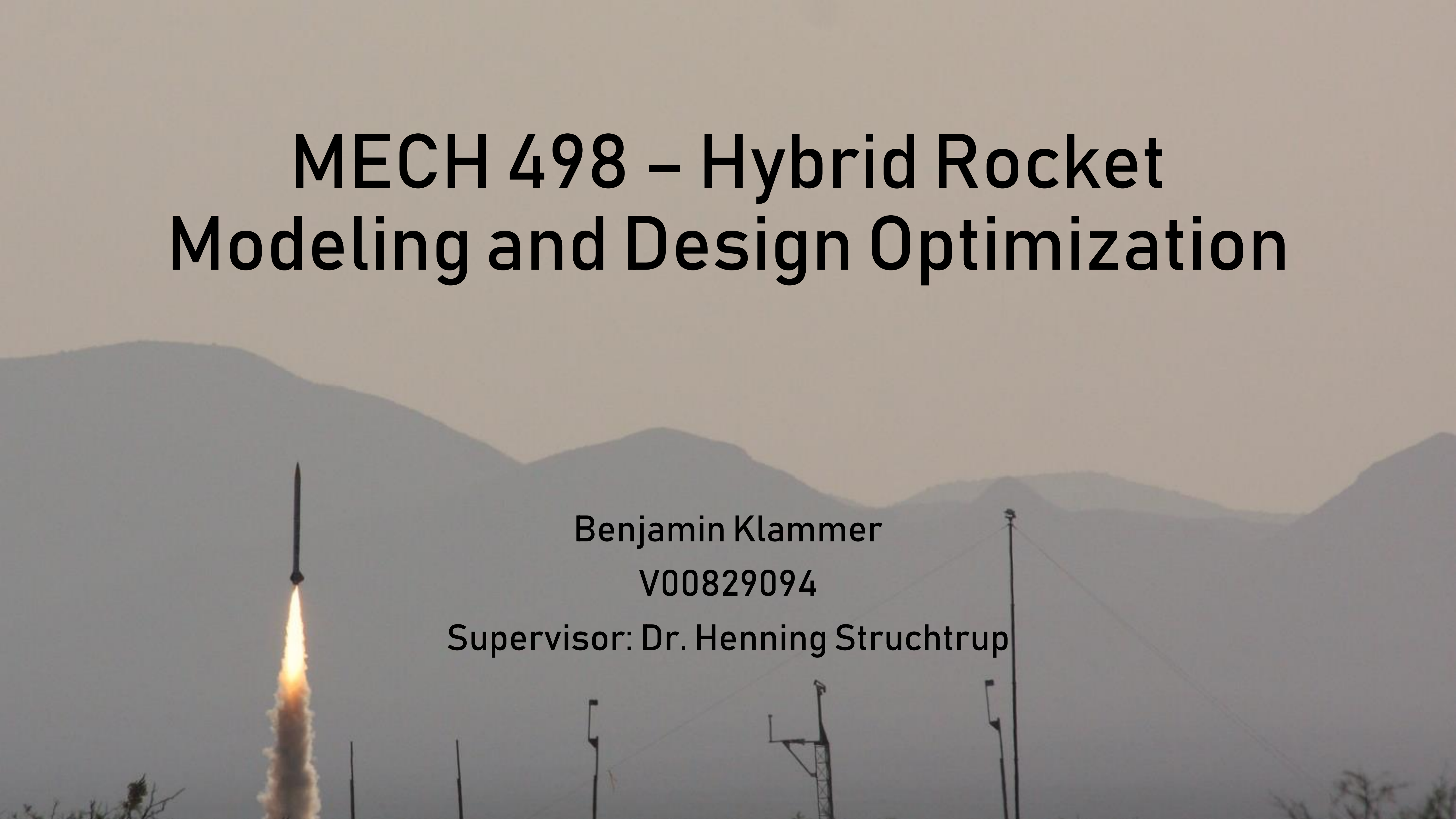
MECH 498 – Hybrid Rocket Modeling and Design Optimization



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Introduction

- Developed numerical model of hybrid rocket implemented in MATLAB
- Performed design optimization using built-in MATLAB functions
- Conducted global sensitivity analysis

UVIC ROCKETRY

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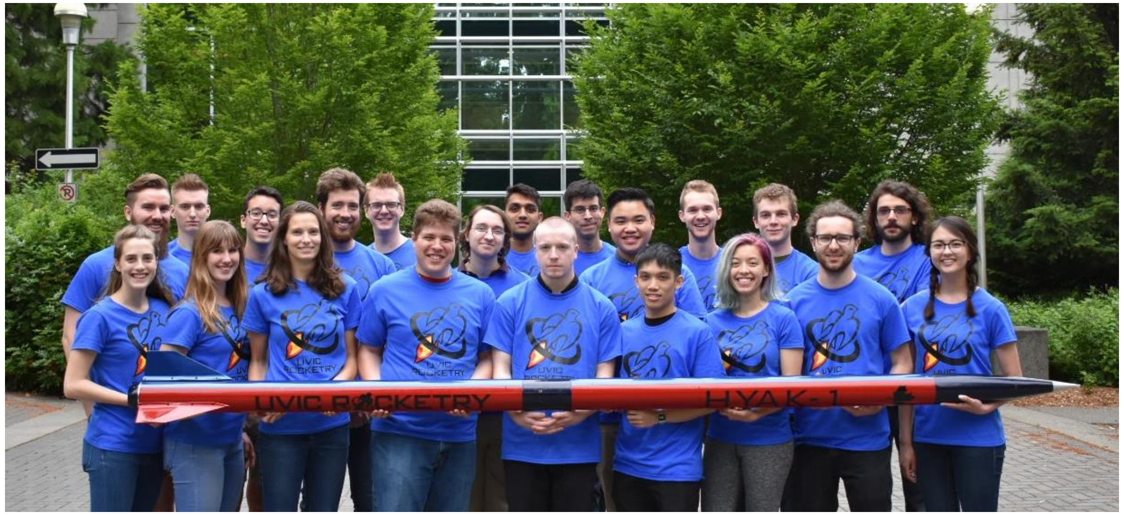
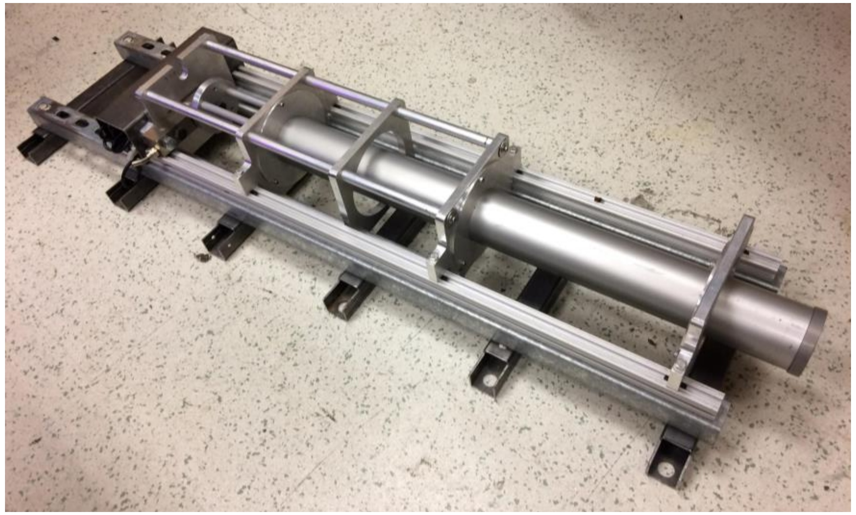
UVic Rocketry (UVR)

- Student-led extracurricular engineering team
- Developing paraffin-N_2O hybrid rocket
  - Currently obtaining approval for testing of half-scale motor
- Launching hybrid rocket (Ramses-1) at Spaceport America Cup in June 2020
  - Competition goal is to reach as close to 10,000ft AGL as possible





