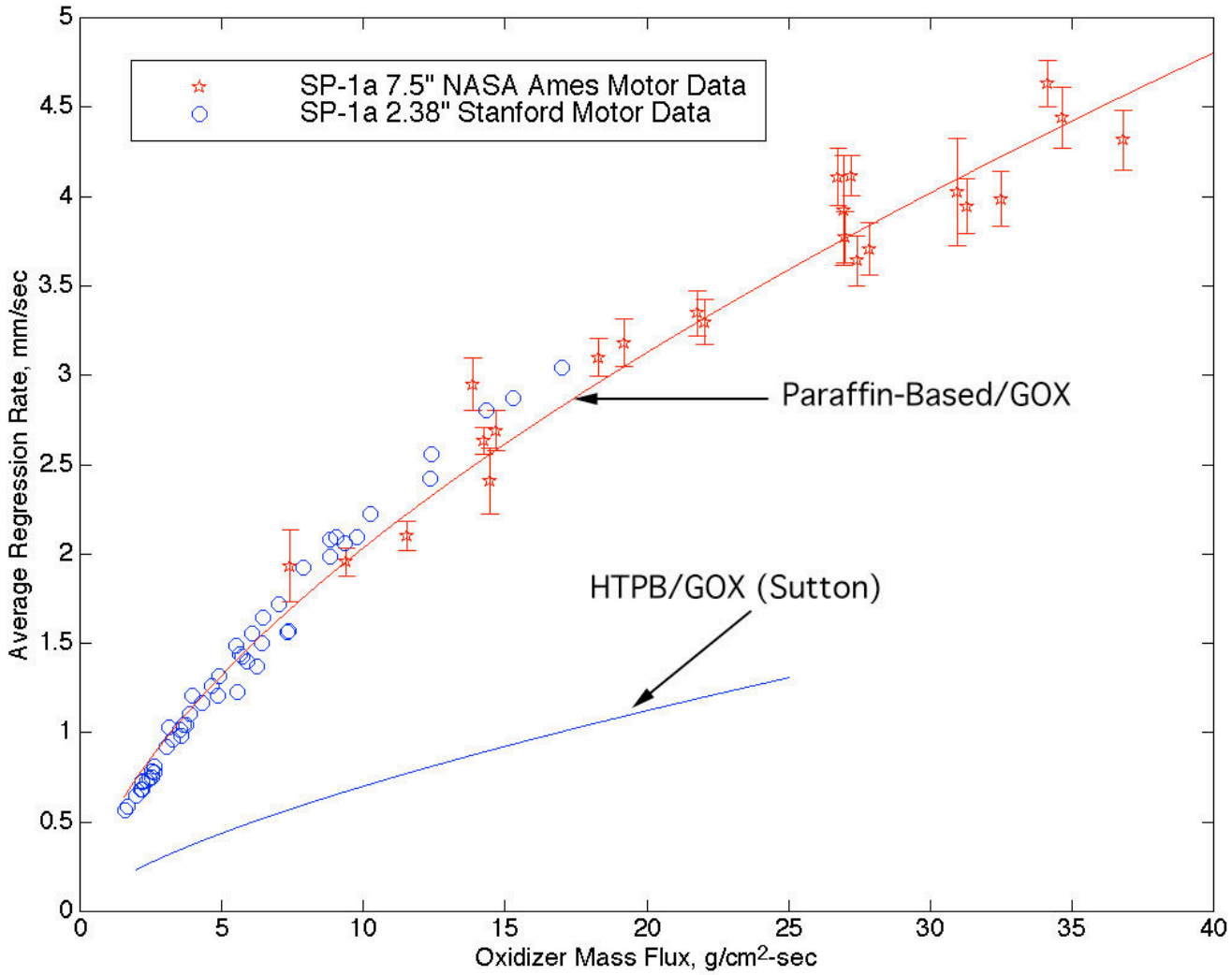
Typical Coefficients

In this case a is given such that r is in m/s and G_{ox} is in kg/m^{2}s, $\dot{r} = aG_{ox}{}^{n}$

Fuel		a ($\times 10^{-5}$)	n	Typical O/F
PMMA	GOx	2.11	0.62	1.3-1.6
	N_2O	5.813	0.5	2.05-2.6
Paraffin	GOx	11.7	0.62	2.15-2.65
	N_2O	15.5	0.5	3.45-4.4
HDPE	GOx	2.34	0.62	2.15-2.65
	N_2O	4.89	0.5	3.45-4.4
HTPB	GOx	3.043	0.681	1.9-2.2
	N_2O	8.7785	0.5	5.95-7.75



Regression Rate vs. G_{ox}



Karabeyoglu, et al. “Scale-up tests of High Regression Rate Liquefying Hybrid Rocket Fuels.” AIAA-2003-1162

Combustion Mechanisms Change with Mass Flux

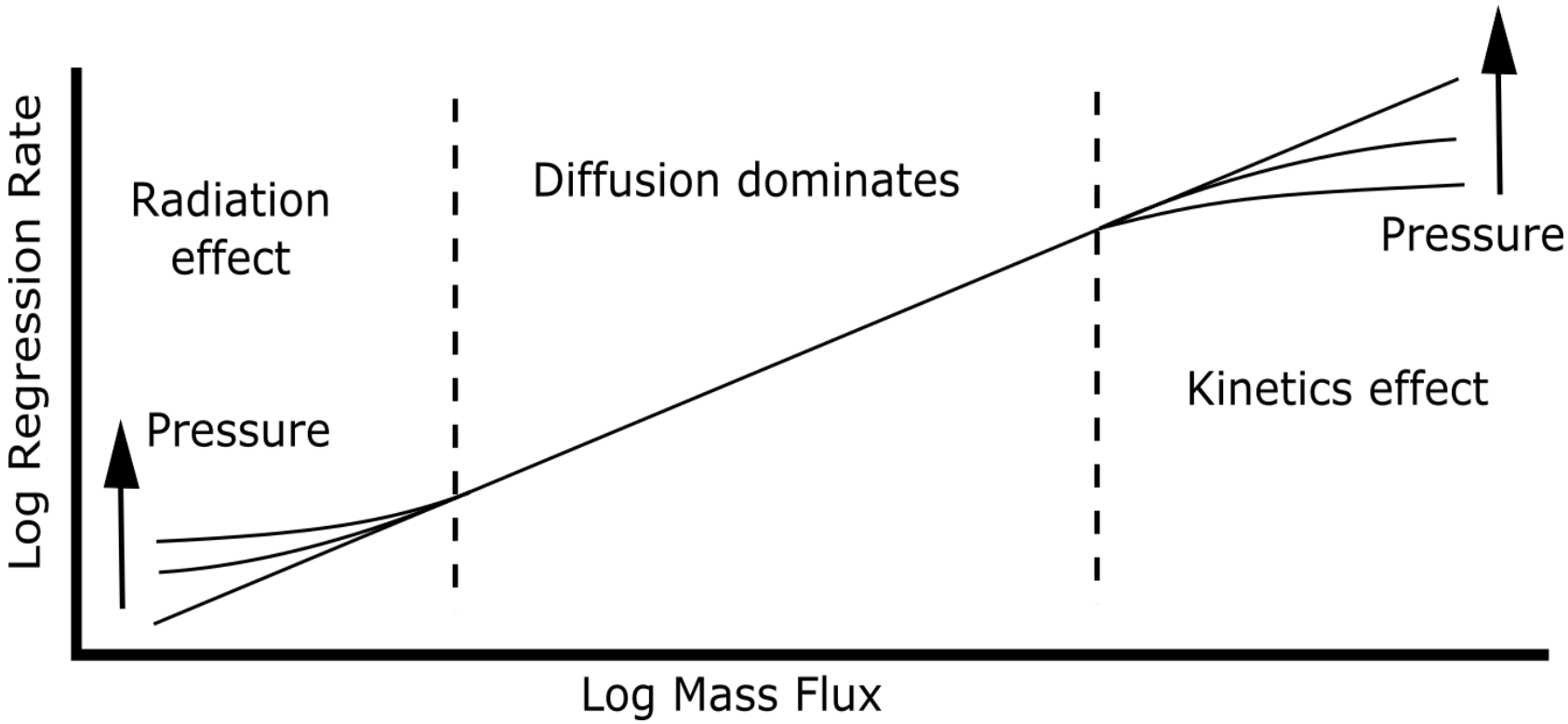
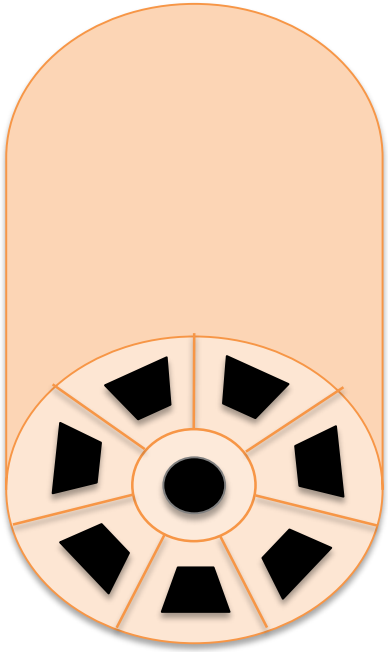


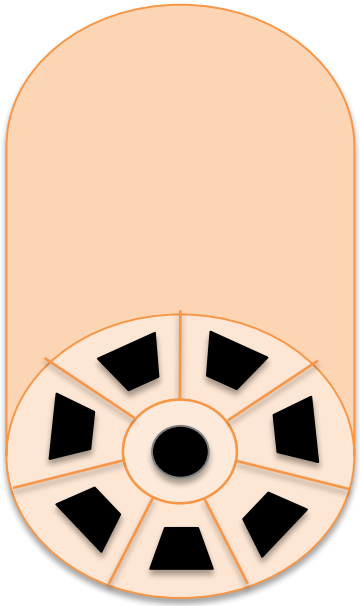
Figure derived from figure in Humble and Henry (1995).
Source: Humble, R. W., Henry, G. N., Larson, W. J., et al., Space propulsion analysis and design, Vol. 1, McGraw-Hill New York, 1995.

