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Turbocharger Jet Engine Build and Engineering Analysis

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Independent Study Report: Turbocharger Gas Turbine Engine

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Contents

1 Abstract

A turbocharger jet engine combines knowledge of fluid mechanics, thermodynamics, heat transfer, machine elements and jet propulsion. This paper describes how we built a jet engine using a truck turbocharger. We review the history of jet engines, types of jet engines, how we built our jet engine and the engineering analysis behind it. We calculated specific thrust, specific fuel consumption, and the adiabatic flame temperature using concepts we learned from jet propulsion and thermodynamics classes. We machined an air handling system out of stainless steel and built a fuel system, cooling oil system and ignition system to run the engine.

2 History of Jet Engine

2.1 Early Stage and Brayton Cycle

The modern day jet engine wasn't developed until the early 20th century, and didn't see proper use until the latter end of WWII. However, the concept of jet propulsion and its precursors had been around for centuries. The first known use of jet propulsion is the aeolipile which was built around 150 BCE [1]. The aeolipile was a device that generated steam and funneled it through two nozzles. This pressurized steam was then used to spin a sphere on its axis [1]. The aeolipile was not used for any practical operations, and was just seen as a novelty [1]. Jet propulsion didn't see any practical applications until the Chinese invention of the rocket in the 7th century during the Tang Dynasty [2]. The Chinese used jet propulsion in the form of fireworks fora multitude of festivals and celebrations. These fireworks are a mainstay in Chinese culture to this day [2]. Later in the 13th century, during the Song Dynasty, the Chinese also invented gun powder which spurred the invention of modern day firearms and rocketry [2]. Soon after the Mongol invasions of China, the rocket technology spread throughout the world [2].

The real use of jet engines in airplanes would come in the form of hybrid engines. Prior to World War II,airplanes were propelled by propellers that were attached to a simple gas turbine engine or a rotary engine. These engines were not too different than the engines that you would see in a car. They would translate the linear movement of the pistons into a rotational movement of the propellers. These propellers would drive air backwards and over the wings, which would then generate the lift. However, at the turn of the 20th century, engineers realized that the efficiency of the propellers takes a severe dip when as they approach higher speeds. When the airplanes started to approach the speed of sound, the engine efficiency would drop. This inevitably led to the onset of research into the jet engine [3].

2.2 Present Technology

Engines developed during WWII were plagued with an extremely low fuel efficiency, which severely hindered their ability to contribute to the war efforts [4]. Fuel efficiency of the turbojet engine became a priority to the world and efficiency. An American company called Pratt and Whitney used a German engineering design to increase the fuel efficiency in 1948. This fuel efficiency issue was mitigated by introducing another compressor to the design. The new compressor was added directly behind the first, and they were powered by their own turbines [5]. The newly included compressor provided greater compression for better performance. This engine was named the "J-57" and it helped usher in the age of commercial airlines, allowing the transport of people across the country and across seas in record time. It also entered service with the U.S. Air Force in 1953. The J-57 hada twin-spool axial flow configuration that had an all-axial, 9-stage low pressure compressor, and a 7-stage high pressure compressor, with annular, 8 flame tube combustors, and an all-axial, single stage high pressure turbine, and a 2-stage low pressure turbine. This engine could produce a maximum thrust of 12,030 lbf, a thrust to weight ratio of

3.44, with a specific fuel consumption of 0.909. The J-57 found military applications in the American F-100 Super Sabre, which was the first production aircraft to exceed the speed of sound in level flight, the Conair F-102 Delta Dart, and the Navy's Chance Vought F8U-1, which set the first official speed record in excess of 1,000 miles per hour [6]. The commercial version of the J-57, the JT3, was used in the Pan American World Airways Boeing 707, the Boeing 727, and the Douglas DC-8. In October 1958, the Pan American World Airways Boeing 707 introduced American jet travel by making its inaugural flight from New York to Paris using 4 JT3s rated at 13,000 lbf of thrust each [6]. The next major innovation in turbojet engines came in 1964 when General Electric created the world's first high bypass turbofan engine, the GE TF39 [7]. This engine incorporated bypass which allows most of the air that it intakes to bypass the turbojet engine entirely. Normally all of the turbine's power output is consumed by the compressor and all of the air is taken through the system and thrust is produced. In a bypass system, the turbine produces far more power than is necessary to operate the compressor and the leftover energy is then used to power a ducted fan which accelerates air to the back of engine and out the nozzle without ever having been taken through the engine itself. This design drastically increases the efficiency and power output of the engine as most of the thrust comes from the bypassed air. The GE TF39 contained an axial, 2 stage fan, a 16 stage high pressure compressor, annular combustors, as well as an axial, 2 stage high pressure turbine, and a 6 stage low pressure turbine. It was rated for 43,300 lbf of thrust, a thrust to weight ratio of 5.4 and a specific fuel consumption of 0.313 lb/lbf-hr [7]. The newest innovations in jet engine aviation have been focused on reducing the environmental impact and the inclusion of new modern day materials to reduce weight and increase performance. For example, the new General Electric GEnx and LEAP engines have been able to reduce the carbon dioxide emissions by 15%, increase the efficiency 15%, and decrease the amount of noise generated by 30%. They achieved this by using carbon fiber for the fan blades which allowed them to cut the number of blades by half, and using new age lightweight ceramic composites for the turbine blades which allowed the Boeing 787 Dreamliner to set new distance and speed records on a round-the world flight recently [8]. Another innovative technology recently introduced has been Adaptive Versatile Engine Technology (ADVENT) which allows current jet engines to switch between high performance modes and high efficiency modes, which was not possible before [8]. Through regulation of the air flow through a second bypass duct via adjustable fans, ADVENT engines can prioritize thrust or efficiency depending on the situation (e.g.: takeoff vs. cruising). This, in conjunction with heat new heat resistant materials and additive manufacturing techniques, allows some aircraft to decrease fuel consumption by 25%, increase operating range by 30%, and increase thrust by 10% [8].

3 Types of Jet Engines

Brayton Cycle:

Jet engines are able to produce thrust by operating on the open Brayton cycle. Jet engines operate on the open Brayton cycle because it is able to handle large volumes of air to produce large amounts of thrust. Figure 2 below shows the open Brayton cycle process:

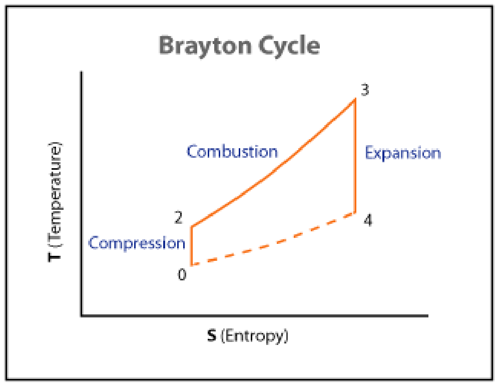

Figure 2: Brayton Cycle Process [9]

The Brayton cycle starts at point 0 when air flows into the jet engine. The air then flows through the compressor in an isentropic compression process to state point 2. At state point 2 the gas is combusted. The combustion occurs at a constant pressure and the gas reaches state point 3. The gas then flows through the turbine state 4 which is an isentropic expansion. The process from state 3 to 4 is what generates the thrust of the jet engine. As shown in Fig. 2, the expansion process is much larger than the compression process. As the pressure and temperature of state 3 increase compared to state 2, more thrust can be generated. The line from state 4 to 0 is dashed because the cycle is open. After state 4, the products flow out of the nozzle and into the surroundings. The process restarts when air flows in at state point 0. The design of jet engines allows the open Brayton cycle to operate continuously, processing large amounts of air and generating significant thrust [10].

3.1 Turbojet

Though there are multiple types of jet engines, the turbo jet is the core of most jet engines. Figure 3 shows a diagram of the turbo jet engine.

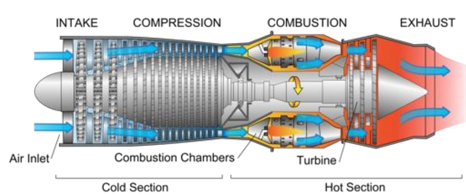

Figure 3: Turbo Jet Engine Diagram [11]

Air flows into the turbo jet engine and then through the compressor, which from Fig. 2, is the process from state 0 to 2. In the compressor, the large volume of air from the intake is forced into a smaller volume, increasing the pressure. The high pressure air then enters the combustion chamber where fuel is injected and ignited. The products then flow through a turbine and out the exhaust. When flowing through the turbine, some of the thrust is used to turn the compressor. This reduces the overall thrust, but helps make the cycle more efficient. In order to only extract a small amount of thrust, the turbine section is smaller. The rest of the energy is converted into the kinetic energy through a nozzle which then produces thrust [3].

