ALTERNATIVE POWER UNIT FOR LIGHT, COMMERCIAL AIRCRAFT: DESIGN AND PERFORMANCE MODELING

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DECLARATION OF ORIGINAL WORK

I herewith declare the originality of this thesis report as my own, unaided work with respect to its literature content in general, the computational codes in particular and the resulting conclusions presented herein.

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(Signature of candidate)

______ day of __________ (year) ___________

ABSTRACT

Developments in the field of microturbine technology and gas turbine driven aircraft has been progressing without much progress in light aircraft predominantly propelled by piston engines. Because of inhibitive maintenance and overhaul costs of such however, propulsion via a gas turbine engine has been proposed with the potential of eventually replacing current engine configurations. Subsequently, the objective was to conceptually design a replacement gas turbine engine in the 150 kW range.

A selection of case studies was used to illustrate the changing technologies to illustrate the technological viability of micro-gas turbines for light aircraft. Advantages and disadvantages of both engine types were discussed and a concise description of gas turbine operations and its components was given.

A brief overview of fundamentals as well as the transmission layout was also supplied. Three configurations were isolated, namely the single spool design, a twin spool design featuring a free power turbine and the effect of a fuel conserving recuperator.

Calculations were performed using Microsoft Excel, which proved sufficient in effectively calculating complex formulae - even under the necessary iterative feed-back conditions the design process demanded.

Eventually, variable-specific design criteria were derived regarding the three engine types. Because fuel consumption still proved inhibitive, the effect of recuperation was investigated which yielded a very competitive engine - should the possibility of recuperator technology exist on time.

As a result, one particular recuperated, single spool gas turbine engine was successfully identified. Having met all the design criteria sufficiently, this preliminary prototype design was numerically described and put within context of principal, peripheral working components such as a compatible gearbox layout.

In Dedication to my absolutely wonderful wife, Rosemary Anne

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LIST OF SYMBOLS

Symbol

Definite angles	α, β, θ
Density	ρ
Diameter	Ø
Efficiency	η
Entropy	S
Flow coefficient	Φ
Heat ratio	γ
Nominal rotor loss coefficient	ξ_R
Nominal stator loss coefficient	ξ_S
Pi	π
Power	**P**
Pressure	P
Pressure-absolute	P_0
Reaction ratio	ψ
Stress	σ
Temperature	T
Temperature-absolute	T_0
Torque	τ

NOMENCLATURE

Acronym

AB	Afterburner
ACC	Advanced Clearance Control
ACM	Air Cycle Machine
AD	Aeroderivative
AE	Aircraft Engine (General)
AF	Air Flow
AFR	Air Fuel Ratio (also A/F)
ANN	Artificial Neural Network
APU	Auxiliary Power Unit
BHP	Brake Horse Power (= HP)
BPR	Bypass Ratio (Turbofans; also BR)
CA	Civilian Airliner
CCGT	Combined Cycle Gas Turbine
CDA	Controlled Diffusion Airfoils
CFD	Computational Fluid Dynamics
CHP	Combined Heat & Power Generation (ie. Cogeneration)
CI / D	Compression Ignition (Reciprocating Diesel Engine)
CNG	Compressed Natural Gas
COGEN	Cogeneration
CR	Compression Ratio (SI/CI only)
E3	Energy Efficient Engine
ECU	Energy Conversion Unit
ECU	Electronic Control Unit
EGSL	Energy and Gas-dynamic Systems Laboratory
EGT	Exhaust Gas Temperature
ESFC	Equivalent Specific Fuel Consumption
ESHP	Equivalent SHP (incorporates jet thrust)
EXP	Experimental
FADEC	Fully Automated Digital Engine Control
FF,FC	Fuel Flow, Fuel Consumption
FOD	Foreign Object Damage
GG	Gas Generator
GPU	Gas Power Unit
GST	Genset (Industrial Power Gen; usually AD's)
GT	Gas Turbine
GTS	Gas Turbine Starter
H	Helicopter
HEV	Hybrid Electric Vehicle (eg. a MT as a battery charger)
HHV	Higher Heating Value
HP,LP	High Pressure, Low Pressure (ie. stages/spool)
HPT/C	High Pressure Turbine/Compressor
HVAC	Heating, Ventilation and Cooling
ISA	International Standard Atmosphere
LA	Light Aircraft
LCA	Light Commuter Aircraft
LCD	Liquid Crystal Display
LHV	Lower Heating Value
LMA	Light Military Aircraft
LPT/C	Low Pressure Turbine/Compressor
LRU	Line Replaceable Unit

LSA	Light Surveillance Aircraft
LTA	Light Transport Aircraft
MA	Military Aircraft
MAP	Marine Application
MBT	Main Battle Tank
MDV	Mechanical Drive Unit
MGT	Micro-gas Turbine
MTB	Microturbine
MTBF	Mean Time Between Failure
MTJ	R/C Model Turbojet Engine
MTP	R/C Model Turboprop Engine
MTS	R/C Model Turboshaft Engine
NASA	National Aeronautical Space Agency (USA)
NEOF	No Evidence of Failure
NGT	Nozzle Guide-vane Temperature
NGV	Nozzle Guide Vane
OGE	Out of Ground Effect
P/W	Power to Weight Ratio
PP / PM	Power Plant / Prime Mover
PR	Pressure Ratio (GT compressors only)
RAF	Royal Air Force
R&D	Research and Development
REC	Recreational
SA	Surveillance Aircraft
SFC	Specific Fuel Consumption
SHP	Shaft Horse Power (= HP)
SI	Spark Ignition (Reciprocating Petrol Engine)
SL	Sea Level
SP	Shaft Power
SST	Supersonic Transport
T/W	Thrust to Weight Ratio
TA	Transport Aircraft
TAS	True Air Speed
TBO	Time Between Overhauls (hrs)
TE	Thermal Efficiency
TEDANN	Turbine Engine Diagnostic Artificial Neural Network
TF	Turbofan Engine
THP	Thrust Horse Power
TJ	Turbojet Engine
TP	Turboprop Engine
TS	Turboshaft Engine
UA(C)V	Unmanned Aerial (Combat) Vehicle
UL	Ultralight Aircraft
USAF	United States Air Force
VLA	Very Light Aircraft
VTOL	Vertical Take Off and Landing

1 INTRODUCTION

Hailed as one of the greatest inventions of the previous century, the modern day aircraft jet engine has its inventive roots as far back as 1791. John Barber of England patented his first gas turbine design - ironically about a century before the necessary materials, designs, tools and manufacturing processes available made building one technically possible. In the early 1900's however, the gas turbine engine once again hit the drawing board as a possible power generation alternative. The first aircraft gas turbine engine though was independently yet concurrently developed in the 30's by both Frank Whittle, a young Royal Air Force officer and engineer and by Hans Joachim Pabst von Ohain who was a doctoral candidate in physics and aerodynamics at the University of Goettingen. Frank Whittle patented his design in 1932, but by 1938 Hans von Ohain and his mechanic Max Hahn had designed, built and test flown their first jet aircraft, and on the 27th August 1939 the von Ohain's engine propelled Heinkel He 178 took into the skies - almost two years, before the Gloster E 28/39 in 1941 rotated off a British runway. [20]

1.1 Background

Today's overwhelming aircraft maintenance costs of piston driven, 80 to 150 kW engines make the possession of a private plane prohibitive to most commercial aircraft pilots and enthusiasts alike: A major service on an engine alone often exceeds the original purchase price of an aircraft, and replacement costs can currently exceed R100,000 for even the smallest four cylinder, normally aspirated engine. Common sense dictates, that this factor alone inhibits a large population group of pilots and enthusiasts alike from owning and maintaining their own, private aircraft. [18]

A secondary motivating factor is the global decline after years of recession in the light aircraft industry market, predominantly because of:

- fundamentally outdated propulsion technologies,
- new trends and demands such as environmentally friendlier, quieter and more efficient engines,
- a requirement for more cost effective power plants in general.

Since light aircraft sales dropped sharply in the late '70s, the private aviation sector at large would in the last couple of decades probably not have been noticed at all if it would not have been for the ultralight and the very light aircraft recreation industry. However, in recent years developments in gas turbine, reciprocating and even possibly in light diesel engine technologies all show potential as prospective, new generation powerplants of the new millennium.

1.2 Gas Turbine Applications and Important Achievements

Ever since the beginning stages, the gas turbine engine has undergone gradual, but significant changes and improvements in all aspects of its performance limits – from engine noise reduction to an improved, specific fuel consumption (SFC) to ever enhanced thrust-to-weight ratios (T/W). The following examples will highlight some of such more recent achievements applicable to turboshaft/turboprop applications.

1.2.1 The Rooivalk helicopter

Helicopters as other rotorcraft like VTOL-winged aircraft also make extensive use of the latest in gas turbine technology. VTOLs usually require only pure shaft power to turn their lifting surfaces such as rotor blades needed for lift, steering, propulsion and directional stability at the required speeds. Converting the chemical fuel energy into shaft output power via gas turbine units called turbo-shaft engines use of such units is evident in the majority of all rotorcraft today - with some minor exceptions. Two major reasons for such are the inherent reliability of gas turbine engines due to the small amount of internally moving parts, and the higher thrust to weight ratios achieved than in other types of comparative engine such as in reciprocating prime movers.

The South African 'Rooivalk' attack helicopter built and designed by Denel Aviation as a case in point started its operation at the SAAF in July 1999. Its two 'Makila' 1K2 turbo-shaft engines generate a combined 2,243 kW of rated take-off thrust, enabling the helicopter to travel at 150 kts (278 km/h), to sustain a maximum climb-rate of 2,620 ft/min (47.9 km/h) and obtain a hover-ceiling out of ground effect of 17,900 ft (5,456 m). [40]

1.2.2 The DC-3 Dakota

Somewhat similar to turboshafts in principle is the turboprop gas turbine engine. Apart from generating a large portion of thrust in shaft power, a certain percentage of combustion gases, the proportion depending on design, is expanded in a nozzle to atmosphere for added thrust. A good example of aircraft utilising such technology can be found in airborne transport, whereby large amounts of power are required at subsonic velocities with the least amount of loss in propulsive efficiency. The propeller naturally lends itself as the preferred choice due to its nature of moving large amounts of air-mass at low enough velocities - typically below 640 km/h - thereby generating well needed thrust relatively efficiently. It is somewhat ironic, that gas turbine engines in the form of gas generators are still being utilised for the task of turning the airscrew, and the similarity between a turbo-shaft and a turboprop engine should be intrinsically apparent.

One of the most famous aircraft and the predecessor of all modern transport is undisputedly the world-renowned DC-3/C-47/R4D 'Dakota'. This rather remarkable aircraft is the product of the Douglas Aircraft Company and was first flown on 17th December 1935. Since the early days, its two radial Pratt & Whitney Twin Wasp 1830-75 I/C engines have been replaced by the PT6A-65AR turbo-prop units manufactured by the same company, and the ensuing results were remarkable. Apart from the fact that the turbo props featured a vast maintenance improvement their performance improved remarkably as well. The maximum, useful load capacity increased from 8,620 to 11,800 lbs (3,910 - 5,352 kg) thereby featuring a 37% increase in carrying capacity. Its normal cruising speed at 12,000 ft (3,658 m) increased from 155 to 195 knots (287 – 361 km/h) with an increase in fuel consumption of only about 30% while the maximum climb rate improved from 1,200 to 1,560 ft/min (21.95 – 28.53 km/h). In other words, although the fuel consumption increased moderately its performance levels improved remarkably - yet with a reduction in maintenance costs. [13]

1.2.3 The C-130 Hercules

A typical all-purpose transport is the Lockheed Martin Aeronautics Company-manufactured C-130 'Hercules' military airlifter. This aircraft has been exported to over 65 countries; with delivery of the 219 C-130As' having begun in December 1956. 1,600 of the over 2,200 C-130s' delivered are still in service today, and the latest generation aircraft, the C-130J featuring even LCD's, holds 54 performance-world records. Apart from a new propulsion system, the latest aircraft has been designed for search and rescue, weather reconnaissance, aerial refueling, combat delivery and electronic combat tasks' mission requirements. The latest, powerful Rolls-Royce AE2100D3 turbo-prop engines generate 29% more thrust with an increased fuel efficiency of 15% with respect to the earlier production aircraft, the retiring C-130E, resulting in a 21% faster aircraft and a climb rate, which has subsequently been halved. Both the cruising altitude and the range of the C-130Js' have improved by 40%. [23]

1.2.4 The An-22 Antheus

A good example of size extent of turboprop applications would undisputedly be the Russian Antonov An-22 'Antheus' heavy equipment military transport. Observing the sheer size of the former Soviet Union, it is presumably understandable that its inhabitants have a somewhat different design philosophy than the one found in the West: Huge distances, often badly prepared airstrips and commonly encountered sub-zero temperatures require aircraft which are huge, strong and rugged all at the same time. The An-22 with its four Kutznetsov NK-12 turboprops rated at 15,000 shp (11,186 kW) each features a set of contra-rotating, four bladed propeller blades per engine, all of which are capable of sustaining a take-off weight of 250,000 kg (550,000 lbs); 100 tonnes of which is the payload itself. With a range of 10,950 km (6,800 mi), the difference in approach towards airborne transport design should become apparent; quite a feat for an aircraft, which had its maiden flight in 1965. [6]

Despite noticeable advances of the jet engine over the last five decades and their ever increasing popularity in aeronautics, piston aircraft are still predominantly represented in light commercial aircraft with seating arrangements of up to 6 people besides the crew. It is therefore the aim of this report to introduce the possibility of economically implementing gas turbine technology of modern standards into currently existing airframes of aircraft, which up to now are largely still piston engine driven.

1.3 The Gas Turbine versus the Four-stroke Reciprocating Piston Engine

Both types of engine display certain similarities yet also some marked differences. Both kinds are air breathing, internally combusting power plants which embrace similar thermodynamic processes, namely the intake, compression, power and the exhaust processes. One fundamental difference however is in the method with which these processes are mechanically executed. In a gas turbine, these events happen continuously yet successively simultaneously. In the reciprocating piston engine however they proceed within a well-timed and purely successive fashion. While in the gas turbine engine, every process is performed individually by a specific component such as compression via the compressor stage(s), combustion inside a combustor can arrangement and so on, in the reciprocating counterpart all events are executed by only the same set of mechanical components, namely the piston-cylinder arrangement with its related but specific sub-systems, such as the cam- and crankshaft assemblies, the carburetor system and so on.

Furthermore, looking from a thermodynamic perspective at the pressure-volume and temperature-entropy diagrams of these two cycles it becomes apparent that both cycle types follow a different path. The reciprocating process follows the Otto cycle while the gas turbine process follows the Brayton cycle. In other words, while the Otto cycle resembles essentially a constant volume process, the Brayton cycle embraces constant pressure cycle behaviour. As the thermal efficiency of heat engines is a function of the difference in maximum and minimum operating temperatures, it becomes apparent that maximizing the operating temperatures for the combustion process would result in a more effective heat engine in that more energy would become available.

In a reciprocating engine, very high pressures are attainable within the combustion space above the cylinder head. One of the reasons are the oil lubricated compression rings surrounding the piston which prevent any leakage past them, and pressures in the region of 7,900 kPa (ie. 68 atm.) are intermittently attainable prior to the expansion stroke. Furthermore, combustion happens just before the piston has completed its compression stroke, and most of the power stroke is rather gradually applied as the piston moves down along its power stroke to the bottom dead centre (BDC) position. Excessive local and thermal stresses are thereby avoided, and the cyclic nature aids in preventing overheating of the internal components – keeping in mind, that only one out of every four successive strokes in a four-stroke engine is subject to relatively high internal combustion pressures. Another reason for high pressure allowances is the heat-resistant design of the engine block itself which incorporates an effective radiator and oil circulation system for heat removal and lubrication.

Axial compressor stages in a gas turbine engine on the contrary limit the compression pressures to pressure ratios of about 25 to 30 : 1 for technologically advanced engines. This equates to only about 2,500 kPa (ie. 24.7 atm.) at best, yet still results in relatively high combustor inlet temperatures. Although hot combustion inlet air requires less fuel to be burned thus improving the SFC, the results of this paper suggest that the consequent reduction in the temperature gradient between compressor and combustor exits results in a moderate reduction in shaft output power. Also, heated compressor air within the final compressor stages increases the compressor shaft power required, and inter-cooling which is currently only possible in multiple, axial compressor design stages might have to be incorporated.

The tendency for a more powerful engine would then lead to burning more fuel, thereby raising the combustor exit temperature available for expansion. However, the first turbine stage and the nozzle guide vanes are subject to a continual inflow of combustion gases in the region of about 950°C. Unlike in a reciprocating combustion engine, the first wheel of turbine blades which converts the combustibles to useful torque is subject to continuous thermal, torsional, axial and even high centrifugal stresses. It is therefore ultimately the first turbine stage which limits and therefore decides upon the maximum, sustainable combustor exit temperature the gas turbine unit can accommodate. Some tip leakage is also not completely avoidable because of the tip clearance necessary between the turbine blade tips and the shroud or casing. This is due to radial elongation of the blades subject to thermal expansion, centrifugal forces and ultimately creep which is time-dependent, plastic deformation due to the abovementioned stresses under the influence of heat.

1.4 Motivation for Gas Turbine Engine Selection

Recent advances in GT technology specifically make the employment of a gas turbine engine more and more favourable. Mention will be made therefore of three primary examples where gas turbines have been sufficiently improved. In order of importance, they are the following:

1.4.1 Microturbine technology

By the end of the 70's, steam turbine power plants which were on occasion coupled to a nuclear power supply and designed for electricity generation in the region of 1 GW reached plant efficiencies of approximately 34%. In the 80's though, combined cycle power plant technology became increasingly popular which introduced the possibility of fundamentally higher efficiency, low capacity power generation.

By matching the thermally more efficient gas turbines - rather than a centralized nuclear power station - with the steam turbines, smaller yet commercially viable and competitive power plants in the range of 100 – 200 MW became feasibility, and plant efficiencies of up to 55% were thereby attainable. [21, 39]

'Distributed Generation', a term used for decentralised power generation of 5 MW and less, utilises the services of such new technological advances. In particular, microturbine technology, which features high-speed gas generators in the range of 15 – 350 kW and incorporates a recuperator for waste heat recovery from its exhaust, offers localized power supply and higher reliability. Because of the inherent need to invest in a self-sustaining power supply grid for each separate machine, it stands to reason that transmission and distribution costs for such units would however be significant. [21, 39]

Further application potentials under development are as small-scale energy cogeneration plants for households or for other CHP (combined heat and power) configurations operating at 70 – 80% thermal efficiencies, for hybrid electric vehicles (HEVs) as battery chargers and for the resource recovery market. ('Capstone' Microturbines for example already operate in HEVs.) Apart from an increase in plant efficiency, advantages are lower emission pollutants and substantially reduced maintenance requirements. Additionally, microturbine plants offer the promise of a lightweight, compact design with a better operating reliability due in part to lower levels of vibration-induced stresses. In essence, they are lighter, more compact, more reliable, and operate with no vibration and less noise than conventional piston engines. [10]

1.4.2 Recuperator and Recirculation technology

Heat exchanger technology in turboprop engine design dates back as far as to the 60's. In 1965 the Allison jet engine company had successfully developed and employed a regenerative turboprop for the use in a U.S. Navy based aircraft which validated the feasibility of airborne heat exchanger units - even by 60's standards.

At the University of Florida the Energy and Gas-dynamic Systems Laboratory (EGSL) has reported on their latest development of more efficient, environmentally friendly gas turbine engines for application in naval vessels, tanks, helicopters and even as small power plants in general. What his engineers in essence did was simply to incorporate a heat exchanger – also referred to as a recuperator, an addition common in microturbines – to reduce the fuel flow into the engine as well as to introduce a recirculating exhaust system for the reduction in emission levels. The result was a compact design with lower operating and maintenance costs. A further advantage was a better part load efficiency. It is generally accepted knowledge that gas turbine engines operate very efficiently at its optimal design load at a predetermined engine speed. Should the load vary and thereby the turbine speed change, operating efficiencies would deteriorate rapidly, thereby rendering the average engine less efficient. This occurs, because the compressor and turbine blading are mechanically fixed and only operate efficiently at the optimal design speed – unless a variable pitch mechanism is incorporated. However, modifications conducted at the EGSL allowed the gas turbine engine to operate within 80% of its power range because of the heat exchanger releasing the waste heat at low power levels. Availability as a power plant in the 30 – 100 kW range is being investigated. [30]

1.4.3 Case study: New generation Main Battle Tank gas turbine engine development*

The Russian T-80 main battle tank (MBT) family of gas turbine engines enabled tactical, operational and technical superiority of such tanks "over the best domestic MBTs powered by diesel engines and foreign MBT's powered by diesel or gas-turbine engines." [34] Whether this statement is true or not, the superiority of gas turbines compared to their reciprocating diesel or petrol engine counterparts with respect to performance, operating conditions and maintenance intensity should become in this particular case study nevertheless somewhat more apparent:

One of the latest in Russian MBT gas turbine development, namely the T-80U, is the upgraded version of the 1,000 hp (745.7 kW) GTD-1000T gas turbine engine. Although significantly more powerful than equivalent diesels of the day, the GTD-1000T displayed excessive noise levels, a much higher fuel consumption, a greater heat signature and a lower operating reliability in the diverse and extreme operating environments common to MBT's. As a result, the 1,250 hp (932.1 kW) GTD-1250 was developed which mirrored modern advances in hi-tech aircraft gas turbine development. This particular power plant features the monoblock layout which is comprised of the engine itself, the air cleaner, engine and transmission cooling system's oil radiators, the transmission oil pump, the generator and starter motor, the cooling (blowing) system for both electric units and engine compartment and the compressed air system. To compensate for the excessively high fuel consumptions, an 18 kW

GTA-18 APU powerpack is introduced to supply power predominantly when the engine is turned off.

(* **Note:** Due to adverse, environmental aircraft engine operating conditions it stands to reason to observe and study the behaviour of gas turbines in similar, adverse environments in order to form a balanced opinion of its applicability in general. Although the following case studies relate to turboshaft applications like in our case, it has to be remembered that these are far more advanced than we require, and although aero-derivatives themselves these particular engine classes often feature bypass ratios, multiple axial compressor and turbine stages, intercooler technologies and so on. Also, no stringent space and weight limitations do in general exist - unlike with our model. However, emphasis can still be placed on the potential of future GT engine technology.)

1.4.4 Main Battle Tank engine advantages

Introduced into series production in late 1985, it demonstrates the already well-known advantages inherent in gas turbine behaviour over its reciprocating rivals - in addition to improved performances due to innovative design features and modifications to the original GTD-1000T power plant. Some such features are briefly described below:

(i) This particular engine is structured around a three-shaft configuration, namely a two spool radial flow compressor with each shaft possessing its own radial compressor and a free turbine driving the power shaft, which is equipped with its own, internal power gearbox. The two radial flow compressor stages compared to mixed flow designs have the advantages of large gas-dynamic stability margins of low- and high-pressure turbo-compressors, a maintenance free air cleaner and the stability of basic operating parameters by sharing the compression load in a balanced method.

(ii) Additionally, an adjustable, free turbine nozzle assembly has been incorporated as a torque-suppression device in order to reduce the engine's output shaft speed. The variable pitch nozzle vanes can be rotated from 0° for zero braking to 120° for a braking force of up to 0.6 times the peak power produced without any additional braking taking place.

(iii) For increased power, the generator gas temperature has been raised, and more effective systems and units have also been installed. Fuel consumption for example has been improved by 25% via a whole set of innovative solutions.