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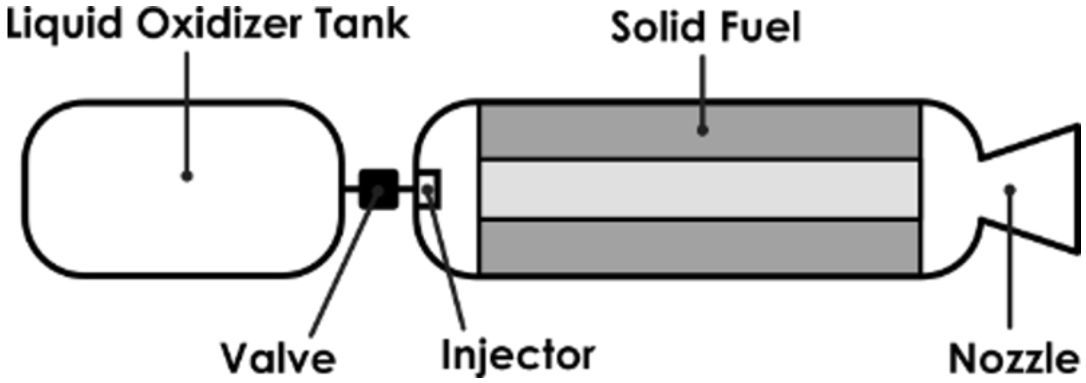
Hybrid Rockets

- Liquid oxidizer, solid fuel
- Simpler than liquid, higher performance than solid
- Safe, environmentally friendly
- Historically underdeveloped



Problem Definition

Need:

- UVR requires a conceptual design of a flight motor in order to begin detailed design

Goal:

- Develop a numerical model of a hybrid rocket and perform design optimization and sensitivity analysis to select an optimal design

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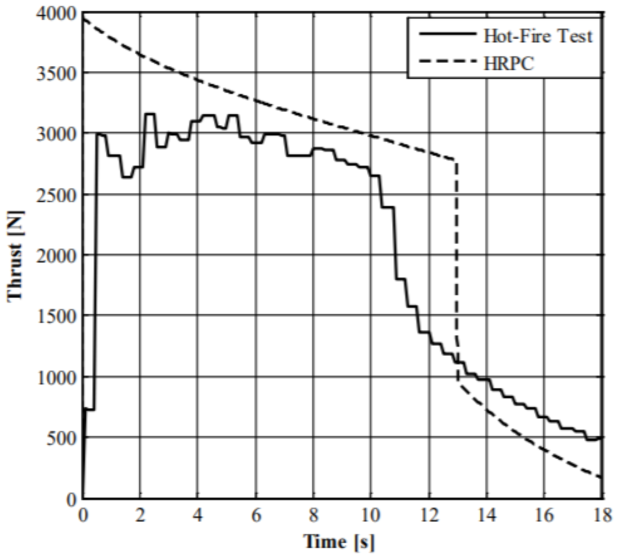
Literature Review

- Stanford/NASA Ames
  - Discovered liquefying fuels, averaged regression rates [2] [3]
  - Nitrous oxide injection and pressurization [4] [5]
- Several other modeling efforts, CFD [6][7][8][37]
- Genetic algorithm commonly used in conceptual design [10][11]
- Uncertainty-based design optimization of hybrid rocket [12]



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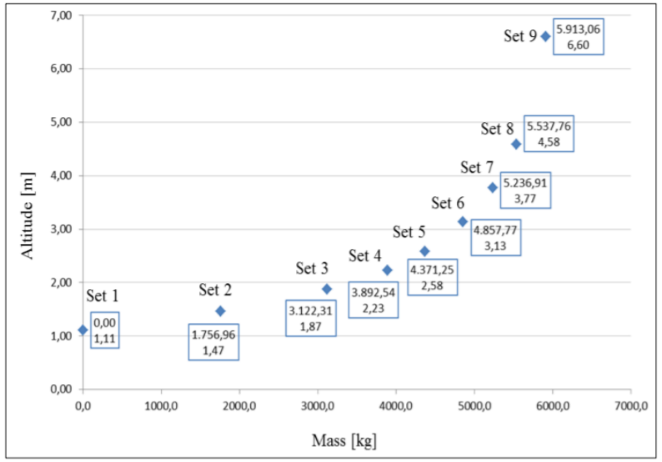

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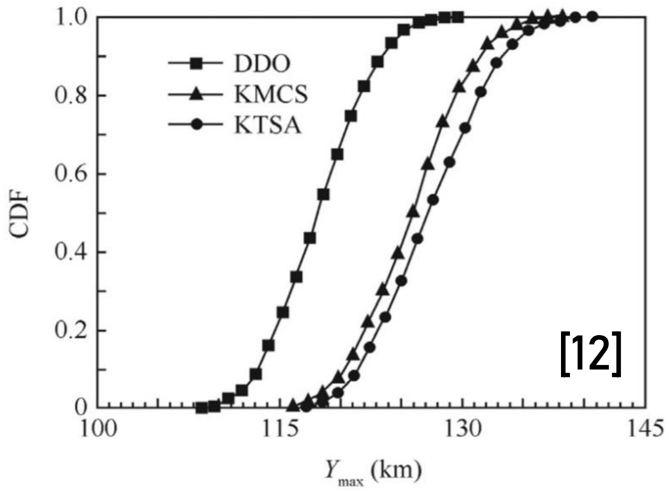

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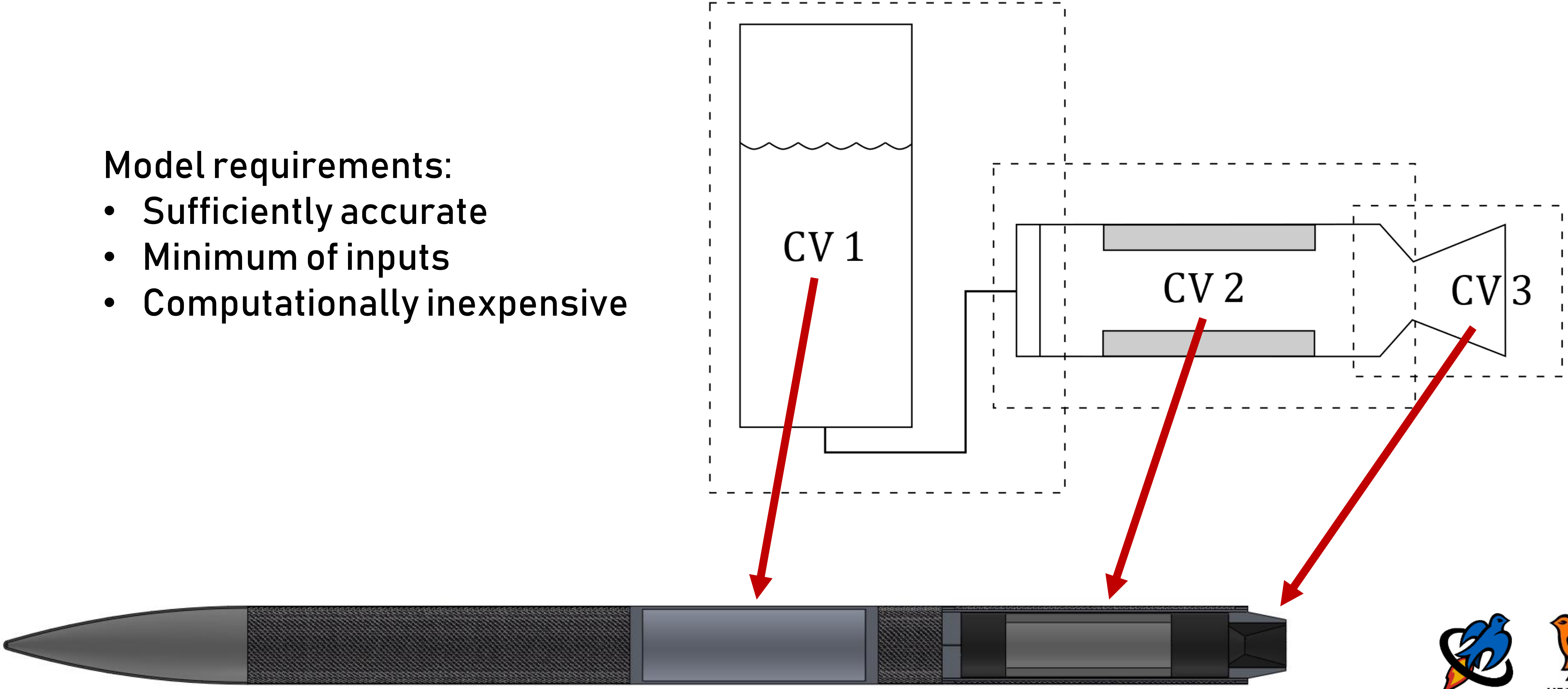
Modeling – Overview

