Spring 2024
Technical Report

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Flight-Scale Hybrid Engine
Propulsion

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1 Technical Report

2 System Overview

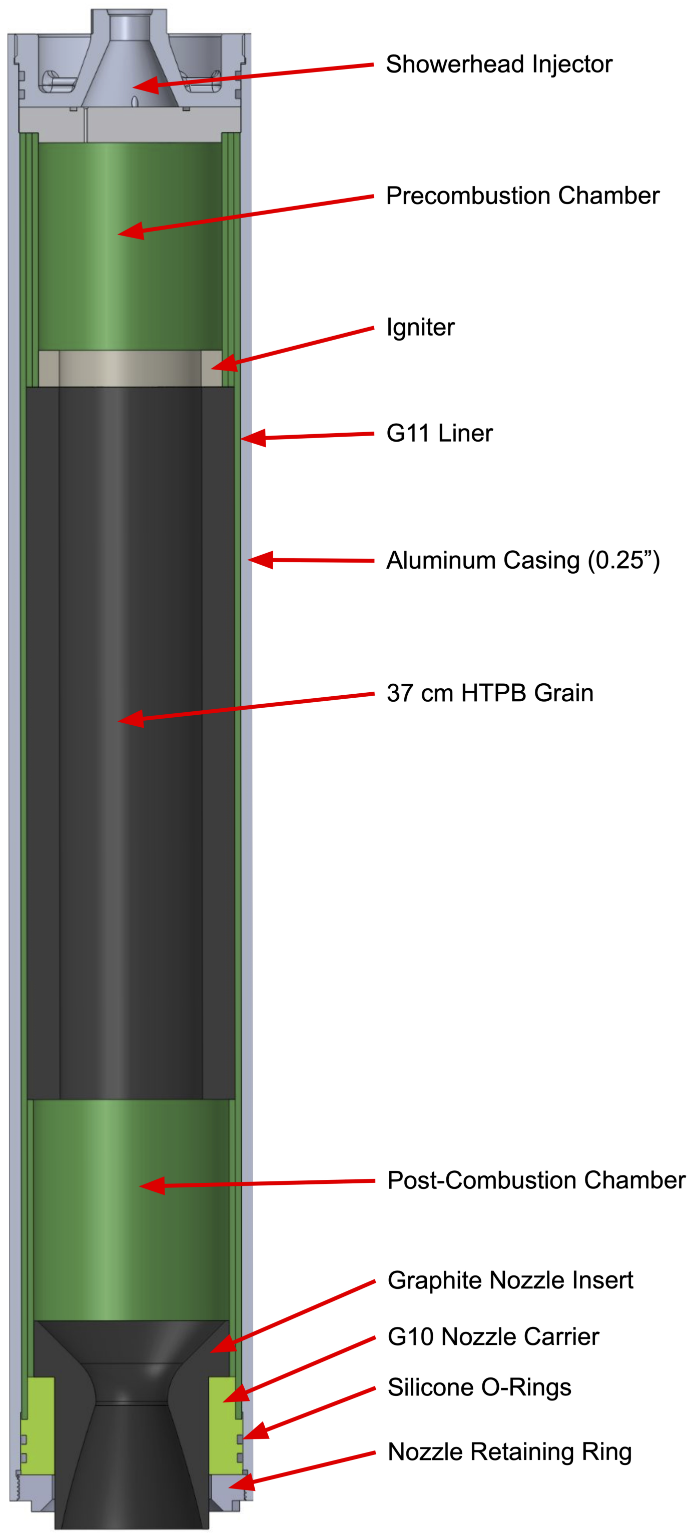

Figure 1: Hybrid Motor Cross Section

2.1 System Purpose

- The purpose of project is to design, manufacture, and test a hybrid engine to compete in the 2025 Spaceport America Cup 10K SRAD Hybrid category. Before this engine is flown, it must be thoroughly tested while collecting data on key parameters. This engine follows the successful hotfire of the 250lbf hybrid engine in January 2024. From analyzing the results of this test, we have decided to focus on improving the nozzle, injector, and lining.

  These variables include:

  - Thrust
  - Run tank Pressure
  - Run tank Temperature
  - Oxidizer Flow rate
  - Combustion chamber pressure

2.2 Functional Requirements – Flight Hardware

PROP 1 Launch vehicles entered the IREC shall use non-toxic propellants.

Safety Critical, Reference: IREC DTE 5.5

PROP 2 Teams shall comply with all rules, regulations, and best practices imposed by the authorities at their chosen test location(s).

Safety Critical, Reference: IREC DTE 5.20

PROP 2.1. Teams must complete required testing by May 1st of the competition year. Teams unable to make this deadline can switch to COTS propulsion.
Internal Requirement

PROP 3 SRAD propulsion systems shall successfully (without significant anomalies) complete an instrumented (chamber pressure and/or thrust), full scale (including system working time) static hot-fire test prior to the IREC.