(iv) Interestingly however, the GTD-1250 is exempt of a heat exchanger. This is the case because its operating conditions are so complex causing the engine ratings to be too unstable as to validate a heat exchanger which requires somewhat more predictable operating conditions for efficient operation. Additionally, the extra weight added to the vehicle would apparently eliminate the effect of reduced fuel consumption rates.

(v) A special multi-component coating applied to some of its components enables the GTD-1250 to operate on various kinds and grades of fuels.

(vi) Finally, the engine features various innovations, which remove intake dust and dust sediments from inside the power plant.

Advantages of the GTD-1250 over its diesel engine equivalent are numerous:

- A specific power of 19.8 kW/tonne (26.9 hp/tonne) improves travelling speeds, tactical mobility and maneuverability,
- Rapid engine starting qualities at low ambient conditions improves combat readiness,
- Reduction in number-types of fuels, oils and lubrication,
- Much lower heat transfer rates to the engine oil,
- Improved mobile fire effectiveness due to low levels of engine vibration, smoother curvilinear motion and optimal movements,
- Less crew fatigue due to low vibrations and reduced noise as well as smoother torque transfer,
- Ability to traverse soils and terrain featuring low carrying capacities due to even torque application and turbo compressor gas coupling,
- 50% reduction in maintenance intensity (and therefore costs), while seasonal maintenance services have become obsolete,
- Reduction in monoblock replacement time (and therefore overhaul costs),
- Multi-fuel capability. [34]

The American counterpart to the T-80U is undisputedly the M1 family of Abrams MBTs manufactured by General Dynamics Land Systems (GDLS), as these themselves are propelled by gas turbine engines. Looking again at the 70's and 80's technology, the M1 Abrams is equipped with a 1,500 hp (1,118.6 kW) Honeywell/Lycoming Textron AGT-1500 engine, and while the later Russian T-90 MBT class of tanks have since then been fitted with diesel units once again, the U.S. Army experienced little problems with their M1 AGT-1500s: During the Gulf War in the 90's during Operation Desert Storm, 2,000 M1's traveled 3,000 km without a single engine failure thereby illustrating, that a good gas turbine engine design can display high levels of reliability with low maintenance requirements - even under non-static, real time operating conditions: Quite a feat for a modified helicopter engine. Also, instead of converting back to conventional diesel engine like the former USSR for their latest in M1A2 tank technology, Honeywell International Engines and Systems and General Electric have been co-assigned for the development of the LV100-5 second generation gas turbine which is to ultimately power the Crusader self-propelled artillery system and serve as updated retrofit units for a large number of outdated Abrams tanks. It is reportedly anticipated, that the LV100-5 which features 43% fewer components than the AGT-1500 will reduce both operation and support costs by a factor of three while increasing the MTBF by a factor of four (ie. by 400%). [1, 12]

To look at comparative figures, a tabular representation of some selected specs for both the T-80U and the M1 and their engines are given in table A3 and 4 of appendix A. [12]

1.4.5 Case study: Luxury liners and environmental friendliness

For the first time in the history of passenger liner operation aero-derivative type gas turbine engines have been introduced as some vessels' propulsion units thereby replacing the conventional diesel engines - just as diesel once replaced propulsion via steam.

According to the manager of advanced marine programs at GE marine engines Ohio Carl Brady, cruise ships are typically propelled by four to five 500 rpm, medium-speed, 8 to 10 MW diesel engine units. At the time of writing, six of the Royal Caribbean's Voyager and Millennium-class cruise vessels were each powered by a pair of LM2500+ GE marine engine gas turbine aero-derivatives - with a simple cycle efficiency of 39% and a single steam turbine unit per vessel. The LM2500+ gas turbine engine itself which is rated at 25 MW output power is an upgrade of the General Electric LM2500 aero-derivative featuring a 23% compressor airflow improvement at minimal combustor temperature increase. The LM2500 in turn is a product of the commercial CF6 and the military TF39 advanced turbofan family of aircraft engines. [16]

The waste heat from the aero-derivatives is used in the steam turbine unit which is coupled to generators to produce electricity for ship services such as water heating and air conditioning. Such an arrangement is referred to as the COGES, an acronym for 'Combined Gas Turbine and Steam Turbine Integrated Electric Drive System', and combined-cycle efficiencies between 45 - 50% can be attained with this type of cogeneration.

Another popular configuration employed in cruise vessels is the 'Combined Diesel and Gas Turbine configuration or CODAG in short whereby an LM2500+ is operated in conjunction with either two, four or even more diesel generator sets. [16]

Also, according to William K. Reilly - an Environmental Protection Agency administrator - turbine technology reduces the amount of environmental pollution in that turbines generate reduced amounts of sludge and oil waste compared to diesel engines and lastly but not least reduced air pollution levels. Nitrous oxide levels are reduced by 80% and sulfur oxides by 98% thereby eliminating the need of specialized exhaust and selective catalytic reduction equipment otherwise necessary for the diesel engine alternative. [44]

Furthermore, according to Brady of GE Marine Engines the reduction of the power density because of the remarkably higher power to weight ratios of gas turbine engines relative to the reciprocating alternatives enables a remarkable space and weight savings. The power density for a thermally and acoustically insulated gas turbine unit is approximately 400 kW/m^3 compared to 80 kW/m^3 for a medium speed, marine diesel engine. (To be noted is the fact, that although the power density for a modern aero-derivative can be as high as 1,500 kW/m^3 without an insulating enclosure, 400 kW/m^3 is still five times better than for an equivalent diesel engine unit.) [41] In summary, overall advantages of the above gas turbine COGES-arrangement thus include lower noise and vibration levels as well as remarkable space savings and enhanced environmental emission reductions.

1.5 Proposal Outline

1.5.1 Proposed Design Objective

The proposal thus put forward is the conceptual design of a modern gas turbine engine with all its improvements as a turboprop unit for light passenger aircraft thereby offering an alternative to the normally aspirated, reciprocating internal combustion engine which has propelled the average, light aircraft so far. It is therefore intended to propose a replacement unit for standard, normally aspirated engines with only slight modifications to the current airframes of light aircraft such as the Cessna 172 and the Piper PA-28 group of airplanes which require engine power ratings ranging from approximately 84 to 150 kW. The specific aim of this project is therefore to establish academic credibility for such an engine design - especially with respect to overall performance and maintenance friendliness. Manufacturing costs will only be touched on briefly; bearing in mind that the factor of scale effect would most likely reduce manufacturing costs of the previously underutilized production methods for currently more exotic types of propulsion systems.

1.5.2 Proposed Design Approach

To achieve the abovementioned objectives, a simplified turboprop engine has been proposed which will in essence be a small centrifugal gas turbine running a fixed pitch propeller via a single reduction step gearbox, indicating the need for a relatively low-speed engine (ie. 12,000 – 15,000 rpm). The power setting to the propeller in turn will be controlled by an adjustable, slightly modified hydraulic coupling of which there are many commercially available (ie. Voith tubo couplings, Vorecon multi-stage drives, etc.). For design purposes, the product design data of a commercial Voith Turbo-coupling will be integrated. The fluid coupling will regulate the propeller speed thereby allowing for a simple, fixed pitch propeller and a permanently engaged, single reduction gearbox layout, as such an arrangement minimizes the complexity and the amount of moving parts of the engine. A 'heavy' radial compressor or alternatively a separate flywheel-unit will additionally function as an energy storage device for an even torque supply and a constant engine speed even at low power loading. Furthermore, the overall dimensions of a conventional piston engine compartment with a standard 150 kW engine appear to be wide enough to accommodate a compressor-wheel diameter of at least 450 mm, which would be sufficient as an initial estimate to allow operation at a low enough rotational speed (ie. specifically 12,500 rpm).

The feasibility of such an engine design will be demonstrated mathematically by the use of a Microsoft Excel spreadsheet program to back up assertions with the following calculations:

- Thermodynamic and compressible flow equations for engine behaviour augmented with guideline-design data [14, 36, 47]
- Gear design [38]
- Rigorous Heat Ratio and Specific Heat determination [47]

Regarding the software, the advantages of using Microsoft Excel as the modeling tool specifically are in essence:

- Low software-costs compared to advanced, specific modeling software

- Excellent availability as an MS Office add-on to Windows distributed worldwide
- Adaptability and flexibility of the software program by anyone familiar with Excel

For proper analysis, aircraft specifications of models currently in service will be utilised for comparison purposes namely the Pilatus 'ASTRA' propulsion system, APU and GTS data where available, selected turboprops and some helicopter turboshaft engines. The computations employed for such optimizing calculations are freely adjustable to either ISA (International Standard Atmosphere: 101.325 kPa at 15°C) conditions as well as to local conditions pertaining to high altitude-airport requirements, posing some of the most severe operating conditions for aircraft (except excessively cold weather conditions); with their low air densities and high static, ambient temperatures. In fact, design altitudes 1,800 m above sea level equate to an equivalent density altitude of about 4,500 m on a fine, hot day in Johannesburg thereby making proper engine design somewhat more critical for local operating conditions. (It is generally understood, that many aircraft such as the first German all-metal plane Junker JU-88, the French supersonic transport Concorde, the Russian Mig-29 fighter plane and even motor vehicles such as the latest in for example VW motor car makes have been, and are still being performance tested domestically because of such extreme local operating conditions.)

For calculation purposes the spreadsheet utilised features an intrinsic failsafe-design philosophy ruling out major runtime errors usually common with any spreadsheet usage of such a nature. Therefore, by changing any specific design variable the program produces an updated, optimized set of design data useful for further investigation and modeling.

2 ADVANTAGES AND DISADVANTAGES OF GAS TURBINE ENGINE SELECTION

2.1 Advantages

2.1.1 Simplicity

From section 1.3 it should have become apparent that the functioning of a gas turbine engine is principally much simpler than having to consider the reciprocating stress behaviour of each and every working component in a reciprocating engine. For each cylinder assembly the piston arrangement is in essence comprised of seven basic functional components held together mechanically which all reciprocate and move vigorously. Stress levels induced due to high, cyclic pressure and force loadings on each component are therefore relatively high and this is discussed in section 1.3. Additionally, because of the comparatively large number of individual components piston engines are generally comprised of – i.e. about 50 parts - the probability of failure of one specific component is relatively high compared to two or three evenly rotating components in an equivalent gas turbine arrangement.[39]

2.1.2 Smoother engine running characteristics

Engine torque fluctuations in the reciprocating, internal combustion engine are caused primarily by tangential force variations, which are acting on the crankpins imparted by the pistons during the power stroke. Because only the fourth stroke in a 4-stroke engine supplies power to the crankshaft while the other three strokes (ie. the suction, compression and exhaust strokes) require marginal power input, the cyclic nature of such an engine should thus become apparent. Although the engine's flywheel levels out torque fluctuations to a large extent it exhibits significant mass, adding stress to related engine components. Alternatively, an intrinsically smooth running engine not requiring a flywheel such as a GT engine which produces a constant amount of torque with minimal torque fluctuation imparts much lower peak stresses onto its transmission system and all of its related components.

On the contrary, the gas turbine engine due to its rotational nature and continuous combustion process is largely exempt of such additional stresses. This intuitively prolongs engine and component life and reduces service and overhaul demands significantly. [21, 39]

2.1.3 Increased reliability, extended engine and transmission life

Recent advances in microturbine technology facilitate the incorporation of HEV systems into commercial transports such as passenger buses. With reference to the 'Capstone' marketing website of their latest HEV driven urban buses, internal combustion engines (with their respective generators) have a life expectancy of 12,500 hours. In comparison the 'Capstone' microturbine can operate for at least 20,000 hours. Also, observing that the 'Capstone' microturbine has no fluid storage, replacement and disposal requirements for lubricating oils and antifreeze unlike in piston engines improves its reliability further. The same applies to the external, thermal management system requirements (such as a radiator) which are absent in the microturbine but not so in the piston engine arrangement. [10]

Due to the even running characteristics and their subsequent operating behaviour as mentioned before, it stands to reason that component life of transmission systems, couplings and gear trains which are driven by the gas turbine engine are subsequently all improved upon.

2.1.4 Lower maintenance costs

The inherent simplicity and even running characteristics of GTE translate in comparatively lower maintenance demands. According to the 'Capstone' marketing website, service costs are reduced by 70% compared with conventional power plant.

Additionally, the time between overhauls (TBO) for piston-driven Lycoming engines are a recommended 2,000 operating hours while for the Pratt & Whitney PT6A range of turboprop engine the basic TBO varies from 3,500 – 4,500 hours for the 500+ SHP range, and up to 7,000 hours for some 1,800+ SHP engines.

This maintenance friendliness became apparent in both the Dakota DC-3 upgrade as well as in the case study on the Russian MBT GTD-1250 gas turbine engine (See appendix A). The U.S. LM-2500 marine engine only requires an expected hot section maintenance repair interval of 12-15,000 hours on its current production units. For every 10,000 service hours only 40 hours of corrective maintenance are required. [10, 16] For a relatively far less complex engine such as proposed by this particular report, lower maintenance costs still can - and should - safely be envisaged.

2.1.5 Better performance per unit weight

One key feature of the air breathing gas turbine, especially in aircraft propulsion, is its superior thrust to weight ratio compared to any reciprocating rival in its class. A gas turbine driven turbo-prop engine generates about twice the propeller thrust power than an equivalent mass reciprocating engine would. Although turbojets themselves exhibit very high SFC figures, a turbo prop engine benefiting of both the higher power to weight ratio and the high propeller thrust efficiency is not only an option, but a solution already employed for decades in the aviation industry – despite the added weight of a speed reducer such as a gear box. Such superior thrust-to-weight ratio trends are attributable to various factors:

- Due to the continuous operating characteristics the critical working pressures in gas turbine engines are kept relatively low – referred to in section 2.1.2. Such behaviour lowers engine-block structure and component strength requirements, thus yielding a lighter engine and turbojets are about 1/3rd in weight with respect to an equivalent size piston engine.
- On account of the absence of reciprocating components which present cyclic inertia limitations and stresses, higher operating speeds are possible thereby improving on the possible power which can be produced.
- As a consequence of the absence of high levels of vibration, such stress factors can largely be ignored thus reducing the supporting airframe weight of the aircraft.

Some figures in chapter 5.1 depict performance level trends of various engine types, and the improved thrust and power to weight ratios of gas turbine units with respect to the piston driven configuration in general is perceivable. A note of caution here though: Diesel engine designs with their inherent reliability - featuring much lighter but stronger hi-tech materials and better designs - are in the not-so-distant future potential entrants into the aviation engine niche market.

2.1.6 Design flexibility

Although not immediately apparent, because of the far reduced, overall number of components (approximately 1/3rd of such in comparable reciprocating variants), the production process requirements are reduced and therefore less time consuming. Because each specific component is fulfilling its own specific function unlike in the piston engine scenario, gas turbines are more straightforward in principle and therefore faster to design, test and to manufacture. Another spin-off due to its inherent nature is the relative ease, with which the GTE can be either scaled up or down for changing, and specific power demands regarding preliminary engine matching designs. The reason for this is once again the specific nature of each component which is therefore easier to analyse and adapt – unlike with the moving components in a reciprocating power plant.

2.1.7 Continually advancing microturbine and gas turbine engine technology

Concerted efforts are being made in developing the microturbine concept with respect to performance, as well as lower emission levels for a host of possible future applications. Even spin-offs in the aeronautical and even the commercial gas turbine R&D fields will eventually also positively influence microturbine development. Some such advances along with progress achieved to date are briefly discussed below:

Recuperators: Whenever employed in modern-day microturbine applications, recuperator technology enables gas turbines to be run at far more manageable SFCs inasmuch that they are increasingly becoming a desirable alternative in small-scale, distributed power generation. Although an unwanted weight addition at large, recuperator design is already being incorporated into aeronautical concepts: M-Dot Aerospace, a U.S. based company under government funding recently developed a 94 hp twin-spool turboprop engine called the TPR80 for UAV application purposes. Their specific design can be produced with or without a recuperator. This indicates, that recuperated aero-GTEs are not an issue of the future any more, and it is this very technology which will ultimately facilitate the expansion of gas turbine approaches into the various fields of transportation on an economically sensible and sustainable, large scale. [28]

Foil Air Bearings: Different types of patented air-bearings in the Microturbine industry further enhance mechanical reliability and turbine operating efficiencies, thus resulting in machinery which is on par or even superior in many aspects to the reciprocating alternatives with respect to performance criteria. Having been employed and improved upon since the '60s this well proved technology holds great promise for modern APU/GTS, GST, ACM and other MTB applications. The reversed multilayer journal bearing concept for example offers both a high degree of rotational stability while the additive relative movements between foil and shaft produces high levels of Coulomb damping, thereby mitigating shock loads. With a tenfold reliability improvement since 25 years ago, foil air bearings in general offer improved bearing load capacities, less wear, better full range operating stability and improved reliability compared to the conventional ball and race journal bearings. Thus, engine life is improved upon while furthering efficiency levels and it should be a matter of time before foil air bearings will be mass produced. [2]

Other Areas: Engine improvements also emanate from the maturing, commercial turbofan market. The latest in GE turbofan design such as the continuously evolving GE90 family of engines features the most powerful civilian turbofan to date. The GE90-115B rated at 115,000 lbs (52,163 kg) of thrust not only incorporates all-composite, 3rd dimension pin reinforced carbon fiber fan blading, but also the development of stronger, new mid-fan shaft material. Further areas of improvement incorporate airblast fuel nozzles, single crystal turbine blade materials and advanced HPT active clearance control optimization methods. Pratt and Whitney with their latest PW4000 turbofan on the other hand incorporate key technologies such as hollow, shroudless fan blades featuring a no life limit, segmented floatwall combustor liners, radial gradient turbine vanes, 2nd generation single crystal turbine blading and the latest in FADEC (Full Authority Digital Electronic Controls) technology. Additionally, controlled diffusion aerofoil technology is developed and implemented globally. [8, 41]

Materials: Microturbines themselves however are undergoing R&D efforts in order to improve their performance levels, reduce emission pollutants and eliminate inherent weaknesses. One of these disciplines is specifically in the field of materials development of e.g. smart ceramics and polymer-fiber reinforcements. Fiber reinforced ceramic matrix composites for example have been introduced commercially as early as 1999 in the Solar Centaur 50 engine as a low nitrous oxide combustor liner for a cleaner combustion. Special CNC-manufacturing techniques had to be adopted for this type of product.

A high priority is development taking place in high-temperature materials; also researched are monolithic ceramics such as the more creep resistant Si-Tu toughened silicon nitride applicable specifically in turbine vanes and their blades. Honeywell Advanced Ceramic Components recently publicised an advanced, creep resistant material labeled as AS-950 [20]

Software Development: The latest in software development aiding in the design for the latest in the GE90 class of engines features the application of 3-D high pressure compressor aerodynamics, and CFD software in general is increasingly strongly employed for very accurate fluid dynamics behaviour modelling. Emphasis is placed on, for example lean-burn combustors featuring advanced controls; some European programs focus on advanced computational methods specifically. Software and electronic design also feature highly in the newly implemented FADEC modules which optimise engine running conditions continuously thereby improving cycle efficiencies, and are in particular implemented in the more recent generation of GTE developments. [8, 20]

2.1.8 Environmental friendliness

With reference to the above case studies involving environmental implications as well as emission friendliness, inherent environmental advantages of gas turbine engines should with respect to impact levels have become more evident. Nitrous oxide levels are reportedly reduced by 80% and sulfur oxides by 98%, but especially promising apart from reduced emission pollutants in, for example, large marine applications is the promise of thermal efficiency improving COGEN CCGT/CHP integration of systems. []

With respect to futuristic aircraft engines though, recuperator technology holds the promise of improving environmentally friendly emission levels even further than inherent GT technology already does. Due to recuperation's ultra low nitrous oxide pollution levels being generated, ie. ½ to 1/3rd of its equivalent diesel, propane or CNG-run counterparts, environmentally friendly engines should futuristically become feasible in an ever widening variety of applications.

2.1.9 Multiple fuel-type flexibility

Not so obvious is the flexibility in fuel types which gas turbine engines can be adapted to operate on. With only minor modifications the intrinsic simplicity and continuous nature of the gas turbine engine enables about any type of liquid or gaseous fuel from methane gas to high flashpoint jet fuels to be burned. It has been established in the '80's that fuels with higher calorific values compared to for example methanol like kerosene and gas oil managed to fuel gas turbine engines more effectively. [51]

The conventional diesel or petrol engine on the other hand would be confronted with a problematic situation involving pre-ignition (backfiring), overpowering/under-powering as well as mismatched carburetor designs - just to name a few incompatibilities. A good example of the contrary would be the ATG-1500 MBT gas turbine: Primarily run on kerosene or diesel based fuels such as DF-1, DF-2, JP-4/5/8 or kerosene, in emergencies at the risk of engine damage for extended use however gasoline, avgas or mosgas can also be employed. [12]

2.2 Disadvantages

2.2.1 Higher specific fuel consumption levels

Higher thrust-to-weight ratios bring with them the consequence of higher fuel consumption levels noticeable predominantly at part load or sub-optimal power settings. Further, the nature of propeller driven equipment is such as to naturally facilitate higher propulsion efficiencies at moderate, sub-sonic air speeds. To operate an airscrew therefore with a gas turbine instead of a more fuel economic piston driven engine appears somewhat counterproductive unless some solution can be found with respect to the notoriously high fuel flow rates at off-design operation. Rightfully so, until the more recent fundamental advances in gas turbine technology were realized prospects of a gas turbine propelling light aircraft at mach 0.3 with cost effective components and minimal maintenance needs was deemed rather inappropriate. However, the possibility of a much less maintenance intensive powerpack with higher levels of operating reliability thereby promising savings in overhaul and maintenance costs might offset the higher fuel cost of the proposed gas turbine.

Furthermore, developments in material and recuperator technologies, the possible incorporation of a different design philosophy such as a free power turbine addition and the possibilities available for transmitting the shaft torque from the engine to the propeller are all rather valid contributing factors for researching the possibility of GTE propulsion instead. The fact, that most modern helicopters are turboshaft driven – including the small ones - should in itself suggest something about the viability of the gas turbine engine with respect to our own intents and purposes.

2.2.2 Initial capital outlay, manufacturing and production costs

Currently, gas turbine engines are more expensive to purchase than comparative counterparts: They require high-speed, heat resistant bearings and rotating components, which addresses problems such as imbalances, centrifugal stresses, metal creep and so on, and one requires state-of-the-art technologies and high-tech materials for their manufacture and production. However, once practical, operational feasibility for such an engine has been established such expenses would drop with the amount of units produced according to some factor of scale. In other words, as the demand increases, large-scale production requirements would reduce the unit cost making this technology more affordable. The necessary requirement however is a viable market demand with realistic money saving incentives when it comes to operating such a type of engine on a medium to long term basis.

2.2.3 Higher noise levels

High noise level concerns are a problem as long as research has not adequately achieved to solve this particular problem. Noise emanates, amongst others, from jet exhaust as well as interestingly from the compressor intake aperture. High noise levels have already been studied with respect to helicopter applications for decades, and certain countermeasures such as frequency-specific sound absorbent materials are already being implemented in different applications. Inverse frequency sound suppression devices, hydraulic reflective and absorptive accumulators and the like may all be employed in time, but such technologies require more research. In marine applications though, and mainly because of little weight concerns, gas turbine power packs are already enclosed within a sound insulating casing.

2.2.4 A phenomenon called 'humming'

As a result of the recent industrial, predominantly US trend towards distributed, small scale GST utility and industrial power generation, advances in the microturbine industry were both speedy and presumably just as uncontrolled. Manufacturing industries focus on product delivery for the starving, deregulating energy market rather than on trouble shooting still present, inherent problem areas and functional in-congruencies which are still apparent with this rather new and evolving technology.