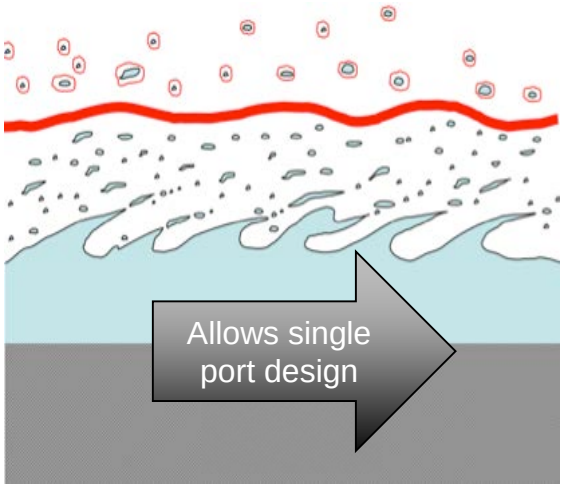
Disadvantages of Multiports

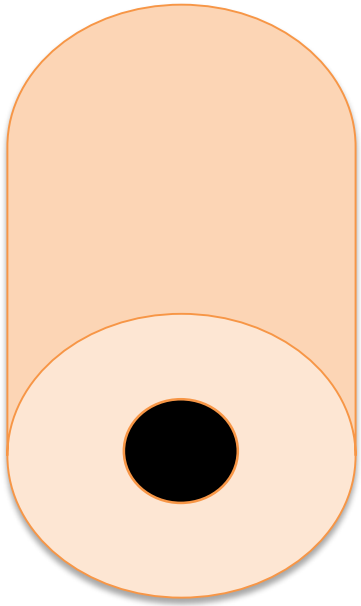
- Excessive unburned mass fraction (5-10%)
- Complex web support
- Compromised grain structural integrity at end of burn
- Poor volumetric loading
- Uneven burning of individual ports
- Requirement for substantial pre-combustion chamber or individual injectors for each port

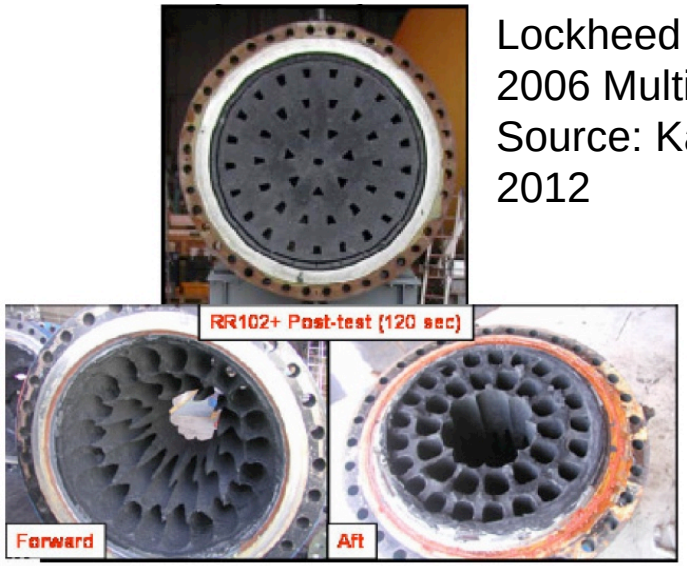


Evolution of Hybrid Rockets









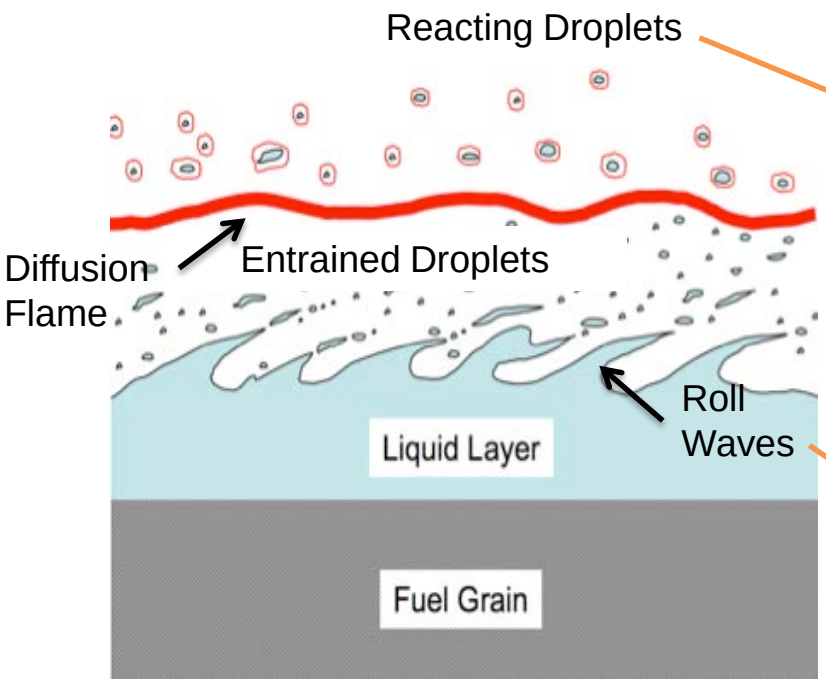
Lockheed Martin
2006 Multiport Test
Source: Karabeyoglu, 2012

Peregrine Motor Test, NASA Ames, Source: Aerospace America 2011.

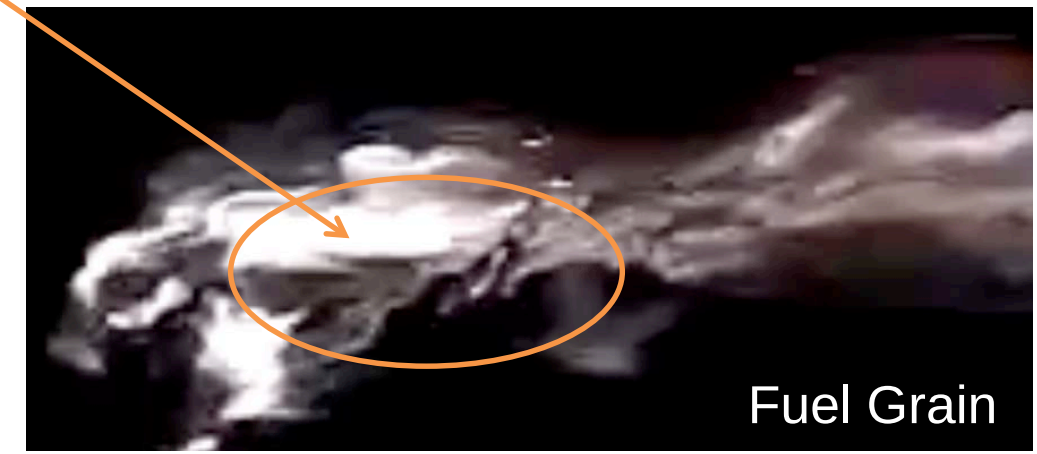
Increasing Regression Rate

- Add oxidizer to the fuel
  - Chemical reactions can now occur on the surface
  - Reduces safety and introduces a pressure dependency
- Add metal particles to the fuel (micron or nano sized)
  - Increases radiative heat transfer
  - Increases cost and difficulty of processing
- Swirl Injection
  - Increases resonance time in combustion chamber and local mass flux
  - Increased complexity, often varies over burn and scaling may be difficult.
- Find a fuel that burns faster