Safety Critical, Reference: IREC DTE 5.22

PROP 4 SRAD propulsion systems shall successfully complete 2 of 3 static fire test before being used on competition rocket

Safety Critical, *Internal Requirement*

PROP 5 All pressure-containing hardware interfaces should be verified with FEA.

Internal Requirement

PROP 6 Metallic cases shall be made of non-ferrous ductile metals such as 6061 aluminum alloy

Reference: IREC DTE 5.7

PROP 7 To help prevent misfires, all teams shall use dual or dual-headed motor igniters for launch.

Reference: IREC DTE 5.9

PROP 7.1. The ignition system shall provide sufficient heat flux to ignite the motor within 0.5s.
Internal Requirement

PROP 7.2. The upper bulkhead shall contain the combustion gases entirely.
Internal Requirement

PROP 7.3. The igniter system shall include utilize redundancies including 2 initiators.
Reference: IREC RSOP 4.4

PROP 8 The thrust-optimizing nozzle shall be used to accelerate exhaust gases.

Safety Critical, *Internal Requirement*

PROP 8.1. Structural components of the nozzle shall be fully reusable.
Internal Requirement

PROP 8.2. Ablative components of the nozzle shall be functional for 2+ firings.
Internal Requirement

2.3 Performance Requirements – Flight Hardware

PROP 9 SRAD and modified COTS pressure vessels constructed entirely from isotropic materials (e.g. metals) shall be designed to a burst pressure no less than 2 times the maximum expected operating pressure, where the maximum operating pressure is the maximum pressure expected during pre-launch, flight, and recovery operations.

Safety Critical, Reference: IREC DTE 6.23

PROP 10 SRAD and modified COTS pressure vessels shall be proof pressure tested successfully (without significant anomalies) tested to 1.5 times the maximum expected operating pressure for no less than twice the maximum expected system working time, using the intended flight article(s) (eg the pressure vessel(s) used in proof testing must be the same one(s) flown at the IREC). The maximum system working time is defined as the maximum uninterrupted time duration the vessel will remain pressurized during pre-launch, flight, and recovery operations.

Safety Critical, Reference: IREC DTE 6.25

PROP 11 Launch vehicles entered in the IREC shall not exceed an installed total impulse of 9,208 pound-seconds (40,960 Newton-seconds).

Reference: IREC RR 2.4

PROP 12 The assembled motor (including all fasteners) shall have a diameter of less than 5.35 inches.

Internal Requirement

PROP 12.1. The motor casing shall have a diameter of 5 inches.
Internal Requirement

PROP 13 The motor casing shall not be longer than 42".

Internal Requirement

PROP 14 The Propulsion system shall provide a thrust to weight ratio of at least 5:1.

Reference: IREC DTE 5.23

PROP 15 The pressure drop across the injector shall be at least 40 percent.