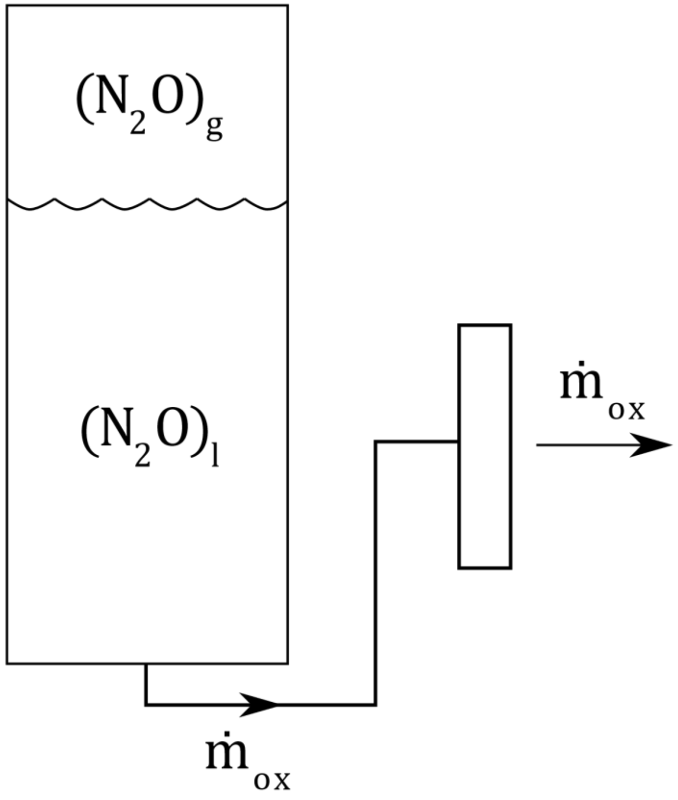
Model requirements:

- Sufficiently accurate
- Minimum of inputs
- Computationally inexpensive



Modeling – Oxidizer Tank

- Initially two-phase saturated fluid (liquid and vapour) in thermal equilibrium
- Two differential equations (mass and energy)
- Mass flow (out) determined by injector model
- Enthalpy determined from saturation properties
- Iteratively solve tank temperature from volume constraints



Mass Balance:

$$\frac{dm_{tank}}{dt} = \dot{m}_{ox}$$

Energy Balance:

$$\frac{dU_{tank}}{dt} = \dot{m}_{ox} h_{outlet}$$

Vapour Mass Fraction:

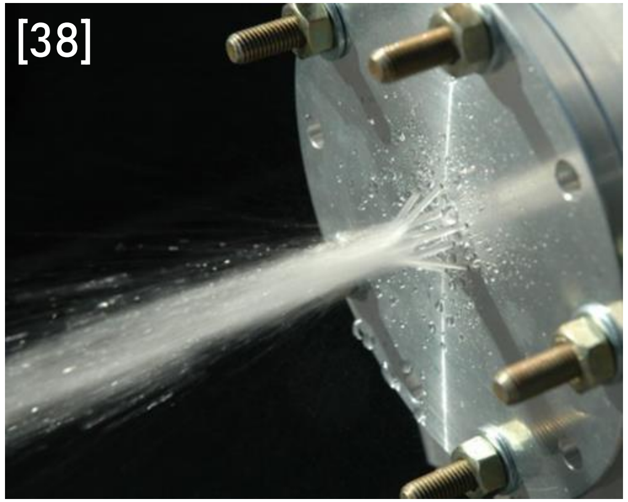
$$x_{tank} = \frac{U_{tank}/m_{tank} - u_{liq}}{u_{vap} - u_{liq}}$$

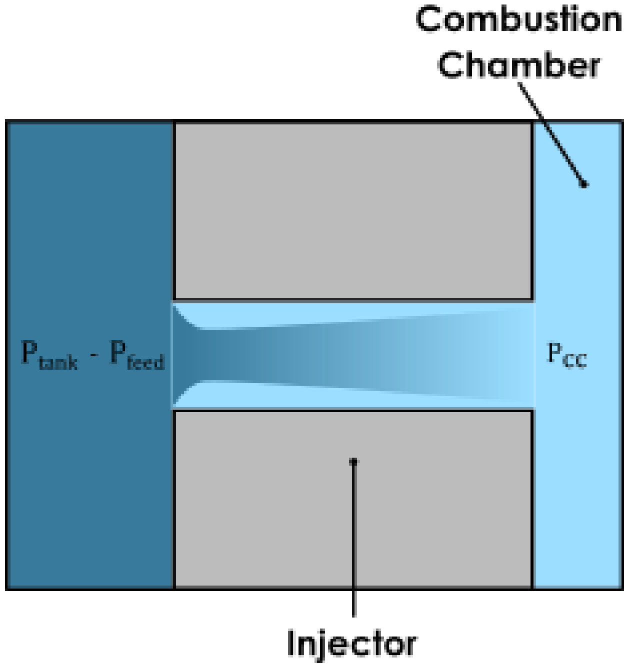
Tank Volume Constraint:

$$V_{tank} = m_{tank}\left(\frac{1 - x_{tank}}{\rho_{liq}} + \frac{x_{tank}}{\rho_{vap}}\right)$$

Modeling – Injector

- Boundary between feed system and combustion chamber
  - Crucial for combustion efficiency and stability
- Single phase incompressible model used for mass flow
- Empirical term captures model simplifications





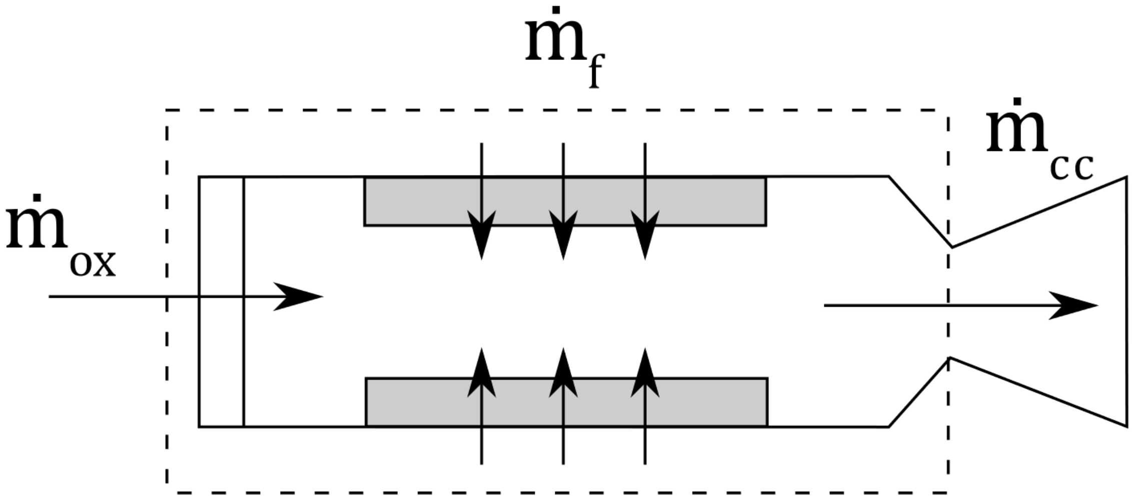
Oxidizer Mass Flow:

$$\dot{m}_{ox} = C_d A_{inj} \sqrt{2\rho_1 (p_{tank} - p_{cc} - p_{feed})}$$

Modeling – Combustion Chamber

Combustion Chamber Functions

- Houses fuel grain and igniter
- Location of combustion
- Downstream of injector, upstream of nozzle