High Regression Rate (Liquefying) Fuels



Cantwell, Karabeyoglu, & Altman, 8th ISICP Cape Town, SA, 2009



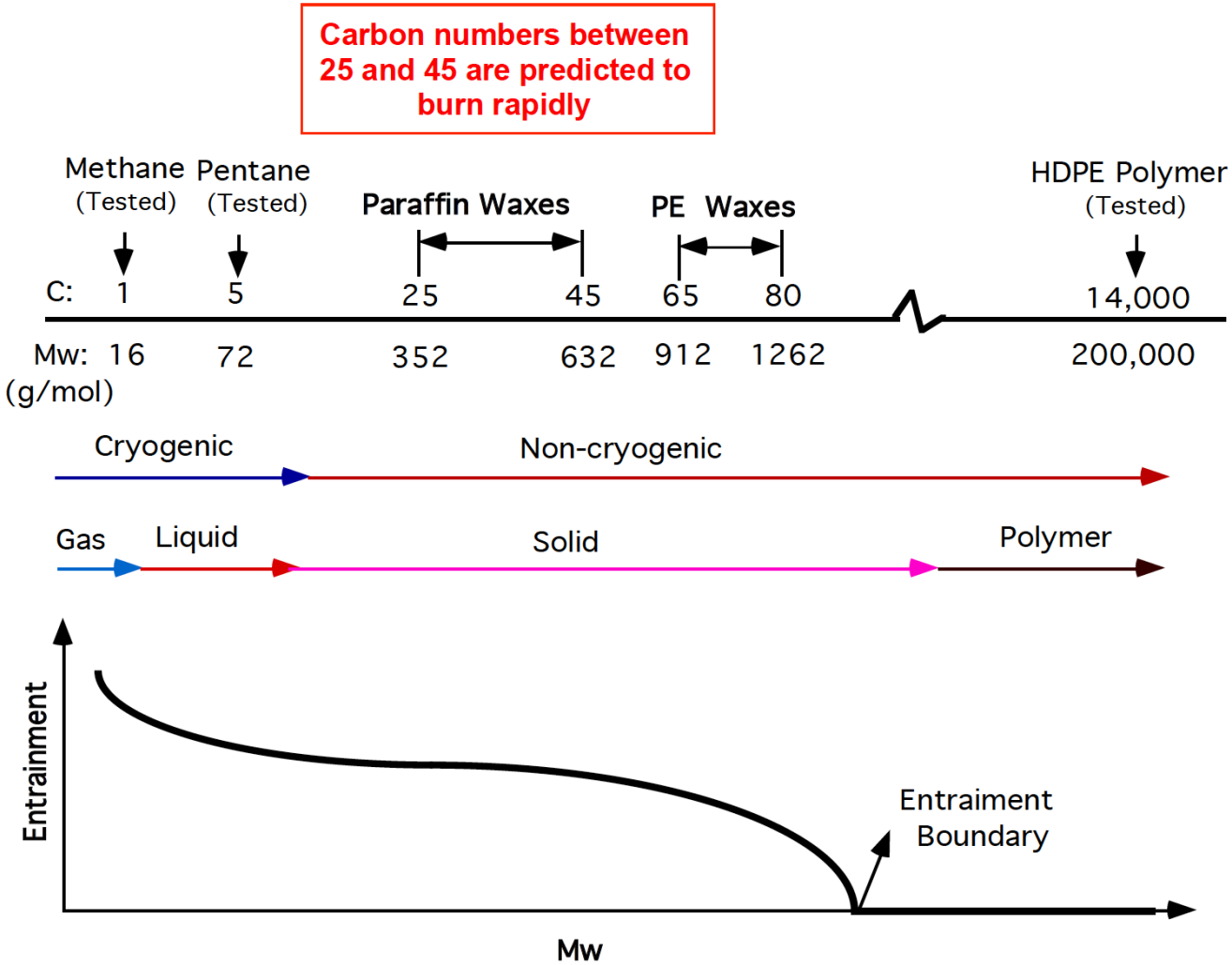
Chandler, Ph.D. Dissertation, 2012

$$\text{Mass transfer due to entrainment} \propto \frac{\text{Dynamic pressure}^{\alpha} \times \text{LL thickness}^{\beta}}{\text{LL surface tension}^{\pi} \times \text{LL viscosity}^{\gamma}}$$

Liquefying Fuel Hybrid Combustion Theory Development

- Marxman's original Diffusion Limited Model was modified to include mass transfer through droplet entrainment
- The following steps are required for this modification
  - Estimate film thickness
  - Stability of the liquid film
  - Scaling for the entrainment mass transfer
- Entrainment results in the following:
  - Reduced heating requirement for the entrained mass
  - Reduced "Blocking Effect" due to two phase flow
  - Increased heat transfer due to the increased surface roughness

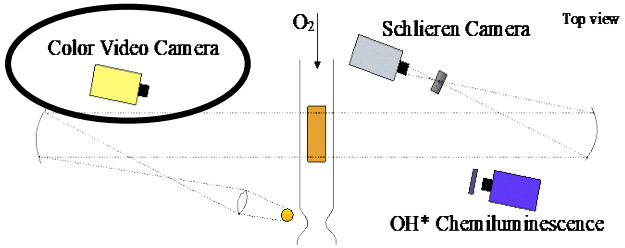
Alkane Waxes C_nH_{2n+2}



From Prof. Brian Cantwell's 283 Course Notes

Comparison of fuels at elevated pressure

Elizabeth Jens, Stanford 2015



P = 460.9 kPa (66.8 psi)

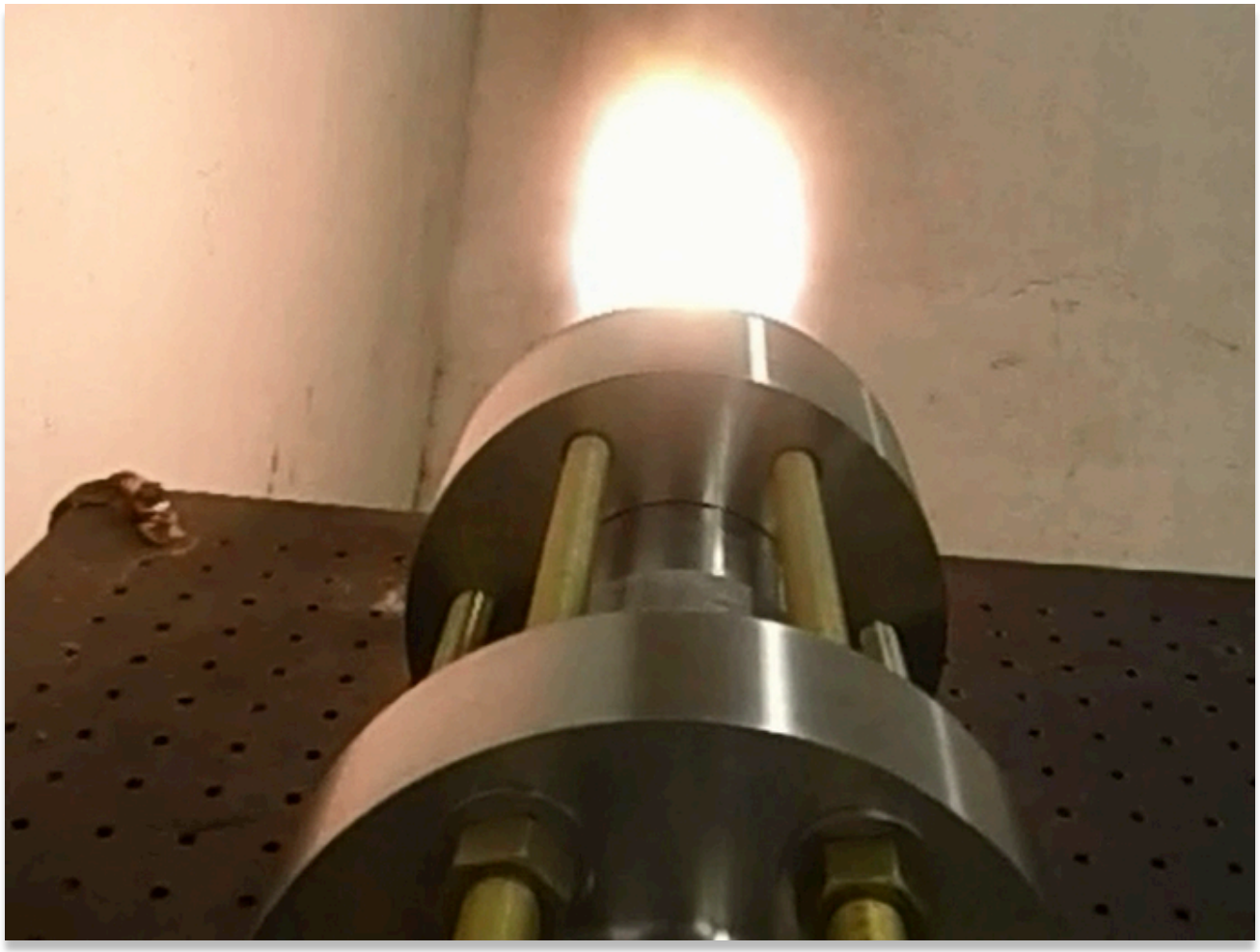
P = 545.9 kPa (79.2 psi)

P = 437.4 kPa (63.4 psi)

P = 492.2 kPa (71.4 psi)

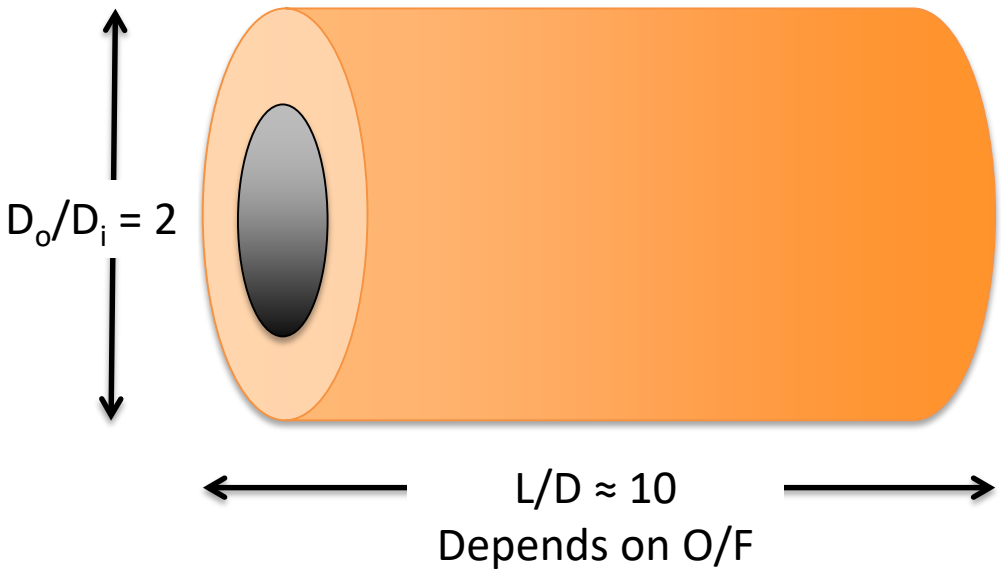
P = 429.9 kPa (62.4 psi)

