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Lab Scale Test System for a Brayton Cycle Engine

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Aerospace science plays a critical role in today's society, it is important that engineers of all types understand the basic theories of such prevalent technology. The purpose of this project is to build a lab scale engine, which provides engineering students with a powerful tool in learning fundamental and advanced engineering topics in Thermo-Fluid field. This completed system allows for hands on education, research and experience in the field of turbomachinery, thermodynamics, heat transfer and fluid mechanics.

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Introduction

The aerospace industry is under a constant state of improvement to make either the manufactured product more efficient, reliable or cost effective. The entire industry is reliant on its engineers to ascertain the knowledge of the jet engine cycle. Hence, it is critical for the students who are actively pursuing the field of aerospace have solid knowledge of the jet engine cycle. It has been demonstrated by numerous studies, that a combination of theory and hands on experience is a critical component of engineering education [1-2].

In an effort to give students the full benefit that they can gain from in-depth laboratory activities, departments devote money and effort to purchase or upgrade lab facilities. However, many universities are facing budget constraints, which have limited their ability to pursue the lab equipment, especially the high cost gas turbine machinery. This project intercedes the gap between a class room education and hands on experience in a cost effective method.

Background

The foundational theory that supports this project is formally known as the 'air-standard Brayton cycle' or in a more accustomed term the 'Jet Engine cycle'. The basic air-standard Brayton cycle utilizes three basic components those of which are also utilized in this

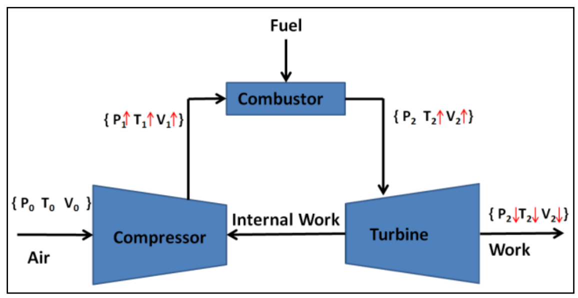


Figure 1 Air-standard Brayton cycle depicting the necessary components[3]

The Brayton cycle begins with the compressor on the turbocharger where air enters and is pressurized as it interacts with the compressor wheel, this compressed air then flows to the combustor chamber where it is mixed with fuel and ignited. During the process of igniting the fuel-air mixture the pressure inside of the combustion chamber remains at a constant pressure since the chamber has a constant flow in and out at all times. This air, now pressurized and heated expands against the turbine of the turbocharger producing work and power for the system. As the hot gasses turn the turbine, the turbine conversely provides power to the compressor of the turbocharger resulting in a system that is self-sustaining.

The air-standard Brayton cycle can also be described graphically using diagrams that show the pressure-volume changes at all four stages of the system and a diagram showing the temperature-entropy changes.

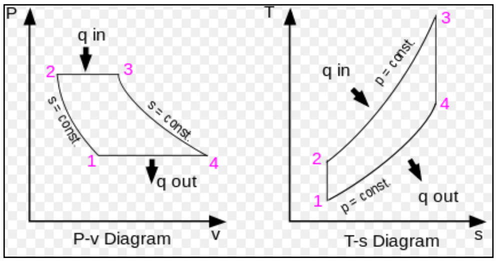


Figure 2 Graphs depicting pressure-volume and temperature-entropy changes during the four stages for this system

The first graph in Figure 2 shows the pressure and volume changes during the systems four stage cycle beginning with air entering the compressor. This cycle begins at stage one and progresses to stage two where the air is compressed as a results of the compressor. At stages two and three the graph shows heat being added to the system as 'q in', from stages three to four the air expands against the turbine releasing pressure bringing the air back to its original atmospheric pressure and temperature as shown by 'q out' thus bringing the cycle to a close where it then starts again.

The second graph in Figure 2 describes the temperature and entropy changes that the system undergoes through its four stages. Air enters the compressor at stages one and two where its temperature is raised as a result of being compressed. From stage two the air enters the combustion chamber where it is heated shown as 'q in' in a constant pressure process leading the cycle to stage three its peak temperature point. From stage three to stage four the air expands against the turbine and is cooled lowering the air temperature to stage four of the cycle. This hot air is cooled in the atmosphere where it is then brought back in to the compressor at stage one bringing the cycle to its starting point where it then begins again [4].

In an operational jet engine the engine is divided into four components, the compressor, combustion chamber, turbine and nozzle as shown in Figure 3.