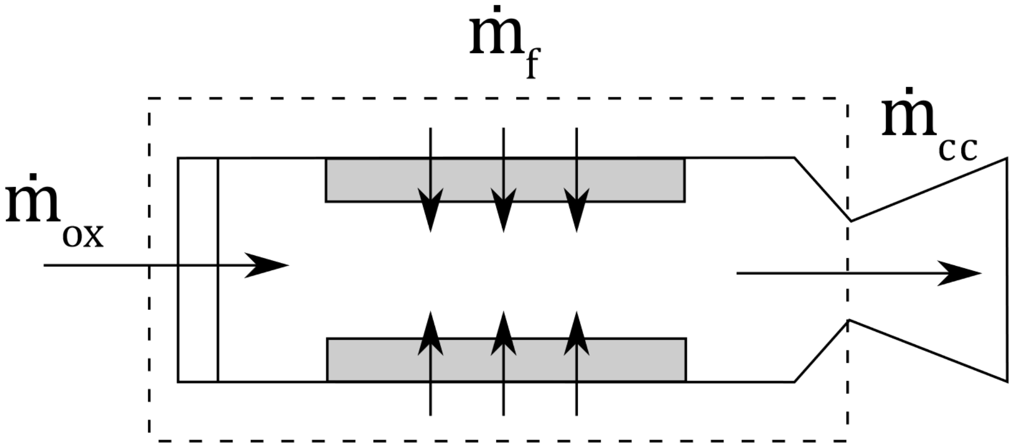


Modeling Assumptions

- Working fluid is homogenous ideal gas
- Flow is steady in chamber
- Properties are constant in chamber
- Combustion happens instantaneously
- Combustion is complete

Modeling – Mass Flow

- Fuel mass flow determined by
  - Mass flux through port
  - Fuel and oxidizer combination
- Ideal nozzle with correction used to determine pressure



Oxidizer to Fuel Ratio:

$$O/F = \frac{\dot{m}_{ox}}{\dot{m}_f}$$

Fuel Mass Flow:

$$\dot{m}_f = 2\rho_f \pi r L(aG^n)$$

Stagnation Pressure:

$$P_0 = \frac{\dot{m}_{cc}}{\zeta_d A_{th}} \sqrt{\frac{T_0 R}{k} \left(\frac{k+1}{2}\right)^{\frac{k+1}{k-1}}}$$

Chamber pressure:

$$P_{cc} = P_0 \left(\frac{T_{cc}}{T_0}\right)^{\frac{k}{k-1}}$$

Chemical Equilibrium with Applications

- CEA is a thermodynamic code written by NASA
- Calculates thermodynamic properties of combustion
- Allows for changes in oxidizer-fuel combination
- Lookup table created as function of P_{cc} and O/F

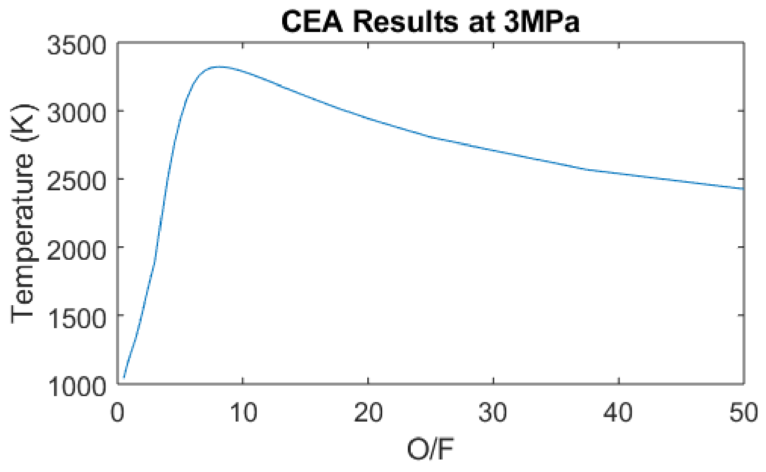
Sample CEA Input Code [14]

```
reac
        oxid N2O    wtfrac=1    t(k)=298.15
        fuel C32H66(a)    wtfrac=1    t(k)=298.15

prob    case=MuleSimData1  hp  p(bar)=30  o/f=9

output  siunits   massf   short

end
```



CEA Combustion Properties:

$$[T_{CEA}, \rho_{cc}, c_{p,cc}, k_{cc}] = CEA(P_{cc}, O/F)$$

Modeling – Nozzle

- Ideal nozzle with correction factor
  - Ideal gas with constant specific heats
  - No heat transfer
  - No shockwaves
  - Isentropic expansion
- Exit properties and thrust calculated using exit Mach number, upstream properties

Exit Pressure:

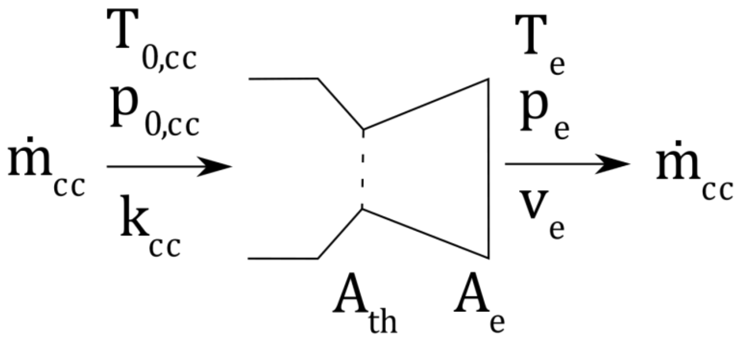
$$P_{exit} = P_0\left(1 + \frac{k-1}{2}M_{exit}^2\right)^{-\frac{k}{k-1}}$$

Exit Velocity:

$$v_{exit} = M_{exit}\sqrt{kRT_{exit}}$$

Nozzle Thrust:

$$F = \zeta_{C_F}(\dot{m}_{cc}v_e + (P_e - P_{atm})A_e)$$



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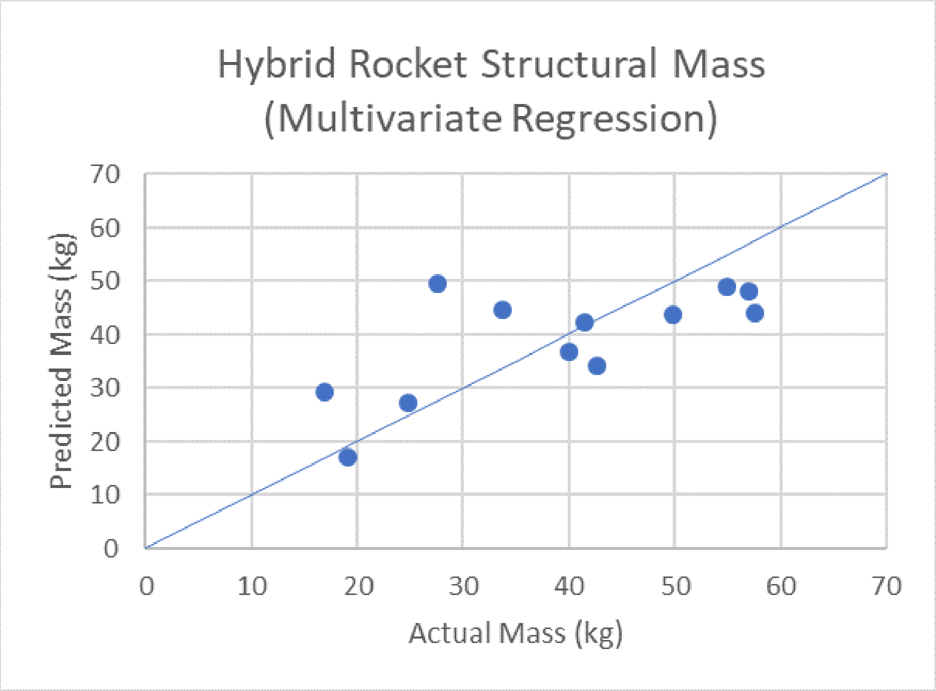
Modeling – Mass Estimation

- Rocket structural mass estimated from rocket outer diameter and propellant mass
- Experimentally fitted to twelve similarly sized hybrid rockets [16-27]
- Difficult to estimate accurately since heavily dependent on detailed design



Structural Mass Estimation:

$$m_{structural} = a_1 m_{propellant} + a_2 OD + a_3$$

Liftoff Mass:

$$m_{liftoff} = m_{structural} + m_{propellant} + m_{payload}$$

Modeling – Trajectory Analysis

- Used MATLAB function (Suborbit) developed by UVR member
  - Single degree of freedom trajectory analysis
  - Input is thrust and mass curves, output is altitude, velocity, and acceleration curves
  - Validated against 27 solid rocket motors (mean error 5.7%) [28]
- Use average drag coefficient values
- Corrects thrust for pressure term (varies with altitude)

Aerodynamic Drag:

$$F_D = C_D A \frac{\rho v^2}{2}$$

Vertical Acceleration:

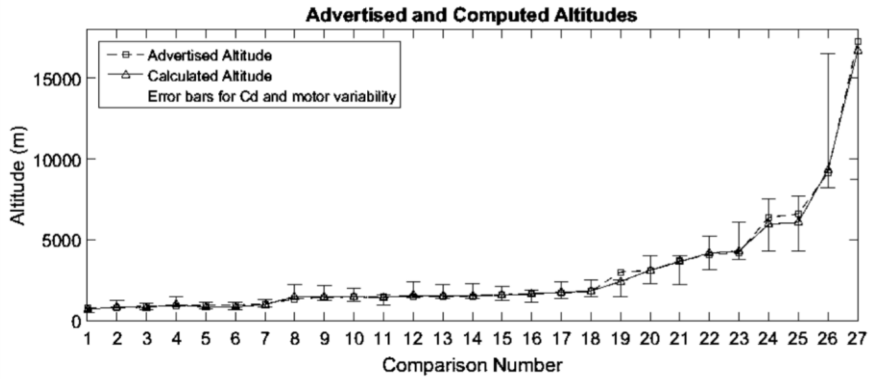
$$a = \frac{F_T - F_D}{m} - g$$

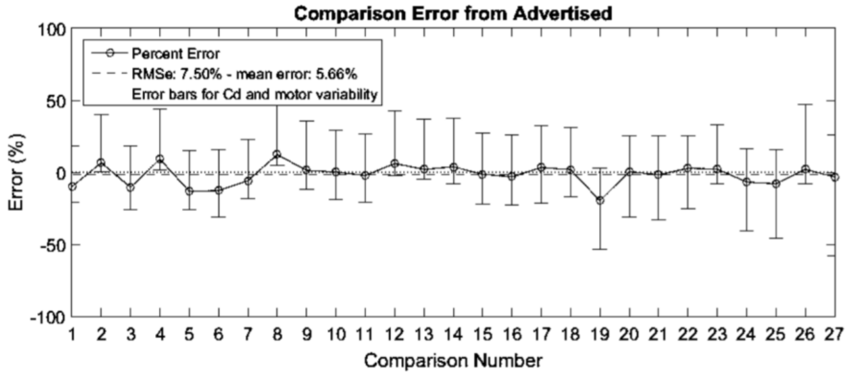
Vertical Velocity:

$$v = \int a dt$$

Altitude:

$$y = \int v dt + y_{launch}$$





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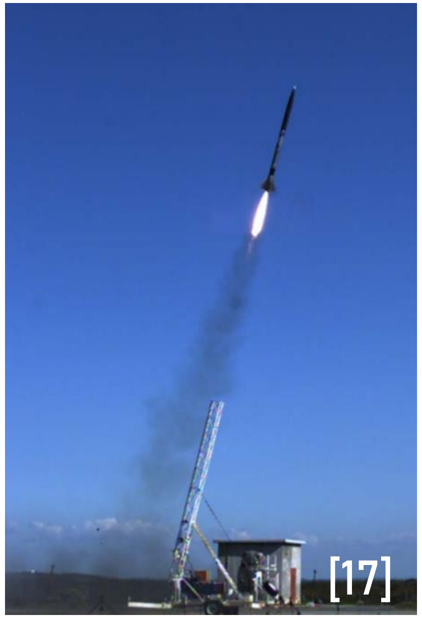
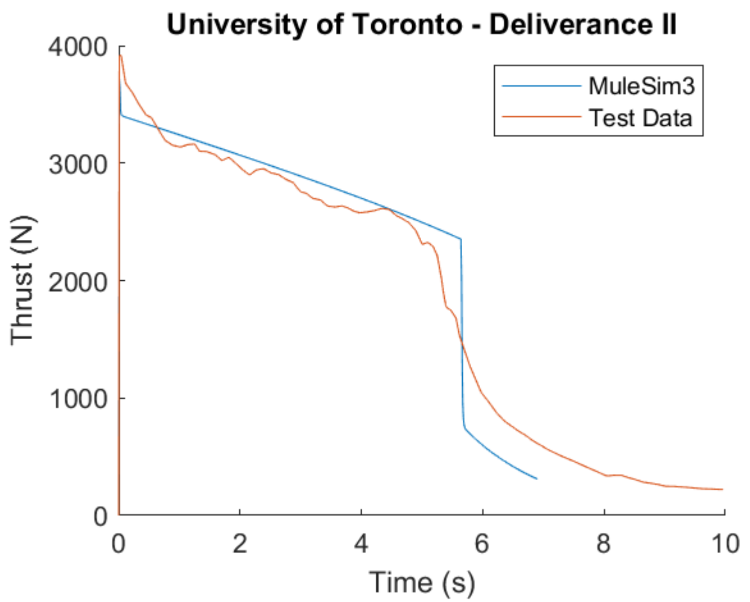
Model Validation

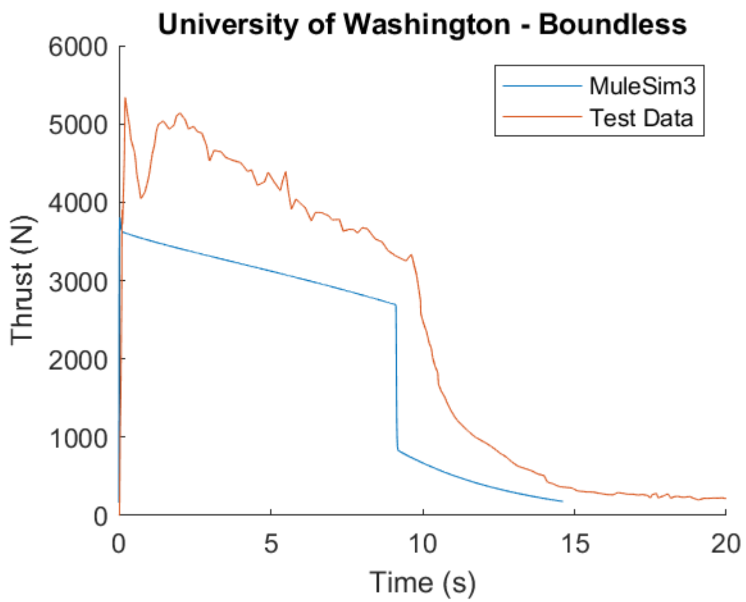
- Only hybrid motor model validated, since "Suborbit" already validated
- Compared various parameters from test results in literature to model output
  - Tank pressure, chamber pressure, motor thrust
  - Only paraffin-N_2O with self-pressurized tanks
- Three rockets examined
  - Boundless – University of Washington [16]
  - Phoenix 1A – University of KwaZulu-Natal [17]
  - Deliverance II – University of Toronto [18]

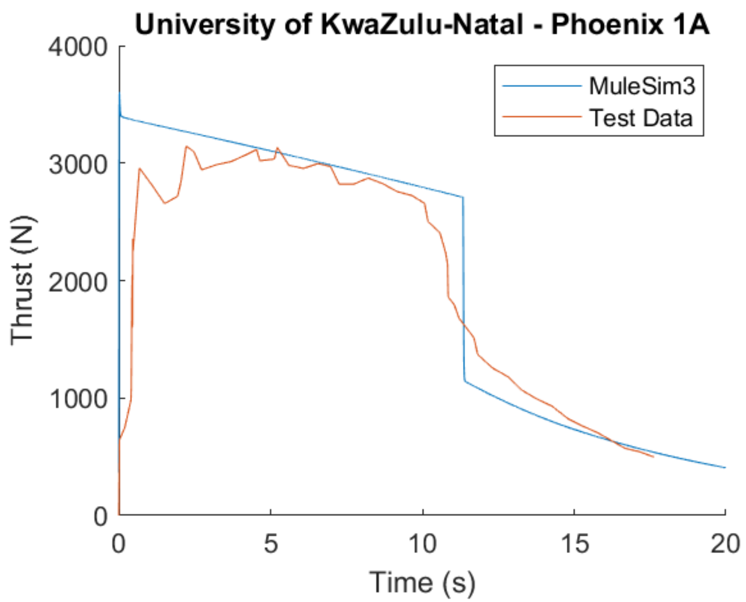


Model Performance Metric: $$e_F = \frac{MAE}{\mu} = \frac{\sum\left|F_i - F_{model,i}\right|}{\sum F_i}$$

Model Validation – Results







Model Error	Deliverance II	Boundless	Phoenix 1A
Tank Pressure	1.55 %	15.0 %	-
Combustion Chamber Pressure	10.7 %	25.7 %	12.3 %
Thrust	7.55 %	32.8 %	12.2 %