How to design a Hybrid Rocket



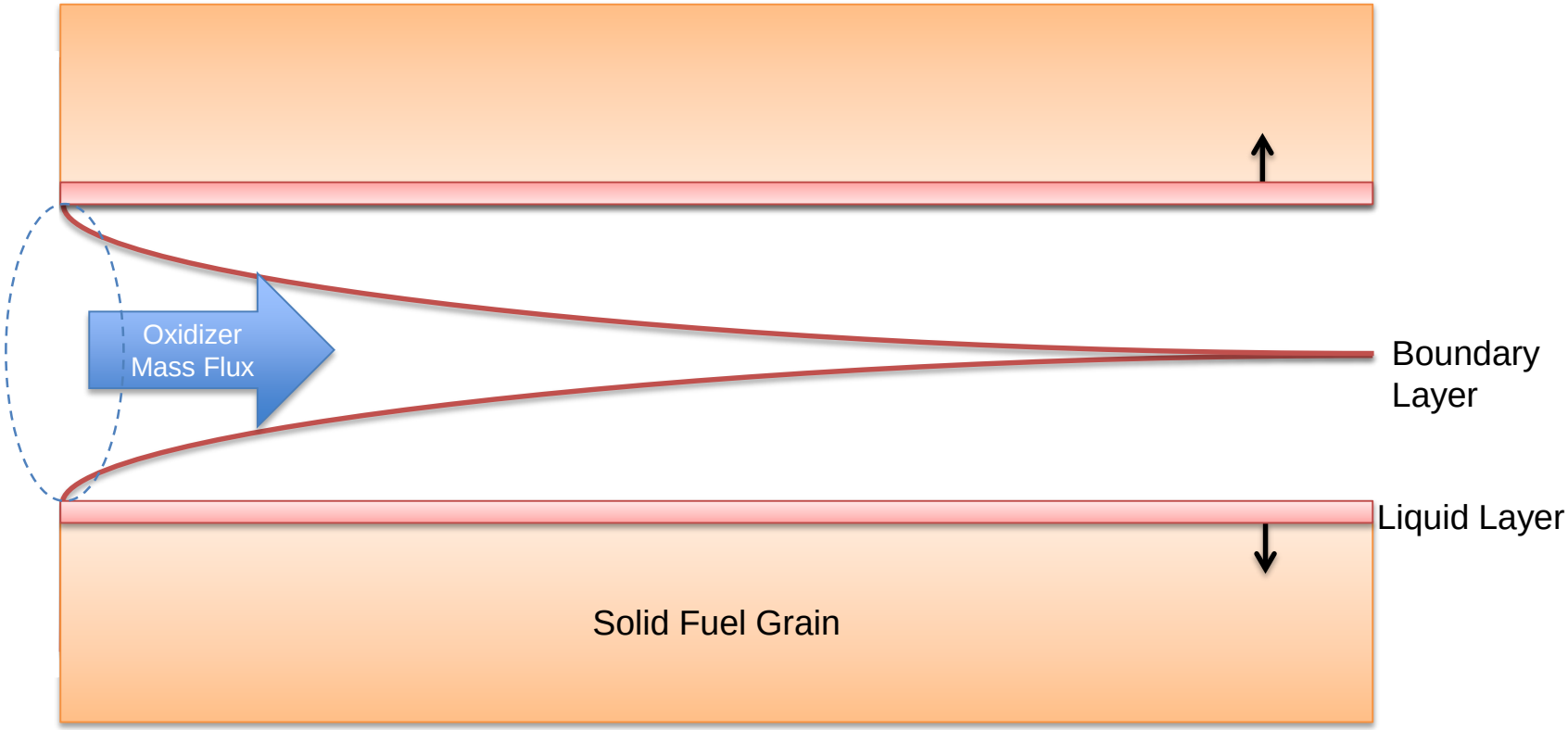
Hybrid Design Process

- Propellant selection ➔ Isp, optimum OF
- Rocket equation ➔ Mass of propellant
- Size of fuel grain can be determined because grain dimensions are more or less fixed



Hybrid Combustion

- Combustion takes place within the boundary layer because the fuel and ox are stored separately
- Hybrids are unique in that the regression rate depends on oxidizer mass flux (mass of oxidizer/cross sectional area of fuel port).
  - Oxidizer mass flux decreases with time
    - There is a range of oxidizer mass fluxes that produce repeatable and stable combustion.
  - O/F shift for all propellant combinations with $n \neq 0.5$ due to the port opening



Hybrid Combustion Continued

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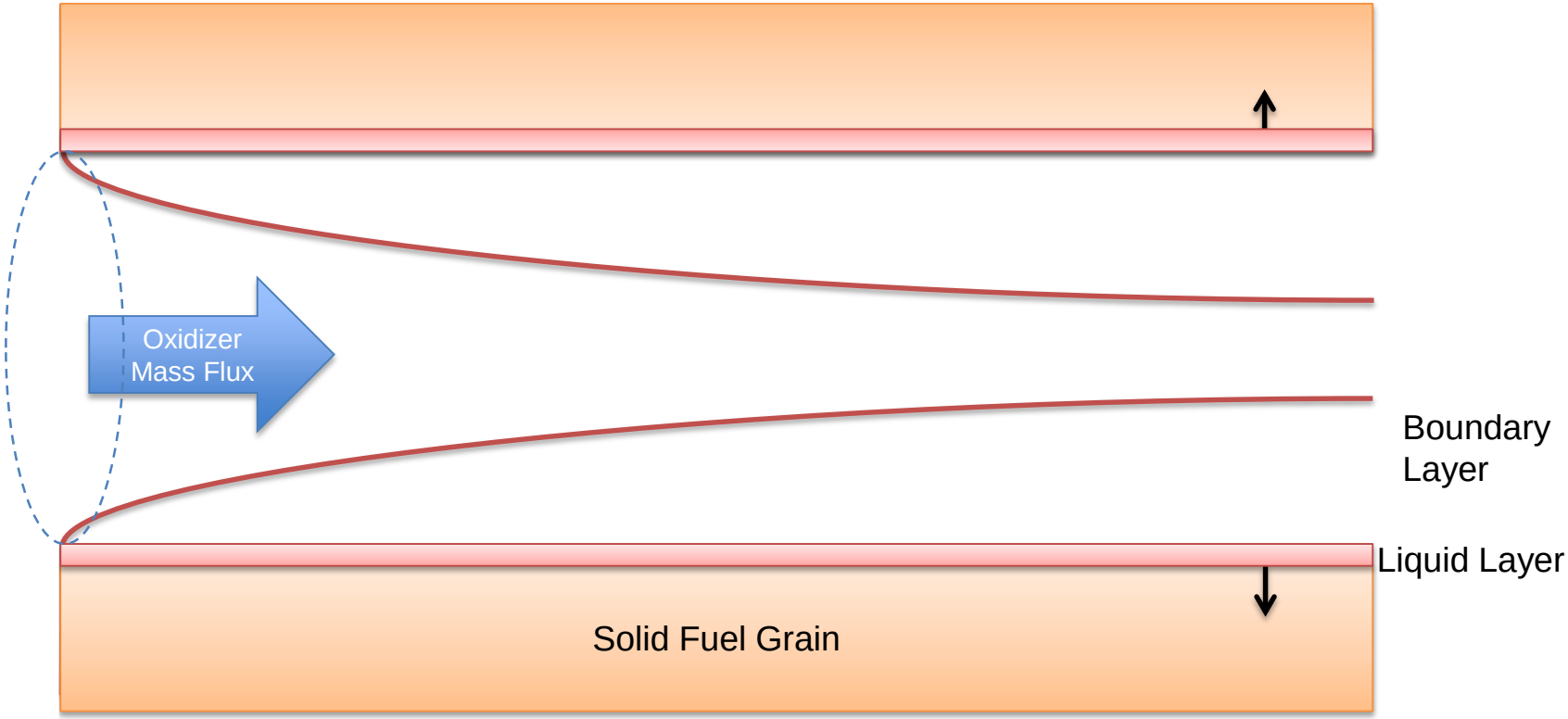
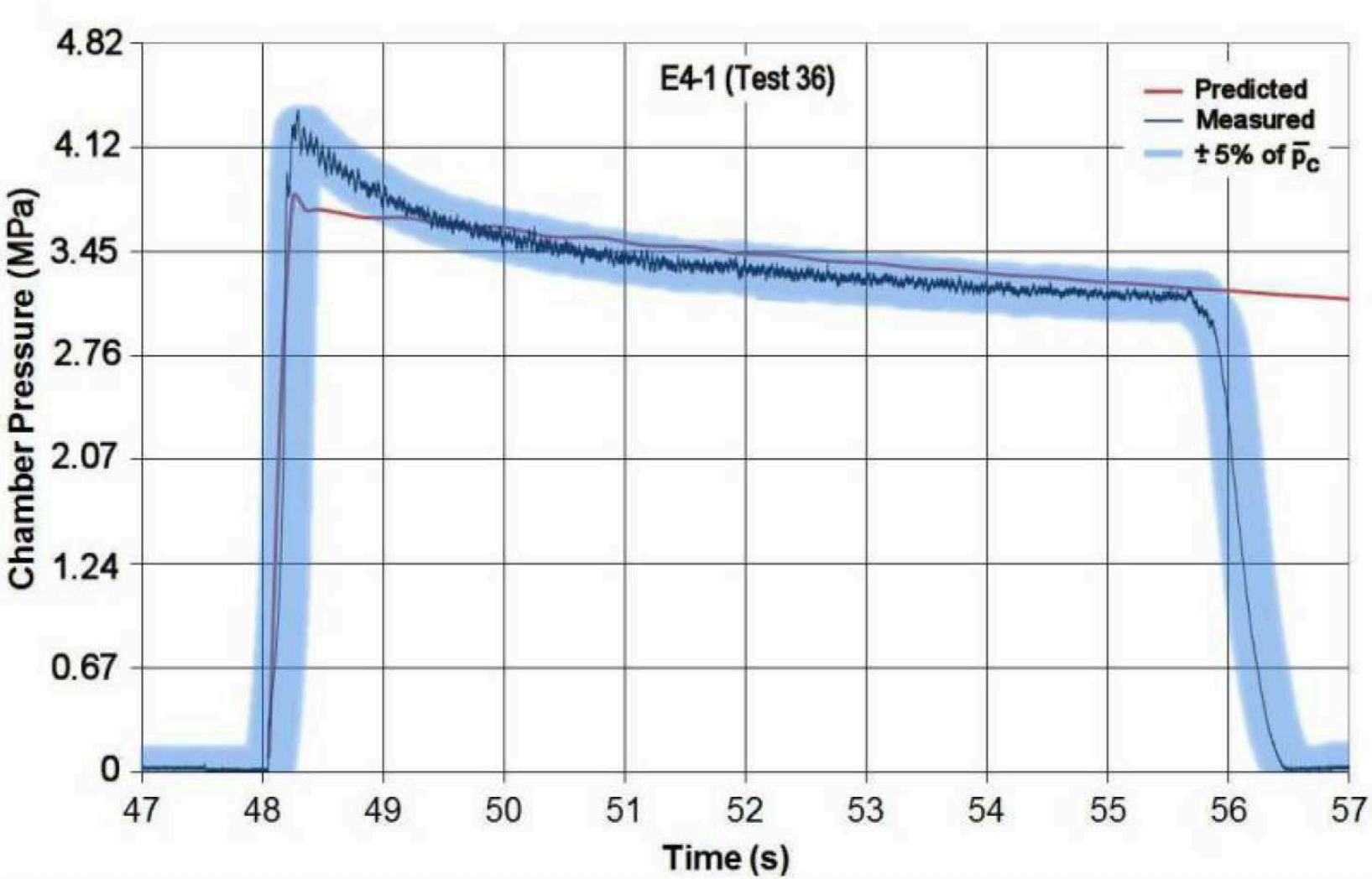