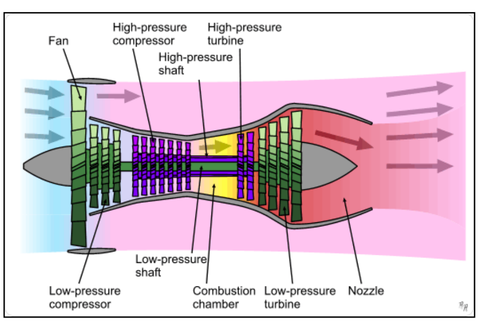


Figure 3 various components of the standard jet engine [3]

The components of a jet engine are the same as used to define the Brayton cycle and prove its operational theory is based on the Brayton cycle theory. The Turbojet project, as proposed by the authors of this proposal, has the same basic components of the jet engine shown in the figure below.

From the Figure 3 it is clear that the difference between the two systems is that the Turbojet does not have multiple stages of compression or expansion as does a jet engine and the air flow is not axial as observed in a jet engine. However, both systems utilize the same basic components and both operate under the same theory, the Brayton cycle. As this project utilizes the same components and theory found in an actual jet engine the conclusion of this project will bring about a wonderful opportunity to experiment with the various components of the Brayton cycle and provide students and faculty members a unique opportunity to both gather data, improve design and performance traits of the Brayton cycle.

Design of the lab scale turbo-jet engine

A commercially designed and manufactured gas turbine or jet engine is often relatively lightweight, and compact. Such design attributes are in line with producing a machine with a high power to weight ratio and aerodynamics for use on aircraft or other applications where space and weight are critical design parameters. The system described here is designed as a teaching and research platform. Thus, the design attributes are on the contrary to a commercial engine. Moreover, the design and manufacture of a gas turbine which is lightweight, and aerodynamic is extremely expensive. The engine described in this document was designed and manufactured to the following attributes to enhance its primary purpose as a teaching and research platform; Easy implementation of data collection equipment, future modification improvement, visibility of all components, durability, mobility, relative ease of use, maintenance, relative low cost, and safety. The following paragraphs will be brief description of the major system components.

The turbo jet platform is comprised of the main engine components and support components. The main engine components include the turbocharger (compressor and turbine), combustor, and thrust nozzle. The support components include the control, lubrication, fuel, and starting systems. As well as the frame which the entire system is built on.

Compressor and Turbine: The rotor and stator parts of turbo jet engine are very complex and expensive components to design and manufacture. Thus, a turbocharger, which already contains all of these components in the form of radial compressor and turbine connected via a common shaft with hydrodynamic journal bearings was utilized. Moreover, the turbocharger stator parts provide flanges and connections which are simple and adaptable to a simple combustor and thrust nozzle. A Borg Warner turbocharger (PN 4P2060) was selected. The turbocharger was originally designed for a Caterpillar model 3406 diesel engine. This unit was chosen for the system due it simplicity (no waste gate or variable turbine geometry) as well as it relative low cost compared to other models. Compressor and turbine performance maps were generously provided by Borg Warner.

Combustor: A simple combustor conceptual design was selected based on current simple proven designs [5]. The combustor was custom fabricated by the university with the assistance of local experts using 304 and 316 stainless steel. The completed combustor consists of the casing, flame tube, cap and intake. An exploded view of the combustor is shown in Figure 4. Combustor detailed design will be addressed in a proceeding section.

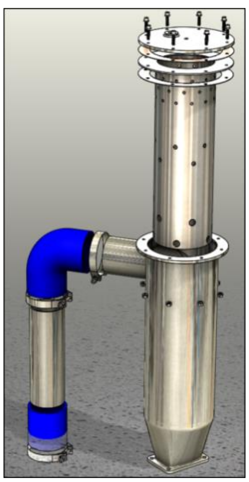


Fig.4 Combustor System 'Exploded View'

Thrust Nozzle: Like the combustor, the thrust nozzle was based off current simple proven designs. The most critical design parameter of the thrust nozzle is the nozzle exit cross sectional area. The nozzle exit cross sectional area must be selected as to produce thrust but not place to much back pressure on the turbine which could overload the thrust bearing in the turbocharger. An outlet area of a circle with a diameter of 3in was selected. This area is equal to the area of the turbine inlet flange. The thrust nozzle is constructed of stainless steel and was fabricated at the university from three different components. A converging section of 20 gage 304 stainless steel. A straight section 16 gage 304 stainless steel and a flange designed to connect to the turbine with a v-band flange. The converging section is 14in long with a 5" diameter inlet and a 3" diameter outlet. The nozzle is shown in figure 5.