3.2 Turboprop

The next type of jet engine is the turbo prop engine. This engine has the core of a turbo jet, but the turbine and compressor also connect to a propeller in front of the engine. Figure 4 shows the turbo prop engine.

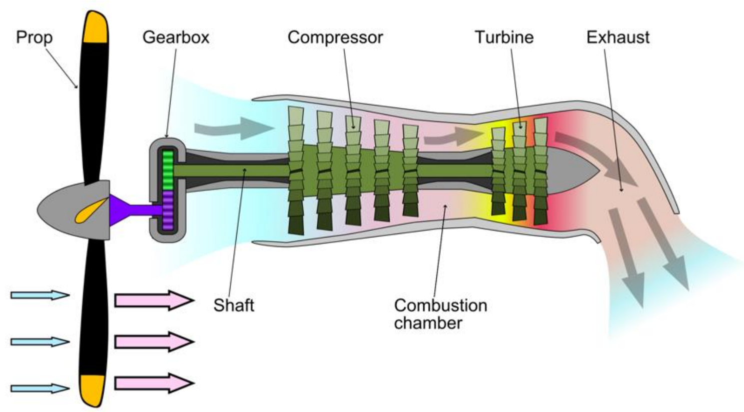

Figure 4: Turbo Prop Engine Diagram [12]

The thrust from the turbo jet engine is used to drive the turbine which powers both the compressor and propeller. As a result, there is very little thrust left over to power the turbo jet itself. The turning propeller generates almost all of the thrust and also helps to force extra air into the core of the engine. The turbo prop is most efficient at low speeds because the propeller becomes less efficient at higher speeds [3].

3.3 Ram Jet

Though it operates on the Brayton cycle, the ram jet engine does not function like any of the other jet engines. Figure 5 shows a diagram of a ram jet engine.

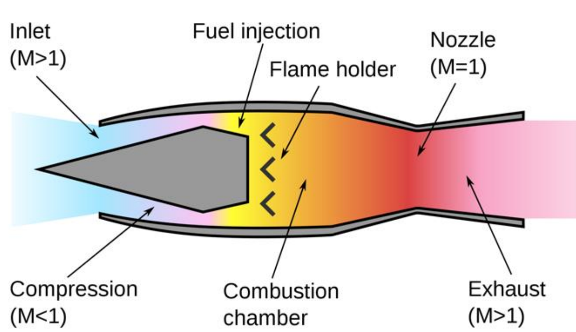

Figure 5: Ram Jet Engine Diagram [13]

From Fig. 5, the ram jet does not look like any other jet engine because it has no moving parts. At very high speeds, the amount of air flowing through a shrinking volume in the inlet increases the pressure enough for combustion to take place. This removes the need for a compressor and turbine, which allows all of the thrust to be used to propel the aircraft. However, as a result of having no compressor, the ram jet is only able to operate at very high speeds [3].

3.4 Turbofan

The type of jet engine we chose to analyze is the turbofan jet engine. Figure 6 below depicts the turbo fan engine.

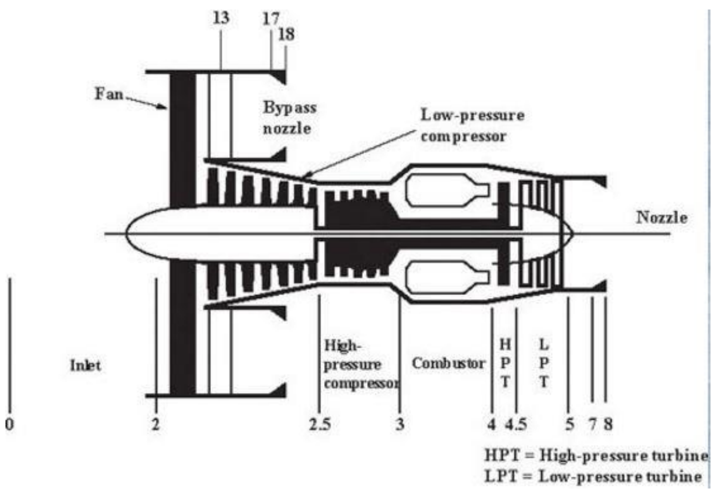

Figure 6: Turbo Fan engine diagram [14]

From Fig. 6, the core of the turbo fan is the turbo jet engine. However, the turbo fan has a large fan in front of the compressor. This turbo fan is designed to draw in a large volume of air, and most of the air passes around the turbo jet core through a bypass. The fan helps force a larger amount of air into the turbo jet, which increases thrust, and the bypass is what makes the turbo fan efficient. The fanis used to increase the speed of the air through the bypass. This only increases the overall speed by a small amount, but the volume of air is very large. The small increase in speed of the large volume of air produces a large thrust, without sending more air through combustion process [3].

4 Turbocharger Jet Engine Design

4.1 Overview

For a turbocharger jet engine to run, a fuel system, cooling oil system, ignition system, and air handling system were needed. Because compressors and turbines are extremely hard to build, we obtained a truck turbocharger through a generous donation from Cummins. Combustion chamber and air pathways components were either bought or fabricated using stainless steel for its thermal and structural durability. The fabrication process dealt with milling and turning the parts in a lathe or a mill. Several design challenges are handled to build this engine. This section will provide the procedure and parts required to build this engine and the way we dealt with the engineering challenges in the process.

4.2 Air Handling System

As mentioned above, air handling system components were built with stainless steel. The most important part of the air handling system is the combustion chamber which consists of the inner liner and outer casing. The main component that needs careful design is the inner liner as this piece contains the combustion process and affects the performance of combustion chamber the most. Since the combustion stage is divided into three zones (Primary zone, Secondary zone, Tertiary zone), there are 3 different sizes for holes placed along the liner [15]. ThePrimary zone is where air enters the inner liner to mix with fuel and ignite. Air enters the secondary and tertiary zones to cool the hot temperatures produced in the primary zone. The sizes of the combustion liner holes in each zone were suggested in a book by Schreckling [15].

The sizes of the holes are calculated as follows:

Sizing Holes for Primary Zone

The total area of holes in Primary zone is suggested to be 30% of compressor inlet area.

Inducer area = πR^2 ,where $R_{Inducer}$ = 32 mm

= 3318 mm^2

Total area of holes in Primary zone = Inducer area * 0.3 = 995 mm^2

There are 24 holes in Primary zone ⇒ 995mm^2/24= 41.5 mm^2

Area = πR^2 ⇒ $R_{Primary\ hole}$=3.63mm = 0.1428 inches

Sizing Holes for Secondary Zone

Total area of holes in Secondary zone has to be 20% of compressor inlet area.

Total area of holes in Secondary zone = Inducer area * 0.2 = 663 mm^2

There are 8 holes in Secondary zone ⇒ 663mm^2/8= 82.9 mm^2

Area = πR^2 ⇒ $R_{Secondary\ hole}$=5.13mm = 0.2019 inches

Sizing Hole for Tertiary Zone

Total area of holes in Tertiary zone has to be 50% of compressor inlet area

Total area of holes in Tertiary zone = Inducer area * 0.5 = 1659 mm^2

There are 8 holes in Tertiary zone ⇒ 1659mm^2/8= 207 mm^2

Area = πR^2 ⇒ $R_{Tertiary\ hole}$=8.12mm = 0.319 inches

Usually, the exact locations of the holes is decided experimentally. When the engine is started and combusted, hot spots leave a marks on the surface of inner liner. Additional holes can be drilled on the marked spots.

The entire length of combustion chamber is suggested to be 6*$D_{compressor}$ [15]. Therefore, the entire length is calculated to be 400 mm.

The diameter of inner liner is suggested to be 2*$D_{compressor}$ [15]. Therefore, the diameter of the inner liner is calculated to be 130 mm. It is suggested that the gap between the casing and inner liner is $D_{inducer}$/2. Therefore, the gap between the inner liner and casing wis calculated to be 32 mm. Table 1 shows the dimensions calculated.