Typical Thrust Profile



Zilliac, et al. “Peregrine Hybrid Rocket Motor Development,” AIAA-2014-3870

Common Hybrid Fuels

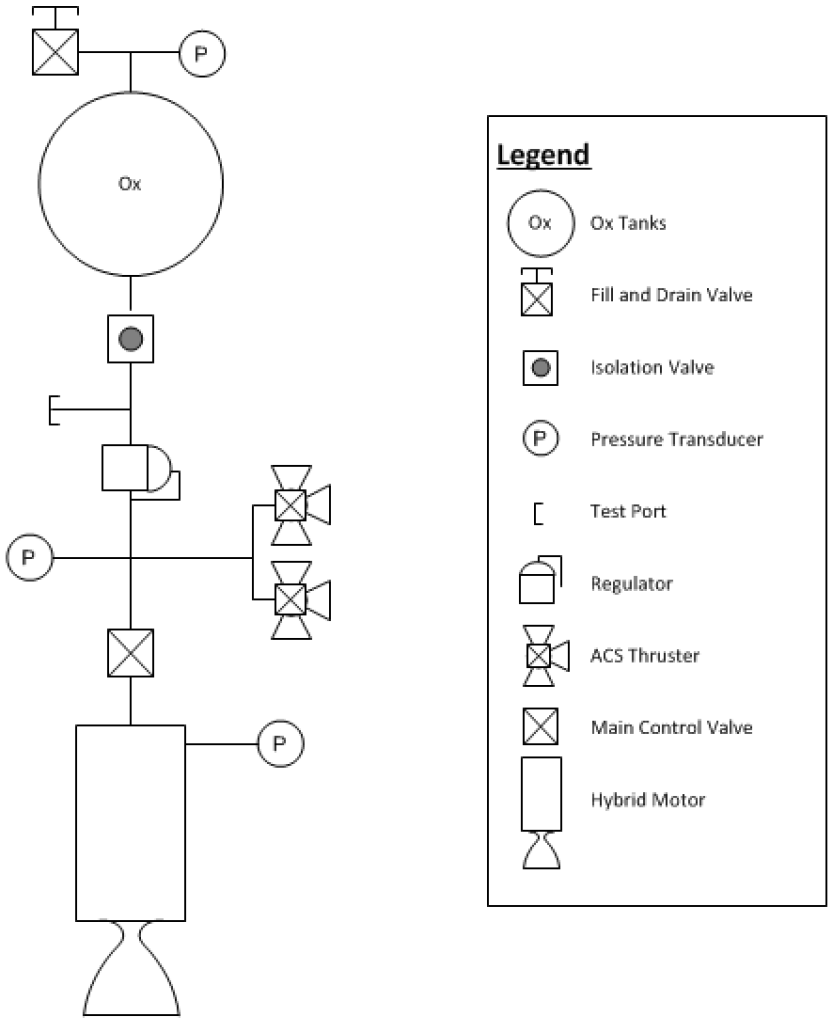
	A.K.A.	Benefits	Issues
Paraffin	Candles	High regression rate fuel, high performance, good low temperature performance	Melting point is low, requires strength additives
HTPB	Rubber	Highest heritage	Often needs multiple ports
HDPE	Food containers Milk Jugs	High performance	Hard to form
PE Wax	Polyethylene	Moderately high regression rate	
ABS	Plastic, Legos, pipes	Can 3D print fuel grains, arc-ignition mechanism developed	Moderate performance
PMMA	Acrylic, contact lenses	High strength, clear grain (can see into combustion)	Moderate performance

Oxidizers

- Same list as for liquid bi-propellants

	Benefits	Issues
Oxygen (O_2)	High performance, possible to get from ISRU	LOx is cryogenic and GOx is low density
Nitrous Oxide (N_2O)	Self Pressurizing	Low performance
Nitrogen Tetroxide (N_2O_4) aka MON-3, MON-25	Good performance, space storable, thought to be hypergolic, can add NO to reduce boiling point	Hazardous
Hydrogen Peroxide (HP)	Somewhat storable, low performance	Decomposes over time
Nitric Acid (RFNA and IRFNA)	Can be hypergolic	Lower boiling point than NTO
Fluorine (F_2)	High performance	Cryogenic, extremely toxic, reacts with almost anything

Typical Gas Hybrid System



What would a liquid system look like?

