TASK HISTORY	TASK TITLE Full Scale Preliminary Design	TASK NUMBER 6	PROJECT IREC Hybrid
TASK HISTORY AUTHOR Connor Latham	TEAM LEAD Avery Moore		TASK DOCUMENTATION Preliminary Design of a full scale rocket with V2 engine
DATE 8/20/18	MILESTONE Prelim tank sizing and engine sizing done	REVIEWER'S INITIALS AM	

Purpose

To display the process of design for a full scale rocket with tankage sizing and engine sizing. This was done in order to provide an approximation of what we might need to get from Luxfer if they decide to give us a tank.

Background

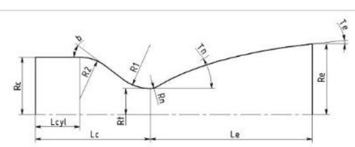
When Avery, Marty, Dalton, and Connor went to Magnesium Elektron (a child company of the group known as Luxfer), Bruce Davis mentioned possibly being able to talk to a company within Luxfer that manufactured pressurized gas cylinders. These include composite overwrapped tanks, which would be great for a flight tank since they are lightweight and incredibly strong. If a tank like this is obtained, it would save the team hundreds of dollars.

A preliminary sizing also helps to begin the design process and offers a step in the right direction for sizing various parts of the rocket. This includes general dimensions like the height and diameter of the rocket, as well as specifics like engine fuel grain diameter.

The software known as Rocket Propulsion Analysis was utilized to determine mass flow rates of the two fuels. Using this, an engine file was created. This engine file simply listed a constant thrust for a specified period of time. This file was then uploaded to Openrocket and a generic geometric rocket shape was created. A mass was also set based on previous rocket designs from within ISS. Various numbers were adjusted like engine burn time and rocket mass until a reasonable and physical representation was found that reached requirements.

Results

The engine characteristics were determined first. The engine specifications are determined from an analysis by the Rocket Propulsion Analysis software. RPA only requires you to input the thrut you want, the chamber pressure, the fuels, and the OF ratio. A screenshot is shown below of the data the program output.

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Thrust and mass flow rates
-------------------------------------------
   Chamber thrust (vac):    2.40950    kN
 Specific impulse (vac):  245.27884     s
   Chamber thrust (opt):    2.22037    kN
 Specific impulse (opt):  226.02632     s
   Total mass flow rate:    1.00172   kg/s
Oxidizer mass flow rate:    0.83477   kg/s
    Fuel mass flow rate:    0.16695   kg/s

Geometry of thrust chamber with parabolic nozzle
-------------------------------------------
    Dc =   76.46  mm       b =   30.00 deg
    R2 =   99.68  mm      R1 =   14.39  mm
    L* = 4292.17  mm
    Lc =  304.80  mm    Lcyl =  224.64  mm
    Dt =   19.19  mm
    Rn =    3.67  mm      Tn =   20.67 deg
    Le =   58.87  mm      Te =    8.00 deg
    De =   48.27  mm
 Ae/At =    6.33
 Le/Dt =    3.07
Le/c15  =  107.55 % (relative to length of cone nozzle with Te=15 deg)