One of these particular drawbacks is an effect called humming Due to new and stringent environmental regulations and also to specifically lower microturbine nitrous oxide (NO_x) emission levels, premixed lean combustion systems were being employed. As an unwanted side effect, combustion induced pressure oscillations were produced causing the equipment to emit an audible 'hum', a clear indication of the engine being subject to unacceptable, natural frequency levels of oscillation. As with any unwanted cyclic pressure frequency waves these can, if left uncontrolled and insufficiently damped, develop into the turbine experiencing catastrophic, internal cyclic stress levels not intended for the engine. The hum develops gradually into an audible howl which indicates progressive, internal engine damage leading ultimately to self-destruction. [2. 21]

Because the phenomenon of combustion induced oscillations is as yet not clearly understood, the end user of the equipment is strictly advised to rather adhere to reduced, external power demands in order to preserve the machine, and it is hereby apparent that solutions still have to be found and researched; suggesting that not enough R&D has gone into the current, existing designs as yet.

3 FUNDAMENTAL GAS TURBINE ENGINE TECHNOLOGY AND ITS OPERATION

Standard jet engines are comprised of five major components. These are the intake and exit ducts, the compressor unit, the burner/combustor section and the turbine assembly.

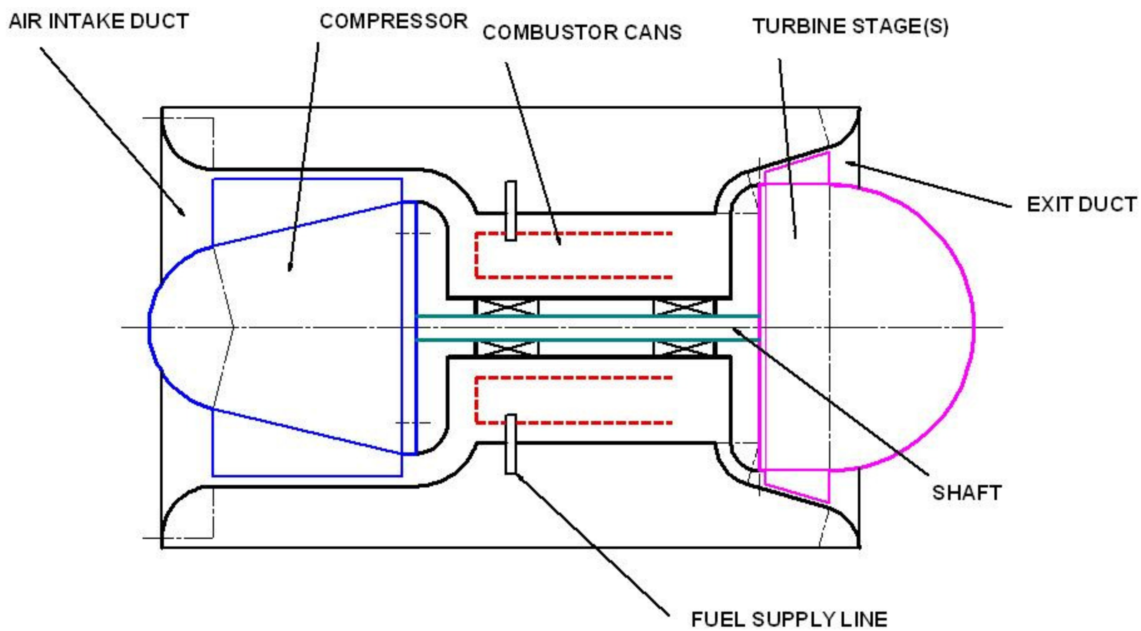


Figure 3.1 Fundamental gas turbine engine component layout

3.1 The Gas Power Cycle Process

A cycle is a process, which begins and ends at the same set of conditions. In this instance for example, the engine is an open gas power cycle, in other words a cycle which obtains its energy from a gas and which begins and terminates 'open ended' at its surrounding, atmospheric environment. The generally accepted, and thermodynamically sound, approach to the theoretical modeling of the gas turbine process is called the Brayton cycle. It is an open, constant pressure heat-engine process, and its operation is usually well explained in almost every heat engine text. In order to enable continuous heat engine operation without running the danger of embarking upon some nonsensical, perpetual motion approach certain requirements for operation have to be met. These incorporate specific, successive cyclic components necessary for the operation of any kind of heat engine, which are the following:

a) The working fluid is such a medium which facilitates heat transfer, absorption or release and containment of the different kinds of energy states which have to take place in a cyclic process. In gas power cycles the incoming air is the working medium and the quality and quality of oxidants present in it will contribute towards the quality of combustion which takes place, and consequently the amount of power which is thereby realised.

b) A heat source (such as the fuel burner) supplies the correct amount of (heat) energy required for combustion. Without the correctly matched energy supply no, or little, work can be produced. The nature of the fuel itself therefore plays a significant role and incorporates properties such as the amount of heat energy released per unit of fuel, its ignition temperature, the combustion rate and specific combustion requirements and characteristics of the fuel.

c) A heat sink which sheds 'unnecessary' heat from the cycle is also required: If the net heat transfer and the net work done are not numerically equal, then either no cyclic process can be established, or alternatively an already existing cycle will cease to function. In other words, a cyclic process stipulates that the overall, internal energy state of the working fluid for a particular cycle must be zero. In this particular situation the cycle is open ended indicating, that the surrounding atmosphere behaves as the sink. The intake temperature - or energy content - is atmospheric while the engine exhaust condition is well above the intake energy condition, and heat is rejected into the atmosphere.

d) A compression device (namely the compressor wheel), which supplies energy to the working air in the form of pressure thereby forcing the working medium into the burner section. This perpetuates the necessary energy exchange which takes place. The compression device needs mechanical work supplied to it in order to do the compression work and a significant shaft power component from the turbine output spool imparts into the compressor the necessary mechanical energy.

e) An expander (in this instance the turbine stage(s)) extracts as much as possible of the high energy content gas from the burner thereby converting expanding gases across its blades to shaft rotational momentum, and therefore into mechanical power. This device is in contact with the hot combustion gases exiting from the combustor. Its purpose is to expand the combustibles at high rotational speeds down to near atmospheric conditions for maximum propulsion efficiency. Its blade design is therefore critical for the net performance of the engine.

As explained under point d) above, the compressor-turbine stage turns the compressor via a connecting spool thereby transferring a portion of the power extracted by the expansion of the combustibles across its stage(s), while the remaining turbine shaft power attends to external load requirements. This is the method of how a single spool turboshaft gas turbine engine supplies its shaft power. By running the compressor which in turn pressurises the incoming air, the cycle is perpetuated as long as the inlet and exit conditions remain operationally favourable and as long as enough oxygen is available in the intake air and fuel is being supplied for combustion. Below is a graphical representation of the temperature-entropy diagram, which illustrates in thermodynamic terms the open-ended Brayton heat engine cycle:

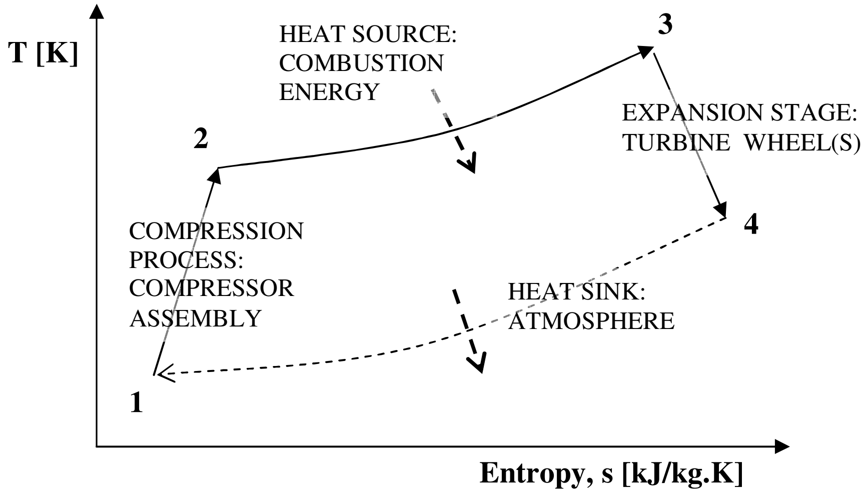


Figure 3.2 Brayton cycle T-s diagram

3.2 Component Description

3.2.1 Inlet duct and compressor stages

The incoming air enters the engine at atmospheric conditions and at the relative flight velocity of the aircraft. The inlet duct of the intake nacelle itself is usually designed – and especially in supersonic air-breathing engines – to retard the air velocity to well below sonic conditions before it enters the compressor wheel for pressurization. Some local pressure is lost in the ducting, and because of the retardation process the compressor wheel inlet state can be taken to be at stagnation conditions.

Different types of compressor designs exist, namely radial, axial and even mixed flow compressors which are a combination of axial and radial compressor stages incorporated into one design. Due to the applicability with respect to this paper, only the single radial (or centrifugal) compressor popular in small gas turbine engine APU's will be focused on. The reasons for such are simplicity of design and therefore reduced manufacturing and maintenance costs. Also a reduced overall length at the expense of a larger engine diameter results in a more compact, rectangular composed engine layout. Additionally, axial compressor blades can stall at low speeds thereby causing destructive, aerodynamic vibrations to take place which jeopardises the structural integrity of the compressor. [17, 47]

Although thermally less efficient a radial compressor has a higher compression ratio than an axial compressor unit. However, a two-stage centrifugal arrangement mounted in series can substantially increase the pressure ratio compared to a single wheel - with a moderate loss of space and simplicity yet an increased reduction in compressor efficiency.

Compressor efficiency deteriorates because of higher inlet pressures and temperatures entering the second compressor stage thereby limiting impeller exit conditions. This occurs due to temperature limitations of the high pressure impeller as well as elevated dangers of induced stall, especially at off-design performance. This is also the case in axial compressors.[52]

From a thermodynamic efficiency perspective, elevated compressor inlet temperatures also increase compressor work substantially, usually necessitating intercooling prior to the air entering the high pressure wheel which in turn adds an unacceptable level of complexity and additional, unwanted weight.

The centrifugal compressor wheel is designed to compress the incoming air to a pressure of approximately 4.5 times that of the intake pressure. For structural reasons, 4.5 is the maximum allowable pressure ratio for radial aluminum impellers, but can be improved upon if a titanium alloy impeller is used.[47] This proves however insufficient for larger turbojet propulsion plants which usually employ a continuous array of multiple axial flow compressor stages of varying dimensions resulting in higher thermal efficiencies at smaller intake areas.

To protect the centrifugal compressor vanes from corrosion, a special coating is applied. Additionally, some small gas turbine units feature bleed-air offtakes for both turbine blade and hot-end component cooling. In modern units, bleed-air can contribute to air bearing pressurisation purposes as well. The schematic diagram of a centrifugal compressor in figure 3.2.1 below highlights the basic components of such an intake assembly: [17, 47]

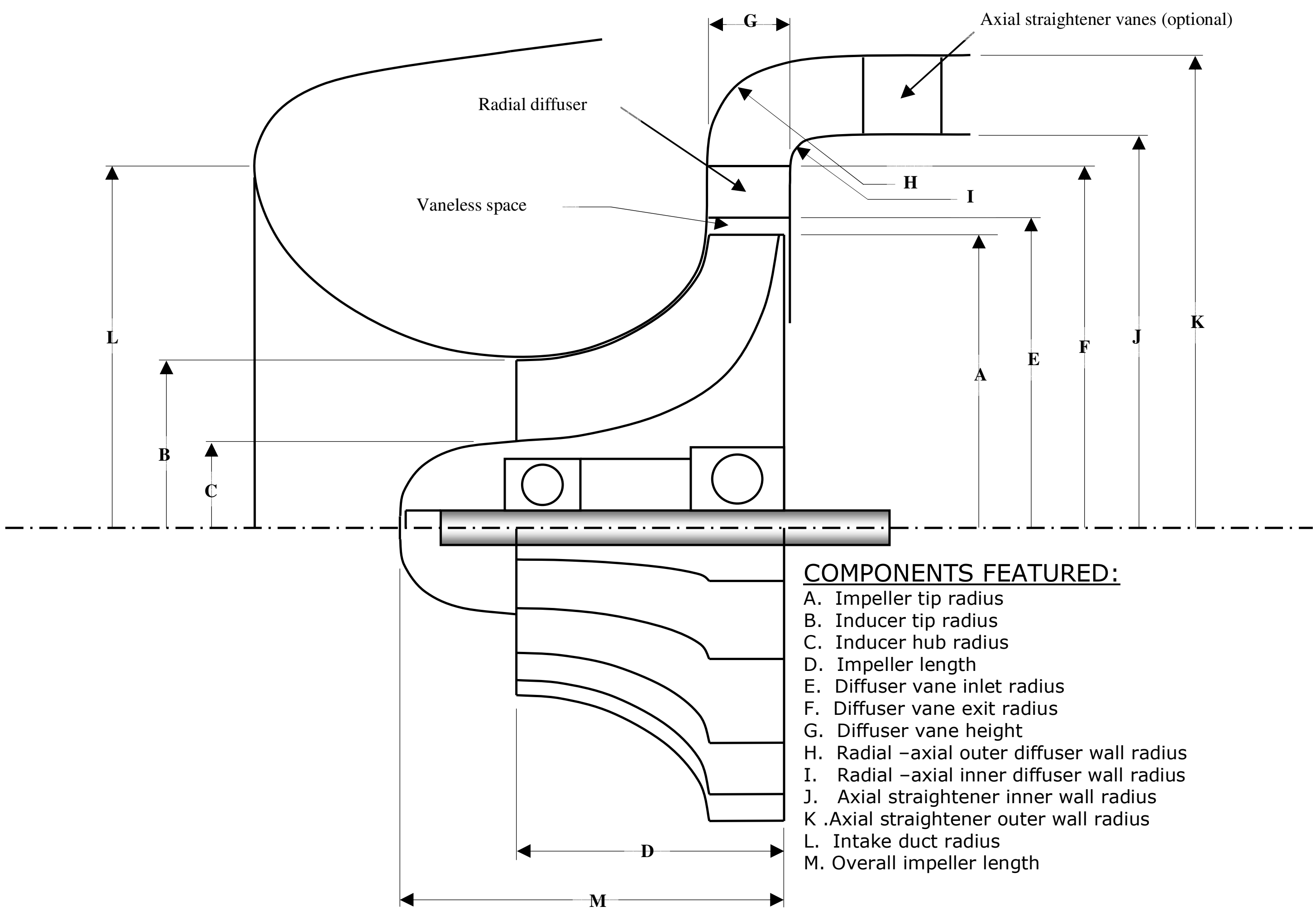


Figure 3.3 Radial, single stage compressor diagram

3.2.2 Burner / Combustion stage

The pressurised air (due to prior compression also at somewhat elevated temperatures) is supplied to combustion chambers at almost isobaric conditions. Burning fuel releases large amounts of heat energy and thus converts either to very high pressures or large volumetric (kinetic) expansions - depending on whether the system is physically contained (i.e. has a quasi-statically 'fixed' boundary or allows for continuous expansion. Thus rapidly expanding, kinetic energy is generated in the combustion process, which is thereafter utilised for the expansion process across the turbine blades via their respective stages, thereby converting chemical energy into shaft rotational power.

The combustion chambers themselves are composed of heat resistant metal sheeting called liners arranged in either can or annular type fashion. These contain strategically placed holes (ie. orifices), which allow the high pressure, oxygen-rich compressor air to enter.

Can-type combustion chambers are cylindrical in shape and are mounted singularly, in pairs or in larger numbers proportional to engine size and fuel supply along the outer radial periphery of the engine. As a consequence the overall engine diameter is increased. However, such an arrangement allows for easier access to the individual cans for service and replacement purposes.

The annular combustion chamber is preferable in smaller gas turbine and microturbine applications due to simplicity and complies with the configuration selected in this paper. An annular unit, which still enshrouds the axially situated spool(s), compacts radial dimensions of the engine but restricts access to the combustion chamber itself due to its shell-type outer wall design, which can only be accessed by dissembling half of the engine. [17]

The combustion fuel is injected into the chambers via spray nozzles at one particular end of the respective chambers. If the fuel is injected in an annular combustor in the reverse direction relative to the overall airflow, it would constitute a reverse-flow annular type arrangement. The fuel supplied via a gear or small piston pump is usually governed by a mechanical- or in more recent years by an electronic fuel management system because the exact amount is crucial for startup and acceleration in preventing surging and flameout within the engine. [17]

Although gas turbine engines can operate on almost any type of liquid and gaseous fuels available which is emphasized by the example on the MBT Avco Lycoming ATG-1500 gas turbine engine, one should design for jet fuel or avgas applications because of their availability at current airports. The ATG-1500, which can operate on diesel, kerosene based fuels, kerosene and even on gasoline (ie. petrol) and certain types of gases for a short period in the event of an emergency is because of this type of flexibility especially well suited for military operations. [12]

3.2.3 Turbine stages

Once exiting the burner section the hot combustion gases undergo expansion to lower pressures and temperatures across one or more turbine stages in the 'hot end' section of the engine. For micro-turbines in particular two specific turbine types are being employed:

For the axial flow turbine design, every stage is composed of both a stator ring and a turbine wheel, and both parts are each made up of oriented and curved blades designed to operate at one particular, optimal set of inlet gas conditions and at a fixed rotational speed. The degree of conversion of heat and pressure to useful torque via a kinetic energy increase and/or expansion at the expense of local pressures across a turbine stage demarcates the degree of reaction of that particular stage. In a mixed impulse/reaction turbine stage for example both partly kinetic and partly pressure energy are converted from the hot exhaust stream to rotational shaft momentum, and these occur due to a change of direction and due to gas expansion (i.e. acceleration). These take place in a controlled manner because of operating-specific profiling of the blades in the stator and rotor sections, which are operating co-dependently. Ultimately, whatever the degree of reaction the final, or only, gas turbine wheel expands the hot combustibles entering it down to almost ambient pressure. The slight pressure difference accounts for any possible exit duct losses, ensuring that for fixed area exit nozzles the combustibles exit into the atmosphere fully expanded, thereby maximising the propulsive efficiency.

The radial inflow turbine wheel is almost the inverse of the radial compressor wheel design inasmuch operation is concerned. Once redirected across a nozzle ring onto the intake throat of the turbine wheel, the hot combustion gases expand and travel both first radially inwards and then axially in a mixed-flow type of flow path. This converts the change in tangential flow direction to rotational momentum and thereby generates shaft rotational power.

A multi-stage axial turbine construction however offers higher expansion efficiencies at the expense of smaller, mechanical tolerance limits and more intricate turbine blade design. Axial turbine stages are usually employed in APU's and subsequently in all larger gas turbine machinery - with the limitation that for APU's and similar a maximum of only two turbine stages are incorporated into this paper's particular design.

For turboprop applications whereby the propeller is of fixed pitch nature, it is usual to operate a propeller at speeds ranging from zero to approximately 2,500 rpm for maximum propulsive thrust. Because a hydraulic coupling takes care of the speed variations it is imperative to supply a constant shaft speed at optimal torque which is therefore also the GT core engine design speed. Because gas turbines usually run at inhibitively high speeds requiring well needed, mechanical gear reducing equipment such as a mechanical or planetary gearbox, it is sometimes conventional to include with the core gas turbine a secondary turbine called a power turbine. As a consequence, a bi- or in larger applications even multiple shaft arrangements are possible.

While the first, so-called compressor turbine is direct-coupled to the compressor wheel via its own shaft and is designed to expand just enough of the combustion gases to keep the compressor powered and turning optimally, the power turbine on the other hand operates somewhat independently. Being direct-coupled to the external load (usually through its own gearing) onto its own, separate output shaft, the power turbine expands as much of the remaining gas as the compressor turbine did not need to. Usually being of larger diameter, the net result of such is a lower output-shaft speed while featuring an increase in supplied torque. The compressor-turbine arrangement therefore behaves entirely as a gas generator for the second, larger diameter power turbine, and in this particular application would drive the propeller via a now far more manageable gear reduction. This type of 'gearing' is nowadays predominantly found from miniature gas turbine turboprops for remote controlled model aircraft enthusiasts to the vast majority of aviation turboprop and turboshaft requirements such as found on UAV's, on most helicopters and even on the Pilatus 'Astra' airplane. Some disadvantages of such an approach are:

- Increase in core engine size and weight,
- Increase in manufacturing costs due to an additional turbine and its inter-turbine ducting,
- More complex power turbine speed-control system requirement.

Another deciding factor of whether a free power turbine should be employed is the application requirement of the engine itself. Because a jet engine produces almost no torque below almost full operating speeds, thus premature external load applications especially in single spool designs can cause overheating and thereby severe engine damage. Because an additional load manifests itself as an elevated exhaust gas temperature (EGT), it is therefore advantageous to be able to permanently monitor temperature increases above the norm. Thus, if a single spool turboshaft is employed the external load can only be applied once the engine has reached its self sustaining condition at about 25 – 30% of its rated speed, thereby making auxiliary equipment such as gear reduction and a high-speed or centrifugal clutch an unwanted necessity. The free power turbine arrangement on the other hand does not require such equipment, because the free power turbine, which carries the load can remain permanently attached to its external load such as the LP compressor spool of a larger aviation GT engine without necessarily interfering with the critical startup procedures of the gas generator itself. It therefore stands to reason, that gas turbine starters (GTS) or jet fuel starters are all of twin-spool, free power turbine design, while APUs' are in general of single-spool arrangement. While it is therefore not possible to utilise an APU as a starter unit without some serious, auxiliary modifications, some GTS units can once fitted with power turbine governing systems be easily employed as APUs and even GST's (i.e. gensets) driving loads such as AC generators. It is therefore common practice in the aircraft industry to utilise GTSs as APUs as well once the main engine reaches its self-sustaining level. Should a variable or even no load at any stage be applied to the free turbine shaft though, a mechanical governor is usually necessary for the prevention of overspeeding of the free power turbine. [17]

As an endnote, it is worth mentioning, that turbine stage design is of paramount importance in determining nozzle guide-vane temperatures (NGTs) and therefore engine performance limits. Turbine wheels, discs and rings are thus generally manufactured from heat resistant materials such as Nimonic or crystallized, high strength steels and their alloys in larger applications. [17]

3.2.4 Exit duct

Depending on the type and size of an engine, the exit or hot end portion is specifically designed in meeting certain needs in conjunction with important and particular design parameters as supplied by that particular type of power plant. A brief description of selected types of exit duct with their respective engines of interest are supplied below:

(Note: In turbojet and some military turbofan engines, thrust generation (also labeled thrust augmentation, afterburning or reheating) is predominantly achieved separately by incorporating an additional section of afterburning tailpipe into which additional fuel is being injected and burned. This method of thrust augmentation can generate between a 50–80% increase in thrust and is largely feasible because of the rather lean fuel to air mixture exiting via the turbine(s) from the burner section, thereby leaving a lot of unburned oxygen behind. As a side effect, much higher (typically doubled) specific fuel consumptions have to be tolerated, and therefore limits the use of afterburning.

In order to achieve reheat, the afterburning tailpipe houses the fuel spray bars which inject the fuel, and a set of flameholders which stabilize and restrict this secondary combustion to within the temperature resistant reheat area before expansion across the variable geometry exhaust nozzle (often a 2-3 position nozzle) takes place. When accelerated across a variable area exit nozzle down to atmospheric pressures, the gas particles are sped up considerably, and additional thrust is thereby generated. This approach to high speed dashes and to necessary take-off thrusts is due to the high rate of fuel burning slowly phased out of modern, military fighter aircraft technology, and advances are being made to attain such high levels of thrusts via alternate yet rather ingenious technological means beyond the scope of this project.)