Figure 5 depicting nozzle utilized

Lubrication System: A lubrication system was designed and fabricated to lubricate and cool the turbocharger hydrodynamic bearings. The Lubrication system is comprised of the following major components. Oil pump, pump drive motor, reservoir, cooler/radiator, and sensor/distribution manifold. The system was constructed almost entirely using commercial off the shelf components. The systems is pressurized via a SHURflo GPBNV gear pump connected to a 120VAC 3/4 HP motor to produce approximately 5 GPM of flow with 10w-30 oil. The cooler was fabricated using a 6300 Btu/hr. radiator and two 290 cfm 120VAC fans. A distribution and sensor manifold was designed and fabricated to allow monitoring of oil pressure and temperature as well as control the flow rate to the turbocharger via a needle valve and bypass line. All pressurized lubrication system tubing was constructed with ASTM B75 3/8" copper tubing (1/2" OD, .402" ID, .049" Wall). All tubing connections were made via AN type 37° brass flare fittings. All pipe thread fittings were sealed using GASOILA PTFE sealant. The turbocharger bearing return is plumbed using 1in steel and PVDF tubing.

Control System: A control system was designed and constructed to control and monitor the critical functions of the system such as oil pressure and temperature, cooling fans, and fuel flow. The system is powered via 120VAC and 12VDC. The oil pump motor, cooling fans, blower, and 12VDC power supply/battery charger are powered via 120VAC. The LPG gas solenoid valve, engine monitoring instruments, and ignition coil are powered via 12VDC. One key component of the control system is an electronic fuel solenoid valve which allows for near instantaneous engine shut down and stoppage of fuel flow via a standard emergency stop switch.

Fuel System: LPG (propane) was chosen as the fuel source for the engine because it is readily available in a pressurized and atomized state. A liquid fueled engine (jet-a or kerosene) would require a fuel pump and vaporization system which would be more costly and complicated. The fuel system was designed using NFPA 58 standards. The system is comprised of the following components: A 100Lb propane tank connected to a single stage welding style regulator. A flexible fuel line which connects the tank and regulator to the test stand. On the test stand the flexible fuel line is connected to an electrically controlled solenoid valve. The solenoid valve feeds 1/4 in 45° flare fitting copper tubing. Finally, the copper tubing is connected to the fuel nozzle placed in cap of the combustor.

Starting System: A relatively simple starting system was designed for the engine by employing a simple electric blower to force air through the system and start the cycle. A simple sparkplug and ignition coil was implemented to provide initial system ignition during startup.

Frame: A steel frame was designed and fabricated to support the entire system. The frame houses the entire system except for the propane fuel tank. The frame was constructed using 1in square steel tubing and angle iron. The frame is equipped with lifting casters which allow the entire system to be mobile as well securely fixed during testing. The frame also includes a blast shield constructed of high impact flame retardant Makrolon® UL972 polycarbonate.

Construction of the Combustor

The primary goal of a combustion system on any gas turbine driven aircraft is to increase the thermal energy of the inflowing gas stream via combustion; this process is an exothermic chemical reaction between the injected hydrocarbon fuel and the oxygen drawn in by the engine. The combustor design can be separated into five primary sections,

1. Flame Tube or Flame Holder
2. Primary Zone
3. Secondary Zone
4. Dilution Zone
5. Annulus Flow Region

The design of a combustor chamber in any air craft is generally the same, however the given dimensions vary from aircraft to aircraft depending on specific needs.

Figure 6 shows schematically the principal features of a main burner and illustrates the general pattern of recirculating and mixing flow patterns. These features are present in both axisymmetric and annular main burners. Inflowing air enters the main burner at station 3.1. Because the airstream velocity leaving the stator of the last compressor stage is undesirably high, the flow must be diffused to a lower subsonic velocity. This is done by the expanding shape of the inner and outer casing, which is the pressure vessel of the main burner. The entering airflow is diffused to station 3.2, which is by definition the reference station for the main burner. A "snout" or splitter stabilizes the diffusing airstream and divides it for distribution to the liner and annulus. The central part of the divided airstream flows through an air swirler into the primary zone, where it mixes with atomized and/or vaporized fuel and with recirculated, partially burned gases. The remaining air flows into the inner and outer annulus, then flows into the liner through various holes and cooling slots punched or drilled into the walls of the liner.