Paraffin Hybrid Test

PMMA Hybrid Test

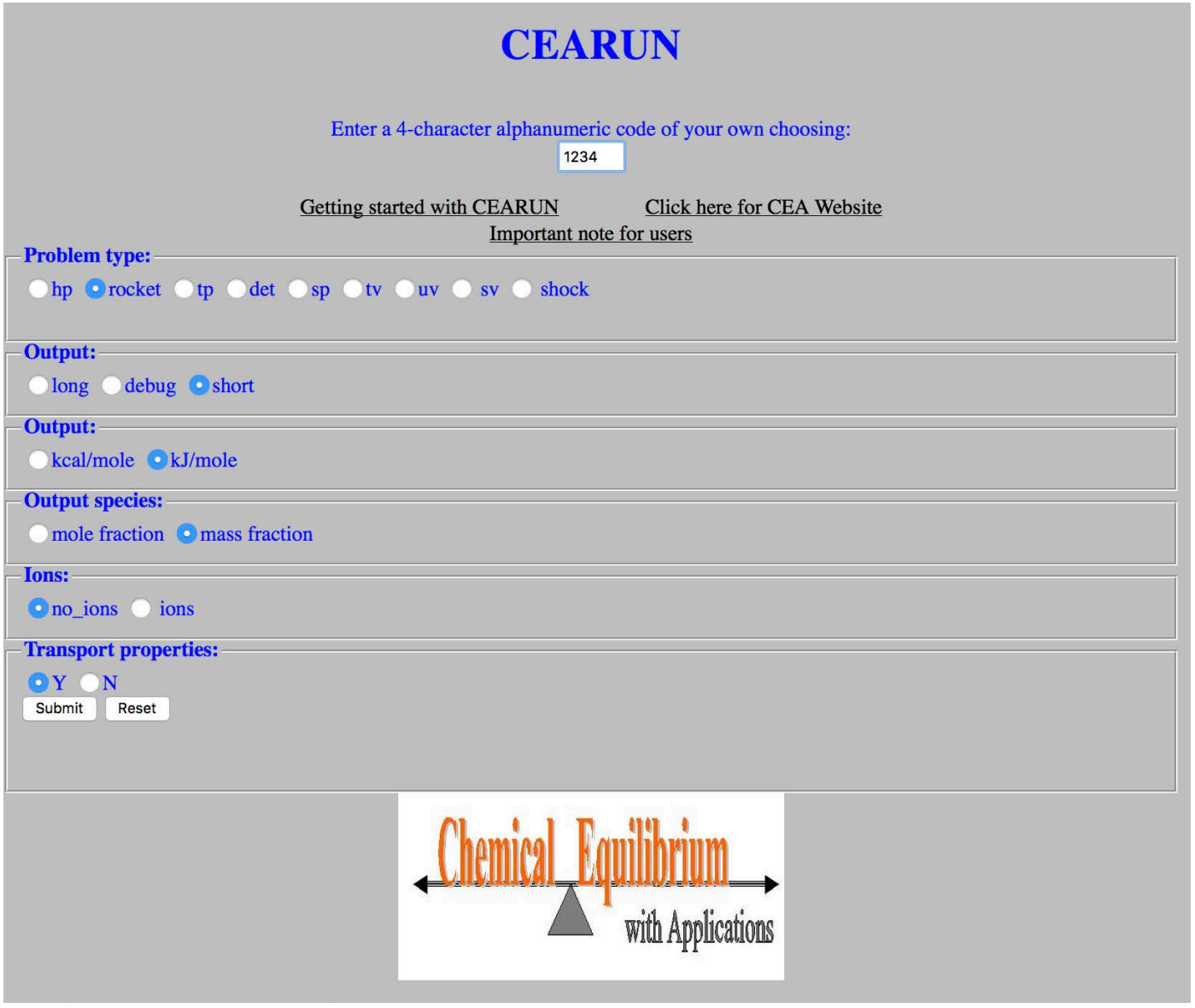
Chemical Equilibrium with Applications (CEA)

- CEA was developed by Bonnie McBride at NASA Glenn
  - http://www.grc.nasa.gov/WWW/CEAWeb/ceaHome.htm
- This code will determine the chemical reactions that occur in the combustion chamber
  - Outputs important design variables such as: Isp, c*, temperatures, velocities, gamma, etc.
  - Can be used to find the optimal O/F ratio

CEA

- Equilibrium vs Frozen Calculations
  - Frozen: no reactions in the nozzle, freezes values to the throat conditions
  - Equilibrium: continuous adjustment during the nozzle expansion

CEA, Step 1


CEARUN
Enter a 4-character alphanumeric code of your own choosing:
1234
Getting started with CEARUN
Click here for CEA Website
Important note for users
Problem type:
hp rocket tp det sp tv uv sv shock
Output:
long debug short
Output:
kcal/mole kJ/mole
Output species:
mole fraction mass fraction
Ions:
no_ions ions
Transport properties:
Y N
Submit
Reset
Chemical Equilibrium
with Applications

CEA, Step 2

Rocket input parameters

c*, Isp, Ivac: ◉ effective velocity ○ sec
◉ infinite area combustor ○ finite area combustor
◉ equilibrium ○ frozen ○ equilibrium and frozen both

You have two options for specifying chamber pressures:

Option(1): specify a low value, a high value, and an interval(16 maximum):

250 | 500 | 50

○ MPa ○ KPa ○ atm ○ bar ◉ psia

Submit Reset

Option (2): Fill boxes with up to 16 values:

250

○ MPa ○ KPa ○ atm ○ bar ◉ psia

Submit Reset

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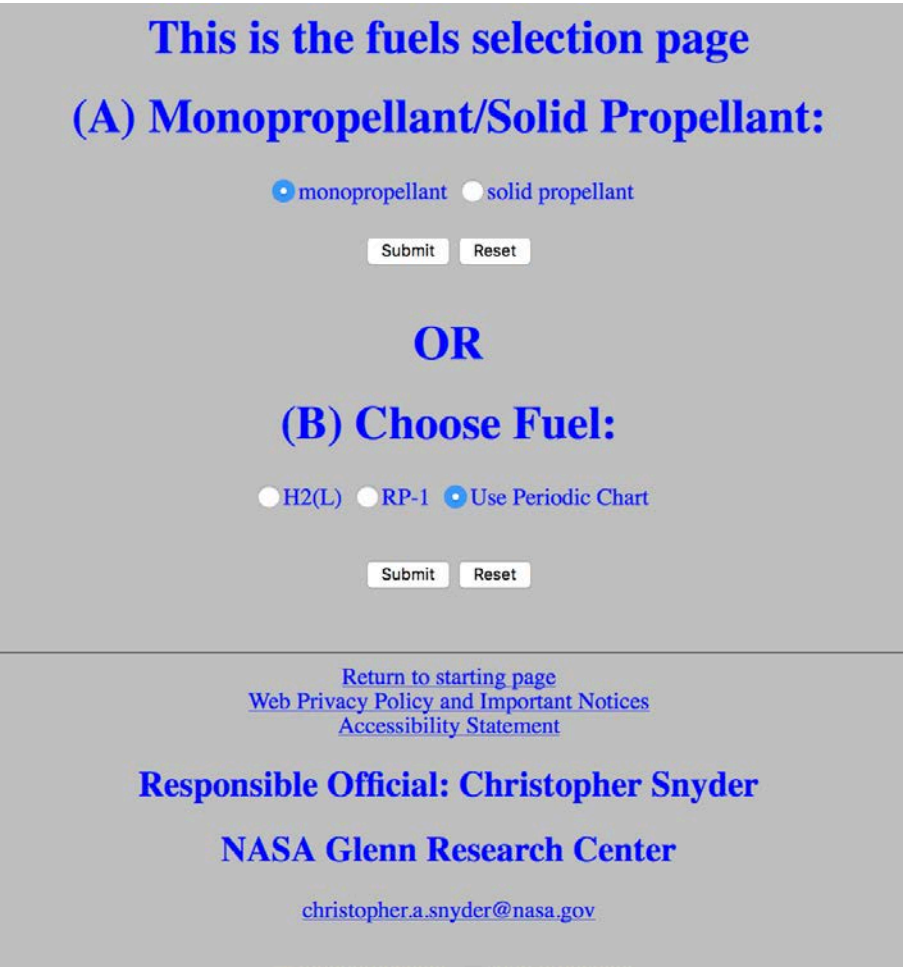
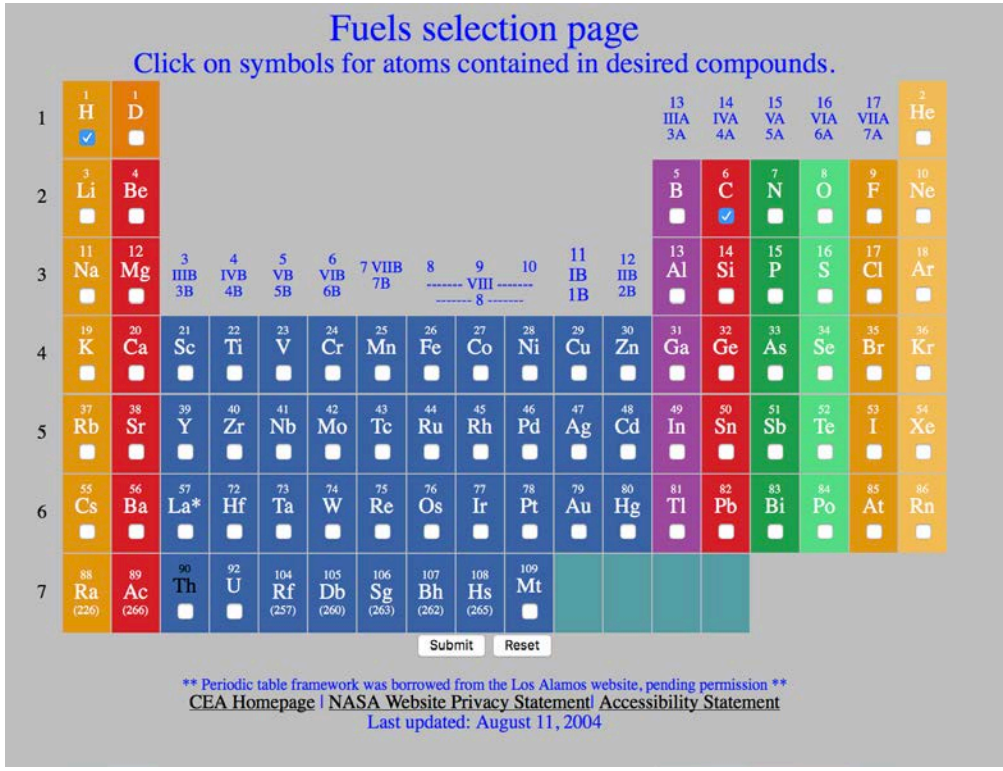
CEA, Step 3





- For Hydrocarbons, select H & C, all combinations in the database will come up
- You'll have to select fuel first, then do the same thing for the oxidizer. Can do multiple of each

CEA, Step 4

Choose your relative O/F values

You have two options for specifying relative values:

Option(1): specify a low value, a high value, and an interval(16 maximum):

1.5 | 2.5 | 0.1

O/F (mass) phi

Submit Reset

Option (2): Fill boxes with up to 16 values

(all empty= 1):

O/F (mass) phi wt% fuel

Submit Reset

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CEA, Step 5

Please fill in desired values for the following fields:

Pc/Pe, subsonic area ratios, or supersonic area ratios:

(Blank boxes are OK)

Digits and decimal points only!!