Table 1: Dimensions of inner liner and outer liner and details of air holes on inner liner

Component	Quantity	Radius
Primary Holes	24	0.1428 inches
Secondary Holes	8	0.2019 inches
Tertiary Holes	8	0.319 inches
	Length	Diameter
Inner Liner	380mm	4 inches
Outer Casing	400mm	6inches

Apart from the inner liner, other parts consist of outer liner, 90° degree tube connections, and flanges. 90° degree tubes were purchased from McMaster-Carr. The overall air handling system is shown in Fig. 7.

Figure 7: Overall Air Handling System

Figure 8 shows the inlet into the combustion chamber from the compressor and combustor outlet which is connected to the 90 degree connector tube. To make this piece, we drilled a hole on the side and welded a 2 inch tube to the hole. The hole is intentionally placed off-center to create swirl which aids the mixing process in the inner liner. Another tapered piece is welded to gradually reduce the diameter of the air pathway. A connection flange is welded at the outlet of combustion chamber for connection with the 90 degree connector tube.

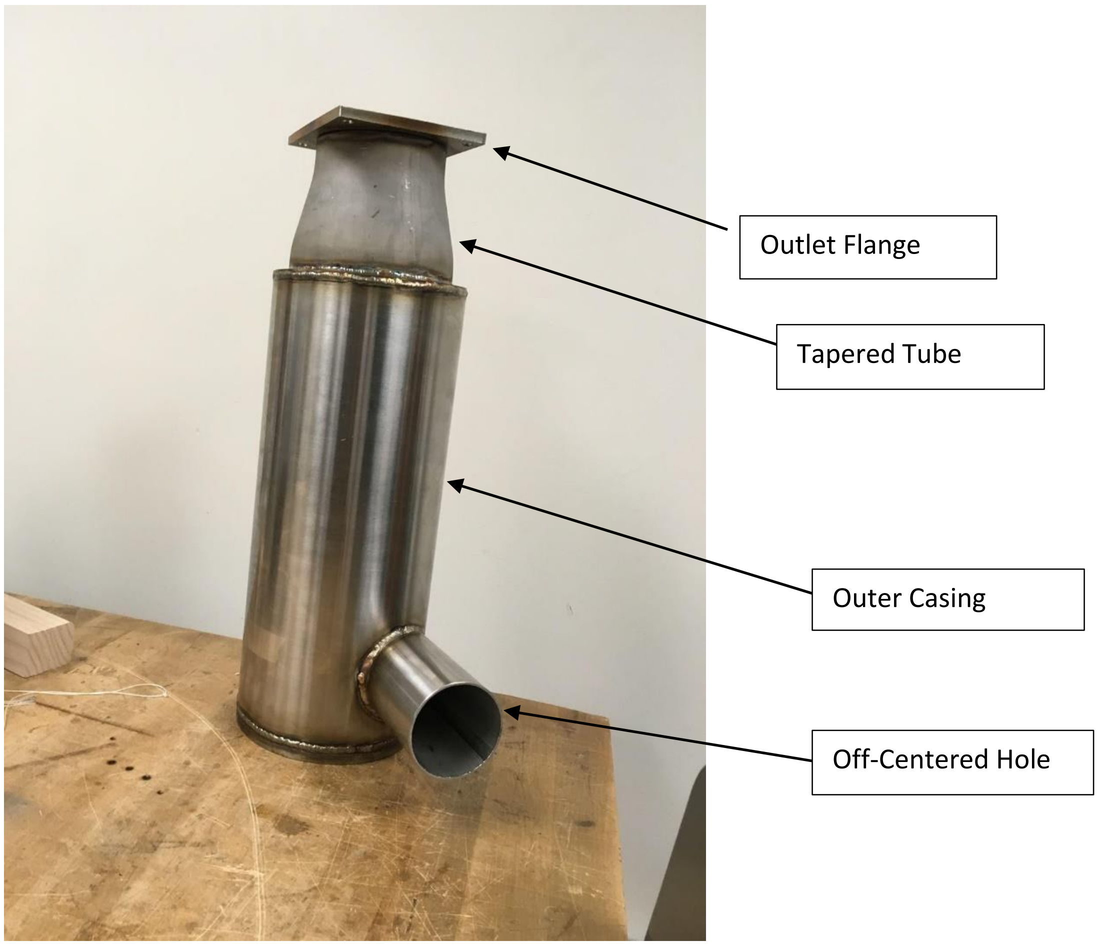

Figure 8: Combustion Chamber Outer Liner

Figure 9 is the inner liner where the combustion process occurs. The hole sizes are calculated above. A stainless tube with 4 inches in diameter and 380 mm in length was found at a scrapyard. It was then drilled with a mill. Drilling or milling a stainless steel is a difficult process and appropriate rotational speed of cutting blade has to be used.

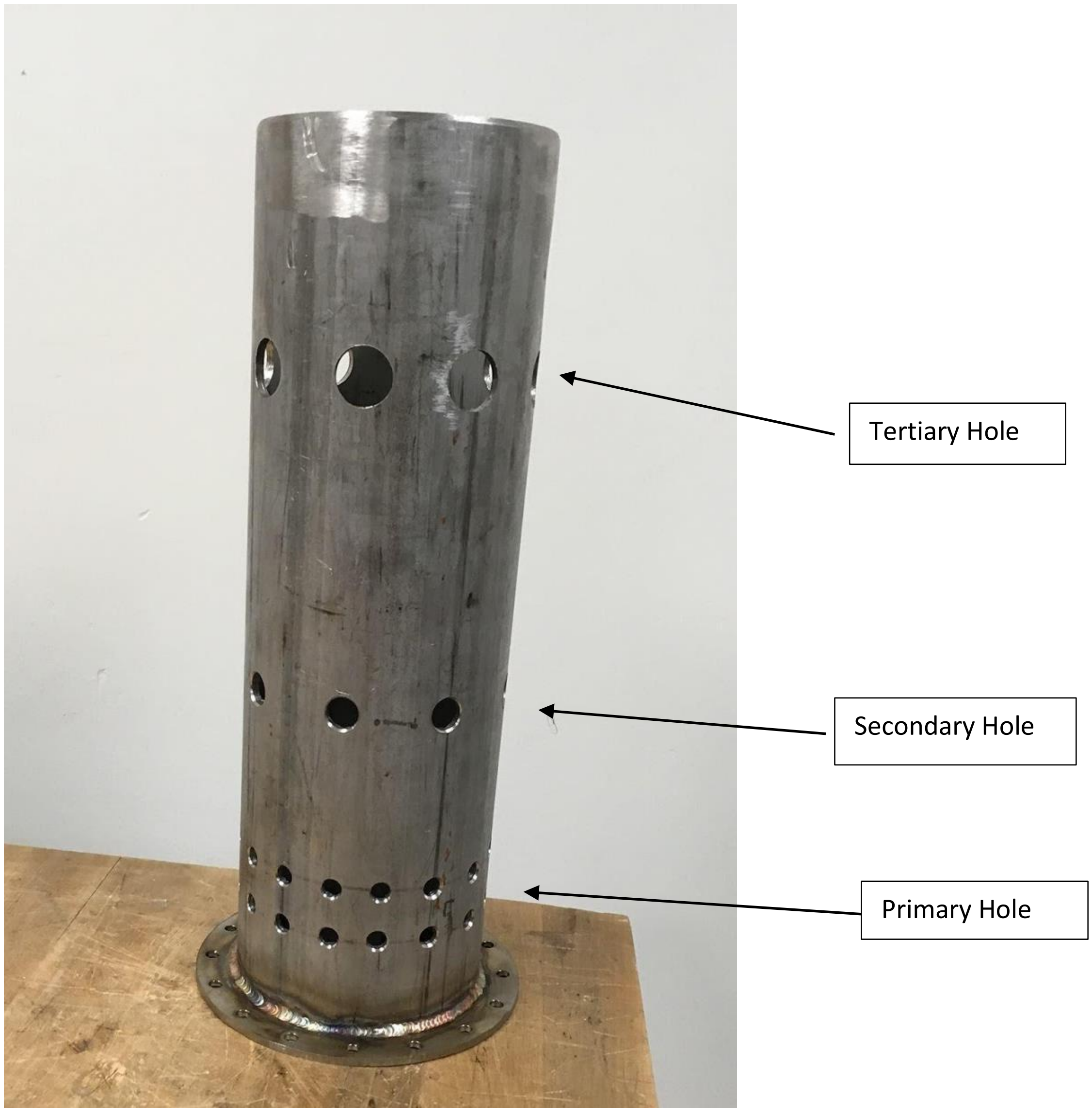

Figure 9: Combustion Chamber Inner Liner

Figure 10 shows the turbocharger compressor and turbine. Figure 11 shows the compressor outlet and thermally resistant rubber coupling that connects the 90 degree tube and the compressor outlet. The 90 degree tube and compressor cannot be bolted or welded because ofmaterial reasons. Figure 12 shows the 90 degree tube that connects the compressor outlet and the combustion chamber. It is welded with the inlet of the combustion chamber.