  Mass =   -0.40  kg

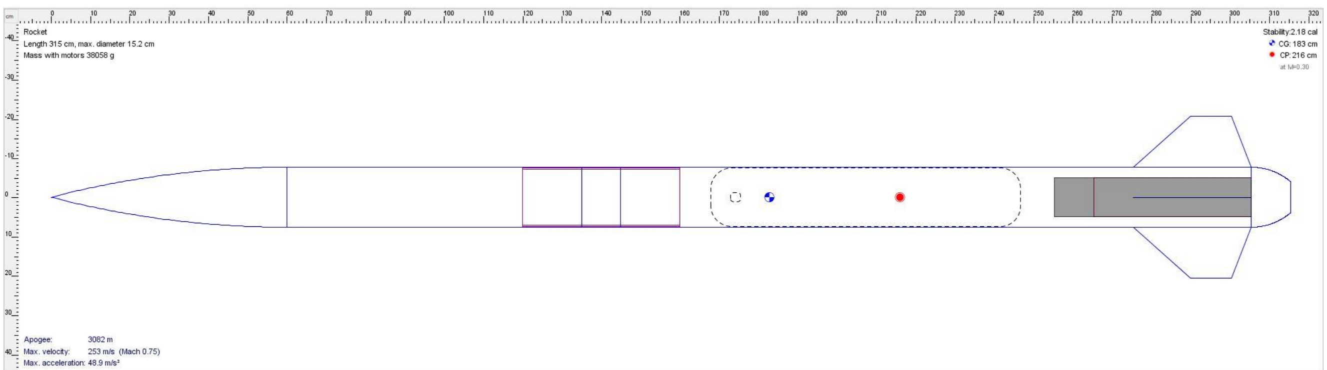
  Divergence efficiency:    0.99157
        Drag efficiency:    0.96223
     Thrust coefficient:    1.61237  (vac)
```

The data from this analysis is shown in the table below.

Parameter	**Value**
Thrust	2224 N / 500 lbf
Chamber Pressure	5.17 MPa / 750 psia
Chamber Temperature	2796 K
OF Ratio	5
Ox Flow Rate	.8347 kg/s
Mass Flow Rate	.167 kg/s

Using these characteristics an engine file was created. An engine file is a specific format for a file that is compatible with rocket simulation. It consists of some engine parameters and then the thrust over time of the rocket. After many iterations of burn times, the final one was burn time of 5.42 seconds in order to reach an apogee of 10111 feet with the following rocket characteristics.

The rocket is shown below in a still obtained from OpenRocket.



A table listing all the important parameters is shown below.

Parameter	**Value**
Length	3.15 m / 10.33 ft
Diameter	.152 m / 6 in
Mass	38.06 kg / 83.91 lbs
Max Altitude	3082 m / 10111 ft
Max Velocity	253 m/s / 565 mph / .75 Mach
Max Acceleration	48.9 m/s^2 / 4.98 G
Stability Margin	2.18 (Not to be trusted, I randomly made the rocket stable, did not know mass of different components)

The rocket would be the largest ISS has ever built. It has a very large booster tube due to the large tank size. The tank size is dictated by technical specs from the Luxfer website for a tank that is 31 in long and has a 5.9 in diameter.

The final table with all predicted data is shown below:

Parameter	**Value**
Rocket Characteristics	
Length	3.15 m / 10.33 ft
Diameter	.152 m / 6 in
Mass	38.06 kg / 83.91 lbs
Max Altitude	3082 m / 10111 ft
Max Velocity	253 m/s / 565 mph / .75 Mach
Max Acceleration	48.9 m/s^2 / 4.98 G
Stability Margin	2.18 (Not to be trusted)
Engine Characteristics	
Thrust	2224 N / 500 lbf
Chamber Pressure	5.17 MPa / 750 psia
Chamber Temperature	2796 K
OF Ratio	5
Ox Flow Rate	.8347 kg/s / 1.84 lb/s
Fuel Mass Flow Rate	.167 kg/s / .368 lb/s
Burn Time	5.42 s
Ox Mass	4.52 kg / 9.965 lbs
Fuel Mass	.905 kg / 1.995 lbs
Ox Volume	.005825 m^3 / 355 in^3
Fuel Volume	.00101 m^3 / 61.36 in^3
Ox Tank Volume	.00901 m^3 / 550 in^3
Ullage (Vapor Volume)	36% of tank volume
Fuel Grain Diameter	98 mm / 3.85 in
Fuel Grain Length	~6 in (don’t trust this, too short I think)

Lessons Learned

The most important lesson is the one that always is at the start of every design session. That is to set certain variables as constants and design from there. The design process is iterative and frustrating because of it. Come back to it later if it turns out to be too annoying.

Also, this rocket is going to be a behemoth. And a lot of it will be NOS. NOS is just not a great oxidizer. It has some really nice properties as a rocket oxidizer. But it isn’t the fact that it has one oxygen. The OF is shifted very far right, and even this design assumes a non-optimal OF of 5 and not the optimal of around 8.4.

The most important lesson is that we need to test these design results. Some seem non-physical and against what one may intuit.

Impact Statement

1. Test to find out more data for NOS/paraffin engines. This will help the team decide if we need to shift OF down to around the 3 range or not. This would help balancing the paraffin to NOS ratio.
2. Continue to iterate on this design, and don't take it at face value as the final solution to this problem. We need to use many more software tools and simulations to find out if what I (one person) did was proper and correct.
3. Ask me questions about this process. I probably didn't catalog all of it perfectly. Just ask.
4. **Don't rely on these numbers. As stated above this was based on the little amount of data that we have so far about our engine and the final engines thrust and efficiency. Base your actual structure and design on a larger amount of data. This was just an exercise to help us decide which tank to purchase.**