Internal Requirement

3 250 lbf Hotfire Results

Figure 2: 250 lbf Hybrid Engine Test

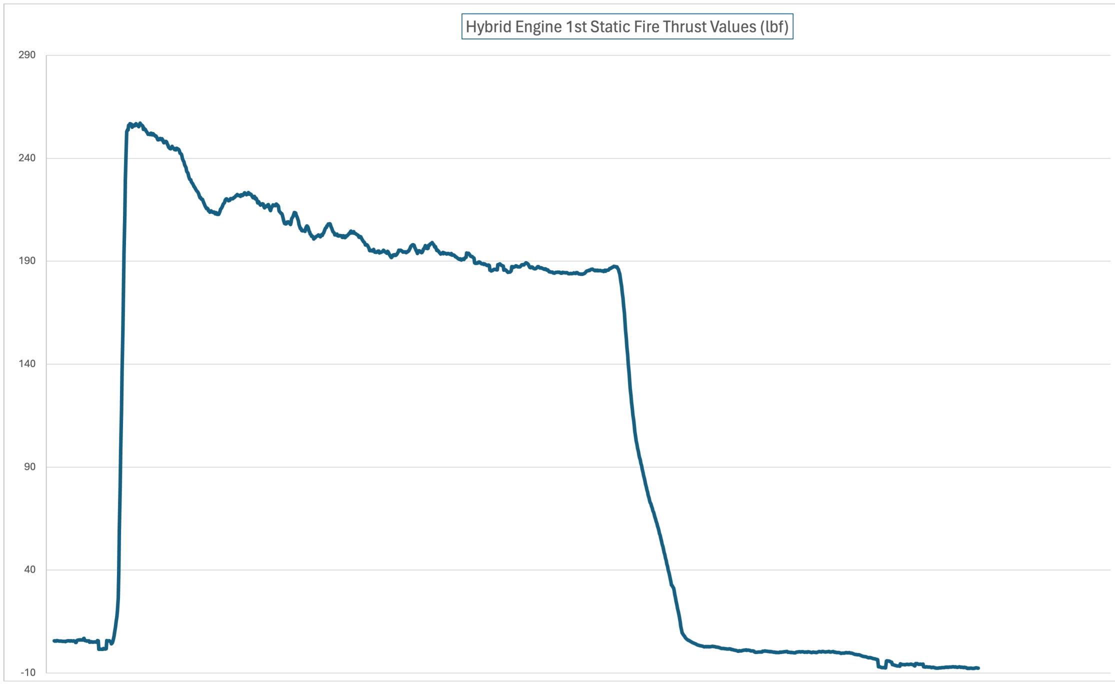

Figure 3: 250 lbf Hybrid Engine Thrust Curve

- The 250lbf hybrid engine produced around 260lbf of peak thrust, and performed very closely to as predicted by the hotfire Matlab simulation script. After disassembling the engine, several issues were noted.
  - Pre-Combustor Burn Through
    - It seems that the liner burn through was most severe above the fuel grain which is most likely due to a recirculation zone bringing hot combustion products upstream by the liner. This is good because it helps combustion stability but bad because it burns through the liner very quickly.
  - Post-Combustor Burn Through
  - Nozzle O-Ring degradation
  - Post-Combustor Burn Through
  - Heavy Nozzle Fouling

Figure 4: Pre-Combustor Burn Through

Figure 5: Nozzle O-Ring degradation

Figure 6: Nozzle Fouling

Figure 7: Post-Combustor Burn Through

4 Design Changes

- The Flight Scale Hybrid engine is different from the 250lbf engine in several ways:
  - Casing Enlargement
  - Nozzle Retention Ring
  - Phenolic Nozzle
  - Full Graphite Nozzle
  - Injector
  - Thicker Liners

4.1 Combustion Chamber Liners

- Due to the burnthrough of the 250lbf engine, the liner thickness was increased in the pre and post combustion chambers. Instead of one layer of 0.125" G11 fiberglass tube, three layers of 0.125" G11 were used in the precombustor, and two layers were used in the postcombustor. The section insulated by the grain was not changed, as there was no burn through in this area. The HTPB grain is still cast directly into the outer liner. The post combustor liners were glued together with high-temp epoxy, and the precombustor liners were glued with 5-minute epoxy. All the liners should be glued with high-temp epoxy if possible. In the future, the liners should be ordered to size and then sanded, due to the approximately 0.005" gap between liners which resulted in gaps during the gluing process.

4.2 Fiberglass Nozzle Carrier

Figure 8: Fiberglass Nozzle Carrier and Liner

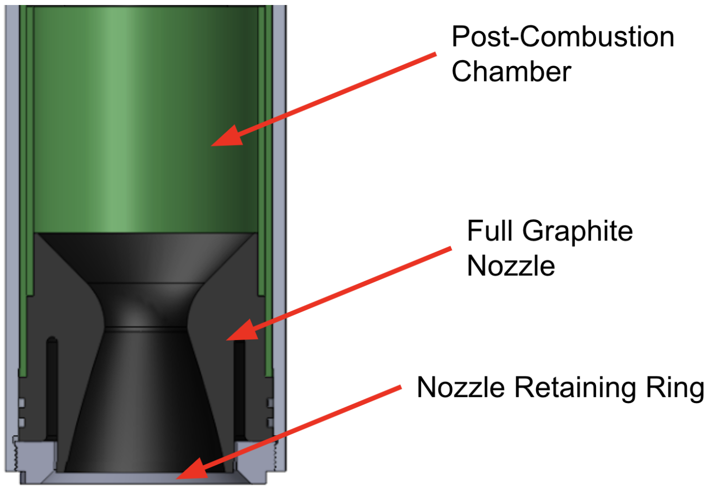

Figure 9: Full Graphite Nozzle

- Due to the Nozzle O ring degradation of the 250lbf engine, the nozzle was changed from full solid graphite to a G10 carrier with a graphite insert. The G10 is much less thermally conductive than graphite, so it will prevent excessive heat transfer to the casing and Nozzle O-rings.
- A more insulating full graphite nozzle was also developed. This nozzle features a cutout to prevent heat transfer to the O-rings and casing. However, this nozzle was not used in this iteration of the engine due to uncertainty about real-world performance. The nozzle should be destructively tested on a hydraulic press to verify actual strength.

5 Full Scale Hotfire Results

- Test one:
  - The first test of the full scale hybrid engine was not a success due to the failure of the igniter to ignite the engine. The igniter was smaller than the igniter for the 250lbf version due to the fact that the inner diameter of the grains is larger, and the inner diameter of the liners in the precombustor remained the same. Another issue that could have resulted in the failure to ignite was the failure of the electrical ignition system. The igniter did not ignite when the ignition button was pressed due to the battery being dead. The ignition wires were then stuck directly into another battery which did end up igniting the igniter, however at this point the test was already not progressing as required by the procedure. The test should have been scrubbed when the igniter failed to light, and recycled for another day. The igniter burned until there was visual confirmation of black smoke, and then after a 3 second count down, the MAV was opened. The engine did not start burning until all the liquid nitrous had

been exhausted. The gas phase burn was quickly extinguished by activating the engine purge system and closing the MAV.