For turboprop engines, the exit duct is generally designed to aid in thrust augmentation apart from the propeller whereby not all the combustion gases are expanded across the last turbine stage because the remaining expansion is left to an exit nozzle; just as achieved by a turbojet power plant.

For turboshafts as in the case of this particular proposal, the combustion gases are converted as much as possible to shaft power in that the waste gases do principally not supply any additional thrust augmentation. For this particular reason, the exit duct is not designed for expansion but rather for diffusion to atmospheric conditions with minimal exit pressure losses. In helicopter applications, such exhaust gases are sometimes (re)directed upwards into the main rotor blade slip-stream for achieving a reduced heat signature. However, the still relatively ‘warm’ exiting gases (ie. +/- 500 °C) should not be allowed to inhibit any other feature and/or functionality of the aircraft’s design.

3.2.5 Advances in auxiliary equipment, bearings and lubrication

To support the shaft as a rotating unit adequately, some distinctive solutions have been employed for years while more modern solutions have only begun to gain acceptance. The ball and needle bearing and the preloaded, angular contact bearing for example feature two distinct types of traditional, high-speed spool supports.

Both types need adequate lubrication achieved typically via jets, or orifices, within a closed circulating oil system. An oil pump usually of the gear type pressurizes the oil supply and returns it to its supply reservoir. This is achieved either under its own gravity or via a scavenging process entailing a secondary pump of larger capacity. As a consequence, more effort for oil circulation is required. Effective filtration ensures the sustained absence of any abrasive, particulate matter. Oil systems are either of dry sump or wet sump design. [17]

Oil sealing is accomplished by either labyrinth seals or by spring loaded carbon seals. The oil itself utilised is usually of synthetic nature and has to be of specific grade and viscosity. Useful specifications for example are the MIL-L-7808 and MIL-L-23699, which respectively feature viscosities of 3 centistokes and 5 centistokes respectively. [17]

For this report's design, oil seals are implied which simplifies calculations with respect to power losses due to oil circulation. However, some auxiliary power is still required for air pressurization. This contribution though has for brevity's sake been ignored. A more recent approach though is air-foil technology already implemented with great success in some larger, commercial engines.

Gas turbine starter motors for APU-sized engines and similar are fundamental in spooling the engine up to a self-sustaining condition. These are usually intermittently rated, heavy-duty electric motors, pull chord starters, hand-cranks or even of hydraulic nature. Ultimately, the engine needs to attain its light-up speed of about 10-20% of its rated rpm at which point the spark plugs begin firing and the engine becomes self-sustaining. [17]

For initial fuel ignition to take place, a high-energy ignition system is usually employed. A capacitor which is charged to a voltage of about 3 000 volts by a solid-state DC inverter stepping up a 24 volt supply discharges into the spark plug in which a special, sealed spark gap facilitates the corona discharge process. [17]

A high-energy spark breakdown at the rate of one or two per second ignites the jet fuel such as kerosene (called paraffin in the USA). Automotive-type ignitions are sometimes used as ignition systems whereby a high voltage (about 20 000 – 30 000 volts at a low current) discharges across a conventional spark gap. When placed in the path of a torch igniter, it provides a flame adequate in igniting the fuel. Hand started engines make use of a small generator. [17]

4 FUNDAMENTAL ENGINE AND TRANSMISSION LAYOUT

4.1 Gas Turbine Engine Type Selection

The fundamental engine-transmission system layout is depicted in figure 4.1 below featuring the gas turbine unit, which in turn is coupled to the flywheel and the mechanical gearbox. A hydraulic, variable (i.e. speed-decreasing) transmission propagates the torque further and enables the engaging and disengaging of the fixed-pitch propeller.

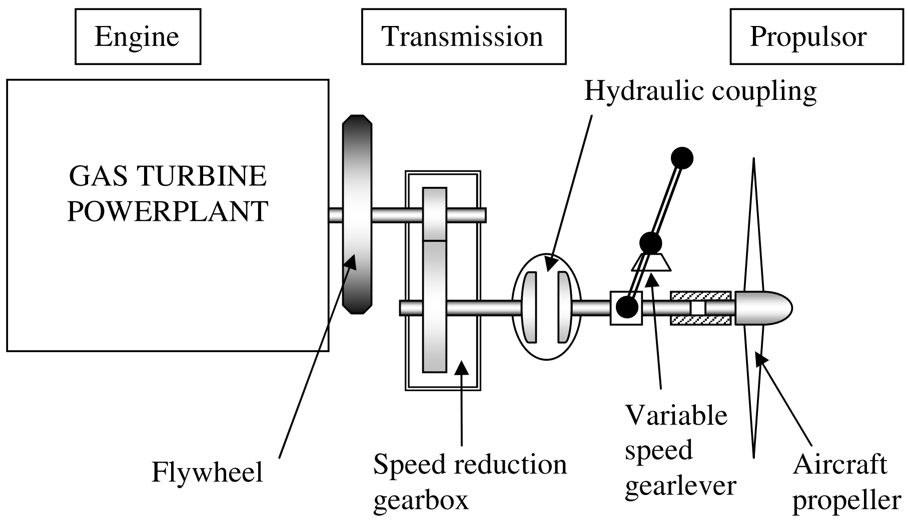


Figure 4.1 General transmission layout

The actual GT engine configuration is a function of essentially three specific proposals: Either a single spool or twin spool are to be selected in conjunction with, or without, a recuperator as is outlined below:

4.1.1 <u>Single spool design</u>

The engine itself is of simple cycle, radial compressor and single turbine stage arrangement whereby the necessary compressor power is derived from the turbine itself. This particular arrangement is called a singe-spool design – with both the compressor and the turbine being fixed onto the same shaft and both subsequently rotating in the same direction and at the same speed. After absorption of the majority of generated turbine power by the compressor wheel, the remaining or 'excess' power available after operating losses have been taken into account is the engine output shaft power. This, in turn, is transmitted through the mechanical and hydraulic transmission system to the propeller. All the combustion gases supplied by the burner are thus expanded to ambient pressure conditions across the single turbine-stage available. A 'heavy' compressor wheel design incorporates the energy-storing flywheel.

4.1.2 Gas generator-free power turbine

The engine is composed of two basic constituents. The gas generator is made up of a radial compressor-compressor turbine arrangement, while the free power turbine is attached to its own, free shaft (or spool) leading to the adjacent transmission system. Both turbine stages are associated with each other via an inter turbine duct, which transports the exiting compressor turbine combustion gases to the power turbine inlet vanes. In this instance, the compressor turbine expands the combustion gases just enough to power the radial compressor and overcome internal friction. This combination referred to as the gas-generator supplies the free power turbine with the remaining, partly expanded combustion gases. It is these gases, which by expanding to ambient conditions impart the output shaft with shaft rotational power. For this particular arrangement, the design and manufacture of an additional turbine-stage together with its inter-turbine ducting is required, and the flywheel is integral to the free power turbine rotor.

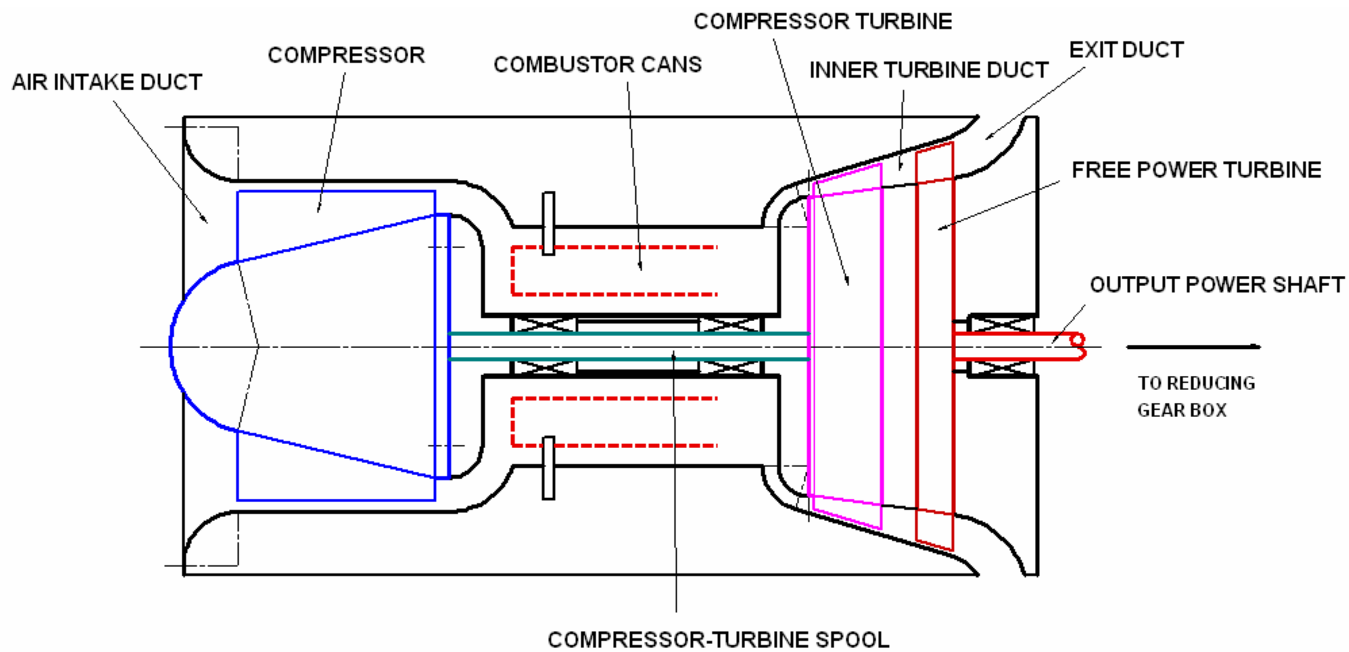


Figure 4.2 Twin-spool gas turbine engine layout

4.1.3 Recuperators

A recuperator - or an air/gas heat exchanger - could be incorporated into the design with either of the two previous arrangements. Such a device is common in ground-based gas turbine technology, because its function entails the improvement of the specific fuel consumption (SFC) of the engine by transferring heat from the exhaust duct to the compressor air prior to it entering the combustion chamber. Unfortunately this process reduces the output power of the turbine because of preheated air entering the combustor, thereby reducing fuel flow and consequently the chemical fuel-energy available. Conversely, the reduced fuel-burn improves the operating efficiency substantially by reducing the SFC to more acceptable levels. For power generation specifically, and even compared to the diesel- and piston engine equivalents with already competitive SFCs', recuperator technology enables gas turbines to operate economically competitively. This explains partly the sudden demand in microturbine technology for deregulated power generation in the last decade in the USA.

With respect to aircraft propulsion units though it has to be mentioned, that futuristically the recuperator emerges to be a promising necessity, and within the scope of this project it will be regarded as such. On the contrary, it has to be borne in mind that recuperators add weight to the aircraft, and in the past lacked the high levels of efficiency necessary to validate such a component. Recuperators are only effective under quasi-static, consistent engine-rating conditions - which does not entirely hold true for aircraft engine applications especially with varying altitudes and fluctuating, ambient weather conditions. Breakthroughs however are already being made and are bound to continue with technologies such as flow control regulator incorporation, futuristically improved and lightweight materials as well as superior heat exchange surface designs. One particular example in mind is the M-Dot Aerospace's 94 hp, TPR80 twin-spool turboprop engine which features a recuperator, but is developed specifically with airborne UAV applications in mind. [28]

4.2 Power Transmission Auxiliaries, and their Design

4.2.1 Engine energy storage and torque transmission

The gas turbine engine is inherently a constant torque device which does not need a flywheel energy storage device. However, when control and stability of optimum engine speed and higher torque transmissions during external load engagement have to be considered some sort of flywheel would prove advantageous. Considering such, still no undue excess strain is placed on the core gas turbine when sudden, higher power demand levels are imposed. In essence, the flywheel:

- maintains optimal core engine speed,
- absorbs the majority of the initial load suddenly imposed onto the engine,
- facilitates, as a result, a more gradual, controlled fuel supply for an increase in fuel burn.

For this particular reason, a 'heavy' compressor (fly-) wheel has been proposed in this design implying the following advantages:

- improving the transmission efficiency,
- saving space,
- reducing overall engine complexity by eliminating the need for a separate, external flywheel assembly.

Such a device is ideally operated in conjunction with varying propeller loading achieved by the degree of engagement or disengagement of the propeller by the variable, hydraulic coupling which has also been proposed. It has to be remembered, that engine speed must remain optimal, even when the load and/or the fuel supply vary. After reengagement of the propeller the power in the flywheel would largely be absorbed initially while additional fuel burn maintains an optimal engine speed. Increased engine power is necessary in order to maintain optimal engine running conditions, because power absorption from the flywheel energy during torque transfer has to be supplemented by added fuel burn from inside of the engine.

At lower power demands during, for example, aircraft descent the flywheel would keep the gas turbine engine running optimally without the need of excess fuel to be supplied. Should the engine be designed on a free power turbine principle, it then stands to reason that the flywheel be incorporated into the free power turbine rotor. This would result in a lighter flywheel because the power turbine is intrinsically larger in diameter than the gas generator turbine. Because an increase in diameter translates into a lower flywheel weight for the same the speed decreases hence increasing the torque output, the 'flywheel' weight might be decreased even more thereby yielding an even lighter engine.

A noteworthy advantage of incorporating a flywheel into gas turbine design is the relatively high operating speed. The proposed design has to operate continuously at one specific speed and at much higher rpm's. This calls for a much lighter flywheel and as a consequence reduces overall engine weight. Due to high centrifugal forces though, great care has to be taken in the dynamic balancing of such a component. Any imbalances within the engine at such high speeds could cause tremendous cyclic stresses leading to induced natural frequencies of vibration, degradation of blade tip clearances and bearing life.

4.2.2 Gear reduction

Speed reduction between the prime mover output shaft and the driven propeller shaft is necessary to reduce the relatively high engine speeds down to approximately 2,500 rpm at best - with a maximum output torque - in order to turn the propeller effectively. As the proposed engine is not running excessively fast but is rather comparable in size, performance and in power output to standard APUs', a single 4:1 or 5:1 spur gear reduction would suffice. In the case of a free power turbine being employed in which the power turbine is already rotating at a reduced speed however a lower gear ratio may be used. If designed appropriately, gear reduction might even prove unnecessary altogether, reducing overall engine size and saving on complexity with the result of making the unit more reliable and more easily adaptable to smaller space requirements.

4.2.3 Variable, hydraulic power transmission

In order to engage and disengage the propeller for the sake of varying levels of thrust - bearing in mind that the airscrew is of a fixed-pitch type – a variable speed hydraulic coupling has been proposed. In essence, a hydraulic coupling is analogous to a torque converter, and great care has to be taken in component matching in that the hydraulic coupling is not allowed to be too heavy, too large or too inefficient at the required operating condition. A lot more goes into the design of such coupling taking torque-demands into account and the like, but for this proposal an adequately matched device on a conceptual level can give a very good idea of very specific coupling requirements as applied to aircraft propulsion later on. One objective would obviously be weight reduction without loss of either functionality or reliability of the component.

As the basic Voith "Type T" turbo coupling is lightest and most compact in its class it will be considered as a slightly modified, variable-speed version for this particular case.

In general, the T-type incorporates the following features:

- Basic design of all turbo couplings with constant fill.
- Application mainly in drives where overload protection and damping of torsional vibration is required.
- Operating fluid usually mineral oil.

For the sake of component matching, the supplier's tables will be consulted. [45]

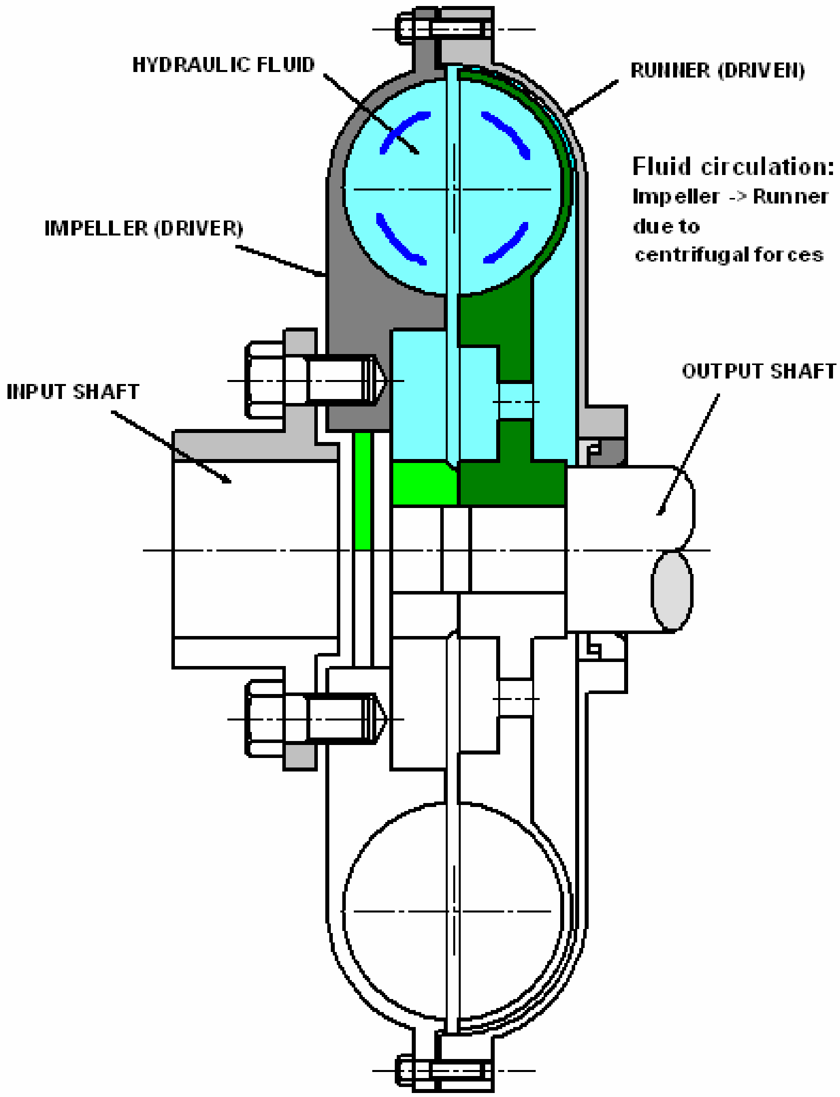


Figure 4.3 Schematics of a Hydraulic Coupling

5 ENGINE DESIGN PROCEDURE

In this chapter comparative aero-engine performance tables are displayed, the engine design procedures presented and general input variables discussed.

5.1 Comparative Aero-engine Tabulations

The following data sets are for illustrative purposes only with the emphasis on specific data trends rather than on individual values. However, because of the extensive diversity of data sources incorporating various trustworthy websites overall trends can be treated with optimism because distinctive trendlines do visibly exist. Furthermore, random errors in a reasonably large data set tend to average out with only marginal loss in accuracy. Subsequently, the following data can be treated as a rough yardstick for estimates of overall performance figures for the present proposal, and emphasis is placed predominantly on shaft power produced in conjunction with specific fuel consumption (SFC).

5.1.1 Aero-engines in general

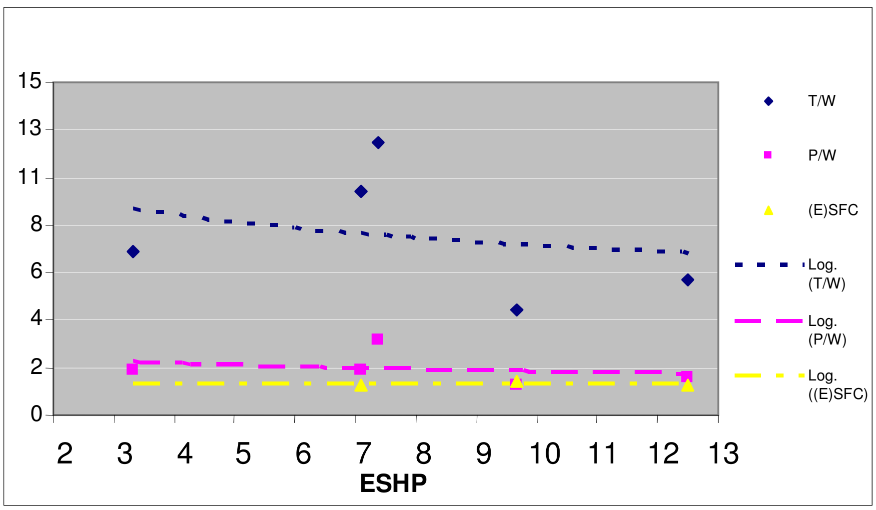


Figure 5.1 Graph of power, torque and SFC versus ESHP for miniature gas turbine engines

From the above graph it is apparent, that:

- SFC is about 1.3 kg/kW.hr common for about all known and tabulated engines.
- The power to weight figures also appear to be consistent at about 2:1 overall.
- ESHP's are in the region from 3 up to 13 for larger units.
- Numerically, the P/W ratios are on average about 1.5 times larger than the specific fuel consumption values.

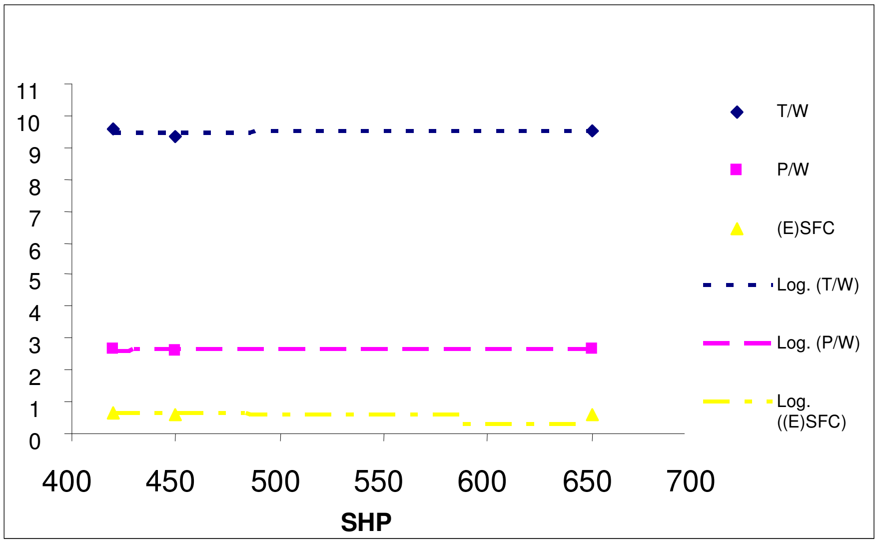


Figure 5.2 Graph of power, torque and SFC versus SHP for helicopter turboshaft engines

In the above figure it is apparent, that helicopter turboshaft engines operating between 400 and 700 SHP in general display already matured and therefore very even trends:

- Fuel consumption levels are at about 0.6 kg/kW.hr.
- Power to weight ratios are numerically 4.5 times as high as the SFC-levels.