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Model Validation – Results

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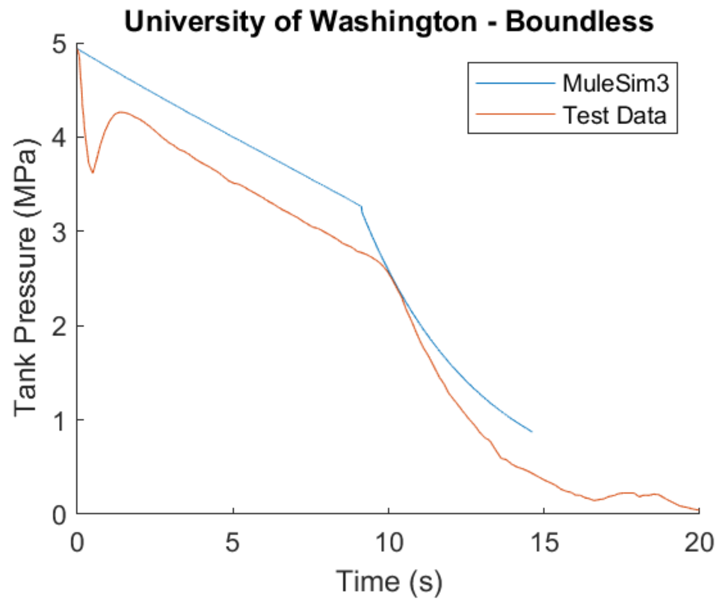
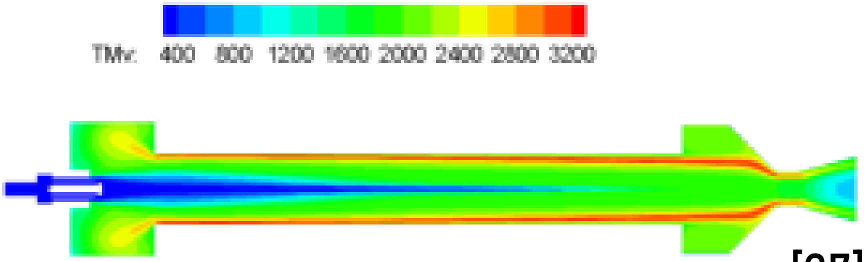


Significant sources of error:

- Uncertainties in input, particularly coefficient of discharge, initial oxidizer mass
- Tank model unable to capture initial transient in tank pressure
- Phoenix 1A tank supercharged with helium
- Did not adjust for different combustion and nozzle efficiencies

Modeling – Limitations

- Does not capture transient effects
  - Initial tank pressure decrease
  - Combustion instability
- No heat transfer
- No axial and radial variation in properties
- Does not analyze non-chemical sources of combustion efficiency
  - Pre/post combustion chambers
  - Injector atomization profile
  - Effects of additives



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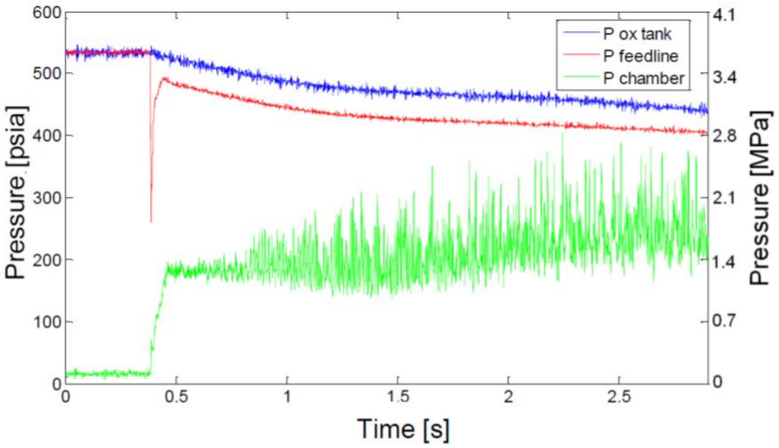

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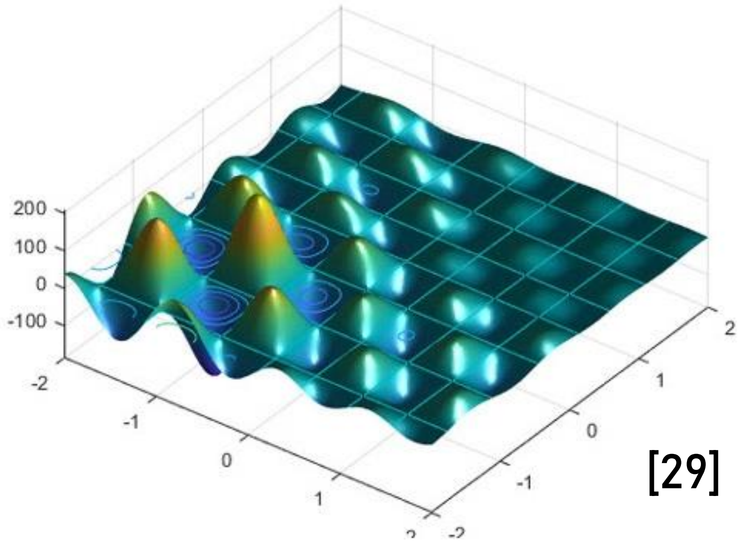
Design Optimization

- Validated model used to maximize specific impulse
- Built-in MATLAB functions used:
  - "fmincon" – interior point (default), trust region reflective, SQP, active set
  - "simulannealbnd" – simulated annealing
  - "ga" – genetic algorithm
  - "patternsearch" – pattern search

- Function not defined over entire design space
- Parameter scaling to improve algorithm performance
- Constraints handled internally
  - Reduce model evaluations



[29]

Specific Impulse:

$$I_{sp} = \frac{I_{tot}}{m_{prop} g} = \frac{v_e}{g}$$

Design Optimization – Design Variables

Parameter	Description	Lower Bound	Upper Bound
Oxidizer Tank Volume	Controls altitude	4 L	14 L
Tank Pressure	Controls oxidizer mass flow	4 MPa	6 MPa
Effective Injector Area	Controls thrust, OF ratio, chamber pressure	10 mm^2	40 mm^2
Fuel Grain Length	Controls OF ratio	0.2 m	0.6 m
Fuel Grain Initial Diameter	Controls mass flux	30 mm	80 mm
Nozzle Throat Diameter	Controls thrust, chamber pressure	20 mm	40 mm
Nozzle Area Ratio	Optimizes thrust	3	10

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Design Optimization – Constraints

- Chamber Pressure
  - Avoid combustion instability and backflow failure mode [30]

$$P_{cc} < 0.8(P_{tank} - P_{feed})$$

- Fuel grain port mass flux
  - High mass flux associated with combustion instability [31]

$$G < 500\frac{kg}{m^2 s}$$

- Outer diameter
  - Use current fuselage mold to reduce costs

$$d_{port,f} + 0.05m < 0.14m$$

- Off-the-rail-velocity
  - Required for aerodynamic stability [32]

$$v_{off-the-rail} > 30m/s$$

- Acceleration
  - Reduces loads

$$a_{max} < 100\frac{m}{s^2}$$

- Altitude
  - Rocket must be close to 10000ft AGL [32]

$$abs(alt_{max} - 3050m) < 100m$$

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Design Optimization – Results

- Genetic algorithm function yielded best results
- Used default function settings

Design Parameter	Value
Oxidizer Tank Volume	10 L
Tank Pressure	6 MPa
Effective Injector Area	17.5 mm^2
Fuel Grain Length	23.9 cm
Fuel Grain Initial Diameter	5.63 cm
Nozzle Throat Diameter	2.16 cm
Nozzle Area Ratio	7.04

Performance Parameter	Value
Specific Impulse	206 s
Maximum Altitude	3143m
Off-the-rail Velocity	30.3 m/s
Maximum Thrust	2606 N
Burn Time (Liquid)	4.81 s
Liftoff Mass	35.1 kg
Total Impulse	10600 Ns