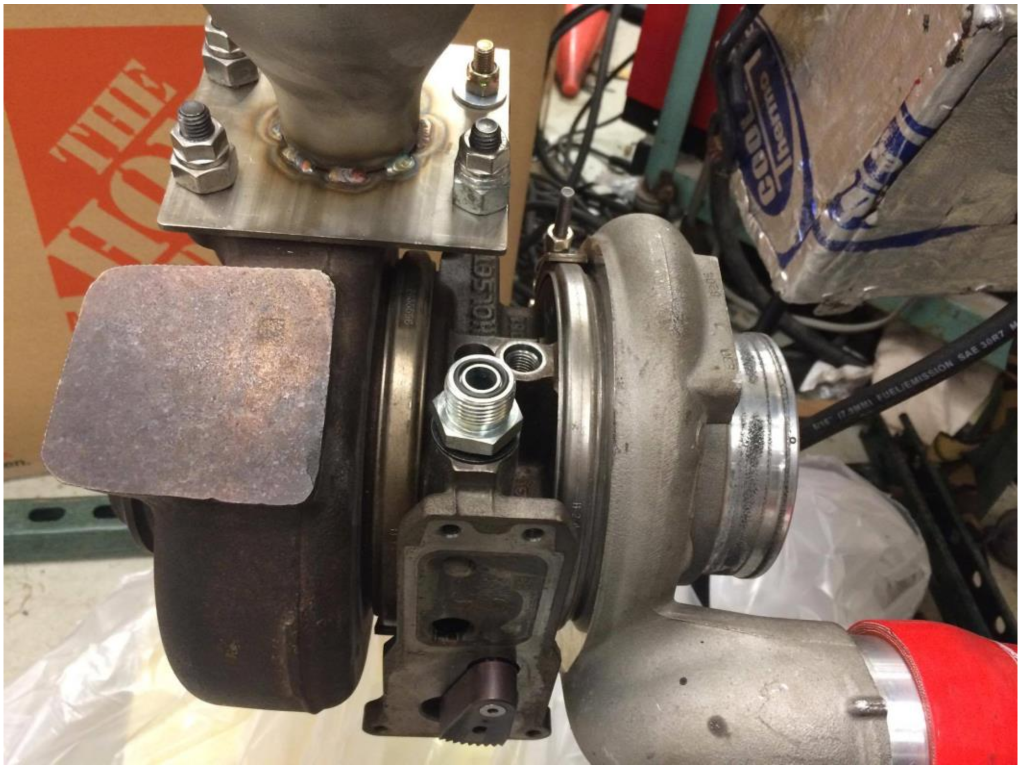

Figure 10: Turbocharger Encasing Compressor and Turbine

Figure 11: Connection between Turbocharger and Combustion Chamber Inlet

Figure 12: 90 degree Inlet into Combustion Chamber

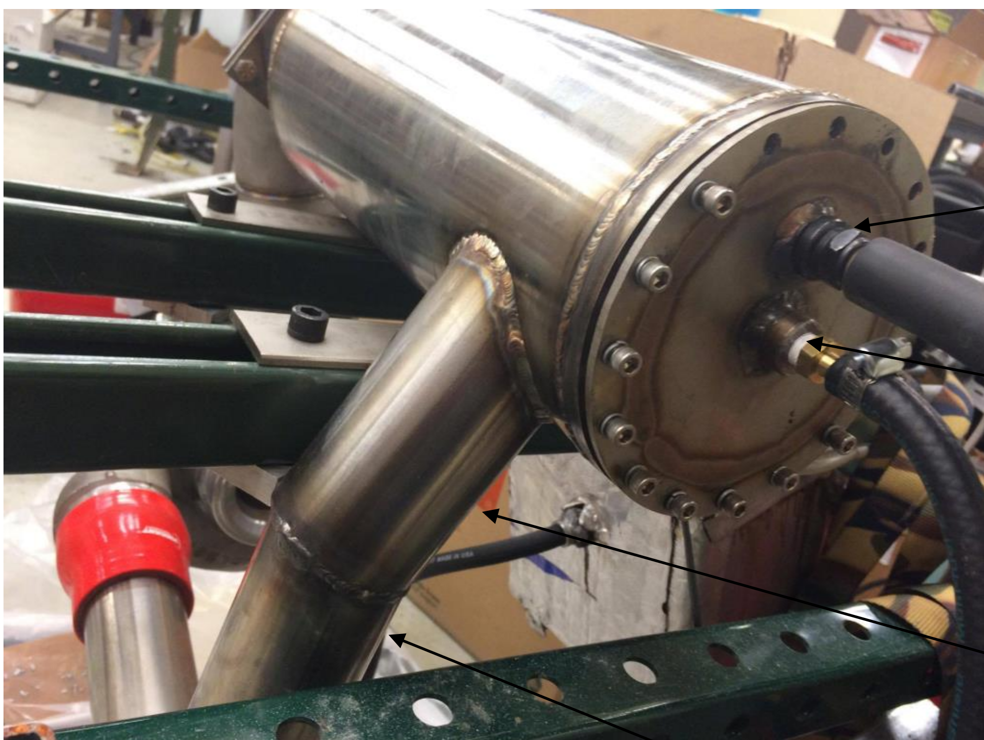

Figure 13: Spark Plug and Fuel Lines

The combustion chamber is mounted to the frame with flanges that are welded to the outside surface of the combustion outer casing, which is shown in Figure 14. Figure 15 shows another 90 degree tube that connects the combustion chamber outlet and turbine. One connection flange is welded at the combustion chamber outlet side of 90 degree tube. Also, another tapered tube is welded at the end of 90 degree tube to reduce the diameter of tube to be able to match the inlet size of turbine. One more connection flange is welded to the tapered piece to fasten bolts between the turbine and tapered piece.