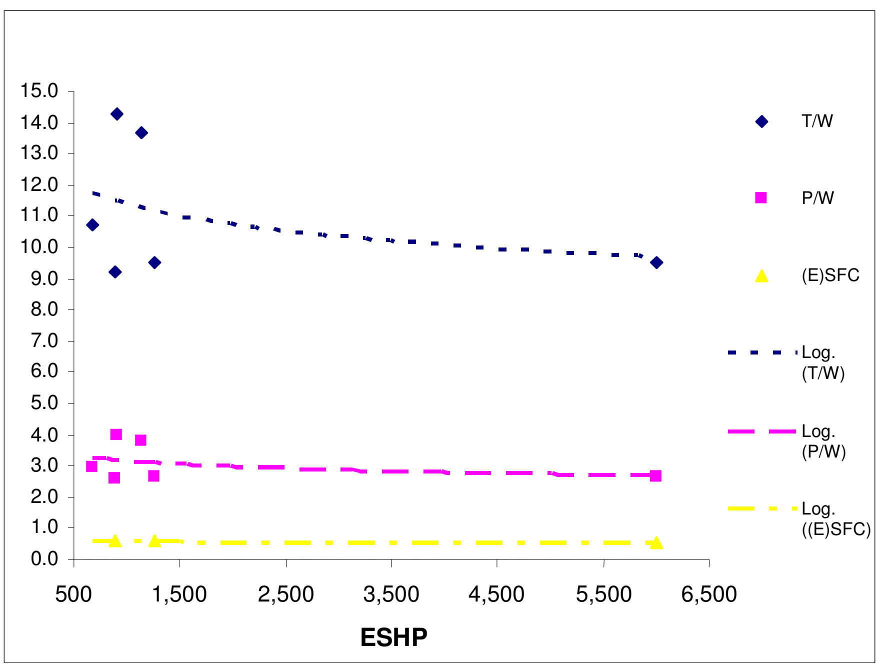


Figure 5.3 Graph of power, torque and SFC versus ESHP for turboprop engines

For turboprop engines between 500 to about 6,500 equivalent shaft horse power ratings, performance levels for different makes of engine display once again across a large engine size spectrum strong and uniform performance trends:

- SFC levels are at 0.5 to 0.6 kg/kW.hr.
- P/W ratios with respect to fuel consumption are 5 times larger.

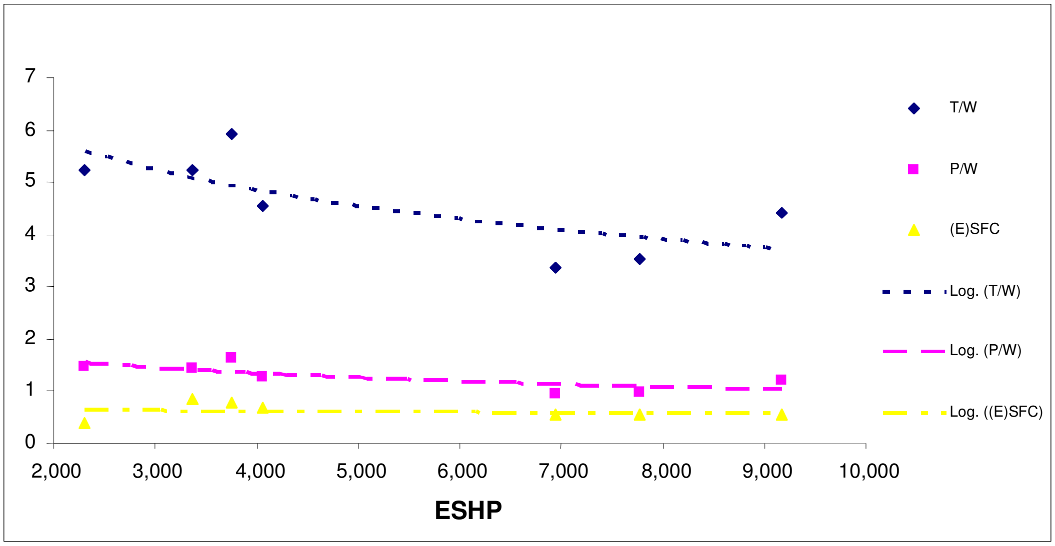


Figure 5.4 Graph of power, torque and SFC versus ESHP for turbofan engines without afterburner

Modern turbofans are designed along turbojet principles with the emphasis on thrust generation via a high-speed fan in addition to the low propulsive efficiency jet-stream:

- A slight degradation in performance is visible with respect to the SFC.
- Degradation is more evident with the power and thrust-to-engine weight ratios.
- Because of the vast quantities of air moved however, relatively large thrust levels can be realised.

From the above performance analyses it is apparent, that overall engine size is a design concern if this project's relatively light engine parameter is to be considered. Careful design is therefore imperative for the engine about to be examined to succeed.

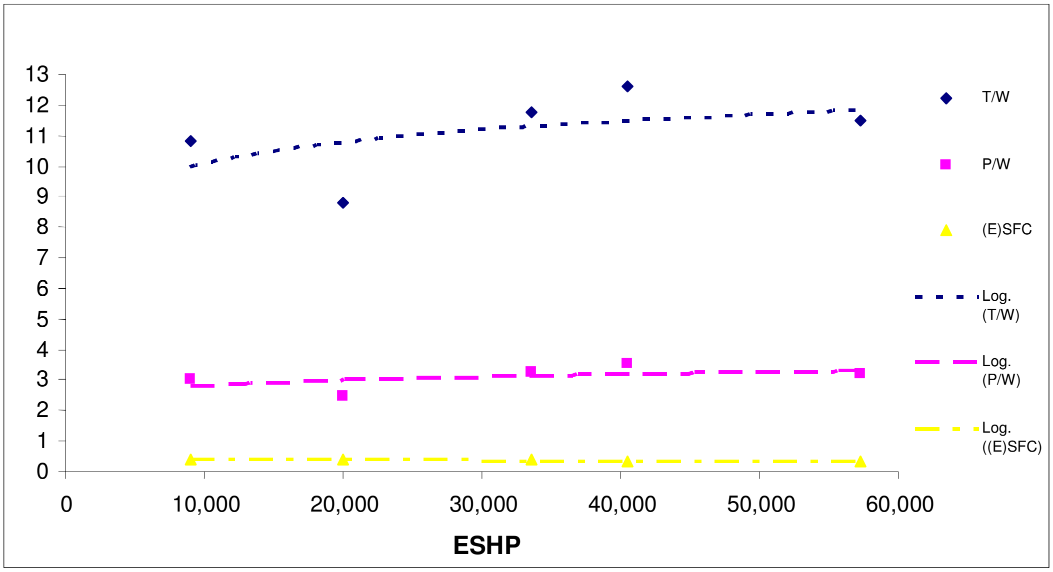


Figure 5.5 Graph of power, torque and SFC versus ESHP for industrial and MAP turbojet engines

Because industrial engines are ground-based units weight is not of significance, and more emphasis is placed on lower fuel consumption levels as apparent in the above:

- At about 0.35 to 0.4 kg/kW.hr, these are the most fuel efficient engines.
- A much improved numerical average P/W versus a SFC ratio of 7.5.

The above explains why most modern industrial gensets are converted aircraft engines called aeroderivatives.

5.1.2 GTS/APU engines

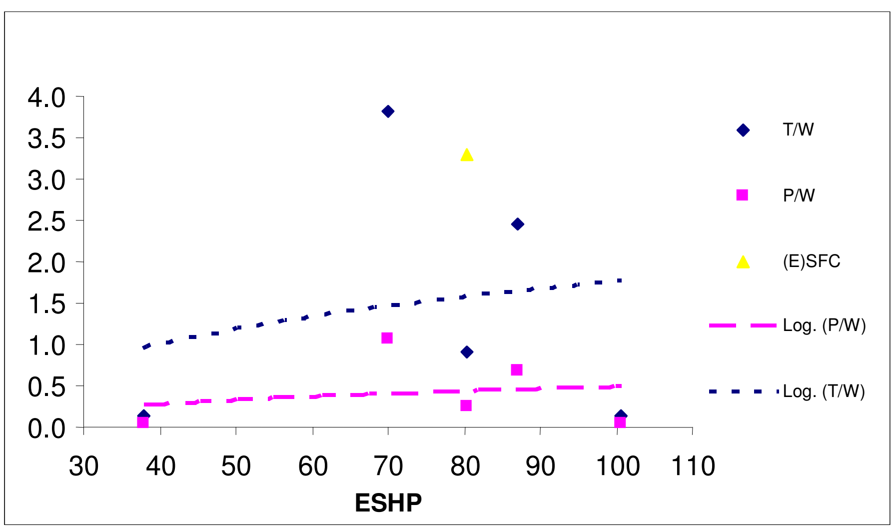


Figure 5.6 Graph of power, torque and SFC versus ESHP for GTS/APU engines

The GTS/APU/GST group of engines belongs to the current range of interest. However, not much is known regarding their performances. Fuel consumption figures are extremely hard to come by, and from the above it is evident, that much confusion exists in this realm of gas turbine engines. Therefore, the SFC value of 3.3 should be treated with caution. It is apparent that, with an increase in engine size and complexity, fuel efficiencies also improve.

5.1.3 IC aero-engines

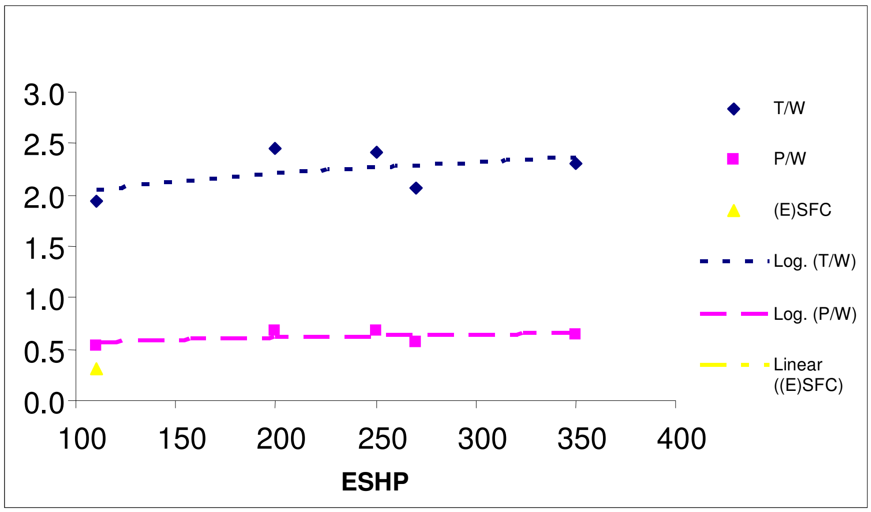


Figure 5.7 Graph of power, torque and SFC versus ESHP for reciprocating petrol aero-engines

Also lacking performance data - especially fuel consumption related - are the piston-driven aero-engines:

- The only SFC value of about 0.3 belongs to the latest generation piston engines.
- On the other hand, the relatively extremely low P/W ratio of only about 0.65 on average indicates the potential of much lighter gas turbine units of equal power with respect to their reciprocating counterparts.

5.2 Design Procedures

In order to derive an optimally designed engine use has been made of data from the internet as well as from existing texts quoted in the reference section.

To derive basic engine requirements needed for the average, light 4-seater aircraft such as the Piper group of planes, models manufactured successfully by leading aircraft companies have been listed in Appendix A. The parameters listed include engine power requirements, propeller speeds and fuel consumption values - with general spreadsheet performance calculations.

In Appendix B already existing gas-turbine specification data have been quoted – ranging from scale model aircraft engines to industrial power generators. Engine performances are compared with the aim in mind to isolate the applicable type of engine for this report's application and to get a basic idea of how and where gas turbines are to date already being employed. All values are with respect to ISA conditions.

In Appendix C more specific performance data for gas turbines are tabulated which are used to plot comparison graphs displayed in section 5.1 for a clearer understanding of engine performance in different size-classes and applications. Data quoted are the Equivalent Shaft Horse Power (ESHP), the Thrust-to-Weight (T/W), Power-to-Weight (P/W) ratios and the Equivalent Specific Fuel Consumptions (ESFC).

Appendix D1 commences with the basic, ambient operating requirements of the engine in question for which two options are available. For comparative reasons, the engine can be designed at ISA conditions but for the purpose of applicability at local 'Highveld' conditions, a much more suited engine would result if local conditions referred to as the 'Operating design Conditions' (ODC) are applied instead. ODC is used for the preliminary engine design.

As a consequence, an altitude of 2,073m ASL at an atmospheric pressure of 82.9 kPa and a temperature of 22.5°C have been chosen. These values concur with a standard summer's day in Johannesburg, Gauteng. As a result a design-density altitude (DDA) of 4,926m ensued. In other words, the engine in question has been designed for operation at an altitude of almost 5 km when compared to ISA conditions.

Table D2 displays the basic engine input parameters selected for this particular design while Table D3 addresses more specifically the separate engine component variable requirements as needed for the engine. Iterative capabilities from this point onwards allows proper engine component matching with respect to input parameters which might appear functional but would not allow a functional, overall engine design to emerge.

In Table D4, the component detail and the engine design process are addressed. Thermodynamic specifics are evaluated to ensure a mathematically sound design, and every critical component is assessed this way. Different configurations are evaluated in detail. The iteration flow chart below is designed to give a good indication of the overall, iterative design processes involved.

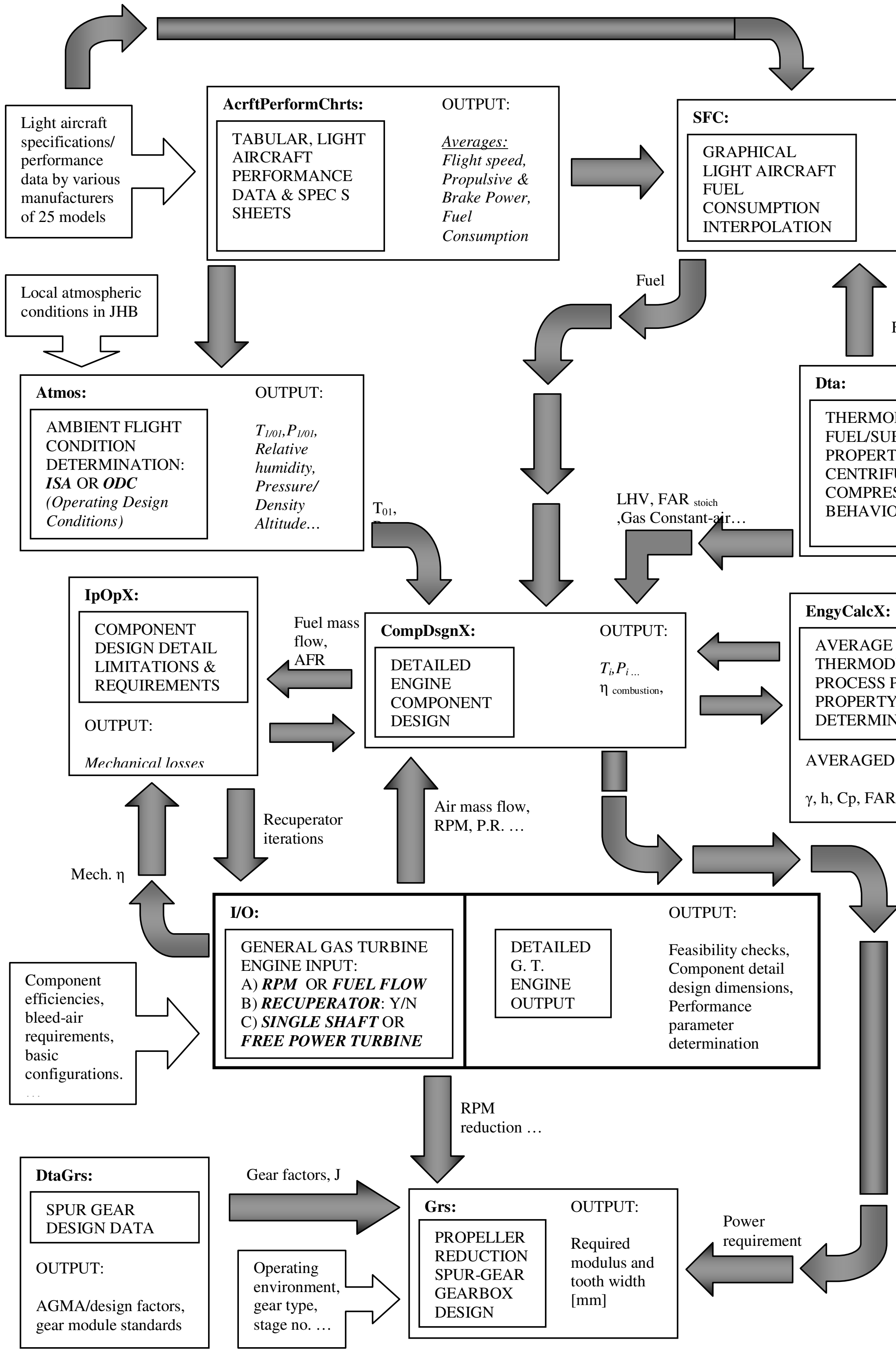


Figure 5.8 Design Iteration flow chart

5.3 Design Parameters

Regarding the basic engine parameter input variables it has to be kept in mind, that engines are rated according to an ISA SLS standard of 15°C at 101.325 kPa. Because of relatively hostile conditions at airports on the South African Highveld compared to ISA, engines designed for local conditions display inevitably different operating behaviour if run under ISA conditions. With respect to the present engine specifically designed for highveld weather, comparisons with foreign engines are not strictly speaking correct as these would inevitably under-perform locally.

5.3.1 <u>Single spool gas turbine</u> [47]

Figure 3.1 depicts the basic component arrangement of a single spool design. An intake pressure loss in the intake duct is arbitrarily assumed to be 0.5% because of the relative ease of manufacture and the functional simplicity of the item. No temperature or stress concerns exist, and as the ambient, free-stream air is in theory slowed down to stagnation conditions at the compressor impeller inlet, friction losses are considered almost negligible in this region.
The radial/centrifugal compressor is designed for a maximum allowable rim speed of 500 m/sec for a cost-effective aluminium alloy impeller appropriate. Such can be improved to 625 m/sec if a titanium alloy impeller is used. To justify the modesty of this thesis's impeller material a high technology level is envisaged for impeller manufacture. For the above impeller design criterion a pressure ratio of 4.5 is achievable but should a higher PR be required for small power applications, a back to back mounting of twin impellers in series almost doubles the PR. [47, pg 183]
Because axial compressors offer higher efficiencies, they are often used in conjunction with radial impellers in more advanced engines.

Pressure losses incurred in general in the system are accounted for by a 3% compressor exit diffuser loss, a 0.5% turbine disc cooling and rim sealing bleed loss, a 0.2% bearing chamber sealing loss, and by a 0.5% high-to-low pressure air system leakage loss. [47]
Additionally, an 8% blade and disc cooling flow offtake at the recuperator air side is introduced between the first stator and rotor stage for turbine blade cooling and additional work generation. [47, pg 226-8]

A mechanical compressor efficiency of 97% is assumed appropriate for preliminary calculations as bearing losses can be safely assumed to be minimal.

The combustor's stator outlet temperature (SOT) is limited due to material, production and ultimately cost margins to a generally acceptable, absolute temperature of 1,400 K. The combustor pressure loss amounts to 3% approximately, while the combustion efficiency is assumed to be 99.9%. [47]

For the only turbine stage the maximum, allowable rim speed is limited - with a polytrophic (ie. the overall stage-) efficiency of 89% - to 400 m/sec while a diagram efficiency of 90% is employed. Such values are substantiated by a combination of centrifugal forces, elevated pressure-expansion exposure and a high temperature environment. The diagram efficiency expresses the energy conversion efficiency of the turbine blades with respect to the pressure and the gas-velocity environment itself and does not include mechanical losses.

Finally, and for completeness's sake, an exhaust-pressure loss of 1% has also been incorporated into the design. [47]

5.3.2 Twin spool power turbine GT

Figure 4.2 depicts the basic component layout of a twin spool power turbine. Unlike with the single spool design, the compressor turbine is limited to a rim speed of 450 m/sec since not all the chemical energy is expanded across it and stresses are thereby reduced. The free power turbine rim speed however is again limited to 400 m/sec for the reason that, although gas expansion is shared between compressor and free power turbine wheel and the operating temperatures are therefore also lower in the secondary stage, the free power turbine is usually of larger diameter, therefore exposing its blades to larger centrifugal forces.

Due to the inter-turbine ducting separating yet channeling the exhaust gases from the first turbine stage to the second one, an approximate pressure loss of 2% and a conduction heat loss component of 4% are subsequently encountered. [47]

Again, polytropic and diagram efficiencies for both the compressor and the free power turbine are taken to be 89 and 90% respectively, which is in line with general observations [47]

5.3.3 Recuperator employment

The recuperator assembly itself is subject to various losses. A recuperator effectiveness of 88% is assumed while further reductions include the recuperator air side inlet duct pressure loss of 3%, an exit duct pressure loss of 1% and an air side total pressure loss of 3%. These variables are general assumptions based on average recuperator design for the sake of this text's preliminary feasibility study, and it is implied that these values improve with maturing technologies in this field. [47]

5.3.4 Mechanical gear reduction

As the turbine output shaft speeds are always in excess of the required propeller speed of approximately 0 to 2,500 rpm variable, a gear reduction unit has to be employed. The reduction ratio is for this text's single-spool unit a function of turbine engine output speed and the maximum propeller speed which is 5:1 for a 12,500 rpm unit. For a twin-spool unit which incorporates a free power turbine, the output shaft speed is in this instance fixed to allow for an external gear ratio of 4:1. In other words, the turbine output speed is fixed at 10,000 rpm but can be varied if necessary; especially, when circumferential free power turbine dimensions become too large.