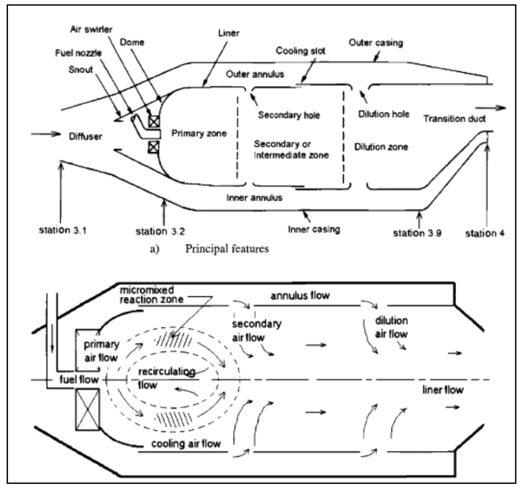


Figure 6 A detailed look at the combustor components [5]

The primary zone is where the action is! Inflowing fuel is atomized, and partially or completely vaporized, by the fuel nozzle. The vaporized fuel is entrained by and mixed into the primary air, which entered through the air swirler. Both the primary air and fuel streams are mixed with partially burned combustion products that are trapped in the recirculation "bubble" in the primary zone. This "backmixing" of partially burned gases with fresh reactants is responsible for the continuous self-ignition process called flameholding, so that an external source of ignition, such as a spark plug, is not required. (However, an external ignition source is required for starting the ftameholding process.) Chemical reaction occurs primarily in the micromixed reaction zone, within which reactants have been mixed to near molecular homogeneity.

From the primary zone the mixture of partially mixed, actively burning, and incompletely burned gases flows downstream into the secondary or intermediate zone, where they continue to burn towards completion while mixing with inflowing air from the secondary holes. Two processes must occur in parallel in the secondary/intermediate zone: 1) the primary zone effluent gases must continue to burn out, and 2) the in-mixing secondary air must "lean out" (reduce the fuel-air ratio of) the liner gas stream. These two processes must be balanced in such a way that the temperature rise which would otherwise occur from continued burnout is offset by a temperature decrease which would otherwise occur as a result of the decrease in fuel-air ratio. Consequently, the liner gases flow through the intermediate zone at essentially constant temperature, and combustion should be complete when the liner gas reaches the downstream end of the intermediate zone.[3]

The dilution zone process, by comparison with the complex chemical and physical processes occurring in the primary and intermediate zones, is a "no-brainer." All that is required of the dilution zone is that any remaining annulus airflow be dumped through the dilution holes into the liner hot gas stream, with just sufficient stirring to avoid hot spots forming on the first-stage high-pressure turbine stators (nozzles). After the hot gases exit the combustor liner at station 3.9, they are accelerated through a converging transition duct until they are choked at the throat of the first stage high-pressure turbine nozzles downstream of station 4.

The design of the combustor chamber for this project follows that of the design of a general gas turbine combustor chamber, utilizing all of the standard components from a combustion chamber with the only discrepancy between this project's chamber and that of a real chamber is the in direction of inflowing air. As stated before the flow through a standard combustion chamber is axial, in this projects design the air enters almost perpendicular to the flame tube allowing air to swirl around the flame tube thus negating the need for a swirler as used in a combustion chamber in a gas turbine.

The dimensions for the outer combustion chamber are based on both chemical kinetics, one dimensional gas dynamics and the physical length of the inner flame tube.

Inside the combustion chamber, round swirling air jets issued from the flame tube are used to mix the inflowing air and propane to the appropriate mixture and to back flow partially burned mixed gases with fresh incoming reactants for continued flame propagation.

The flame tube (liner) has a lower static pressure than the annulus region above the flame tube, this pressure differential induces fluid flow from the annulus into the flame tube where a vena contracta forms inducing an entering air jet (Vj). The figure 7 below depicts this.