Another issue was that the run tank pressure did not achieve the desired value of 850psi during the fill. This is due to the large quantity of nitrous required to fill the upgraded run tank. Although the nitrous supply tank was being heated and had a pressure of xx before filling, the pressure dropped to around 6xx towards the end of the fill. This is due to the nitrous in the supply tank boiling off and reducing the temperature and consequently the pressure of the contents.

- We decided to ignite the engine with a makeshift APCP igniter as specified in the backup plan. During this second attempt to fire the engine, we again did not achieve the desired run tank fill pressure or level. Because the test was already off nominal, we decided to proceed anyways to see how the gas phase burn performed. The engine ignited immediately after the MAV was opened and burned for about 11 seconds. The peak thrust produced was approximately 100lbf which is quite low, but not completely unexpected for a gas phase burn. During this test, data was only being recorded at 2HZ which likely obscured valuable information. This has already been fixed by software.

Figure 10: First Hotfire Attempt

Figure 11: Second Hotfire Attempt

5.1 Testing Procedure Improvements

- 2x Nitrous Supply tanks connected during the fill stage
- Larger Igniter
- More robust electrical ignition system

5.2 Full Scale Matlab Script

6 Theory

Liquid N2O is injected into the combustion chamber by the injector. The combustion chamber is at a much lower pressure than upstream of the injector (850 psi upstream vs 350 combustion chamber) As the liquid N2O oxidizer passes across the injector, some of the liquid N2O becomes gas due to pressure drop. This vapor then accelerates to the local speed of sound, causing a choked flow condition along injector orifice. This causes the mass flow of the injector to only be dependent on the upstream conditions. This also isolates the fluid system from upstream propagating oscillations due to variations in chamber pressure.

The injected N2O then passes through the pre-combustion chamber, where it continues to vaporize. The pre-combustion chamber is just an empty space between the injector and grains.

The oxidizer then reaches the grains, where it mixes with the vaporizing fuel, as shown below. The oxidizer and fuel react, in an approximately 7-1 oxidizer-fuel ratio. The fuel and oxidizer ratio will continue to mix and combust in the post combustion chamber, which is an empty section of the combustion chamber above the nozzle and below the grains. The precombustion chamber features a recirculation zone which circulates hot combustion gases upstream, which decomposes incoming nitrous and promotes stable combustion[1].

The generation of hot gases from combustion create pressure in the chamber (350 psi), and then are accelerated by the nozzle to supersonic speeds, creating thrust.

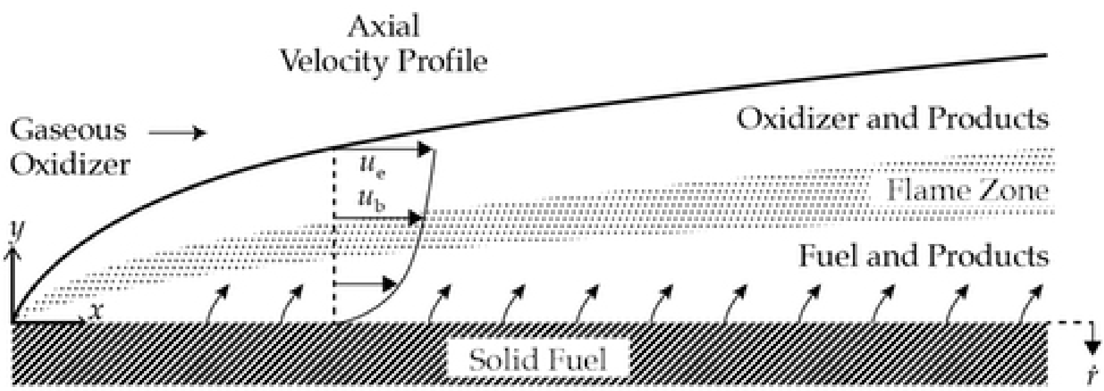

Figure 12: Hybrid Combustion Diagram [3]

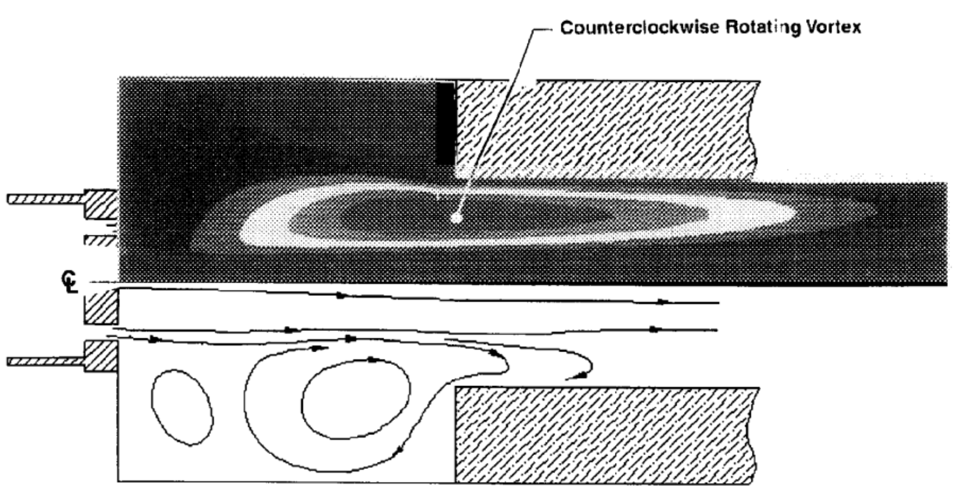

Figure 5 Streamlines and schematic representation of the flowfield for the axial injector.