Subsonic Area Ratio:

Supersonic Area Ratio:

40

Pc/Pe:

Submit Reset

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Accessibility Statement
Responsible Official: Christopher Snyder NASA Glenn Research Center
christopher.a.snyder@nasa.gov

CEA, Output

```
        NASA-GLENN CHEMICAL EQUILIBRIUM PROGRAM CEA2, FEBRUARY 5, 2004
                  BY  BONNIE MCBRIDE AND SANFORD GORDON
     REFS: NASA RP-1311, PART I, 1994 AND NASA RP-1311, PART II, 1996

******************************************************************************


prob case=12342087 ro equilibrium


 ! iac problem
o/f 2.2
p,psia  250
supar 40
reac
 fuel  paraffin wt%=100 t,k=298.15
 oxid O2 wt%=100 t,k=298.15
output trans
output    short
output trace=1e-5
end

             THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM

          COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR

Pin =   250.0 PSIA
CASE = 12342087

            REACTANT                    WT FRACTION      ENERGY      TEMP
                                         (SEE NOTE)     KJ/KG-MOL      K
FUEL        paraffin                     1.0000000  -1860600.000    298.150
OXIDANT     O2                           1.0000000         0.000    298.150

O/F=    2.20000  %FUEL= 31.250000  R,EQ.RATIO= 1.510004  PHI,EQ.RATIO= 1.510004

                CHAMBER   THROAT     EXIT
Pinf/P            1.0000   1.7339   456.78
P, BAR            17.237   9.9411  0.03774
T, K             3438.17  3259.20  1454.09
RHO, KG/CU M    1.3442 0 8.2906-1 7.4114-3
H, KJ/KG         -580.41 -1262.97 -5918.56
U, KJ/KG        -1862.73 -2462.05 -6427.71
G, KJ/KG        -41464.6 -40018.9 -23209.5
S, KJ/(KG)(K)    11.8912  11.8912  11.8912

M, (1/n)          22.293   22.600   23.745
(dLV/dLP)t      -1.03461 -1.02768 -1.00001
(dLV/dLT)p        1.6236   1.5267   1.0002
Cp, KJ/(KG)(K)    6.2570   5.7432   1.9247
GAMMAs            1.1396   1.1385   1.2225
SON VEL,M/SEC     1208.9   1168.4    788.9
MACH NUMBER        0.000    1.000    4.142

TRANSPORT PROPERTIES (GASES ONLY)
  CONDUCTIVITY IN UNITS OF MILLIWATTS/(CM)(K)

VISC,MILLIPOISE   1.0472   1.0090  0.57111

 WITH EQUILIBRIUM REACTIONS

Cp, KJ/(KG)(K)    6.2570   5.7432   1.9247
CONDUCTIVITY     15.1431  13.4312   1.7418
PRANDTL NUMBER    0.4327   0.4314   0.6311

 WITH FROZEN REACTIONS

Cp, KJ/(KG)(K)    2.0202   2.0107   1.7893
CONDUCTIVITY      3.4242   3.2534   1.5985
PRANDTL NUMBER    0.6178   0.6236   0.6393

PERFORMANCE PARAMETERS

Ae/At                      1.0000   40.000
CSTAR, M/SEC               1779.4   1779.4
CF                         0.6566   1.8362
Ivac, M/SEC                2194.7   3423.3
Isp, M/SEC                 1168.4   3267.5

MOLE FRACTIONS

*CO             3.9716-1 3.9508-1 3.3477-1
*CO2            1.1048-1 1.1955-1 2.0597-1
*H              4.2438-2 3.5921-2 2.2127-5
HCO             1.493 -5 8.412 -6 5.338-10
HO2             2.483 -5 1.341 -5 7.178-15
*H2             1.1805-1 1.1827-1 1.8560-1
H2O             2.6883-1 2.8244-1 2.7364-1
*O              9.006 -3 6.160 -3 8.350-11
*OH             4.6073-2 3.6823-2 9.4044-7
*O2             7.911 -3 5.729 -3 5.794-11

 * THERMODYNAMIC PROPERTIES FITTED TO 20000.K

NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF
OXIDANT IN TOTAL OXIDANTS
```

NIST Webbook

- http://webbook.nist.gov/chemistry

Search for Species Data by Chemical Name

Please follow the steps below to conduct your search (Help):

1. Enter a chemical species name or pattern: (e.g., methane, *2-hexene)
2. Select the desired units for thermodynamic data:
   - (•) SI ( ) calorie-based
3. Select the desired type(s) of data:

Thermodynamic Data	Other Data
[ ] Gas phase	[ ] IR spectrum
[ ] Condensed phase	[ ] THz IR spectrum
[ ] Phase change	[ ] Mass spectrum
[ ] Reaction	[ ] UV/Vis spectrum
[ ] Ion energetics	[ ] Gas Chromatography
[ ] Ion cluster	[ ] Vibrational & electronic energy levels
	[ ] Constants of diatomic molecules
	[ ] Henry's Law

4. Press here to search: Search

Oxidizer to Fuel (O/F) Ratio

- Ratio of the mass flow rate of oxidizer to mass flow rate of fuel.
- Typically the mass flow rate of oxidizer is kept constant unless throttling
- O/F Shift: The oxidizer to fuel ratio can shift due to the port opening up during a burn or throttling.
  - There is no O/F shift for n=0.5 due to port opening.

$$\frac{O}{F} = \frac{\dot{m}_{ox}}{\dot{m}_f} = \frac{\pi^n D_p^{2n} \dot{m}_{ox}}{\rho_f \pi \; D_p L \; a \; 4^n \dot{m}_{ox}^n} = \frac{D_p^{2n-1} \dot{m}_{ox}^{1-n}}{\rho_f \pi^{1-n} a \; 4^n \rho_f L}$$

$$G_{ox}(t) = \frac{4\dot{m}_{ox}}{\pi \; D_p^2} \qquad \dot{r}(t) = aG_{ox}^{n} \qquad \dot{m}_f(t) = \rho_f \pi \; D_p L \; \dot{r}$$

Comparison to State of the Art

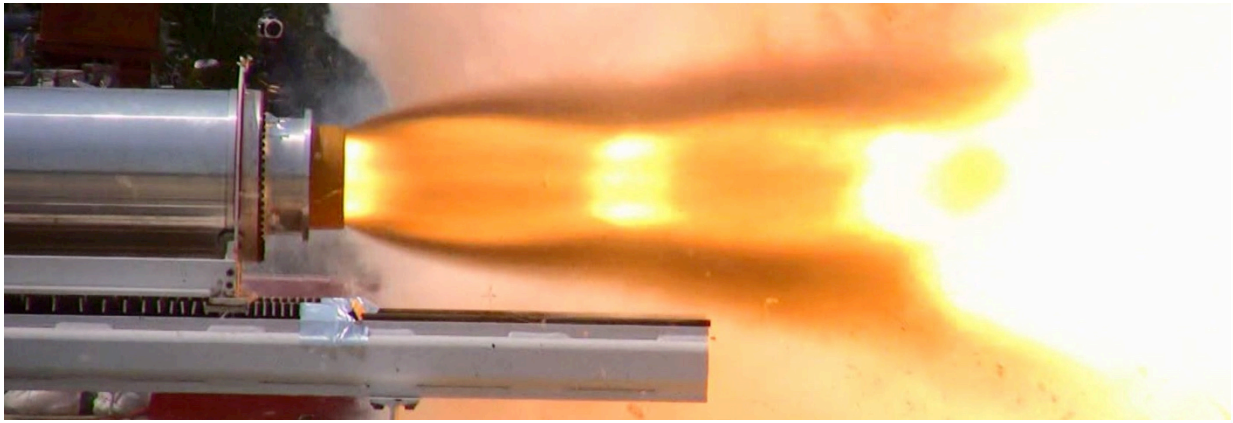
		Hybrid (Paraffin/GO_2)	State of the Art		
			Solid	Liquid (MMH/MON)	Monoprop
Isp (Performance)		330 s	285 s	324 s	240 s
Restart capability		Yes, multiple	N/A	Yes, multiple	Yes, multiple
Throttling		Simple, 10:1	None	Complex, 3:1	Simple, 10:1
Low temperature storage and operations		< -90 C (Predicted)	- 40 C	- 44 C	+ 13 C
L/D Ratio		High (Predicted)	Low	Moderate	Moderate
Safety	Toxicity	Nontoxic	Toxic	Toxic	Toxic
	System Complexity	Moderate	Low	High	Moderate
	Recurring Cost	Low	Low	High	Moderate
Simple ISRU?		Yes	No	No	No