6 RESULTS AND DISCUSSION

6.1 Gas Turbine Data Presentation

From **Appendix F** the most optimised engine ratings for each specific engine configuration have been retrieved and tabulated below:

DATA RANGES RELATING TO APPENDIX F, I a1:

S O N 1:

Gas turbine configuration:	**SINGLE SPOOL**
Cycle condition:	**OPTIMISED**
Recuperator:	**OFF**
Compressor PR:	**4.5**

A **Cycle status:** **SFC OPTIMISED**

Core engine speed:	11,100	rpm	32.015	mm
Compr. turbine rim vel.	410.5	m/sec		
Power turbine rim vel.:	N/A	m/sec		
Shaft power:	2,536.9	kW		
SFC:	0.868	kg/kW.hr		
Air intake:	23.68	kg/sec		
Required, external gear ratio:	4.44	:1		

B **Cycle status:** **TIP SPEED LIMITS**

Core engine speed:	8,750	rpm	25.210	mm
Compr. turbine rim vel.	399.9	m/sec		
Power turbine rim vel.:	N/A	m/sec		
Shaft power:	2,409.5	kW		
SFC:	0.911	kg/kW.hr		
Air intake:	23.69	kg/sec		
Required, external gear ratio:	3.5	:1		

REMARKS:

Optimised for 100% impulse turbine blading design

S R N:

Gas turbine configuration:	**SINGLE SPOOL**
Cycle condition:	**REQUIRED**
Recuperator:	**OFF**
Compressor PR:	**4.5**

A **Cycle status:** **SFC OPTIMISED**

Core engine speed:	4,001	rpm	12.290	mm
Compr. turbine rim vel.:	386.8	m/sec		
Power turbine rim vel.:	N/A	m/sec		
Shaft power:	151.9	kW		
SFC:	14.669	kg/kW.hr		
Air intake:	25.44	kg/sec		
Required, external gear ratio:	1.6	:1		

B **Cycle status:** **TIP SPEED LIMITS**

Core engine speed:	8,000	rpm	25.273	mm
Compr. turbine rim vel.:	399.8	m/sec		
Power turbine rim vel.:	N/A	m/sec		
Shaft power:	152.3	kW		
SFC:	15.003	kg/kW.hr		
Air intake:	25.99	kg/sec		
Required, external gear ratio:	3.2	:1		

DATA RANGES RELATING TO APPENDIX F, I b:

S O Y 1:

Gas turbine configuration:	**SINGLE SPOOL**
Cycle condition:	**OPTIMISED**
Recuperator:	**ON**
Compressor PR:	**4.5**

A **Cycle status:** **SFC OPTIMISED**

Core engine speed:	9,000	rpm	35.282	mm
Compr. turbine rim vel.	404.4	m/sec		
Power turbine rim vel.:	N/A	m/sec		
Shaft power:	2,580.2	kW		
SFC:	0.439	kg/kW.hr		
Air intake:	32.19	kg/sec		
Required, external gear ratio:	3.6	:1		

B **Cycle status:** **TIP SPEED LIMITS**

Core engine speed:	7,800	rpm	30.47	mm
Compr. turbine rim vel.	399.1	m/sec		
Power turbine rim vel.:	N/A	m/sec		
Shaft power:	2,490.5	kW		
SFC:	0.454	kg/kW.hr		
Air intake:	32.11	kg/sec		
Required, external gear ratio:	3.12	:1		

REMARKS:

Optimised for 100% impulse turbine blading design

S R Y:

Gas turbine configuration:	**SINGLE SPOOL**
Cycle condition:	**REQUIRED**
Recuperator:	**ON**
Compressor PR:	**4.5**

A **Cycle status:** **SFC OPTIMISED**

Core engine speed:	8,280	rpm	38.406	mm
Compr. turbine rim vel.:	406.1	m/sec		
Power turbine rim vel.:	N/A	m/sec		
Shaft power:	152.3	kW		
SFC:	6.436	kg/kW.hr		
Air intake:	38.09	kg/sec		
Required, external gear ratio:	3.312	:1		

B **Cycle status:** **TIP SPEED LIMITS**

Core engine speed:	7,100	rpm	32.598	mm
Compr. turbine rim vel.:	399.5	m/sec		
Power turbine rim vel.:	N/A	m/sec		
Shaft power:	162.3	kW		
SFC:	6.12	kg/kW.hr		
Air intake:	37.75	kg/sec		
Required, external gear ratio:	2.84	:1		

DATA RANGES RELATING TO APPENDIX F, II a&b:

S 0 N 2:

Gas turbine configuration:	**SINGLE SPOOL**
Cycle condition:	**OPTIMAL**
Recuperator:	**OFF**
Compressor PR:	**2.5**

A **Cycle status:** **SFC OPTIMISED**

Core engine speed: 26,750 rpm 8 mm
Compr. turbine rim vel. 378.5 m/sec
Power turbine rim vel.: N/A m/sec

Shaft power: 150 kW
SFC: 0.781 kg/kW.hr
Air intake: 1.25 kg/sec

Required, external gear ratio: 10.7 :1

B **Cycle status:** **TIP SPEED LIMITS**

Core engine speed: 26,750 rpm 8 mm
Compr. turbine rim vel. 378.5 m/sec
Power turbine rim vel.: N/A m/sec

Shaft power: 150 kW
SFC: 0.781 kg/kW.hr
Air intake: 1.25 kg/sec

Required, external gear ratio: 10.7 :1

S 0 Y 2:

Gas turbine configuration:	**SINGLE SPOOL**
Cycle condition:	**OPTIMAL**
Recuperator:	**ON**
Compressor PR:	**2.8**

A **Cycle status:** **SFC OPTIMISED**

Core engine speed: 36,186 rpm 11 mm
Compr. turbine rim vel.: 378.6 m/sec
Power turbine rim vel.: N/A m/sec

Shaft power: 152.4 kW
SFC: 0.256 kg/kW.hr
Air intake: 1.459 kg/sec

Required, external gear ratio: 14.47 :1

B **Cycle status:** **TURBINE TIP SPEED LIMITS EXCEEDED**

Core engine speed: N/A rpm N/A mm
Compr. turbine rim vel.: N/A m/sec
Power turbine rim vel.: N/A m/sec

Shaft power: N/A kW
SFC: N/A kg/kW.hr
Air intake: N/A kg/sec

DATA RANGES RELATING TO APPENDIX F, III a:

S O N 3:

Gas turbine configuration:	**SINGLE SPOOL**
Cycle condition:	**OPTIMISED**
Recuperator:	**OFF**
Compressor PR:	**2.4 - 2.7**

A **Cycle status:** **SFC OPTIMISED**
Pressure Ratio: **2.7**

Core engine speed: 12,500 rpm 3.2 mm *
Compr. turbine rim vel. 380.8 m/sec **
Power turbine rim vel.: N/A m/sec

Shaft power: 152.7 kW
SFC: 0.737 kg/kW.hr
Air intake: 1.208 kg/sec

Required, external gear ratio: 5 :1

B **Cycle status:** **TIP SPEED LIMITS**
Pressure Ratio: **2.4**

Core engine speed: 12,500 rpm 5.5 mm *
Compr. turbine rim vel. 382.1 m/sec **
Power turbine rim vel.: N/A m/sec

Shaft power: 150.5 kW
SFC: 1.085 kg/kW.hr
Air intake: 1.777 kg/sec

Required, external gear ratio: 5 :1

REMARKS:

* Care should be taken regarding minimum axial impeller exit depths: If in doubt, a larger value should be chosen- at the expense of some fuel efficiency.

** Both upper and lower tip speed limits are within range!

DATA RANGES RELATING TO APPENDIX F, III b:

S O Y 3:

Gas turbine configuration:	**SINGLE SPOOL**
Cycle condition:	**OPTIMISED**
Recuperator:	**ON**
Compressor PR:	**2.6 - 3.1**

A **Cycle status:** **SFC OPTIMISED**
Pressure Ratio: **3.1**

Core engine speed: 12,500 rpm 3.3 mm *
Compr. turbine rim vel.: 380.9 m/sec **
Power turbine rim vel.: N/A m/sec

Shaft power: 150.4 kW
SFC: 0.314 kg/kW.hr
Air intake: 1.464 kg/sec

Required, external gear ratio: 5 :1

B **Cycle status:** **TIP SPEED LIMITS**
Pressure Ratio: **2.6**

Core engine speed: 12,500 rpm 9.1 mm
Compr. turbine rim vel.: 384.2 m/sec **
Power turbine rim vel.: N/A m/sec

Shaft power: 151.2 kW
SFC: 0.432 kg/kW.hr
Air intake: 3.227 kg/sec

Required, external gear ratio: 5 :1

REMARKS:

* Care should be taken regarding minimum axial impeller exit depths: If in doubt, a larger value should be chosen- at the expense of some fuel efficiency.

** Both upper and lower tip speed limits are within range!

DATA RANGES RELATING TO APPENDIX F, IV:

T 0 N 1:

Gas turbine configuration:	**TWIN SPOOL**
Cycle condition:	**OPTIMAL**
Recuperator:	**OFF**
Compressor PR:	**4.5**

A **Cycle status:** **SFC OPTIMISED**

Core engine speed:	17,000	rpm	30.715	mm
Compr. turbine rim vel.	460.5	m/sec	*	
Power turbine rim vel.:	400.3	m/sec		
Shaft power:	2,320.2	kW		
SFC:	0.547	kg/kW.hr		
Air intake:	14.84	kg/sec		

Required, external gear ratio: 4 :1

B **Cycle status:** **TIP SPEED LIMITS**

Core engine speed:	12,900	rpm	23.475	mm
Compr. turbine rim vel.	448.9	m/sec		
Power turbine rim vel.:	400.5	m/sec		
Shaft power:	2,333.7	kW		
SFC:	0.548	kg/kW.hr		
Air intake:	14.95	kg/sec		

Required, external gear ratio: 4 :1

REMARKS:

Optimised for 100% impulse turbine blading design

* Exceeding the limit

T R N:

Gas turbine configuration:	**TWIN SPOOL**
Cycle condition:	**OPTM/REQU'D**
Recuperator:	**OFF**
Compressor PR:	**4.5**

A **Cycle status:** **SFC OPTIMISED**

Core engine speed:	######	rpm	12	mm	*
Compr. turbine rim vel.:	488.0	m/sec	**		
Power turbine rim vel.:	381.3	m/sec	***		
Shaft power:	157.7	kW			
SFC:	0.535	kg/kW.hr			
Air intake:	1.005	kg/sec			

Required, external gear ratio: 4 :1

B **Cycle status:** **TURBINE TIP SPEED LIMITS EXCEEDED**

Core engine speed:	N/A	rpm	*
Compr. turbine rim vel.:	N/A	m/sec	**
Power turbine rim vel.:	N/A	m/sec	***
Shaft power:	N/A	kW	
SFC:	N/A	kg/kW.hr	
Air intake:	N/A	kg/sec	

REMARKS:

*	Range consistent at approx. 0.43
**	Consistent at 488, exceeding the limit
***	Whole range within limit
****	No power-curve apex

DATA RANGES RELATING TO APPENDIX F, IV:

T 0 N 2:

Gas turbine configuration:	**TWIN SPOOL**
Cycle condition:	**OPTM/REQU'D**
Recuperator:	**OFF**
Compressor PR:	**4.5**

A **Cycle status:** **SFC OPTIMISED**

Core engine speed:	99,538	rpm	11	mm
Compr. turbine rim vel.	483.4	m/sec		
Power turbine rim vel.:	381.2	m/sec		
Shaft power:	149.8	kW		
SFC:	0.513	kg/kW.hr		
Air intake:	0.915	kg/sec		

Required, external gear ratio: 4 :1

B **Cycle status:** **POWER REQUIREMENT EXCEEDED**

Core engine speed:	12,900	rpm	23.475	mm
Compr. turbine rim vel.	448.9	m/sec		
Power turbine rim vel.:	400.5	m/sec		
Shaft power:	2333.7	kW		
SFC:	0.548	kg/kW.hr		
Air intake:	14.95	kg/sec		

Required, external gear ratio: 4 :1

T O Y: **

Gas turbine configuration:	**TWIN SPOOL**
Cycle condition:	**OPTM/REQU'D**
Recuperator:	**ON**
Compressor PR:	**4.5**

A **Cycle status:** **SFC OPTIMISED POWER REQUIREMENT EXCEEDED**

Core engine speed:	29,250	rpm	12.5	mm	*
Compr. turbine rim vel.:	449	m/sec			
Power turbine rim vel.:	384.4	m/sec			
Shaft power:	489	kW	***		
SFC:	0.165	kg/kW.hr			
Air intake:	3.508	kg/sec			

Required, external gear ratio: 4 :1

B **Cycle status:** **TIP SPEED LIMITS** ****

REMARKS:

*	Average value is taken
**	Rotor inlet relative whirl component = -7.4 m/sec, possibly causing diffusion!!!
***	Minimum attainable under these PR conditions
****	The same as in A!

Note: Recuperating demands specific GT designing, and twin spool designs are not easy to recuperate.
Non-recuperated twin spool designs are more efficient than non-recuperated single spool designs.

6.2 Gas Turbine Data Analysis

With reference to Section 6.1 some interesting observations can be made wherefore use will be made of the following reference letters, which occur to the left of the data sets in their respective data presentation tables:

Letter 1: **S,T** [Single or Twin-shaft]
Letter 2: **O,R** [Optimised/Optimal or Required]
Letter 3: **N,Y** [Recuperator employment: No or Yes]
Letter 4 (Where applicable):
1,2,3.. [Any recurring identical digit sets above are differentiated]
Sub-letters: **A,B** [A: SFC-optimised, B: Maximum tip speed limit adhered to]

6.2.1 The single spool, non-recuperated engine

For the single spool, non-recuperated gas turbine engine design, the engine of choice is a selection between SRN-A, SRN-B, SON2-A, SON3-A and SON3-B.

Considering SRN-A and B at first, one can surmise that because of the much more favourable external gear ratio requirement of 1.6:1 unlike 3.2:1 and because of a somewhat better fuel consumption engine SRN-A would be the preferred choice. Distressing however is the extremely high fuel consumption value of 14.669 kg/kW.hr due to the fact, that 150 kW is a sub-optimal power setting for an engine which performs at its best at around 2,500 kW. Reducing the compressor pressure ratio from a maximum of 4.5 to a lower value instead should therefore yield a more favourable, optimised design for our specific power requirements.

With a pressure ratio of 2.5:1 therefore, the SON2-A engine offers the required power at a much reduced SFC value of about 0.781. Although a tremendous improvement in itself, this will not be sufficient with respect to other engine types in its performance class. Another distressing factor is the external gear ratio requirement of 10.7:1, which can prove excessively high for a mechanical gearing device limited by space and weight.

In order to reduce both space and weight, a new approach has been embarked upon in trying to determine the best suited engine design for its operating environment. To be able to accommodate a permanent external gear reduction of 5:1, which in this instance demarcates the maximum allowable limit due to space constraints, engine speed was set to 12,500 rpm. In order to also achieve the required output power of 150 kW, the next consecutive variable to be optimised was the axial impeller exit depth, which indirectly also varied the overall air flow through the engine. This exercise has been repeated for a number of pressure ratio settings as displayed in Appendix F, and the resulting engine designs featured a far improved SFC with respect to the SRN-A and B, with the added advantage of optimal gear requirements. To reduce the fuel consumption to absolutely minimal limits, engine SON3-A therefore was the engine of choice; however with an axial exit impeller depth of only 3.2 mm. With all basic requirements having been met with the exception of a somewhat elevated fuel consumption of 0.737 kg/kW.hr, the only other alternative left is to consider engine recuperation.

6.2.2 The single spool, recuperated engine

The single spool, recuperated choice of engines is a selection of the SRY-A, B, SOY2-A and the SOY3-A and B variants. At a fuel consumption level of 6.436 kg/kW.hr, the adapted SRY-A and B engines optimised for 2,580 kW display very poor fuel efficiencies at lower external loads. It is however noteworthy, that with respect to the SRN-group of engines, recuperation nevertheless still managed to improve SFC levels by about 56%, which in itself adds a lot of emphasis to the concept of recuperation. Apart from such, engines SRY-A and B are relatively similar; with engine B featuring the better SFC of 6.12 kg/kW.hr and a better gearing requirement of 2.84:1 compared to 3.312:1. Additionally engine B - unlike A - does not exceed the stipulated turbine rim velocity stress limitation of 400 m/second, and therefore when putting SFC aside engine SRY-B becomes the natural choice of preference.

Optimising again for our power requirement yields an engine labeled as the SOY2-A featuring a pressure ratio of 2.8:1. Given its incredibly low fuel consumption of 0.256 kg/kW.hr, which is due to added recuperation, it also holds very good promise. With a gear requirement of 14.47:1 though, this option proves impracticable for gearing requirements in general. Nevertheless, such low levels of fuel consumption could possibly justify a multiple gear set as already found on larger, commercial turboprop units. The high mechanical efficiencies of spur gears for example render such an approach reasonably viable as long as weight and space requirements can satisfactorily be met. Because maintainability and initial capital outlay are also requirements to be met, the present work will discard such options altogether, and the only two engines left for discussion are the SOY3-A and B respectively:

The SFC-optimised engine SOY3-A supersedes the SOY3-B only because of a better fuel consumption which exhibits an acceptable 0.314 kg/kW.hr. This takes into account the axial exit impeller depth of only 3.3 mm and all other critical requirements which have been met, such as its preset gear ratio of 5:1 and a shaft power rating of 150.4 kW. With regard to a recuperated, single spool gas turbine engine this design would definitely be the engine of choice. Care must however be taken in that the small axial exit depth parameter might or might not adversely affect efficiency.

In summary, worth reemphasizing are low enough engine core speeds in conjunction with comparable fuel economy and engine size, and all these criteria appear to have been satisfactorily met in the SOY3-A variant.

6.2.3 The twin spool, non-recuperated engine

For the twin-spool designs an external gear ratio requirement of 4:1 has been chosen which tends to enhance engine efficiency. Also, the following non-recuperated candidate can be selected from either TRN-A or TON2-A. It is interesting to observe, that in this instance a higher pressure ratio is required for optimal operation of these engines; preferably the highest mechanically attainable of 4.5:1.

Both the TRN-A and TON2-A engines operate so closely to each other's parameter settings, that they can in essence be considered as the same engine. Because the TON2-A engine however operates with a slightly lower SFC of 0.513 kg/kW.hr it would naturally be the choice of preference. With a preset gear ratio of 4:1, all appears to be within specified limits if it was not for the compressor-turbine rim velocity exceeding its limit by 33.4 m/sec. Also, because of the high primary spool velocity though, presetting the external gear ratio to a low value will require the free power turbine to compressor turbine radius ratio to be excessively large, which it actually is.

6.2.4 The twin spool, recuperated engine

In the recuperated, twin spool engine department the only available choice would be the TOY-A engine, which has been optimised at a very noteworthy SFC of 0.165 kg/kW.hr. Fine-tuning yielded an engine which produced an output shaft power of 489 kW minimal. Although excessive by a significant 339 kW, when considering such low a fuel consumption this powerplant would prove extremely useful for larger power demand requirements. Caution has to be taken though in the design of the engine, because the rotor inlet relative whirl component entering the turbine rotor ring was found to be slightly negative, ie. – 7.4 m/sec. This can cause diffusion and turbine efficiency deterioration; especially at sub-optimal power settings.

From this point onward, the recuperated SOY3-A variant of gas turbine engines shall be chosen as the ultimate and competitive aircraft propulsion design.

6.2.5. Final engine proposal

Table 6.1 Basic engine input parameters

Engine type:		Gas Turbine	
Intake air mass flow:		1.464	kg/sec
Gas Turbine velocity:		12,500	rpm
Intake:			
Intake pressure loss:		0.5	%
Centrifugal compressor:			
Maximum allowable rim speed:		500	m/sec
Compressor technology level:	(vl,l,m,h,vh)	h	
Pressure Ratio:	(For AI <= 4.5)	3.1	: 1
Mechanical compressor efficiency:		97	%
Compressor exit diffuser:			
Compressor exit diffuser pressure loss:		3.0	%
Turbine disc cooling & rim sealing bleed:		0.5	%
Bearing chamber sealing/chamber:		0.2	%
Leakage from high - low pressure air system:		0.5	%
Customer bleed extraction:		0.0	%
Is a Recuperator employed? (Y/N):		y	

Pressure losses:	**Lambda:**	**%tage:**
Compressor delivery to air inlet	0.7 - 1.5	3 - 6
Air outlet to combustor inlet	2.5 - 6	1 - 2.5
Turbine outlet to gas inlet	0.4 - 1.2	2 - 6
Gas outlet	Exh. Duct	Calculation

Recuperator air side:			
Cooling flow offtake position (Inlet/Exit):		inlet	
Recuperator air side inlet duct pressure loss:		3.0	%
Recov'd blade and disc cooling flow offtake:		8.0	%
Recuperator:			
Recuperator effectiveness:		88	%
Air side total pressure loss:		3.0	%
Recuperator exit duct:			
Recuperator exit duct pressure loss:		1.0	%
Combustor:			
Turbine inlet temperature (SOT) T41:		1,400	K
Combustor pressure loss:		3.0	%
Equivalent IC engine fuel flow:	0.013	kg/sec	
Fuel type:	Kerosene		
Combustion efficiency:		99.9	%
Pseudo-station 415 mixing:			
Cooling air addition, doing work:	(Max: 8%)	8.0	%
a) Turbine:			
Maximum rim speed:		400	m/sec
Polytropic efficiency:		89	%
Diagram efficiency:		90	%
b) Compressor & free power turbine:			
Inter-turbine duct pressure loss:	(0.5 - 2.5%)	2	%
Inter-turbine duct heat loss:		4	%
Compressor turbine:			
Maximum rim speed:		450	m/sec
Polytropic efficiency:		89	%
Diagram efficiency:		90	%
Free power turbine:			
Maximum rim speed:		400	m/sec
Polytropic efficiency:		89	%
Diagram efficiency:		90	%
Initial guesses:			
Isentropic efficiency-estimate:		88	%
Gas side inlet temperature 1st guess T6ini.:		900	K

Please select one of the following two options:

RPM input parameter (Y/N):	y	
Mass-flow input parameter:		
Gas Turbine velocity:	**12500**	**rpm**

Gear Design Determinants:

a) Single gas turbine arrangement:			
Propeller speed @ cruise vel.:	2,500		RPM
Optimum gas turbine speed:	12,500		RPM
Gas turbine int'l gear reduction:	1	:	1
External gearing requirement:	**5.000**	**:**	**1**
b) Free power turbine arrangement:			
External gearbox reduction:	4	:	1
Required power turbine speed:	**10,000**	**rpm**	
Gas / Free P. turbine vel. ratio:	**0.8000**		

Basic Design Point Single Spool Calculations:

Recuperator inlet duct - Turbine:

	1,079.6	K
	1,078.1	K --->
T6 Error:	1.5	K
% error:	0.14	%
Compressor input power:	194.8	kW
Capacity WRTP, station 41:	0.218	kg√K/s kPa
Turbine output power:	403	kW
Overall mechanical efficiency:	99.910	%
Available shaft power:	208	kW
FAR:	0.0116	
Fuel mass flow:	0.0155	kg/sec
Margin for improvement:	0.00	kg/s
Relat. fuel flow efficiency:	86	%
P9/Ps9:	1.002	
Exhaust plane area A9:	0.130	m2
Exhaust plane diameter D9:	0.406	m
Specific power/thrust:	142.25	Ns/kg
Spec. fuel consumption:	0.2670	kg/kWh
S. P. thermal efficiency:	31.282	%

Recuperator gas side:		
Recuperator gas side inlet duct pressure loss:	4.0	%
Recuperator gas side pressure loss:	4.0	%
Recuperator exhaust:		
Exhaust pressure loss:	5	%
Exhaust design mach number:	0.05	
Final calculations:		
Jet pipe/exhaust pressure loss:	1.0	%

Table 6.2 Component detail parameters

Gas Turbine velocity:	12500	rpm
Intake mass-flow:	**1.46439**	**kg/sec**

Axial impeller exit depth:	3.3	**mm**
Current gas turbine velocity:	12,500	rpm

Specific Fuel Consumption:	S. Sp:	T. Sp:
(Single spool, Twin spool)	0.316	

FEASIBILITY CHECKS:

Compressor:		***Check:***
Mean inlet mach number:	0.477	OK!
Impeller rim speed:	411.0	OK!
Inducer tip rel. mach number:		0.9->1.3
Inducer hub-tip ratio:	0.979	0.35->0.7
Inducer tip/exit rel. velocity ratio:		0.5->0.6
Shroud-tip ratio:	0.250	
Compressor specific speed:	0.255	
Compressor efficiency:	70.0%	

Required compressor power:	247.7	kW

Recuperator iterations:

a) Recuperator T6-error:	-0.05	%
b) Recuperator T6-error:	-0.02	%

a) Single turbine tip vel.:	**380.9**	OK!
Rotor inlet rel. whirl component:	348.5	OK!
Hub tip ratio:	0.989	0.5->0.85
Reaction ratio (ideal: 0.5):	0.040	OK!
Flow coefficient (ideal: 0.8):	0.762	OK!
Turbine efficiency (t-s):	**82.9%**	
Generated turbine power:	402.2	*Shft Pwr:* **150.4**

b) Compressor turbine tip vel.: **N/A**

Table 6.3 Single spool component detail values

Recuperator inlet duct:

1,169.2 K

1,169.8 K --->

T6 Error:	-0.6	K
% error:	-0.05	%
Compressor input power:	248	kW
Compressor efficiency:	69.99	%
Turbine output power:	398	kW
Mechanical turbine efficiency:	99.00	%
Available shaft power:	150.4	kW

FAR:	0.0099	
Fuel mass flow:	0.0132	kg/sec
Margin for improvement:	0.00	kg/sec
Relat. fuel flow efficiency:	100	%

P9/Ps9:	1.002	
Exhaust plane area A9:	0.172	m2
Exhaust plane diameter D9:	0.468	m

Specific power/thrust:	102.74	Ns/kg
Spec. fuel consumption:	0.316	kg/kWh
S. P. thermal efficiency:	26.45	%

Overall mechan. losses:	0.36	kW
Overall mech. efficiency:	99.910	%

GENERAL COMPONENTS

Bearing type:	Ball bearing	
Bearing race diameter:	65	mm
Lubricating oil:	Medium mineral	
Oil flow rate:	0	l/h

INTAKE/COMPRESSOR

Nacelle intake diameter:	165	mm

Relative inlet shroud angle:	32.0	°
Relative inlet hub angle:	31.4	°
Exducer vane backsweep:	8.39	°
Number of impeller vanes:	30	
Impeller length:	376	mm
Inducer hub radius:	77	mm
Inducer tip radius:	78	mm
Inducer tip velocity:	103	m/sec

Impeller tip radius:	314	mm
Diffuser vane inlet radius:	330	mm
Diffuser vane exit radius:	440	mm
Diffuser vane height:	10	mm
Radial-axial outer diff. wall radius:	443	mm
Axial straightener outer wall radius:	883	mm
Axial straightener inner wall radius:	881	mm
Radial-axial inner diffuser wall radius:	442	mm

COMBUSTOR CAN

Combustor volume:	7.4E-03	m3
Overall can area:	0.0194	m2
Can radius:	0.0787	m
Outer annular radius:	0.1099	m2
Combustor can length:	0.3784	m

TURBINE

Stator exit blade angle:	50.4	°

Stator absolute exit angle:	68.4	°
Stator deflection angle:	68.4	°
Stator inlet/exit blade root radius:	288	mm
Stator inlet blade tip radius:	292	mm
Stator inlet blade height:	5	mm

Stator exit blade tip radius:	291	mm
Stator exit blade height:	3	mm
Average stator pitchline radius:	**290**	mm

Rotor relative exit blade angle:	52.7	°
Rotor deflection angle:	103.1	°
Rotor inlet/exit blade root radius:	288	mm
Rotor inlet blade tip radius:	291	mm
Rotor inlet blade height:	3	mm

Rotor exit blade tip radius:	299	mm
Rotor exit blade height:	11	mm
Average rotor pitchline radius:	**294**	mm

EXHAUST

Exhaust plane area:	0.129	m2

6.2.6. General observations

With respect to the single spool designs and their turbine stage reaction ratios, the most fuel efficient designs were achieved by approaching reaction ratios of zero all-round. In other words, the turbine stages were of a pure impulse design only. For the twin-spool designs on the other hand, a more even "sharing" of the gas reaction on both the compressor and power turbine could be observed, and both turbine rotors were subjected to a reaction of roughly 47 - 50%.