Figure 13: Recirculation in Precombustor [1]

The grains are designed to provide to correct burning surface area to supply enough fuel to achieve a stoichiometric ratio. There will be one cylindrical port grain, because this is the easiest grain to model and to modify.

As the grain burns, the surface area increases, so the fuel oxidizer ratio shifts towards fuel rich. However, the oxidizer flow rate will decrease as the tank empties due to the decrease in vapor pressure caused by the nitrous boiling off.

The oxidizer is supplied by the test stand, which consists mainly of a tank and a valve to control oxidizer flow. One important note is that the pressure and temperature in the run tank will affect the flow rate of oxidizer through the injector.

Parameter	Value
Specific Gas Constant	306 J/(Kg·K)
Combustion Chamber Temperature	3304 K
Specific Heat Ratio of Combustion Products (at 350 psi chamber pressure)	1.1453 (unitless)
Initial Injector Mass Flow Rate	1.5 Kg/s
Exit Velocity	2304 m/s
Peak Thrust	950 LBF
Chamber Pressure	350 PSI
Target Impulse	22000 Ns

Table 1: Engine Parameter Values

These parameters were combined with these fundamental Hybrid Combustion equations[2]:

$$\dot{m}_f = \rho_f A_b \dot{r} = 2\pi \rho_f R L \dot{r} \tag{1}$$

$$\dot{m}_f = \rho_f A_b r \tag{2}$$

$$\dot{r} = a \left(\frac{\dot{m}_0}{\pi R^2} \right)^n . \tag{3}$$

Along with other standard rocket engine equations in Matlab to create a performance estimation script.