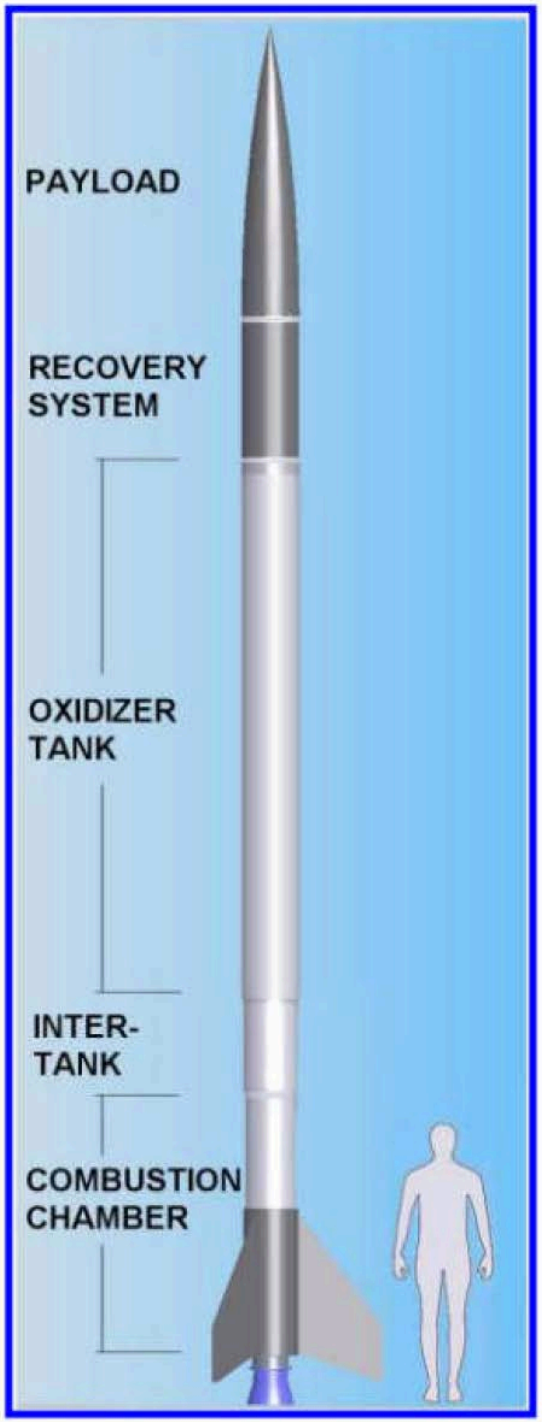
SpaceShipTwo

- Intended for Space Tourism
  - Air launched via WhiteKnightTwo
  - Will carry 6 passengers and 2 pilots to 110 km
- Propellant combination has changed twice from HTPB/N_2O to Nylon/N_2O then back to HTPB/N_2O
  - Reported stability problems

Example: Peregrine Sounding Rocket

- Objective: design, build, test, and fly a stable, efficient liquefying fuel hybrid rocket.
- Specs:
  - Paraffin-based fuel with Nitrous Oxide, Earth storable
  - Size: diameter of 15 in. (38.1 cm)
  - Thrust: 14,000-lbf (62.3-kN)
- Check out one of the hotfire tests at NASA Ames.

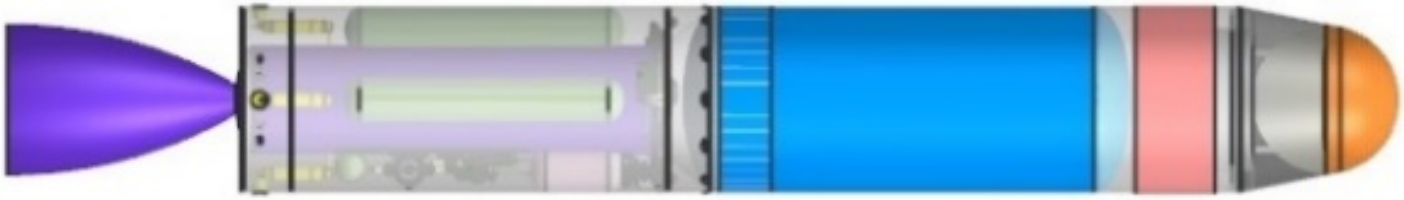




Hybrid Mars Ascent Vehicle

- Benefits
  - Low temperature tolerance
  - Lowest mass option
  - Can be throttled
  - Restartable –Single Stage to Orbit possible
- Challenges
  - Packaging – flexible, but less compact than solids

Artist's Concept of a MAV, Source: NASA



ISRU (for MARS) Hybrid Testing

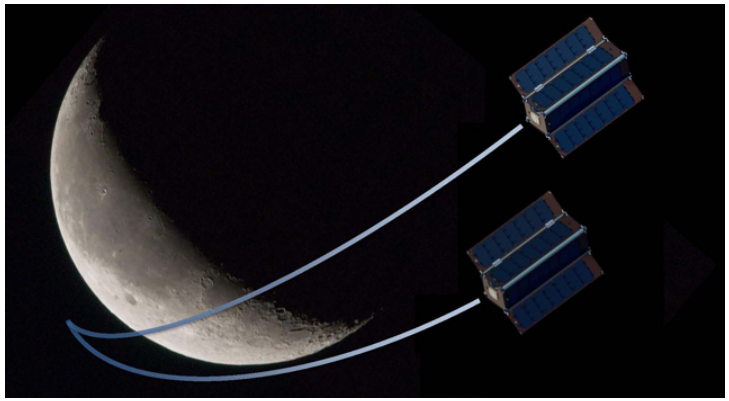
- Filling oxidizer tank at Mars or on the way there
  - Incremental step towards ISRU – convert CO_2 into O_2
  - Byproduct of ECLESS operations
- Hybrid test facility developed at JPL

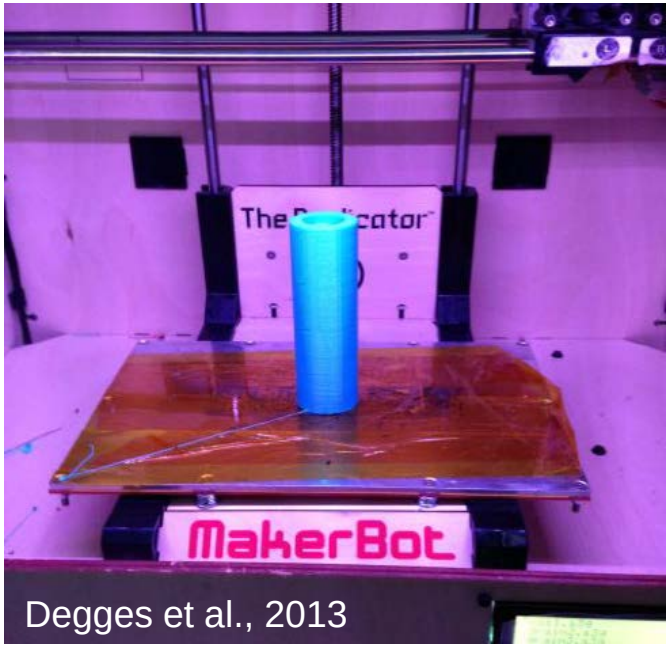
Test of a paraffin/gaseous oxygen motor at the JPL Test Facility

Multiple sizes of combustion chamber tubes

CubeSat/SmallSat Propulsion

- Benefits
  - Capable of high thrust and delta V's required for orbit insertion <u>OR</u> low level thrust for station keeping
  - Ideal for ride sharing: "green," storable propellant options.
  - Many different fuels/oxidizers to choose from
  - Multiple starts/stops possible
- Challenges
  - Igniter development
  - Packaging





Degges et al., 2013

Questions?

Regression Rate Data

Results summary of the average regression rate with oxygen for various fuels									
No.	Fuel	a †	n	No. of Tests	Chamber Pressure Range (MPa)	Average O/F Ratio Range	Data Reduction Technique	Oxidizer Mass Flux Range (g/cm^2-sec)	Ref.
1	Paraffin, SP1A	0.488	0.62	65	1.1-6.9	1.0-4.0	DA	1.6-36.9	15
2	HTPB, (Thiokol)	0.146	0.681	16	-	-	-	3.8-30.2	16
3	HTPB+19.7%AL	0.117	0.956	2	1.2	-	OA	5.1-23.0	17
4	HTPB	0.304	0.527	3	2.0	-	OA	6.2-31.0	17
5	HTPB+20%GAT	0.473	0.439	5	-	-	-	-	18
6	PMMA	0.087	0.615	8	0.3-2.6	-	-	3.3-26.6	19
7	HDPE	0.132	0.498	4	0.7-1.3	3.8-5.9	DA	7.7-26.1	20
8	PE Wax, Marcus 200	0.188	0.781	4	0.5-1.2	2.2-3.2	DA	4.8-15.8	20
9	PE Wax, Polyflo 200	0.134	0.703	3	0.6-1.2	1.6-1.7	DA	4.4-16.3	20
10	HTPB	0.194	0.670	6	-	-	OA	17.5-32.0	21
11	HTPB+13% nano Al	0.145	0.775	12	-	-	OA	16.5-34.2	21
12	Paraffin, FR5560 + 13% nano Al	0.602	0.730	8	-	-	OA	14.5-29.0	21
13	Paraffin, FR5560	0.672	0.600	4	-	-	OA	6.3-12.3	21
14	Paraffin, FR4550	0.427	0.748	3	0.7-?	1.3-1.8	DA	4.3-11.9	20

Regression rate equation: $\bar{\dot{r}} = a\overline{G}_o{}^n x^m$ with $m = 0$

† For use with G_o with units of gm/cm^2-sec, produces an average regression rate in mm/sec.

DA: Diameter Averaged, FA: Flux Averaged, AA: Area Average, OA: Other averaging technique applied

Zilliac and Karabeyoglu, “Hybrid Rocket Fuel Regression Rate Data and Modeling.” AIAA 2006-4504