Regarding the twin spool designs, another discovery was the tendency for these to exhibit ever increasing rotational speeds the smaller the engine became, which is not evidently prevalent with the single spool designs. Typical values for the 150 kW unit were in the region of approximately 100,000 rpm, and on the contrary a perfectly sound 2,320 kW machine operated happily at only around 17,000 rpm core engine speed and at an SFC of 0.547 kg/kW.hr. This is somewhat counterproductive, as the inherent advantage of slower spinning free power turbine wheels was offset by all-round higher rotational speeds in the smaller units.

Additionally, whether an engine is designed with recuperation capability or not, the incorporation of such a device requires engine-specific parameter sets which are often very different from actual working engine parameters. Consequently each engine design should be treated according to the way it has been intended. For recuperator incorporation to be intended it is therefore expected from the onset for it to be engaged continuously, and not just like, for example, an afterburner in special cases would be. Additionally, due to the inherent simplicity of this particular design it lacks differential load setting features made possible by, for example, variable pitch guide vanes and the like. A simple engine, thereby although more reliable and robust, lacks the flexibility of more complex, self adjusting features for continually optimised operations.

With regard to the design process itself, trial and error once again re-emphasized the need to go about the design process in a systematic way. In other words, in order to design some machinery it behoves the designer to account for the parameter requirements first, such as required engine speed, in order to design the machine accordingly. Otherwise, the most optimal engine designs might still fall short of their general operating requirements, whether super-optimised or not. As a consequence maybe an efficient but not an effective design solution can be found.

6.3 Secondary Components' Design

6.3.1 Gearbox design

Table 6.4 Single spool gas turbine spur gear design

General input parameters:

Gear type:		Spur
Number of gear stages:		Single stage
Gearbox rpm reduction:		5.000 : 1
Tooth shape:		Full depth involute
Manufacturing method:		Milling
Factor of Safety:		1.5
Preferred modules:		**4, 5, 6, 6, 8, 8**
Next choice:		4.5, 5.5, 7, 7, 9, 9
Module:		5.5 mm
Pressure angle:	(20^o, 25^o only!)	25 °
Reliability:		95 %
Gas turbine running behaviour:		Uniform
Gear Train running behaviour:		Uniform

Quantity:	Formulae:	
Addendum:	a =	5.5
Dedendum:	b =	6.875
Tooth thickness:	t =	2.75
Minimum number of pinion teeth:	Np =	12
Minimum number of teeth / pair:	Np+ NG =	24

Tooth width: 80.5 mm
(ie. a Permissible Tooth Width!)

Engine Specification:

Engine type:	Gas Turbine	
Max. power:	150.4	kW
Spd @max.pwr:	12,500	rpm

Propeller/gear speed : 2,500 RPM

Material Selection:

Pinion:	Material:	2.5%NiCrMo	steel
(on engine shaft)	Hardness:	331	BHN
	Treatment:	Hardened, tempered	
Gear Wheel:	Material:	3%CrMo	steel
(on transmission shaft)	Hardness:	269	BHN
	Treatment:	Hardened, tempered	

p. 664, Mechanical properties of steels:

UTS - pinion: 1,075 **MPa**

UTS - gear: 850 **MPa**

Bearings:

Bearing type: Rolling bearing

Design Philosophy: Emphasis on:
Low mass
Small size
High pinion rpm
-> to minimise pinion dia.

OUTPUT DETERMINANTS:

AGMA-geometry factor determination:

Minimum no. pinion teeth available:	Np =	13
Gearbox rpm reduction:		5
Pressure angle:		25 °

Teeth on gear wheel: 65.00

Interpolator: (T-G02/3)	1	y-coord
	6	x-coord
50	**85**	**x/y-lbls**
0.36138	**0.36572**	**x/y-vals**

AGMA J - Factor, Pinion: 0.36324

Minimum no. gear teeth available:	NG =	65

Interpolator: (T-G02/3)	19	y-coord
	2	x-coord
1	**17**	**x/y-lbls**
0.3675	**0.50683**	**x/y-vals**

AGMA J - Factor, Gear: 0.471997

Strength calculations:

Pinion	UTS: 1,075 MPa J factor: 0.36324	**Product:**	**390.4829**
Gear	UTS: 850 MPa J factor: 0.472	**Product:**	**401.1979**

Note: Because the Pinion from the above comparison is the weaker member, our design will be based on it.

Pinion:

RPM pinion:	12,500		
RPM gear:	2,500	**Angular velocity:**	**1,309.0 rad/sec**
Input power:	150,448 W	**Torque:**	**114.9 Nm**
Module:	5.5 mm		
Pinion teeth:	13	**PCD:**	**71.5 mm**
		Tangential velocity:	**46.80 m/sec**
Torque:	114.9 Nm	**Force:**	**3,215 N**
UTS:	1,075 MPa	**Endurance limit:**	**537.5 MPa**

Spur-gear design factors:

Dynamic Factor Kv:

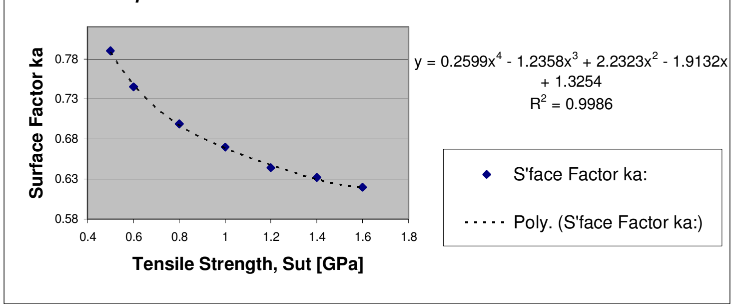
Manufacturing method: Milling

Tangential velocity: 46.80 m/sec

Dynamic Factor Kv:
0.1136 - Barth Eqn mod.

Surface Finish Factor Ka:

Graph SG-G01: Surface Finish Factor:



Tensile strength:	1.0750 GPa	**Surface Finish Factor Ka:** **0.6603**
Size Factor Kb:		
Module:	5.5 mm	**Size Factor Kb:** **0.902**
Reliability Factor Kc:		
Reliability:	95 %	**Reliability Factor Kc:** **0.868**
Temperature Factor Kd:		
Temperature:	298.3 K	**Temperature Factor Kd:** **1.0**
Stress Concentration Factor Ke:		
Incorporated into J - ie. = 1 for Spur gears!		**N/A**
Miscellaneous Effects Factor Kf:		
Tensile strength:	1.0750 GPa	**Miscellaneous Effects Factor Kf:** **1.330**
Overload Correction Factor Ko:		
Gas turbine running behaviour:	Uniform	**Overload Correction Factor Ko:**
Gear Train running behaviour:	Uniform	**1.00**

Load-distribution Factor Km:

Select one of the following characteristics for the support only:

Accurate mountings,small bearing clearances, minimum deflection, precision gears (y/n) ?: a) **y**

Less rigid mountings, less accurate gears, contact across full face (y/n) ?: b) **n**

Accuracy/ mounting such, that less than full face exists (y/n) ?: c) **n**

Approximate gear width: **150** mm

Load-distribution Factor Km:
1.4

Tooth-width calculations:

Modulus:	5.5	mm	**Max. recommended tooth width:**	**86.4 mm**
Surface Finish Factor Ka:	0.660			
Size Factor Kb:	0.902			
Reliability Factor Kc:	0.868			
Temperature Factor Kd:	1.0			
Stress Concentration Factor Ke:	N/A			
Miscellaneous Effects Factor Kf:	1.330			
Overload Correction Factor Ko:	1.00			
Load-distribution Factor Km:	1.4			
Endurance limit:	537.5	MPa		
Factor of Safety:	1.5		**Allowable tensile stress:**	**175.98 MPa**
Modulus:	5.5	mm		
AGMA Geometric J-factor:	0.3632			
Allowable tensile stress:	175.98	MPa		
Dynamic Factor Kv:	0.1136			
Force:	3,215	N	**Tooth width:**	**80.5 mm**

ie. a Permissible Tooth Width!

6.3.2 Variable coupling proposal [45]

With reference to Voith Turbo Coupling product description brochures, hydraulic coupling sizes T, TV, TVV and TVVS are all readily available. Therefore with reference to the Voith Turbo sizing chart, a shaft input speed for the impeller of about 2,500 rpm after prior speed reduction via the gearbox yields with an anticipated 2.5% of slip an output speed of 2,438 rpm which is acceptable for estimation purposes.

The brief tabulation below displays the two couplings' general specifications with the emphasis on their compatibilities with respect to this particular design intent:

Table 6.5 Voith Transmission Couplings

Coupling size:	**366**	**422**
Coupling type:	"T"	"T"
Outer Ø (mm):	424	470
Maximum length:	357	391
Coupling length:	198	218
Weight (kg):	52	78

For an input speed of 2,500 rpm at a rated output power of 150 kW, the best coupling of choice would be the 422 size of coupling. Downscaling slightly however due to the rated power of 150 kW being rather generous with respect to the maximum propeller power needed (of approximately 130 kW on average), the next best size of 366 is also worth of consideration, seeing that it weighs remarkably less and is consequently somewhat smaller.

7 CONCLUSION AND RECOMMENDATIONS

7.1 Conclusions

As technology of materials, production methods and of engine designs improve, gas turbine propulsion methods continue to expand into an ever widening range of applications. Apart from the majority of aircraft propulsion systems and utility/industrial power generation modules in operation, gas turbine engines are already found in various kinds of other applications. These include uses in marine vessels as their primary movers, as auxiliary power units and as gas turbine starters (such as APU's) in both commercial and military aircraft. For distributed power generation of electricity they can be found as small and compact microturbines in the 15-300 kW range. In road transport they are employed in trucks, in buses and lately in passenger sedans as emerging hybrid-electric vehicle technology appearing to be an emerging trend for the future. They are employed in new generation remote controlled drones and U(C)AV's and in small-scale turboprop and turboshaft aircraft in general. Even as turbochargers in high performance cars, they have for many years already found successful application.

To date with only some minor exceptions, gas turbine engines have not been employed in light aircraft such as in the Cessna and Piper family of aircraft. Reasons for such are relatively high specific fuel consumptions in comparison with their piston-engine counterparts, higher acquisition costs and some technological challenges attached to designing gas turbine engines in the 75 to 150 kW range. However, turbo-shaft engines for helicopter applications have had huge successes so far, and even light aircraft such as the Pilatus PC-7 Mk.II turboprop-Turbo Trainer which employs a Pratt & Whitney Canada PT6A-25C engine exhibiting a rated engine power of 522 kW (700 shp) are successfully propelled by gas turbine-driven turboprops.

Specifically with the initial concept approach of this text in mind, a feasible engine design with application potential in light aircraft in general could successfully be isolated. This design was labeled the recuperated, SOY3-A variant of gas turbine engines and has the following operating specifications:

DESIGN PARAMETERS

Gas Turbine Engine-type:	Recuperated single spool, SFC optimised
Engine Pressure Ratio:	3:1
Core Engine Design Speed:	12,500 rpm @ impeller exit depth = 3.3 mm
Shaft Output Power:	150.4 kW
Specific Fuel Consumption:	0.314 kg/kW.hr
Air Intake Rate:	1.464 kg/sec
External Gear Ratio:	5:1 to a fixed pitch propeller

OPERATING CONDITIONS

Forward Aircraft Velocity: 260 km/hr
Design Density Altitude (ASL): 4,926 m

Specific advantages of gas turbine technology with respect to the piston driven power pack include a higher degree of reliability, lower maintenance requirements, fuel-type flexibility unheard of in piston engine technology and higher power ratings for the same engine weight. All these factors contribute favourably towards a more economical engine which is easier to maintain and more reliable to operate. Initial production costs - however daunting they might seem - can over time with an increase in demand by the public sector and with technological advances reduce to manageable levels according to a factor of scale – and proportional to demand in general.

In particular in the field of light aircraft traditionally propelled by piston driven internal combustion engines, it has herewith been successfully demonstrated, that it is possible to not only match, but to even exceed existing performance levels via jet engine replacement units, and reasons for such are as numerous as convincing.

7.2 Recommendations for Future Work

With respect to the promises gas turbine technology appears to hold for light aircraft, research and development in this field of aviation is imperative. Specific areas requiring further research in general include, but are not limited to the following:

- Higher fuel efficiencies can be achieved by research into improvements of recuperator as well as emerging intercooler and recirculation technologies.
- Research into the formation of induced, oscillatory vibrations has to be also more clearly understood and studied.
- Efforts should be made in the reduction of microturbine tip clearance losses.
- Research is required in the region of more cost-effective manufacture of critical turbine stage components such as the heat resistant, fully crystalline turbine blades and in components' tolerance limits for the reduction of pressure losses.
- Other disciplines needing more research and yet made little mention of, are in the fields of improved noise reduction emanating from the compressor wheel and the jet exhaust.
- Heat and sound insulation needs to be studied more which should result in more user-friendly, more efficient engines.
- More emphasis should also be placed in advanced software development incorporating ANN's and in improving existing CFD software for design work.
- Electronic FADEC control systems for microturbine application would result in higher operating efficiencies, user friendliness and engine reliability.

REFERENCES

1. Aerotech News. (no date). U.S. Army selects Honeywell-General Electric engine to power self-propelled artillery, tanks. INTERNET. http://www.aerotechnews.com/starc/2000/092900/Honeywell_GE.html Cited 2003.
2. Agrawal Giri L., The American Society of Mechanical Engineers (1998). Foil air bearings cleared to land. INTERNET. http://www.memagazine.org/backissues/july98/features/foil/foil.html Cited 2003
3. Allison Engine Company. Product info sheets. Allison GMA 2100 turboprop. (1992).
4. Baer-Riedhart, Jenny. NASA, Dryden Public Affairs. (2/1/2001). XB-70. INTERNET. http://www.dfrc.nasa.gov/gallery/movie/XB-70/index.html Cited 2002.
5. Bennett, Chris. Lucas Aerospace CR.201 APU. INTERNET. http://www.cbjet.clara.co.uk/index/index.html Cited 2003.
6. Bird Publishing - Aviation Marketplace. (Updt. 31/05/2003). A Guide to Russian Airliners. INTERNET. http://www.bird.ch/Russians/An22/AN22P02.html Cited 2003.
7. Bluefox9. (1999). v2500. INTERNET. http://www.alair.com/Commercial/v2500.html Cited 2002.
8. Bokulich, Frank. SAE International. (no date). Preparing for the future of aircraft engines. INTERNET. http://www.sae.org/aeromag/features/aircraftengines/index.htm Cited 2003.
9. British Airways. (no date). The Concorde. INTERNET. http://www.britishairways.com/concorde/index.html Cited 2002.
10. Capstone Turbine Corporation. (2002). Applications: Hybrid Electric Vehicles. INTERNET. http://capstoneturbine.com/applications/hevAppDetails.asp Cited 2002.
11. Concorde SST. (June 2000). Concorde Technical Specs. INTERNET. http://www.concordesst.com/powerplant.html Cited 2002.
12. Chris (Sgt), The Commander's Hatch. (1998). M1. INTERNET. http://members.aol.com/panzersgt/ Cited 2003.
13. DC3 Aviation Museum. (1995). Aircraft Comparisons: Turbo-Prop vs. Radial. INTERNET. http://centercomp.com/cgi-bin/dc3/gallery?25000 Cited 2003.
14. Dixon S.L. (1998). Fluid Mechanics and Thermodynamics of Turbomachinery, 4th Edition.
15. Everett Aero. (Updt. 2/5/2003). Plessey Dynamic Solent GTS. INTERNET. http://www.everettaero.com/solent.html Cited 10/6/2003.
16. GE Aircraft Engines. (no date). Cruise Control. INTERNET. http://www.geae.com/spotlight/cruisepower.html Cited 2003.
17. Global Internet. (no date). How it works: Small gas turbine engine (APU). INTERNET. http://www.users.globalnet.co.uk/~spurr/sec.htm Cited 2003.
18. Gruning, Norman. Ex-aircraft co-owner and initiator of research. Private conversation in January 2002.
19. Honeywell International. (2002). LV100-5 engine. INTERNET. http://www.lv100.com/ Cited 2003.
20. Langston, Lee S. (1999). Flight and Light. INTERNET. http://www.memagazine.org/supparch/mepower00/flight/flight.html Cited 2003

21. Langston, Lee S. (2001). Good Times with a Double Edge. INTERNET. http://www.memagazine.org/supparch/mepower01/withedge/withedge.html Cited 2003.
22. Leland R., Haynes (Msgt USAF, Ret.). (15/04/1996, Rev.:22/03/2001). J-58 Pratt and Whitney Engine. INTERNET. http://www.wvi.com/~lelandh/image2~2.html Cited 2002.
23. Lockheed Martin Aeronautics Company. (2002). C-130 Hercules — The 20th Century's Greatest Airlifter Will Also Be the 21st Century's Greatest. INTERNET. http://www.lmaeronautics.com/products/airmobility/c-130/index.html Cited 2002.
24. Lusser L., Bird Publishing – A Guide to Russian Airliners. INTERNET. http://www.russians.bird.ch/Tu144/TU144P01.html
25. Lux, Boris (Dr.). (1997). The Thrust SSC. INTERNET. http://www.luebeck.com/~lux/re_lx/tssc/ts9701.htm Cited 2002.
26. (Textron) Lycoming Aircraft Piston Engines(current). Piston Engine Selection Guide. INTERNET. http://www.lycoming.textron.com/main.jsp?bodyPage=productSales/engineSelectionGuide/ Cited 2003.
27. Military Parade Ltd (2000). Gas-Turbine Power Plant for the T-80u Main Battle Tank. INTERNET. http://www.milparade.com/2000/39a/03_01.shtml Cited 2002.
28. M-Dot Aerospace. (no date). Complete Stand-Alone Engineering Department. INTERNET. http://www.m-dot.com Cited 2003.
29. Orlando, J. (Major) et al. (10/2/1994). An Artificial Neural Network System for Diagnosing Gas Turbine Engine – Fuel Faults. INTERNET. http://www.emsl.pnl.gov:2080/proj/neuron/papers/kangas.mfpg94.html Cited 2003.
30. Orlando, Steve. Science Daily Magazine. (18/05/1998). Modified Gas Turbine Engine Cuts Fuel Costs And Lowers Emissions. INTERNET. http://www.sciencedaily.com/releases/1998/05/980518153536.htm Cited 2002.
31. Parker, Samantha. The blackbird reborn, Sky and Telescope, June 1993, pp.16-17.
32. Rolls-Royce. (2003). Fact Sheet: Model 250-C20B/J turboshaft engine. INTERNET. http://www.rolls-royce.com Cited 2003.
33. SAE International- Aerospace Engineering Online. (24/04/2001). Technology update: UAV flies extended-range mission. INTERNET. http://www.sae.org/aeromag/techupdate/11-1999/04.htm Cited 2002.
34. Sauron's Creations. (2000). The T-80 line of MBTs is produced at Omsk Machine-Building Plant. INTERNET. http://armor.kiev.ua:8100/fofanov/Tanks/MBT/t-80.html Cited 2003.
35. Sauron's Creations. (2000). The T-90 Main Battle Tank. INTERNET. http://armor.kiev.ua:8100/fofanov/Tanks/MBT/t-90.html Cited 2003.
36. Sayers A.T. (1990). Hydraulics and Compressible Flow Turbomachines.
37. Sereti Aircraft Engines. (current). Sereti AeroSix – 155. INTERNET. http://www.sereti.net Cited 2003.
38. Shigley J.E. (1986). Mechanical Engineering Design, 1st Metric Edition.
39. Tanner, Chuck. PMA Online Magazine. (no date). Microturbines: A Disruptive Technology. INTERNET. http://www.retailenergy.com/articles/microturbines.htm Cited 2002.
40. The Website for Defence Industries – Army (current). Rooivalk Attack Helicopter, South Africa. INTERNET. http://www.army-technology.com/projects/rooivalk/specs.html Cited 18/02/2003.