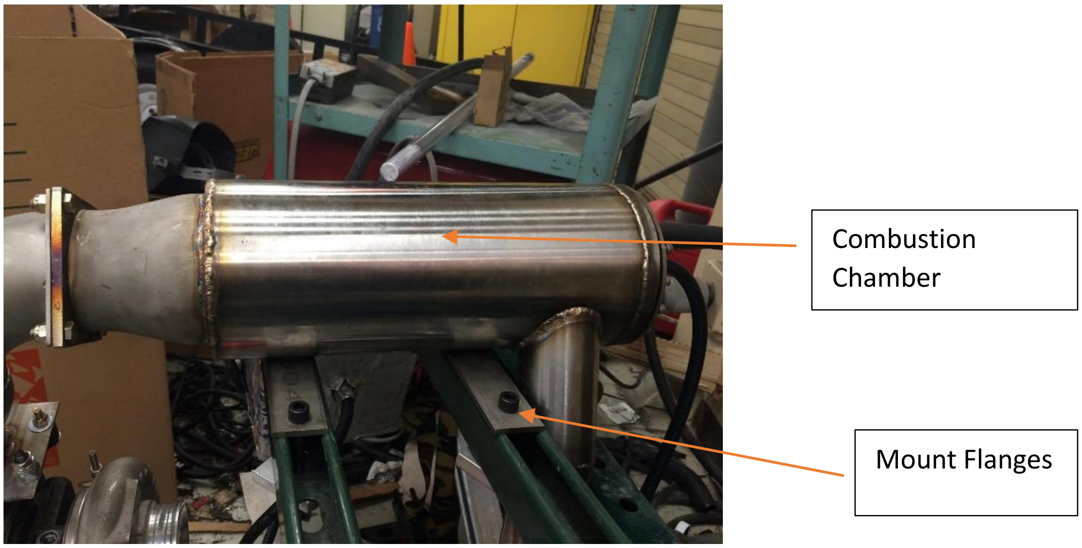

Figure 14: Combustion Chamber Mount to Case

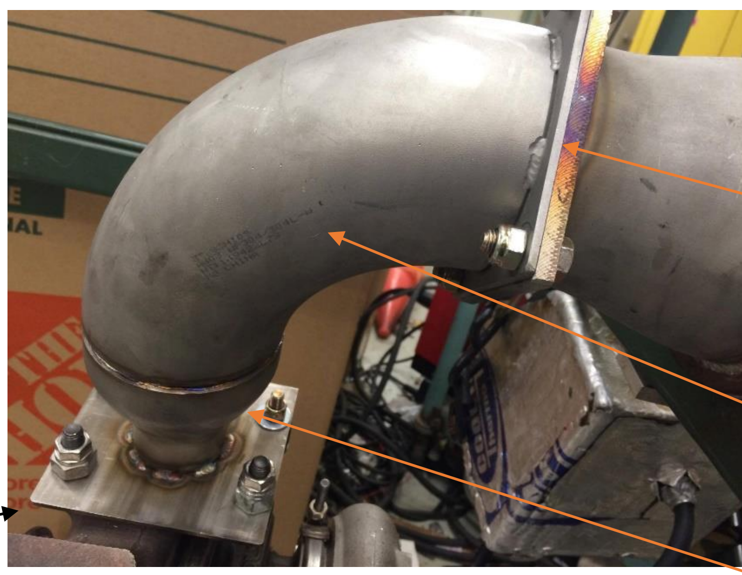

Figure 15: Combustion Chamber Exit 90 degree Tube

4.3 Design Safety

Because jet engines are inherently dangerous, safety was one of our major concerns. The biggest concern was the temperature of the components. From our thermodynamic analysis, we found that the peak working fluid temperature was 1300 K. Downstream of the compressor outlet, we used exclusively stainless steel, which has a max operational temperature of 1400 K. Additionally, the peak temperature is expected near the centerline of the combustion chamber. This margin gives us a small factor of safety because we expect the actual temperatures to be much less than 1300 K because this temperature was obtained for full throttle operation.

The second major safety concern was operator safety while operating the engine. To address this issue, we made the electronics remote from the rest of the engine. The box is attached to the pumps and the spark plug transformer by a small wiring harness that can be extended to any length. This allows for safe and remote operation. In the event of a failure of some kind, the operator can shut down the fuel and oil pumps to prevent any additional damage.

The final safety device are blast shields. These will be two plates of steel that will be located on either side of the turbo charger assembly. In the event of a catastrophic failure of any components in the turbocharger, these shields will deflect shrapnel away from the operators. We do not expect this type of failure to occur, but this will offer assurance in case the worst happens.

4.4 Electrical/Ignition System

In order to supply the fluid pumps with electricity and to create spark inside the combustion chamber, we designed an electrical system which is shown in Fig. 16. The system consists of two main circuits. The first is a simple switch circuit used to control the fuel and oil pumps. We used 2 30 amp solid state relays (RY1, RY2) to actuate the pumps. Each relay was switched on by a small toggle switch (S1, S2) and mounted to a remote project box. This allowed us to remotely activate or deactivate the pumps. An LED is also wired in parallel with the relays off of the toggle switch to indicate that the pumps are on or off.

The second main circuit is a driver for the spark plug. The circuit uses a 555 timer (U1) as a simple oscillator. This circuit then actuates transistor Q1, which shorts the main spark plug transformer (T1) to ground. This sudden short generates a large voltage across the primary of the transformer. The transformer increases this to a few thousand volts, and produces a spark across the spark gap (V2). The frequency of the driver circuit can be adjusted by changing the resistance setting on the trim pot (R4). The entire circuit is controlled by a single momentary push-button switch mounted to the same panel as the toggle switches as the two pumps. To operate the system, press and hold the push button switch. During which, the circuit will produce sparks at the frequency set by the trim pot.

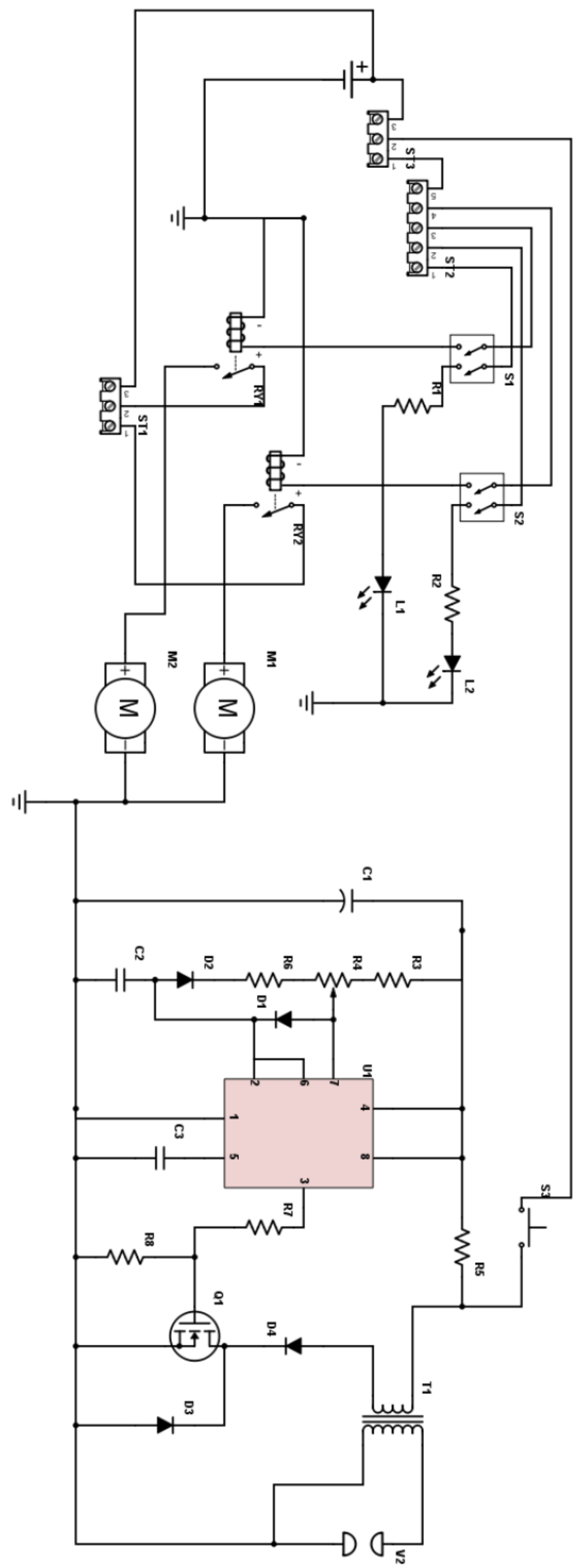

Figure 16: Electric Circuit Diagram

4.5 Fuel Delivery System

The fuel is supplied by a Shurflo 1.5 gpm furnace pump. As discussed earlier, the pump was supplied with electrical current by the main power circuit. To regulate fuel flow, we used a regulator with a return line. The regulator also has vacuum referencing capability. That is, it can adjust fuel pressure based on an air pressure reference so that the fuel head remains the same regardless of chamber pressure. However, because we expected a high chamber pressure, we chose not to use this feature because the regulator was not rated for positive pressure. The regulator is also adjustable via a small set screw to allow for fuel pressure adjustment. This is a critically important feature because the only way to throttle the engine is by varying the fuel flow. The fuel pressure is monitored by a pressure gauge attached to the regulator.

Propane will be used to start the engine because it burns much easier than kerosene. However, kerosene will be used once the system is started because kerosene enables more stable combustion characteristics.

The fuel was stored in a 1 gallon tank fabricated from welded 0.040" aluminum sheet metal. The tank had two 3/8 NPT bungs welded in for the feed and return lines. The feed line was 3/8" ID low pressure line. The line out of the pump to the fuel regulator was also 3/8" ID, but the supply and return lines out of the regulator were reduced to ¼" ID high pressure fuel line. All lines were connected using NPT-hose barb fittings. The hoses were clamped to the barbs using standard hose clamps. At the engine, fuel flow is fed into the back of the combustion chamber near the air inlet. The fuel is forced through a small nozzle to atomize it as much as possible.