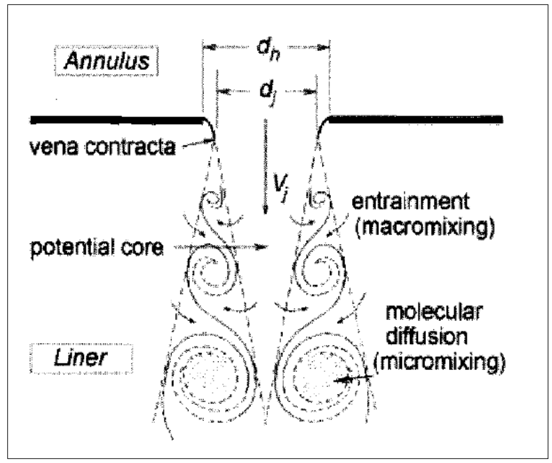


Figure 7 A depiction of the air jets forming inside the flame tube [5]

The picture above shows the inflowing air jet (Vj) into the liner, it also depicts the process of molecular micromixing. The velocity of the incoming jet can be derived from Bernoulli's equation. With the pressure drop value calculated it was determined that with the change in compressor discharge pressure, the average pressure drop between the annulus and liner was between 2-5%. With the determination of the pressure differential value the fluid velocity passing through the liner was found to be about 3 m/s. This value is constant for the compressor's range of pressure and the entire liner as fluid velocity isn't dependent on the diameter of the liner hole, only mass flow rate depends on the entrance area. [3]

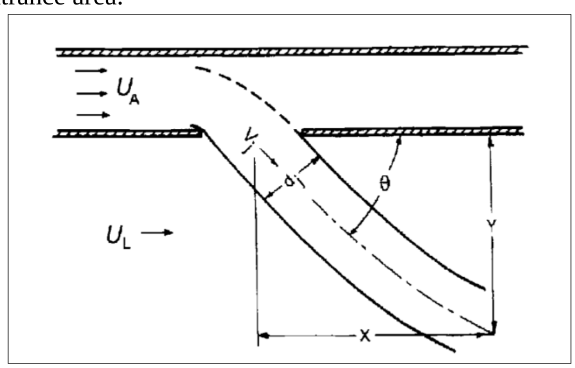


Figure 8 A depiction of the fluid flow from the annulus region into the liner region [5]

The above diagram depicts the flow pattern of air from the annulus into the liner, it is of great importance to study the flow behavior inside the liner. As stated previously the liner is responsible for both macro and micro mixing of the air and hydrocarbon mixture, this is precisely achieved through the flow of air from the annulus to the liner. As the fluid from the annulus enters the jet stream it carries the momentum of the above flow with a axial momentum flux of ρU_a^2 along with the y direction momentum flux ρV_j^2, with respect to a standard Cartesian coordinate system.

By calculating the trajectory of the inflowing jets (utilizing lefebre's equation) we can plot the macro mixing of the liner and determine the proper distances between each zone for proper mixing to occur.

Through research on combustor chamber design it was found that inside the mixing zone (the vortex flow) a significant amount of micro mixing has occurred approximately $10d_j$ downstream of the liner opening[1]. Downstream of the inlet of air the jet grows linearly with the vertical direction at a conical half angle of approximately 7 degrees.

It was determined that the point of adequate mixing is approximately .1m for the primary zone and .24m for the secondary zone which are precisely the distances used in the design of the combustion chamber utilized in this project. As a result of these found distances it was determined that a flame tube liner of 16.5 inches would be an adequate length for this project.

The liner hole size per zone was calculated on the basis of the needed air mass flow rate to initiate the burn (primary zone) with a rich mixture and to complete the burn (secondary zone), hole size for the primary was found to be .25in and likewise the secondary zone .625. Similarly the number of holes was determined by the mass flow rate per hole and multiplying that flow rate until the needed flow rate into each zone was determined.[3]

Analysis

The analysis for this project was done with respect to the entire system in order to determine the overall engine efficiency. This was accomplished through the use of four temperature sensors and one pressure sensor. Consequently in order to measure the entire system efficiency it was necessary to account for the exchange and rejection of heat into and from the system as a whole.

The pressure sensor was placed after the compressor wheel to capture the initial system pressure after passing through the compressor of the turbocharger housing. The four temperature sensors were arranged in a fashion where the temperatures for each component were recorded before and after the air interacts with each component respectively. Two temperature sensors was placed before and after the compressor wheel, to capture the air temperature before and after its interaction with the compressor wheel. A temperature sensor was likewise placed just after the combustor to record the combustor outlet temperature before entering the turbine and finally one temperature sensor was placed after the turbine to measure system output temperature.