6.1 Matlab Script

This script is used to provide an estimate for thrust, chamber pressure, O/F ratio, regression rate, and fuel mass flux over time. The inputs of this script are oxidizer mass flow rate, inner diameter of grain, grain length, throat diameter, and combustion temperature. Outputs are solved numerically and iteratively. The oxidizer flux is simply a linear fit based on Waterloo's hybrid blowdown curve[4]. Note that exit pressure is assumed to be equal to ambient pressure which is not accurate because the chamber pressure decreases significantly during the burn, and the ambient pressure will also be changing as the rocket ascends. Also, the nozzle is assumed to be isentropic.

```
R=8314/27.142;% J/kg*K, gas constant for HTPB and N2O
y= 1.14 ; %specific heat ratio for HTPB and N2O
```

```
pe= 101325; %pa %pamb
%pe=101219.16; %spaceport
%pcp= 350 ;%psi
Tc=3300; %K

pc=380*6894.76; %psi to pa set max chamber pressure

astar=0.00109871; %guess


a=.192; %constant  in aG^n equation for regression rate
n=.675; %constant
id=7.3; %cm fuel grain initial ID
OD=10.795; %cm grain OD
t=7; %burn time in seconds


mox=1900; %oxidizer mass flux in grams
L=37; %cm grain length
rhofuel=1036;%kg/m^3 htpb density

ves=sqrt((2*R*y/(y-1))*Tc*(1-(pe/pc)^((y-1)/y))); %hybrid propulsion book pg 166


%define time vector from 0 to t seconds with dt second interval
dt=0.01;
tvec = 0:dt:t;

gma=y*(2/(y+1))^((y+1)/(y-1)); %gamma
cstar=sqrt(((R*Tc)/gma)); %cstar
sizee=size(tvec);
%efvec=linspace(1,0.76,sizee(2));
oxvec=linspace(1,0.62,sizee(2))*mox;

Aflat=pi*(((OD/2)^2)-((id/2)^2)); %top and bottom areas
%Aflat2=pi*(((OD/2)^2)-((id2/2)^2));
%mfuel=rhofuel*Aflat*L/1e6; %initial fuel mass in kg
mfuel=(rhofuel*(Aflat*L))/1e6; %initial fuel mass in kg

rdot=a*(((4*oxvec(1))/(pi*id^2))^n);%mm/sec regression rate
%rdot2=a*(((4*oxvec(1))/(pi*id2^2))^n);%mm/sec regression rate

fdot=rdot*rhofuel*(pi*L*id+(Aflat*2))/1e4; %g/s ;
%fdot2=(rdot*rhofuel*(pi*L2*id2))/1e4;
OF=oxvec(1)/(fdot); %oxidizer/fuel ratio

OFT=7.20;
```

```
 while OF>OFT+.01 || OF<OFT
 if OF>OFT
     mox=mox-1;
 end

 if OF<OFT
     mox=mox+1;
 end
 oxvec=linspace(1,0.62,sizee(2))*mox;


Aflat=pi*(((OD/2)^2)-((id/2)^2)); %top and bottom areas
mfuel=rhofuel*Aflat*L/1e6; %initial fuel mass in kg

rdot=a*(((4*oxvec(1))/(pi*id^2))^n);%mm/sec regression rate

fdot=rdot*rhofuel*(pi*L*id+(Aflat*2))/1e4; %g/s ;

OF=oxvec(1)/(fdot); %oxidizer/fuel ratio

 end


%initialize vectors for inner radius and OF over time
idvec = zeros(size(tvec));
idvec2 = zeros(size(tvec));

OFvec = zeros(size(tvec));
rdotvec = zeros(size(tvec));
rdotvec2 = zeros(size(tvec));
Lvec=zeros(size(tvec));
fueldotvec=zeros(size(tvec));
thrustvec=zeros(size(tvec));
mdotvec=zeros(size(tvec));
pchvec=zeros(size(tvec));
vesvec=zeros(size(tvec));
impvec=zeros(size(tvec));
spimpvec=zeros(size(tvec));

idvec(1)=id;

rdotvec(1)=rdot;


Lvec(1)=L;
fueldotvec(1)=fdot;
mdotvec(1)=(fdot+mox)/1000;
```

```
OFvec(1)=mox/(fdot);
pchvec(1)=mdotvec(1)*cstar/astar;
vesvec(1)=sqrt((2*R*y/(y-1))*Tc*(1-(pe/pchvec(1))^((y-1)/y)));
thrustvec(1)=((fdot+mox)/1000)*vesvec(1);%+(pe-pchvec(1))*ae;
impvec(1)=thrustvec(1)*dt;
spimpvec(1)=thrustvec(1)/(9.81*mdotvec(1));

if pchvec(1) ~= pc
 astar=(mdotvec(1)*cstar)/pc;
end


pchvec(1)=mdotvec(1)*cstar/astar;
vesvec(1)=sqrt((2*R*y/(y-1))*Tc*(1-(pe/pchvec(1))^((y-1)/y)));
thrustvec(1)=((fdot+mox)/1000)*vesvec(1);%+(pe-pchvec(1))*ae;
impvec(1)=thrustvec(1)*dt;
spimpvec(1)=thrustvec(1)/(9.81*mdotvec(1));


%loop over time vector
for i = 2:length(tvec)
    %calculate the port diameter at each time point using the regression rate
    ↪  formula
    idvec(i)=idvec(i-1)+(2*rdotvec(i-1)*dt/10); %cm
    Lvec(i)=(Lvec(i-1)-(2*rdotvec(i-1)*dt)/10) ;%assume axial regression same as
    ↪  radial regression
       % idvec2(i)=idvec2(i-1)+(2*rdotvec2(i-1)*dt/10); %cm

    rdotvec(i)=a*(((4*oxvec(i))/(pi*(idvec(i)^2)))^n);%mm/s

    %calculate the fuel mass flow rate at each time point using the inner radius
    fueldotvec(i) =
    ↪  1000*(rdotvec(i)/1000)*(rhofuel)*(pi*Lvec(i)*idvec(i)+(pi*(((OD/2)^2)-((idvec(i)/2)^2)
    ↪  %g/s
    mdotvec(i)=(fueldotvec(i)+oxvec(i))/1000;
    pchvec(i)=mdotvec(i)*cstar/astar;
    vesvec(i)=sqrt((2*R*y/(y-1))*Tc*(1-(pe/pchvec(i))^((y-1)/y))); %hybrid
    ↪  propulsion book pg 166

    thrustvec(i)=mdotvec(i)*vesvec(i);%-(pe);%*efvec(i);
    impvec(i)=thrustvec(i)*dt;
    spimpvec(i)=thrustvec(i)/(9.81*mdotvec(i));
    %calculate the OF ratio at each time point using the fuel and oxidizer mass
    ↪  flow rates
    OFvec(i) = oxvec(i)/fueldotvec(i);
end
```

```
astar=(mdotvec(1)*cstar)/pc ; %throat area M^2
td=sqrt(astar/pi)*2*100; %throat diameter in cm
totimp=sum(impvec);
Aflatfinal=pi*(((OD/2)^2)-((idvec(i)/2)^2)); %final top and bottom areas

fuelused=sum(fueldotvec.*dt)/1e3; %fuel mass flux
fremaining=mfuel-fuelused;%final fuel mass in kg
 totalox=sum(oxvec)*dt/1000;
OFavg=(sum(oxvec)*dt/1000)/fuelused;


%run tank for 27°C
oxrho=723.14; %kg/m3
Vox=(totalox/oxrho)*1000; %Liters of liquid oxidizer
Vtank=Vox*1.17; %account for 17% ullage
IDtank=5.5*2.54;  %inner diameter of 6
IDrntank=4.5*2.54;
Ltank=(Vtank*1000)/(((IDtank/2)^2)*pi); %Oxidizer tank length (not including
 ↪  bulkheads)
Lruntank=(Vtank*1000)/(((IDrntank/2)^2)*pi); %Oxidizer tank length (not including
 ↪  bulkheads)

%plot everything
figure
subplot(7,1,1)
plot(tvec,idvec)
xlabel('Time (s)')
ylabel('Inner diameter (cm)')
title('Inner diameter over time')

subplot(7,1,2)
plot(tvec,OFvec)
xlabel('Time (s)')
ylabel('O/F ratio')
title('O/F ratio over time')

subplot(7,1,3)
plot(tvec,rdotvec)
xlabel('Time (s)')
ylabel('Regression rate (mm/s)')
title('Regression rate over time')

subplot(7,1,4)
plot(tvec,fueldotvec)
xlabel('Time (s)')
ylabel('Fuel Mdot (g/s)')
```

```
title('Fuel Mdot rate over time')

subplot(7,1,5)
plot(tvec,thrustvec)
xlabel('Time (s)')
ylabel('Thrust (N)')
title('Thrust over time')

subplot(7,1,6)
plot(tvec,pchvec/6894.76)
xlabel('Time (s)')
ylabel('Pressure (Psi)')
title('Chamber Pressure vs Time')

subplot(7,1,7)
plot(tvec,spimpvec)
xlabel('Time (s)')
ylabel('Specific Impulse (S)')
title('Specific Implulse vs Time')


me=sqrt((2/(y-1))*((mean(pchvec)/pe)^((y-1)/y)-1)); %average mach number

Ae=astar*(((y+1)/2)^(-1*(y+1)/(2*(y-1)))*((1+((y-1)/2)*me^2)^((y+1)/(2*(y-1))))/me);
Te=Tc*(((1+((y-1)/2)*(me^2)))^-1); %Avg exit temperature

fprintf ('\n\nMax Exhaust velocity: %.2f m/s\n', ves);
fprintf ('\n\nMean Exhaust velocity: %.2f m/s\n', mean(vesvec));
fprintf ('\n\nAverage Mach Number: %.2f \n', me);
fprintf ('\n\nAvg Exit Temperature: %.1f K \n', Te);



fprintf ('Characteristic velocity: %.2f m/s\n\n', cstar);
fprintf ('Initial Oxidizer mass flow rate: %.4f kg/s\n', mox/1000);
fprintf ('Initial fuel mass: %.2f kg\n', mfuel);
fprintf ('Initial regression rate: %.2f mm/s\n', rdot);
fprintf ('Initial fuel mass flow rate: %.2f g/s\n', fdot);
fprintf ('Initial oxidizer/fuel ratio: %.2f\n\n', OF);
fprintf ('Fuel sliver remaining: %.2f cm\n', OD-idvec(end));
%fprintf ('Fuel liner remaining: %.2f cm\n', OD-idvec2(end));
fprintf ('Fuel Grain final length: %.2f cm\n\n', Lvec(end));
%fprintf ('Final top and bottom areas: %.4f cm^2\n', Aflatfinal);

fprintf ('Throat area: %.8f m^2\n', astar);
fprintf ('Exit Area: %.8f m^2\n', Ae);
fprintf ('Exit Diameter: %.8f cm\n', 100*2*(sqrt(Ae/pi)));
```

```
fprintf ('Area Ratio: %.8f \n', Ae/astar);

fprintf ('Throat diameter required for 360psi chamber pressure: %.4f cm\n', td);
fprintf ('Mean chamber pressure: %.2f psi\n', mean(pchvec)/6894.76);
%Nozzleload=((2.5^2)*pi-astar*1550)*380 %lbf
fprintf ('Fuel used: %.4f kg\n', fuelused);
fprintf ('Oxidizer used: %.4f kg\n', totalox);
fprintf ('Final fuel mass: %.4f kg\n', fremaining);
fprintf ('Average oxidizer/fuel ratio: %.4f\n\n', OFavg);
fprintf ('Max Thrust: %.4f Newtons %.4f lbf\n',
↪  max(thrustvec),max(thrustvec)/4.448);
fprintf ('Total impulse: %.4f newton seconds\n\n', totimp);

fprintf ('6" OD Flight Oxidizer Tank Length: %.2f cm %.2f in\n', Ltank,
↪  Ltank/2.54);
fprintf ('Run Tank Length: %.2f cm %.2f in\n\n', Lruntank, Lruntank/2.54);


% Specify the file name
filename = 'hybridpressure.csv';

% Save the array to CSV
csvwrite(filename, pchvec);
```