4.6 Oiling System

The purpose of oiling system is to cool the rotating shaft between the turbine and compressor. Without this system, bearings will melt and engine will break down. The oiling system design is quite similar to that of the fuel system. Like the fuel, the oil is regulated via a pressure regulator with a return line. The only major difference is that the oil is fed into the turbo directly. Our oil tank is large with respect to the oil channel and oil line in the system. We therefore decided that additional oil cooling would not be necessary. Also, the oil system is closed unlike the fuel tank. After the oil flows through the turbo charger, it drains out of the bottom at atmospheric pressure into the tank below. The oil is required to drain at atmospheric pressure to make sure it does not push through the shaft seals inside the turbo. The upstream oil pressure must also be regulated carefully to prevent the same problem from occurring.

4.7 Startup Procedure

A guideline to start up the engine is presented below:

1) Start oil pump and confirm 45-psig-system pressure.

2) Start kerosene pump and confirm 100-psig-system pressure.

3) Place blower into inlet and allow turbine to accelerate (about 1 minute).

4) Open combustor drain to purge any residual fuel in combustion chamber (usually empty).

5) Turn igniter on. Allow engine to accelerate until no further acceleration can be achieved with the leaf blower.

6) Open propane valve to pre-ignite cold engine.

7) Turn fuel idle valve on (25 pounds nozzle pressure). Confirm fuel ignition.

8) Close propane valve.

9) Allow engine to accelerate until it tries to pull the blower into the engine.

10) Remove blower from inlet. Turn igniter off.

11) Allow engine to accelerate to idle speed (approximately 25,000 rpm).

12) Open throttle to 80-100 psig for full power.

13) The engine can be accelerated, reduced, or idled with the fuel control.

5 How Jet Engine Works

5.1 How Turbocharger Jet Engine Works

There may be some minor differences between jet engine designs. For example, some engines have one turbine which provides power for the compressor while some engines have multiple stages of turbines and compressors. Also, like all other engines, the turbojet engine is a machine that converts chemical energy into kinetic energy to produce thrust using a Brayton cycle. For this case, however, a radial compressor and a radial turbine are used instead of an axial compressor and an axial turbine because an axial compressor is not only very hard to build but also hard to buy. It was out of the scope of this project to build a compressor and a turbine and these two components were donated from a Cummins.

5.2 Components and Component Performance

A turbocharger jet engine consists of a compressor, combustor and turbine. There must be a nozzle to produce a thrust. But for our purposes, we did not include a nozzle. Refer to Fig. 6 for numbering through each component. Measuring how the flow properties and velocity changes at each component is important in calculating the overall engine performance. Measuring the velocity, pressure and temperature change will enable us to calculate the exit velocity, fuel flow rate, and various flow properties, such as specific heat, at the given stage. These parameters will be used to calculate the specific fuel consumption (fuel efficiency), specific thrust, propulsive efficiency, etc. Before presenting an equation, we need establish a notion of the total condition. The total condition is measured when the given flow slows down isentropically to stagnation (zero velocity). Total conditions of the freestream is simply the temperature and pressure of the atmosphere because, away from the engine inlet, air is relatively stationary. To calculate for the flow property change, we introduce a few variables and equations. π_a is called a ratio of total pressure across a component and the corresponding subscript indicates the component: c is compressor, b is burner and, t is turbine. There is also a ratio relating the the total temperature,τ_a, of the outlet to inlet [3]. Equation 1 and 2 show these two concepts.

$$\pi_a = \frac{total\ pressure\ leaving\ component\ a}{total\ pressure\ entering\ compoenent\ a} \quad (1)$$

$$\tau_a = \frac{total\ temperature\ leaving\ component\ a}{total\ temperature\ entering\ component\ a} \quad (2)$$

Losses in the inlet are due to the wall friction. Wall friction lowers the total pressure so that π_a becomes less than 1. However, the frictional loss is small resulting in π_{inlet} of 0.99 [3]. We can follow a MIL-E5007D correlation to determine π_{inlet} in Eq. 3. As for the total temperature loss, the inlet can be assumed to be adiabatic because the temperature difference between the inlet air and freestream air is negligible [3].

$$\pi_{inlet} = \begin{cases} 1 & M_0 \leq 1 \\ 1 - 0.075(M_0 - 1)^{1.35} & 1 < M_0 < 5 \\ \dfrac{800}{M_0^4 + 935} & 5 < M_0 \end{cases} \qquad (3)$$

The total pressure ratio, π_c, is a very important design parameter that can be changed to affect specific thrust and specific fuel consumption. As the total pressure ratio increases, specific thrust decreases while the specific fuel consumption increases. Therefore, we have to find a balance between the thrust and fuel consumption to determine the total pressure ratio. Although compressors can also be assumed to be adiabatic with a relatively high degree of approximation, real engines heat the fluid a considerable amount simply by compressing the flow and it results in more accurate calculation if isentropic efficiency is included. Also, in a real engine, rather than constant total pressure ratio across the process of compression, pressure rises in a continuous curve throughout the compressor. The varying pressure results in varying isentropic efficiency which makes the determination of isentropic efficiency difficult. To solve this problem, a differential notion is used. The entire compression process is divided into an infinite number of small steps so that isentropic efficiency stays constant for each division. This is a notion of polytropic efficiency. Inclusion of polytropic efficiency, e, in the calculation of overall isentropic efficiency accounts for the varying isentropic efficiency [16]. Efficiency of turbomachinery components can be calculated as follows in Eq. 4 and 5 [3]. γ is the specific heat ratio and h is the enthalpy.

$$\eta_c = \frac{\pi_c^{\frac{\gamma-1}{\gamma}} - 1}{\tau_c - 1} = \frac{h_{t3i} - h_{t2}}{h_{t3} - h_{t2}} = \frac{ideal\ work\ of\ compression\ for\ given\ \pi_c}{actual\ work\ of\ compression\ for\ given\ \pi_c} \qquad (4)$$

$$\text{where } \tau_c = \pi_c^{\frac{\gamma-1}{\gamma e_c}}$$

The turbine also goes through a similar analysis as compressor.

$$\eta_t = \frac{1 - \tau_t}{1 - \tau_t^{\frac{1}{e_t}}} = \frac{h_{t4} - h_{t5}}{h_{t4} - h_{ti5}} = \frac{actual\ work\ of\ turbine\ for\ given\ \pi_t}{ideal\ work\ of\ turbine\ for\ given\ \pi_t} \qquad (5)$$

$$where\ \tau_t = \pi_t^{\frac{e_t(\gamma_t - 1)}{\gamma_t}}$$

However, the total pressure ratio for the turbine can also be easily found by performing a power balance between the turbine and compressor with a mechanical efficiency. Equation 6 shows the power balance between the turbine and compressor. η_m is the mechanical efficiency of a shaft.

$$\dot{m}_c C_{pc}(T_{t3} - T_{t2}) = \eta_m \dot{m}_4 C_{pt}(T_{t4} - T_{t5}) \qquad (6)$$

This equation can be divided by $\dot{m}_c C_{pc} T_{t2}$ to obtain a simple equation 7 for total temperature ratio through the turbine [3]. f is the fuel to air ratio.

$$\tau_t = 1 - \frac{1}{\eta_m (1+f)\tau_\lambda}[\tau_c - 1] \quad where\ \tau_\lambda = \frac{T_{t4}}{T_0} \quad (7)$$

For the burner, the fuel to air ratio will be simply specified to calculate the burner efficiency. An ideal combustion chamber would have a total pressure ratio of 1. However, because of friction in the wall, inefficient mixing and dissociation, the total pressure will drop through the combustion chamber. Application of the first law of thermodynamics to the burner produces Eq. 8 and 9 [3]. f is a fuel to air ratio. For the exhaust temperature from combustion chamber, an adiabatic flame temperature analysis will be done. We will use a lower heating value of kerosene for the adiabatic flame temperature calculation. The adiabatic flame temperature is when there is no work or heat transfer throughout the process of combustion. For our purposes, an equivalence ratio of 2 will be used. h_{pr} is the heating value of the kerosene.

$$\eta_b = \frac{\left((1+f)h_{t4} - h_{t3}\right)}{f h_{pr}} \quad (8)$$

$$\eta_b \dot{m}_f h_{PR} = \dot{m}_4 C_{Pt} T_{t4} - \dot{m}_0 C_{Pc} T_{t3} \quad (9)$$

4.3 Engine Performance

We need to consider which parameters to measure to evaluate the performance of an engine. As mentioned before, thrust and fuel are two design goals that engineers focus on for very obvious reasons. Thrust is measured by a parameter called specific thrust and fuel consumption is measured by specific fuel consumption. Because a turbofan is simply a turbojet engine with an additional fan, equations for a turbofan will be presented. Later in the calculation section, the contribution of fan will be deducted from calculation to show the difference in performance between a turbofan and a turbojet.