The method for measuring system efficiency was through the utilization of the standard air-standard Brayton cycle formula shown below, this function was applied from reference two shown in the paper's reference section.

$$\eta_{th} = 1 - \frac{q_l}{q_h} = 1 - \frac{c_p(T_4 - T_1)}{C_p(T_3 - T_2)} = 1 - \frac{T_1(\frac{T_4}{T_1} - 1)}{T_2(\frac{T_3}{T_2} - 1)}$$

Consequently where the following temperatures represent those depicted in the system efficiency equation shown above.

- T1: Temperature of air before system interaction
- T2: Temperature of air after interaction with compressor wheel.
- T3: Temperature of air after interaction with turbine wheel.
- T4: Temperature of air rejected by system to the environment.

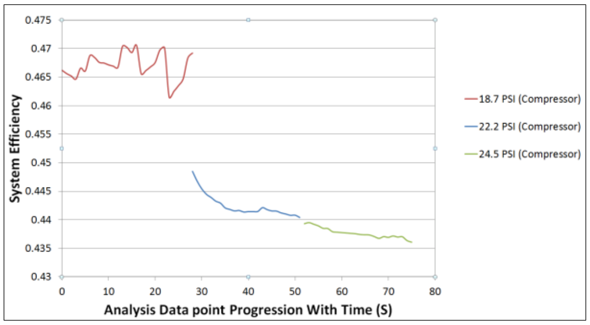
The above efficiency formula works on the basic premise of finding the ratio of heat rejected (q_L) over heat exchange (q_H) of the system. We can express these heat coefficients as temperatures and as a result express the efficiency formula as 'system net work' over 'system heat addition' in the form of various system temperatures as shown previously.

The analysis procedure for determining system efficiency was divided into three system operating points as shown below.

Operating Points:	Fuel Pressure (PSI):	Compressor Pressure (PSI):
1	20	24.5
2	40	22.2
2	60	18.7

Each system operating points were governed by the fuel pressure introduced into the system which in turn regulated the compressor wheel pressure thus identifying the control point for system analysis. At each of the three operating points the data collected by the four temperature sensors and one pressure sensor was recorded and analyzed.

The results for determining system efficiency are expressed in the following graph depicting system efficiency compared to data point collection with time progression.



As shown in the graph, system efficiency ranges from approximately forty-seven percent and forty-three percent. The initial data point collection, referenced at 20 PSI for fuel pressure, is shown to be extremely variant as induced by system instability at the lower power band. Upon an increase in the introduction rate of fuel the system data smoothed out showing an increase in system stability and air flow. The results from the two data point collection zones of forty and sixty PSI fuel pressure show a steady system decrease in efficiency.

The decrease in system efficiency can be explained by understanding which variables change in the overall system efficiency function. Of the temperatures sampled only the temperature leaving the compressor and turbine should experience any change and in this case a rise and decrease in temperature respectively. The room temperature (T1) is assumed to remain constant throughout the system operation, likewise the combustor temperature will also remain a constant, within reason, as the fuel to air ratio remains the same resulting in a constant energy density output. The compressor temperature will experience a rise as induced from a increase in propane flow rate increasing compressor wheel RPM resulting in a higher air compression ratio ultimately increasing air temperature leaving the compressor. Likewise the turbine will experience a decrease in air temperature as its ability to extract energy from the air flow increases with turbine wheel RPM.

Conclusions

Gas turbine technology is an ever increasing field of interest with respect to efficiency, reliability and feasibility to the general public. With this in mind the need for advancement and perfection of the understanding on how gas turbine work is most prevalent in today's world. This project aids in the development of student understanding in the form of offering a lab scale system operating on the air stand Brayton cycle, offering students the opportunity for hands on learning in all disciplines associated with gas turbine technology.

Utilizing standard components, readily available, this project proves the feasibility of developing a gas turbine under the constraints of budgeting for some schools. The foundational principles of the Brayton cycle are relevant in this project providing a direct correlation between the theory of the Brayton cycle and actual real world operation thus bringing the gas between theory and experience.

Upon a full system analysis the operational characteristics of the system under operation were that of a gas turbine in a working state operating at approximately forty-five percent efficiency. This analysis proved that the system operated in the same way as any gas turbine would under loading conditions showing the practicality of the engine.

Gas turbine technology should be explored and understood by students, professors and person(s) with an interest in the field. With the prevalence of gas turbines in our daily lives it's obvious for the need to explore their operations and behavioral characteristics in a laboratory setting. With the understanding of gas turbine technology one can improve upon their functionality, reliability and feasibility allowing for a brighter future.

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