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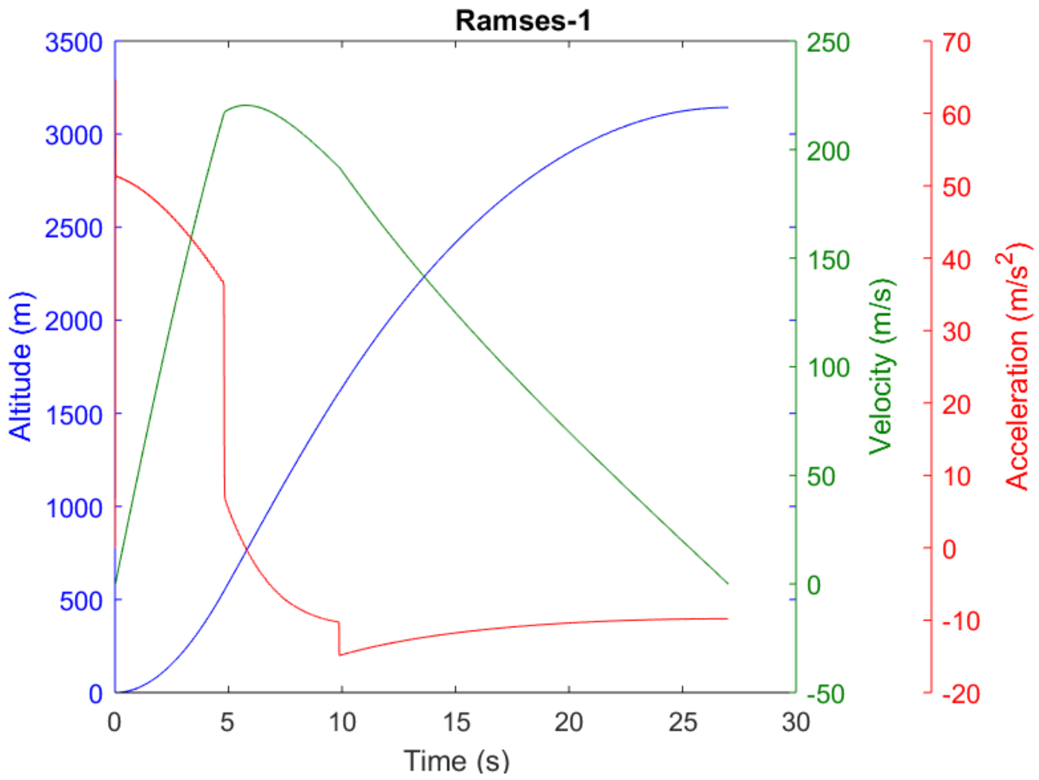
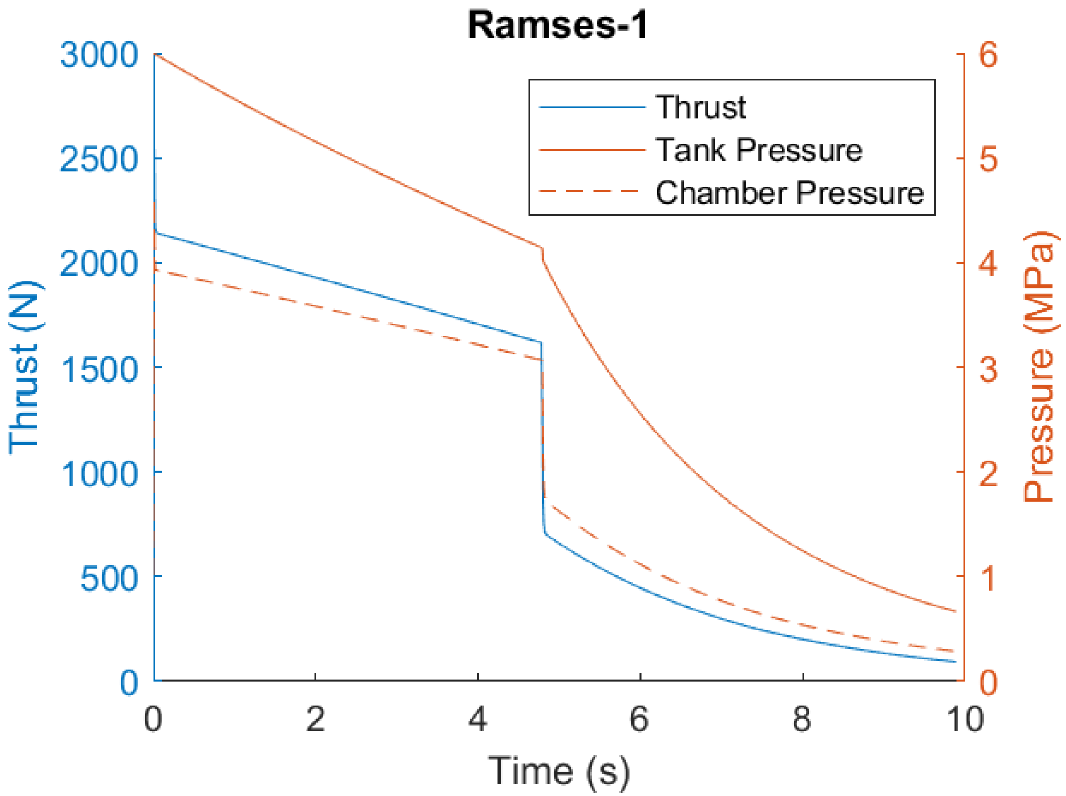
Design Optimization – Results





Sensitivity Analysis

- Global methods preferred
  - High Coupling between parameters

- Elementary effects method selected [33]
  - Requires few model evaluations
  - Global one-at-a-time approach
  - Normally used as screening method

- Compared input effects on:
  - Altitude (most important)
  - Specific impulse (secondary goal)

Elementary Effect:

$$EE_i^j\left(\mathbf{x}^{(l)}\right) = \frac{\left[y\left(\mathbf{x}^{(l+1)}\right) - y\left(\mathbf{x}^{(l)}\right)\right]}{\Delta}$$

Sensitivity Measure:

$$\mu_i^* = \frac{1}{r}\sum_{j=1}^{r} |EE_i^j|$$

Uncertainty Quantification

Parameter	Mean	Deviation	Distribution	Notes
Tank Volume	10 L	0.02	Normal	Machining tolerances
Tank Fill Level	60 %	0.1	Normal	Difficult to measure directly
Initial Oxidizer Tank Pressure	6 MPa	0.05	Uniform	
Effective Injector Area	17.5 mm^2	0.05	Uniform	
Fuel Grain Length	0.239 m	0.01	Normal	
Fuel Grain Initial Diameter	56.3 mm	0.025	Normal	
Nozzle Throat Diameter	21.6 mm	0.075	Normal	Large due to throat regression [34]
Nozzle Area Ratio	7.04	0.14	Normal	Large due to throat regression
Feed system pressure drop	0.1 MPa	0.5	Uniform	Highly dependent on detailed design
Density of Fuel	930 kg/m^3	0.032	Uniform	
Regression Rate Constant	0.155	0.1	Uniform	Standard for paraffin-N_2O [35]
Regression Rate Exponent	0.5	0.01	Uniform	Standard for paraffin-N_2O
Altitude	1400 m	0.007	Normal	Altitude of Spaceport America (SA)
Ambient Pressure	0.867 MPa	0.1	Normal	Atmospheric pressure at SA
Ambient Temperature	301 K	0.02	Normal	Average temperature at SA
Discharge Correction Factor	1.05	0.05	Uniform	Values suggested in Sutton [36]
Characteristic Velocity Correction Factor	0.8	0.1	Uniform	Values suggested in Sutton
Thrust Coefficient Correction Factor	0.9	0.05	Uniform	Values suggested in Sutton
Drag Coefficient Variability	1	0.1	Uniform	