First, thrust is simply the momentum change between the inlet and outlet of the engine plus the pressure forces acting on front end and rear end. For the turbojet assuming frontal area equals the rear area. Specific thrust is how much thrust an engine produces per mass flow rate of air. Equation 10 and 11 show how to calculate thrust and specific thrust [3].

$$\mathrm{F} = (\dot{m}_9 V_9 - \dot{m}_0 V_0) + A_9(P_9 - P_0) \quad (10)$$

$$Specific\ Thrust = \frac{F}{\dot{m}_0} = a_0\left(\frac{\dot{m}_9}{m_0}\frac{V_9}{a_0} - M_0\right) + \frac{A_9 P_9}{\dot{m}_0}\left(1 - \frac{P_0}{P_9}\right) \quad (11)$$

This specific thrust equation can be derived into the equation below which we can use with variables we calculated. Using an isentropic relationship, the Mach number can be found. Specific thrust can be calculated as shown in equation 12 [3]. The fuel to air ratio is usually a value smaller than 0.06 [3].

$$\frac{F_C}{\dot{m}_C} = a_0\left((1+f)\frac{V_9}{a_0} - M_0 + (1+f)\frac{{T_9}/{T_0}}{{V_9}/{a_0}}\frac{1 - P_0/P_9}{\gamma_c}\right) \quad (12)$$

$$\text{where } f = \frac{\dot{m}_f}{\dot{m}_c}$$

All the subscripts have to be changed accordingly when calculating for flow in the core. But equations are the same. A substitution can be done to simplify temperature ratio between the outlet and inlet temp of engine [3].

$$\frac{T_{t9}}{T_0} = \tau_i \tau_c \tau_b \tau_t \tau_n = \tau_\lambda \tau_t \tau_n \quad (13)$$

Specific thrust from the fan and the core can simply be added together to obtain the total specific thrust. From the calculated specific thrust, specific fuel consumption can be easily obtained using equation 14 [3]. The fuel efficiency is usually measured by a parameter called specific fuel consumption (S) in the industry. Specific fuel consumption is how much fuel an engine consumes per unit of thrust it generates [3].

$$\mathrm{S} = \frac{f}{\frac{F}{\dot{m}_0}} \qquad (14)$$

4.4 Design Parameters, Limit, Problem Set Up

The design choices we can make are total pressure ratios through the compressor. If we increase the total pressure ratio through the compressor, higher thrust can be obtained. However, the temperature of exhaust also rises with higher pressure ratio. When designing a product, considering the design limits is very important. For our case, the design limit is often the exhaust temperature from the combustion chamber because it is the highest temperature in the engine. Without this consideration, the turbine will melt and system will break down. Another design choice is the type of fuel. Each fuel has a different heating value. Although a fuel with a higher heating value has more energy to run the engine, a higher heating value also results in higher temperature. A safety factor and storage factor should be also considered. For our purposes, we will use kerosene. The heating value for the kerosene is 35 MJ/kg.

For the reasons in section 5.3, the total pressure ratio and total temperature ratio through the inlet is assumed to be 1.0 [3]. For the state of art technology level compressors, turbines, and fans, polytrophic efficiencies are around 0.9 [3]. The total pressure ratio through a compressor is 4 [3]. This compression ratio was chosen by comparing the rotation speed to a figure in a book by Mattingly [3] on page 679. The usual rpm speed for steady state compressor operation for this jet engine is around 100,000 rpm which is converted to be around 477 m/s [3]. The exhaust pressure from the turbine is not expanded because we do not have a nozzle. Therefore, the pressure will be substantially higher than the freestream value of 101325 Pa. A fuel to air ratio of 0.04 will be used [3].

The combustion chamber exhaust temperature should be less than 1500 K which is the metallurgical temperature limit for nickel-titanium-aluminum alloys. All the inputs are listed in Table 2.

Table 2: Problem Set Up

Variable	Name	Value	Variable	Name	Value
M_0	Freestream Mach Number	0.01	e_t	Polytropic efficiency (turbine)	0.9
P_0	Freestream Pressure	101325 Pa	P_0/P_5	Pressure ratio	0.5
R	Gas Constant	0.287KJ/Kg-K	C_{pc}	Specific Heat at Compressor	1300KJ/kg-K
f	Fuel to Air Ratio	0.04	H_{PR}	Heating Value	35 MJ/kg
π_i	Total pressure ratio (inlet)	1	γ_c	Specific heat ratio at compressor	1.4
η_m	Shaft Efficiency	0.99	γ_t	Specific heat ratio at compressor	1.3
T_0	Freestream Temperature	280	C_{pt}	Specific heat at turbine	1500KJ/kg-K
π_b	Total pressure ratio (burner)	0.99	π_c	Total pressure ratio (compressor)	4
e_c	Polytrophic efficiency (compressor)	0.9	T_{t4}	Combustion Exit Temperature	1200K

5 Engineering Analysis of Turbocharger Jet Engine

5.1 Thrust and Specific Fuel consumption Calculation

First, total temperature ratio is solved at each stage.

$$\tau_{inlet} = 1$$

$$\frac{T_{t3}}{T_{t2}} = \tau_c = \pi_c^{\frac{\gamma-1}{\gamma e_c}} = 4^{\frac{1.4-1}{1.4*0.9}} = 1.553$$

$$T_{t2} = T_0$$

$$T_{t3} = (1.553)(280) = 434K$$

$$\eta_b = \frac{\left((1+f)h_{t4} - h_{t3}\right)}{fh_{pr}} = \frac{\left((1+0.04)1500 * 1200 - 1300 * 434\right)}{0.04 * 35 * 10^6} = 0.934$$

$$\tau_t = 1 - \frac{1}{\eta_m(1+f)\tau_\lambda}[\tau_c - 1] \ \ where\ \tau_\lambda = \frac{T_{t4}}{T_0}$$

$$\tau_t = 1 - \frac{1}{0.99(1+0.04)4.28}[1.553 - 1] = 0.589$$

$$where\ \tau_\lambda = \frac{1200}{280} = 4.28$$

$$\frac{T_{t5}}{T_0} = \tau_i\tau_c\tau_b\tau_t = \tau_\lambda\tau_t = (4.28)(0.589) = 2.52$$

Second, the total pressure ratio is solved at each stage.

$$\pi_t = \tau_t^{\frac{\gamma_t}{e_t(\gamma_t-1)}} = 0.547$$

$$\frac{P_{t5}}{P_5} = \frac{P_0}{P_5}\pi_i\pi_c\pi_b\pi_t = (0.552)(1)(4)(.99)(0.547) = 1.196$$

Using the pressure ratio terms and temperature ratio terms, the Mach number at the exhaust of a core and fan can be separately calculated.

$$M_5^2 = \frac{2}{\gamma_t - 1}\left[\left(\frac{P_{t5}}{P_5}\right)^{\frac{\gamma_t-1}{\gamma_t}} - 1\right] = \frac{2}{1.3-1}\left[(1.196)^{\frac{1.3-1}{1.3}} - 1\right] = 0.281$$

$$M_5 = .53$$

$$\left(\frac{V_5}{a_0}\right)^2 = {}^{T_5}/_{T_0}\left(M_5^2\right)$$

$$\frac{V_5}{a_0} = .841$$

Finally, specific thrust and specific fuel consumption can be calculated.

$$v_0 = M_0\sqrt{\gamma_0 R T_0} = 0.01\sqrt{1.4 * 287 * 280} = 3.35 m/s$$

$$\frac{F_{core}}{\dot{m_{core}}} = v_0\left((1+f)\frac{V_5}{v_0} - M_0 + (1+f)\frac{T_5/T_0}{V_5/a_0}\frac{1 - P_0/P_5}{\gamma_c}\right)$$

$$3.35\left((1+0.04)0.841 - 0.01 + (1+0.04)\frac{2.52}{0.841}\frac{1-0.5}{1.4}\right) = 6.62\frac{N}{kg}.s$$

$$\mathrm{S} = \frac{f}{\frac{F}{\dot{m}_0}} = \frac{0.04}{6 * 397.9} = 6038\, mg/Ns$$

This result may seem surprising in that the engine generates so little thrust. However, this is normal because a drag called additive drag becomes infinitely large to reduce the thrust when the machine is stationary. Also, this turbocharger engine does not have a nozzle which is the main source of thrust in an engine. Therefore, a low thrust is normal.