Simulated output:

```
Max Exhaust velocity: 2328.33 m/s


Mean Exhaust velocity: 2259.57 m/s


Average Mach Number: 2.54


Avg Exit Temperature: 2273.1 K
Characteristic velocity: 1579.30 m/s

Initial Oxidizer mass flow rate: 1.5700 kg/s
Initial fuel mass: 1.90 kg
Initial regression rate: 2.22 mm/s
Initial fuel mass flow rate: 217.76 g/s
Initial oxidizer/fuel ratio: 7.21

Fuel sliver remaining: 1.28 cm
Fuel Grain final length: 34.78 cm
```

```
Throat area: 0.00107763 m^2
Exit Area: 0.00436833 m^2
Exit Diameter: 7.45783514 cm
Area Ratio: 4.05363316
Throat diameter required for 360psi chamber pressure: 3.7042 cm
Mean chamber pressure: 305.82 psi
Fuel used: 1.1712 kg
Oxidizer used: 8.9146 kg
Final fuel mass: 0.7328 kg
Average oxidizer/fuel ratio: 7.6118

Max Thrust: 4162.4909 Newtons 935.8118 lbf
Total impulse: 22849.4960 newton seconds

6" OD Flight Oxidizer Tank Length: 94.10 cm 37.05 in
Run Tank Length: 140.57 cm 55.34 in
```