- 19 uncertain parameters
- Uncertainty based on expected values at launch day

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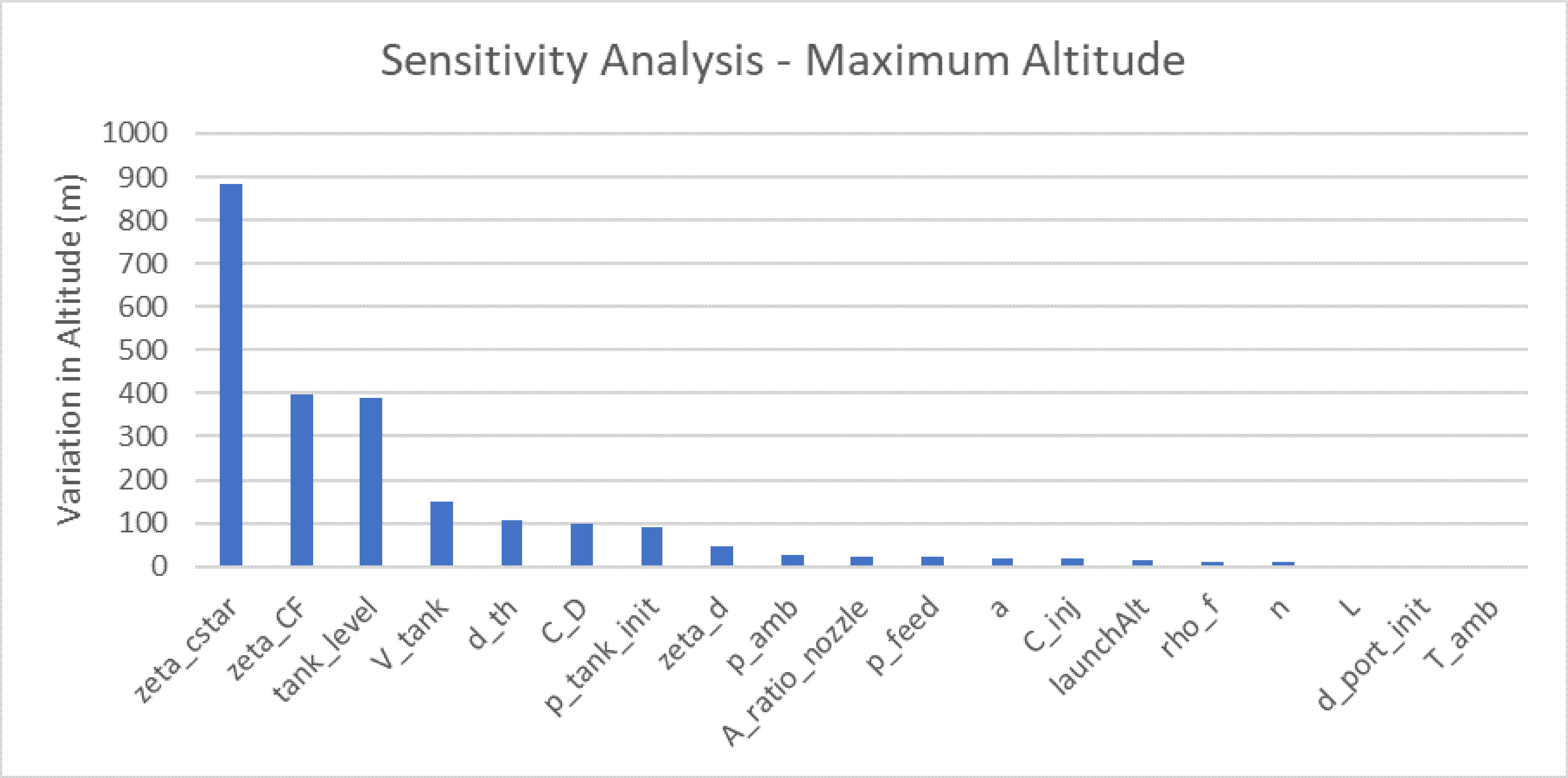
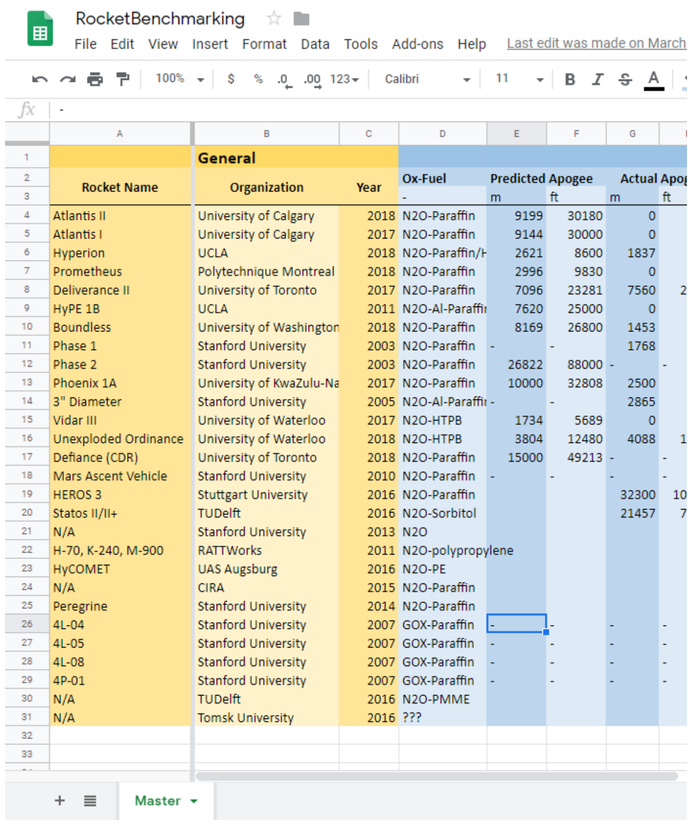
Sensitivity Analysis – Altitude



Lessons Learned

- Accurate hybrid rocket modeling is difficult
  - Hybrid rocket physics not well understood
  - Large errors and uncertainties in measurement and input
  - Empirical testing still dominates
- Record as much as possible
  - Compile similar relevant facts in one place
  - Write report section as soon as task completed
- Understanding algorithms is crucial to using them
  - Problem was initially poorly set up for MATLAB functions
- Your solution is only as good as your problem definition



General							
Rocket Name	Organization	Year	Ox-Fuel	Predicted Apogee		Actual Apog	
			-	m	ft	m	ft
Atlantis II	University of Calgary	2018	N2O-Paraffin	9199	30180	0	
Atlantis I	University of Calgary	2017	N2O-Paraffin	9144	30000	0	
Hyperion	UCLA	2018	N2O-Paraffin/H	2621	8600	1837	
Prometheus	Polytechnique Montreal	2018	N2O-Paraffin	2996	9830	0	
Deliverance II	University of Toronto	2017	N2O-Paraffin	7096	23281	7560	2
HyPE 1B	UCLA	2011	N2O-Al-Paraffi	7620	25000	0	
Boundless	University of Washington	2018	N2O-Paraffin	8169	26800	1453	
Phase 1	Stanford University	2003	N2O-Paraffin	-	-	1768	
Phase 2	Stanford University	2003	N2O-Paraffin	26822	88000	-	-
Phoenix 1A	University of KwaZulu-Na	2017	N2O-Paraffin	10000	32808	2500	
3" Diameter	Stanford University	2005	N2O-Al-Paraffi	-	-	2865	
Vidar III	University of Waterloo	2017	N2O-HTPB	1734	5689	0	
Unexploded Ordinance	University of Waterloo	2018	N2O-HTPB	3804	12480	4088	1
Defiance (CDR)	University of Toronto	2018	N2O-Paraffin	15000	49213	-	-
Mars Ascent Vehicle	Stanford University	2010	N2O-Paraffin	-	-	-	-
HEROS 3	Stuttgart University	2016	N2O-Paraffin			32300	10
Statos II/II+	TUDelft	2016	N2O-Sorbitol			21457	7
N/A	Stanford University	2013	N2O				
H-70, K-240, M-900	RATTWorks	2011	N2O-polypropylene				
HyCOMET	UAS Augsburg	2016	N2O-PE				
N/A	CIRA	2015	N2O-Paraffin				
Peregrine	Stanford University	2014	N2O-Paraffin				
4L-04	Stanford University	2007	GOX-Paraffin	-	-	-	-
4L-05	Stanford University	2007	GOX-Paraffin	-	-	-	-
4L-08	Stanford University	2007	GOX-Paraffin	-	-	-	-
4P-01	Stanford University	2007	GOX-Paraffin	-	-	-	-
N/A	TUDelft	2016	N2O-PMME				
N/A	Tomsk University	2016	???				

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Future Work

- More powerful sensitivity analysis tools
  - Variance-based sensitivity
- Testing of scale motor
  - Validation of model with more detailed data
  - Determination of realistic correction coefficients
- Develop higher fidelity models
  - CFD of injector and propellant tank
  - Combustion stability analysis
- Detailed design
  - Specify rocket architecture
  - Fill and feed system, combustion chamber, tank, etc.
- Uncertainty-based design optimization

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References

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