5.2 Adiabatic Flame Temperature and Thermal Efficiency

Governing Equations of an Ideal Brayton Cycle [10]:

$$\frac{T_2}{T_1} = (r)^{\frac{(k-1)}{K}}$$

$$\dot{Q}_H = \dot{m}C_p(T_3 - T_2)$$

$$\dot{W}_t = \dot{m}C_p(T_3 - T_4)$$

$$\dot{W}_c = \dot{m}C_p(T_1 - T_2)$$

$$\dot{W}_{net} = \dot{W}_t + \dot{W}_c$$

Where:

$$r = Pressure\ Ratio$$

$$k = \frac{C_p}{C_v} \quad k_{air} = 1.4 \quad C_{p,air} = 1.005\ \frac{kJ}{kg * K}$$

$$w_{c,ac} = \frac{W_{compressor}}{\eta_{compressor}}$$

$$w_{t,ac} = \eta_{turbine} * w_{turbine}$$

For our Turbocharger:

$$r = 3.5$$

$$T_1 = T_{atm} = 293°K \quad P_1 = P_{atm} = 101.35\ kPa$$

$$P_2 = 101.35 * r = 354.725\ kPa$$

$$T_2 = 293 * (3.5)^{\frac{1.4-1}{1.4}} = 419.098°K$$

Find Adiabatic Flame Temperature for Kerosene (assume pure dodecane fuel [17]):

Balanced Reaction:

$$2C_{12}H_{26} + 37O_2 \rightarrow 26H_2O + 24CO_2$$

Adding in Nitrogen and assuming that mixture burns in 150% theoretical air to account for the cooling air in the combustor:

$$2C_{12}H_{26} + 55.5O_2 + 208.68N_2 \rightarrow 26H_2O + 24CO_2 + 18O_2 + 208.68N_2$$

The preceding equation represents the ideal combustion of Kerosene. Carbon Monoxide will be present in the products of combustion, yielding the following equation:

$$2C_{12}H_{26} + 55.O_2 + 208.68N_2 \rightarrow 26H_2O + aCO_2 + bCO + cO_2 + 208.68N_2$$

We are not given the relative concentrations of carbon dioxide and carbon monoxide, so we assumed that 95% of the carbon atoms form carbon dioxide, 5% of the carbon atoms form carbon monoxide. This yields the following:

$$2C_{12}H_{26} + 55.5O_2 + 208.68N_2 \rightarrow 26H_2O + 22.8CO_2 + 1.2CO + 19.2O_2 + 208.68N_2$$

From the first law of thermodynamics we have the following equation:

$$Q_{c.v.} = W_{c.v.} + \sum\nolimits_P N_e(\Delta\overline{h_f}^o + \Delta\bar{h}_T) - \sum\nolimits_R N_i(\Delta\overline{h_f}^o + \Delta\bar{h}_T)$$

We assume that the combustion chamber is well insulated and the working fluid does not do any work on the system, so we can conclude that $Q_{c.v.} = 0\ W_{c.v.} = 0$. This simplifies the equation to the following:

$$\sum\nolimits_P N_e\left(\Delta\overline{h_f}^o + \Delta\bar{h}_T\right) = \sum\nolimits_R N_i\left(\Delta\overline{h_f}^o + \Delta\bar{h}_T\right)$$

for all reactants. We also assumed that the outlet temperature would be well above the dew point at atmospheric conditions so that the outlet water is in vapor form.

From A.10[1] the following information was obtained:

$$\Delta\overline{h_f}^o{}_{C_{12}H_{26}} = -290900\ \frac{kJ}{kmol}$$

$$\Delta\overline{h_f}^o{}_{H_2O} = -241826\ \frac{kJ}{kmol}$$

$$\Delta\overline{h_f}^{o}{}_{CO_2} = -393522 \frac{kJ}{kmol}$$

$$\Delta\overline{h_f}^{o}{}_{CO} = -110527 \frac{kJ}{kmol}$$

To find $\Delta\bar{h}_{T,in}$ $[\frac{\mathrm{kj}}{\mathrm{kmol}}]$ For each compound, we will neglect the contribution of the fuel because there is no easily available data for its enthalpy of formation. This is a reasonable assumption because the thermal mass of the fuel compared to that of the gaseous components is small.

$$\Delta\bar{h}_{T,in,H_2O} = 4144.4 \qquad \Delta\bar{h}_{T,out,H_2O}$$

$$\Delta\bar{h}_{T,in,O_2} = 3638.8 \qquad \Delta\bar{h}_{T,out,O_2}$$

$$\Delta\bar{h}_{T,in,CO_2} = 4863.4 \qquad \Delta\bar{h}_{T,out,CO_2}$$

$$\Delta\bar{h}_{T,in,CO} = 3568 \qquad \Delta\bar{h}_{T,out,CO}$$

$$\Delta\bar{h}_{T,in,N_2} = 3559 \qquad \Delta\bar{h}_{T,out,N_2}$$

This yields the following equation:

$$(-290900) + 55.5(3638.8) + 208.68(3559) = 26\left(-241826 + \Delta\bar{h}_{T,out,H_2O}\right) + 22.8\left(-393522 + \Delta\bar{h}_{T,out,CO_2}\right) + 1.2\left(-110527 + \Delta\bar{h}_{T,out,CO}\right) + 19.2\left(\Delta\bar{h}_{T,out,O_2}\right) + 208.68(\Delta\bar{h}_{T,out,N_2})$$

$$16046155.52 = 26\,\Delta\bar{h}_{T,out,H_2O} + 22.8\,\Delta\bar{h}_{T,out,CO_2} + 1.2\Delta\bar{h}_{T,out,CO} + 19.2\Delta\bar{h}_{T,out,O_2} + 208.68\bar{h}_{T,out,N_2}$$

Assuming that the output products are all nitrogen we find the flame temperature to be $2100°K$.

Using Microsoft Excel, a final adiabatic flame temperature of 1922 K was calculated using an iterative procedure. However, this result does not reflect actual temperature of the combustion chamber exhaust temperature because the process is not entirely adiabatic and heat will be dissipated. Also, thehot flame temperature will be lowered with cool air around the inner liner. Therefore, the assumption of a combustion chamber exit temperature of 1200K is reasonable.

Now, we calculate the work required to drive the compressor. For the sake of these calculations we will assume that the compressor and the turbine have efficiencies of about 0.85. These parameters will be measured during testing of the engine.

$$\dot{W}_c = \frac{\dot{m}C_p(T_1 - T_2)}{\eta_{compressor}} = \frac{\dot{m} * 1.005(293 - 419.098)}{\eta_{compressor}} = \frac{-126.73\dot{m}}{.85} = -149.09\dot{m}\ W$$

We will assume that the turbine will expand the combustor exhaust to atmospheric pressure:

$$\frac{T_3}{T_4} = (r)^{\frac{(k-1)}{K}} \quad T_4 = \frac{1922}{(3.5)^{\frac{(1.4-1)}{1.4}}} = 1343.7°K$$

Now we can calculate the Turbine output power:

$$\dot{W}_{t,ac} = \dot{m}C_p(T_3 - T_4) * \eta_{turbine} = \dot{m}(1.005)(1922 - 1343.7) * .85 = 494\dot{m}\ W$$

$$\dot{W}_{net} = \dot{W}_t + \dot{W}_c = 494\dot{m} - 149.09\dot{m} = 344.915\dot{m}$$

The total heat power supplied is given by:

$$\dot{Q}_H = \dot{m}C_p(T_3 - T_{2'}) = \dot{Q}_H = \dot{m}(1.005(1922 - 419.098) = 1510.42\dot{m}\ W$$

This will yield an overall engine efficiency of:

$$\eta_{th} = \frac{\dot{W}_{net}}{\dot{Q}_H} = \frac{344.915\dot{m}}{1510.42\dot{m}} = 22.8\%$$

Summary

We built a real life version of turbo jet engine using a turbocharger from a truck. The system consists of an air handling system, ignition system, fuel system, and oil cooling system. The whole air pathways are made from stainless steel with parts either fabricated in a machine shop or purchased. An engineering analysis shows the thrust generated is 6.62 N/kg s, specific fuel consumption is $6038\ mg/Ns$. The adiabatic flame temperature is 1922 K. The thermal efficiency is calculated to be 22.8 %.

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