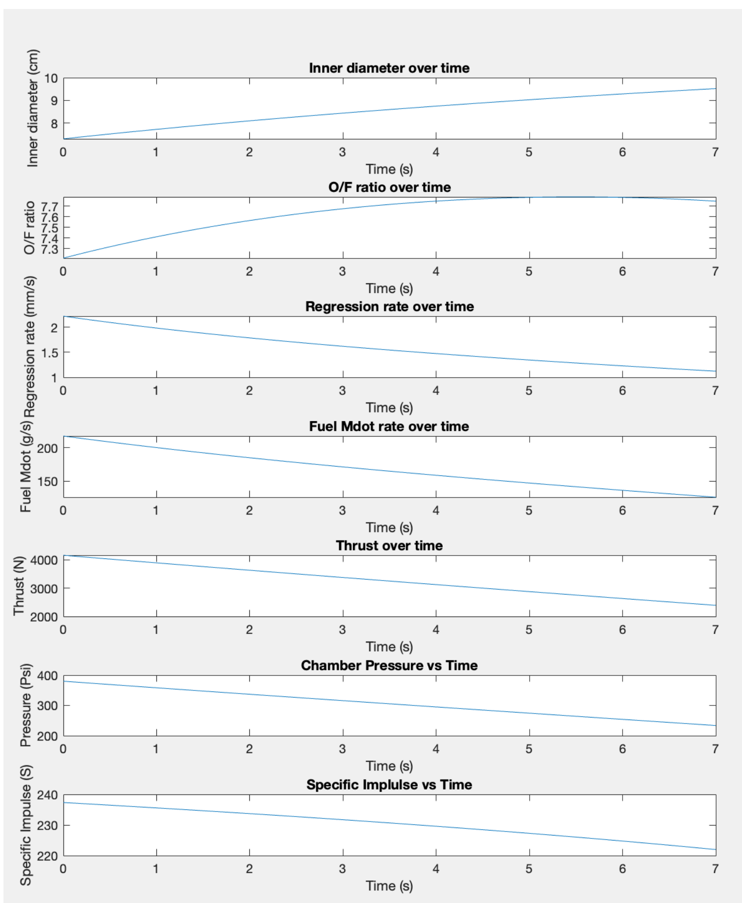

Figure 14: Graphical Output

7 2025 Launch Vehicle Considerations

- The 2025 Launch Vehicle must reach 10,000ft, and must have a thrust to weight ratio of at least 5:1. (see requirements section) The 2024 Launch Vehicle (Javelina) solid motor weighs 51.3 lbs and produces 19040 Ns and 1000lbf peak thrust. This configuration has excess impulse, and therefore around 9.6 lbs of ballast mass must be added to not exceed 10,000ft.

  Without any mass reduction optimization, the current hybrid propulsion system is estimated to weigh approximately 33 kg, or 72.75 lbs fully loaded. This is estimated based on the the following values:

  - Run tank mass- 10.7 kg
  - Oxidizer mass- 8.9 kg
  - Assembled motor mass- 11.5 kg
  - MV74 vent valve mass- 0.64 kg
  - MAV mass- 1.09 kg

  It is estimated that we will not need ballast mass for next year's rocket, so therefore the additional mass of next year's rocket is 72.75-9.6-51.3= 11.85 lbs. The propulsion system mass does not include the MAV electronics or mounting hardware, any piping, or additional run tank fittings, or pressure transducer. However, these masses should be on the order of mass that we would be able to additionally reduce from the propulsion system. The Matlab hybrid script predicts around 22849 Ns for a 7 second liquid burn. This does not include the impulse or oxidizer mass used during the gas phase burn. The ultimate values for the total impulse and thrust developed must be determined with a full duration hot fire test.

8 Conclusions

References

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