41. United Technologies Corp. - Pratt & Whitney (current). Product info sheet. PW4000 Superior Technologies.
42. United Technologies Corp. - Pratt & Whitney Canada. Product info sheets. 250 turboprop and turboshaft model specifications.
43. United Technologies Corp. - Pratt & Whitney. (2001). Products: ST6L's. INTERNET. http://www.pratt-whitney.com/ Cited 2003.
44. Valenti, Michael. The American Society of Mechanical Engineers. (1998). Luxury Liners Go Green. INTERNET. http://siteminer.superstats.com/search.html Cited 2003.
45. Voith Turbo GmbH & Co. KG. (5/2002). Product info sheets. Voith Turbo Couplings. http://www.voithturbo.com Cited 2003
46. Wallace, James. Seattle Post. (19/11/2002). Powerful jetliner engine scores big on first flight test. INTERNET. http://seattlepi.nwsource.com/business/87632_newengine19.shtml Cited 2002.
47. Walsh P.P., Fletcher P. (2000). Gas Turbine Performance.
48. Williams, Dave. (4/2/2000). Engine Weight/Size FYI. INTERNET. http://www.angelfire.com/ar/dw42/engfyi.htm Cited 2003.
49. Williams International. Teletext of product info (current). The Williams/Rolls FJ44 engine. (20/11/2000).
50. Cohen H., Rogers G.F.C., Saravanamuttoo H.I.H. (4th ed.) Gas Turbine Theory.
51. Yousef S.H. Najjar & Mohammed Othman, revised 12 June 1986, Fuel effect on vehicular gas turbine engine performance.
52. N.Goudain, July 2005, Numerical simulation of rotating stall in a subsonic compressor.

BIBLIOGRAPHY

1. Bennett, Ian. (Updated: Feb. 2003) Chapter 4, Driven Equipment. INTERNET. http://www.gasturbine.pwp.blueyonder.co.uk/chapter4.htm Cited 2003.

APPENDIX A

AIRCRAFT PERFORMANCE TABULATION

Table A Aircraft Performance List for Personal/Small Business Single Piston Engine Aircraft, in Order of Engine Performance

			GENERAL DESCRIPTION: Dimensions:								Performance:												DESCRIPTION / REMARKS: **Note:** All aircraft are single engined monoplanes with trycicle undercariages
MANUFACTURER:	MODEL:	NAME:	*Qty of Seats:* [total]	*O/All Height:* [m]	*O/All Length:* [m]	*Wing Span:* [m]	*Wing Area:* [m²]	*Aspect Ratio:* b^2/A	*Empty weight:* [kg]	*T/Off weight:* [kg]	*Engine Type:*	*Brake Power:* [kW @ RPM]	*Fuel* Cons:* [l/hr]	*Propeller Type:*	*Prop. Dia.:* [m]	*Wing Load:* [kg/m²]	*Max. vel.:* [km/h]	*Cruise vel.:* [km/h]	*Rate of Climb:* [m/sec]	*Operat. ceiling:* [m]	*Max. Range:* [km]	Reserve:	
Piper Aircraft Co.	PA-18 95	Super Cub	2	2.02	6.83	10.73	16.6	**6.94**	367	680	Continental C-9012-F	65		2bld fixed pitch			180	161	3.6068	4,801	580		All metal, high-wing monoplane
Robin Aircraft Co.	DR-400/120	Dauphin	3	2.23	6.96	8.72	13.6	**5.59**	535	900	Lycoming O-235-L2A	84	25	2bld fixed pitch			241	215	3.0480	3,658	860		Wood, low-wing monoplane
Robin Aircraft Co.	DR-3000/100	DR-3000/100	2	2.66	7.51	9.81	14.5	**6.64**	580	900	Lycoming O-235	87	**25**	2bld fixed pitch			230	210	2.9972	3,962	1,120		Metal, low-wing monoplane
Piper Aircraft Co.	PA-28 161	Warrior II	4	2.22	7.25	10.67	15.8	**7.21**	613	1,105	Lycoming O-320-A2B	110	30	2bld fixed pitch			235	233	3.2715	3,353	1,185		Metal, low-wing monoplane
Piper Aircraft Co.	PA-22-150	Carribean	4	2.53	6.28	8.93	?		499	907	Lycoming O-320	112	**30**	2bld fixed pitch			224	212	3.6830	4,572	850	R	Metal, high-wing monoplane
PZL WarszawaOkecie	160A	Koliber	4	2.80	7.37	9.75	12.7	**7.49**	607	850	Lycoming O-320-D2A	120	**30**	2bld fixed pitch			220	194	?	3,498	960		Metal, high-wing monoplane
Piper Aircraft Co.	PA-28 140	Cherokee 140	3	2.44	7.38	9.14	14.2	**5.88**	535	885	Lycoming O-320-D2G da	120	30	2bld fixed pitch			232	216	?	4,570	900		Metal, low-wing monoplane
Piper Aircraft Co.	PA-28 151/161	Warrior / Cadet	4	2.22	7.25	10.67	15.8	**7.21**	615	1,106	Lycoming O-320-D2G da	120	30	2bld fixed pitch			235	219	?	4,335	1,025		Metal, low-wing monoplane
Cessna Aircraft Co.	172R	Skyhawk	4	2.72	8.28	11.00	16.2	**7.47**	743	1,111	Lycoming IO-360-L2A	120 @ 2,400	35	2bld fixed pitch	1.91	68.8	228	226	3.6576	4,115	1,272		All metal, high-wing monoplane
Piper Aircraft Co.	PA-28 180	Cherokee Arrow	4	2.44	7.38	9.14	14.2	**5.88**	558	1,089	Lycoming IO-360-A	135	**35**	2bld fixed pitch			245	230	?	5,000	1,165		Metal, low-wing monoplane
Robin Aircraft Co.	DR-400/180	Dauphin	4	2.23	6.96	8.72	13.6	**5.59**	600	1,100	Lycoming O-360-A	135	**35**	2bld fixed pitch			278	260	4.1910	4,717	1,450		Wood, low-wing monoplane
Piper Aircraft Co.	PA-28 R180	Cherokee Arrow	4	2.44	7.38	9.14	14.2	**5.88**	626	1,134	Lycoming IO-360-B1E	135	35	2bld const spd			274	260	4.4450	4,572	1,600		Metal, low-wing monoplane
Robin Aircraft Co.	DR-3000/160	DR-3000/160	4	2.66	7.51	9.81	14.5	**6.64**	650	1,150	Lycoming O-320-D2A	135	30	2bld fixed pitch			270	255	4.4450	4,572	1,490		Metal, low-wing monoplane
Piper Aircraft Co.	PA-28 181	Archer III	4	2.22	7.25	10.67	15.8	**7.21**	752	1,155	Lycoming O-360-A4M	135	40	2bld fixed pitch			246	237	3.3883	4,036	924		Metal, low-wing monoplane
Piper Aircraft Co.	PA-18 150	Super Cub	2	2.02	6.88	10.73	16.6	**6.94**	429	794	Lycoming O-320	150	**30**	2bld fixed pitch			210	185	4.8768	5,791	740		All metal, high-wing monoplane
Piper Aircraft Co.	PA-28 R201 T	Cherokee Arrow	4	2.52	8.33	10.80	15.9	**7.34**	786	1,315	Continental TSIO-360-FB	150	36	2bld const spd			330	320	4.7752	?	1,667	R	Metal, low-wing monoplane
Raytheon Beechcraft	C-33	Debonair	4	2.51	7.77	10.00	16.5	**6.06**	807	1,383	Continental IO-470-K	170		2bld const spd			312	298	4.7244	?	957		Metal, low-wing monoplane
PZL WarszawaOkecie	235A	Koliber	4	2.80	7.37	9.75	12.7	**7.49**	705	1,150	Lycoming IO-540-B4B5	175	51	2/3bld con spd			260	248	?	4,497	958		Metal, high-wing monoplane
PZL WarszawaOkecie	35A	Wilga	4	2.96	8.10	11.12	15.5	**7.98**	870	1,300	PZL-AL14-RA	195		2bld const spd			194	157	4.5974	4,039	510	R	Metal, high-wing monoplane
PZL WarszawaOkecie	2000	Wilga	4	2.96	8.10	11.12	15.5	**7.98**	900	1,400	Lycoming IO-540	225	51	3bld const spd			?	190	?	?	1,500	R	Metal, high-wing monoplane
Cessna Aircraft Co.	206H	Stationair	6	2.83	8.61	10.97	16.2	**7.43**	988	1,633	Lycoming IO-540-AC1A5	225 @ 2,700	65	3bld const spd	2.01	100.9	280	263	5.0190	4,785	1,352		All metal, high-wing monoplane
Cessna Aircraft Co.	206H	Turbo Stationair	6	2.83	8.61	10.97	16.2	**7.43**	1,034	1,633	Lycoming TIO-540-AJ1A	231 @ 2,500	74	3bld const spd	2.01	100.9	330	304	5.3340	8,230	1,320		All metal, high-wing monoplane
Raytheon Beechcraft	A-36	Bonanza	6	2.62	8.13	8.38	16.8	**4.18**	1,040	1,665	Continental IO-550-B	225		3bld const spd			340	326	6.1366	?	1,694	R	Metal, low-wing monoplane
Raytheon Beechcraft	B-36 TC	Bonanza	6	2.62	8.13	8.38	16.8	**4.18**	1,104	1,746	Continental TIO-520-UB	225	59	3bld const spd			394	370	5.3492	?	2,022	R	Metal, low-wing monoplane
Piper Aircraft Co.	PA-46-500TP	Malibu Meridian	6	3.44	9.02	13.11	?		680	2,200	Pratt&Whitney PT6A-42A	373 shaft kW		4bld c/s revers.	2.08		?	485	8.8314	?	1,885		Select metal, low-wing monoplane

NOTE *: Only approximate and/or average values!

Aircraft Performance Calculations of Engine Power Requirements at various Flight Conditions:

Note: All calculations are estimates, and based on maximum take-off weights.

GENERAL DESCRIPTION:

		Dimensions:	**Performance:**							
		Wing Area:	*T/Off weight:*	*Break Power:*	*Propeller Type:*	*Prop. Dia.:*	*Wing Load:*	*Operat. ceiling:*	*Max. Range:*	RES.
MANUFACTURER:	MODEL:	[m^2]	[kg]	[kW @ RPM]		[m]	[kg/m^2]	[m]	[km]	
Piper Aircraft Co.	PA-18 95	16.6	680	65 0 0	2bld fixed pitch	0	0	4,801	580	0
Robin Aircraft Co.	DR-400/120	13.6	900	84 0 0	2bld fixed pitch	0	0	3,658	860	0
Robin Aircraft Co.	DR-3000/100	14.5	900	87 0 0	2bld fixed pitch	0	0	3,962	1,120	0
Piper Aircraft Co.	PA-28 161	15.8	1,105	110 0 0	2bld fixed pitch	0	0	3,353	1,185	0
Piper Aircraft Co.	PA-22-150	?	907	112 0 0	2bld fixed pitch	0	0	4,572	850	R
PZL WarszawaOkecie	160A	12.7	850	120 0 0	2bld fixed pitch	0	0	3,498	960	0
Piper Aircraft Co.	PA-28 140	14.2	885	120 0 0	2bld fixed pitch	0	0	4,570	900	0
Piper Aircraft Co.	PA-28 151/161	15.8	1,106	120 0 0	2bld fixed pitch	0	0	4,335	1,025	0
Cessna Aircraft Co.	172R	16.2	1,111	120 @ 2,400	2bld fixed pitch	1.91	68.8	4,115	1,272	0
Piper Aircraft Co.	PA-28 180	14.2	1,089	135 0 0	2bld fixed pitch	0	0	5,000	1,165	0
Robin Aircraft Co.	DR-400/180	13.6	1,100	135 0 0	2bld fixed pitch	0	0	4,717	1,450	0
Piper Aircraft Co.	PA-28 R180	14.2	1,134	135 0 0	2bld const spd	0	0	4,572	1,600	0
Robin Aircraft Co.	DR-3000/160	14.5	1,150	135 0 0	2bld fixed pitch	0	0	4,572	1,490	0
Piper Aircraft Co.	PA-28 181	15.8	1,155	135 0 0	2bld fixed pitch	0	0	4,036	924	0
Piper Aircraft Co.	PA-18 150	16.6	794	150 0 0	2bld fixed pitch	0	0	5,791	740	0
Piper Aircraft Co.	PA-28 R201 T	15.9	1,315	150 0 0	2bld const spd	0	0	?	1,667	R
Raytheon Beechcraft	C-33	16.5	1,383	170 0 0	2bld const spd	0	0	?	957	0
PZL WarszawaOkecie	235A	12.7	1,150	175 0 0	2/3bld con spd	0	0	4,497	958	0
PZL WarszawaOkecie	35A	15.5	1,300	195 0 0	2bld const spd	0	0	4,039	510	R
PZL WarszawaOkecie	2000	15.5	1,400	225 0 0	3bld const spd	0	0	?	1,500	R
Cessna Aircraft Co.	206H	16.2	1,633	225 @ 2,700	3bld const spd	2.01	100.9	4,785	1,352	0
Cessna Aircraft Co.	206H	16.2	1,633	231 @ 2,500	3bld const spd	2.01	100.9	8,230	1,320	0
Raytheon Beechcraft	A-36	16.8	1,665	225 0 0	3bld const spd	0	0	?	1,694	R
Raytheon Beechcraft	B-36 TC	16.8	1,746	225 0 0	3bld const spd	0	0	?	2,022	R
Piper Aircraft Co.	PA-46-500TP	?	2,200	373 aft k 0	4bld c/s revers.	2.08	0	?	1,885	0

AT LEVEL FLIGHT (MAXIMUM VELOCITY):

				estimate	estimate	estimate	estimate	Efficiency estimates: estimate	estimate	estimate	estimate
		Max. EAS:	*Lift Coeff.:*	*Drag Coeff.:*	*L/D Ratio:*	*Drag Force:*	*Thrust Power:*	*Mech. Effic.*	*Hydr. Effic.*	*Prop. Effic.*	*Brake Power:*
MANUFACTURER:	MODEL:	[m/s]	C_L	C_D	C_L/C_D	[N]	[kW]	%	%	%	[kW]
Piper Aircraft Co.	PA-18 95	50.00	0.2624	0.0210	12.50	533.5	35.57	98%	100%	87%	**41.72**
Robin Aircraft Co.	DR-400/120	66.94	0.2365	0.0189	12.50	706.1	47.07	98%	100%	87%	**55.21**
Robin Aircraft Co.	DR-3000/100	63.89	0.2435	0.0195	12.50	706.1	47.07	98%	100%	87%	**55.21**
Piper Aircraft Co.	PA-28 161	65.28	0.2628	0.0210	12.50	866.9	57.80	98%	100%	87%	**67.79**
Piper Aircraft Co.	PA-22-150	62.22						98%	100%	87%	
PZL WarszawaOkecie	160A	61.11	0.2870	0.0230	12.50	666.9	44.46	98%	100%	87%	**52.14**
Piper Aircraft Co.	PA-28 140	64.44	0.2403	0.0192	12.50	694.3	46.29	98%	100%	87%	**54.29**
Piper Aircraft Co.	PA-28 151/161	65.28	0.2631	0.0210	12.50	867.7	57.85	98%	100%	87%	**67.85**
Cessna Aircraft Co.	172R	63.33	0.2738	0.0219	12.50	871.6	58.11	98%	100%	87%	**68.16**
Piper Aircraft Co.	PA-28 180	68.06	0.2652	0.0212	12.50	854.4	56.96	98%	100%	87%	**66.81**
Robin Aircraft Co.	DR-400/180	77.22	0.2172	0.0174	12.50	863.0	57.53	98%	100%	87%	**67.48**
Piper Aircraft Co.	PA-28 R180	76.11	0.2208	0.0177	12.50	889.7	59.31	98%	100%	87%	**69.57**
Robin Aircraft Co.	DR-3000/160	75.00	0.2258	0.0181	12.50	902.2	60.15	98%	100%	87%	**70.55**
Piper Aircraft Co.	PA-28 181	68.33	0.2507	0.0201	12.50	906.2	60.41	98%	100%	87%	**70.86**
Piper Aircraft Co.	PA-18 150	58.33	0.2251	0.0180	12.50	622.9	41.53	98%	100%	87%	**48.71**
Piper Aircraft Co.	PA-28 R201 T	91.67	0.1576	0.0126	12.50	1,031.7	68.78	98%	100%	87%	**80.67**
Raytheon Beechcraft	C-33	86.67	0.1787	0.0143	12.50	1,085.0	72.34	98%	100%	87%	**84.84**
PZL WarszawaOkecie	235A	72.22	0.2780	0.0222	12.50	902.2	60.15	98%	100%	87%	**70.55**
PZL WarszawaOkecie	35A	53.89	0.4625	0.0370	12.50	1,019.9	68.00	98%	100%	87%	**79.75**
PZL WarszawaOkecie	2000							98%	100%	87%	
Cessna Aircraft Co.	206H	77.78	0.2668	0.0213	12.50	1,281.2	85.41	98%	100%	87%	**100.18**
Cessna Aircraft Co.	206H	91.67	0.1921	0.0154	12.50	1,281.2	85.41	98%	100%	87%	**100.18**
Raytheon Beechcraft	A-36	94.44	0.1779	0.0142	12.50	1,306.3	87.09	98%	100%	87%	**102.14**
Raytheon Beechcraft	B-36 TC	#####	0.1389	0.0111	12.50	1,369.8	91.32	98%	100%	87%	**107.11**
Piper Aircraft Co.	PA-46-500TP	0.00						98%	100%	87%	

Note: All calculations below are estimates, and are based on maximum take-off weights.

REQUIREMENTS AT TAKE-OFF AND MAXIMUM RATE OF CLIMB:

												Efficiency estimates:				Comparisons:		
							estimate	estimate		estimate	estimate	estimate	estimate	estimate	estimate			
		Rate of Climb:	*Climb Vel.:*	*AOA:*	*Horiz. Vel.:*	*Lift Coeff.:*	*Drag Coeff.:*	*Drag Force:*	*Drag wt Comp.:*	*Total Drag:*	*Thrust Power:*	*Mech. Effic.*	*Hydr. Effic.*	*Prop. Effic.*	*Brake Power:*	*Rated B.P.*	*Calc. B.P.*	*Percent. Error*
MANUFACTURER:	MODEL:	[m/sec]	[m/s]	[°]	[m/s]	C_L	C_D	[N]	[N]	[N]	[kW]	%	%	%	[kW]	[kW]	[kW]	%
Piper Aircraft Co.	PA-18 95	3.607	50.00	4.14	49.87	0.2638	0.0211	533.5	481.1	1,014.6	67.64	98%	100%	87%	**79.33**	**65**	**79**	**22.0%**
Robin Aircraft Co.	DR-400/120	3.048	66.94	2.61	66.88	0.2370	0.0190	706.1	401.9	1,108.0	73.86	98%	100%	87%	**86.63**	**84**	**87**	**3.1%**
Robin Aircraft Co.	DR-3000/100	2.997	63.89	2.69	63.82	0.2441	0.0195	706.1	414.1	1,120.2	74.68	98%	100%	87%	**87.59**	**87**	**88**	**0.7%**
Piper Aircraft Co.	PA-28 161	3.272	65.28	2.87	65.20	0.2635	0.0211	866.9	543.1	1,410.0	94.00	98%	100%	87%	**110.25**	**110**	**110**	**0.2%**
Piper Aircraft Co.	PA-22-150	3.683	62.22	3.39	62.11				526.5			98%	100%	87%		**112**		
PZL WarszawaOkecie	160A	?	61.11									98%	100%	87%		**120**		
Piper Aircraft Co.	PA-28 140	?	64.44									98%	100%	87%		**120**		
Piper Aircraft Co.	PA-28 151/161	?	65.28									98%	100%	87%		**120**		
Cessna Aircraft Co.	172R	3.658	63.33	3.31	63.23	0.2747	0.0220	871.6	629.2	1,500.9	100.06	98%	100%	87%	**117.36**	**120**	**117**	**-2.2%**
Piper Aircraft Co.	PA-28 180	?	68.06									98%	100%	87%		**135**		
Robin Aircraft Co.	DR-400/180	4.191	77.22	3.11	77.11	0.2178	0.0174	863.0	585.5	1,448.5	96.57	98%	100%	87%	**113.26**	**135**	**113**	**-16.1%**
Piper Aircraft Co.	PA-28 R180	4.445	76.11	3.35	75.98	0.2215	0.0177	889.7	649.5	1,539.2	102.61	98%	100%	87%	**120.35**	**135**	**120**	**-10.9%**
Robin Aircraft Co.	DR-3000/160	4.445	75.00	3.40	74.87	0.2266	0.0181	902.2	668.4	1,570.7	104.71	98%	100%	87%	**122.81**	**135**	**123**	**-9.0%**
Piper Aircraft Co.	PA-28 181	3.388	68.33	2.84	68.25	0.2513	0.0201	906.2	561.7	1,467.8	97.85	98%	100%	87%	**114.77**	**135**	**115**	**-15.0%**
Piper Aircraft Co.	PA-18 150	4.877	58.33	4.80	58.13	0.2267	0.0181	622.9	651.0	1,273.9	84.93	98%	100%	87%	**99.61**	**150**	**100**	**-33.6%**
Piper Aircraft Co.	PA-28 R201 T	4.775	91.67	2.99	91.54	0.1580	0.0126	1,031.7	671.8	1,703.5	113.57	98%	100%	87%	**133.20**	**150**	**133**	**-11.2%**
Raytheon Beechcraft	C-33	4.724	86.67	3.12	86.54	0.1792	0.0143	1,085.0	739.4	1,824.4	121.63	98%	100%	87%	**142.65**	**170**	**143**	**-16.1%**
PZL WarszawaOkecie	235A	?	72.22									98%	100%	87%		**175**		
PZL WarszawaOkecie	35A	4.597	53.89	4.89	53.69	0.4659	0.0373	1,019.9	1,087.7	2,107.6	140.51	98%	100%	87%	**164.80**	**195**	**165**	**-15.5%**
PZL WarszawaOkecie	2000	?										98%	100%	87%		**225**		
Cessna Aircraft Co.	206H	5.019	77.78	3.70	77.62	0.2680	0.0214	1,281.2	1,033.4	2,314.6	154.31	98%	100%	87%	**180.99**	**225**	**181**	**-19.6%**
Cessna Aircraft Co.	206H	5.334	91.67	3.34	91.51	0.1928	0.0154	1,281.2	931.9	2,213.1	147.54	98%	100%	87%	**173.05**	**231**	**173**	**-25.1%**
Raytheon Beechcraft	A-36	6.137	94.44	3.73	94.24	0.1787	0.0143	1,306.3	1,061.0	2,367.3	157.82	98%	100%	87%	**185.10**	**225**	**185**	**-17.7%**
Raytheon Beechcraft	B-36 TC	5.349	109.44	2.80	#####	0.1393	0.0111	1,369.8	836.9	2,206.7	147.12	98%	100%	87%	**172.55**	**225**	**173**	**-23.3%**
Piper Aircraft Co.	PA-46-500TP	8.831										98%	100%	87%		**373**		

PS.: Pressure Altitude has not been incorporated in the preliminary calculations as yet, which might explain the slight discrepancies between the calculated and the actually rated values! These calculations are only estimates, laying the groundwork for further study and research.

Average Value:
-0.111

AVERAGE VALUES:

L/D ratio:	12.5%	fixed.	
A/craft Weight:	*Wing Area:*	*Max. vel.:*	*O/All Effic.:*
[kW]	$[m^2]$	[km/h]	%
1,170	15	260.35	0.85

For Maximum Rate of Climb:			
Total Drag:	*Thrust Power:*	*Calc. B.P.*	*Rated B.P.*
[N]	[kW]	[kW]	[kW]
1,658	111	129.67	149.33

Fuel:
Fuel Cons.:
[l/hr]
38.80

APPENDIX B

ENGINE SPECIFICATION TABLES

Table B1 Aero-engine Performance Comparison Sheet at ISA, SLS

Engine Application:						**Engine Description:**		
No.	**Type:**	**Application:**		**Qty:**	**Date:**	**Model:**	**Description:**	**Manufacturer:**
1	MTJ	REC EXP	Scale model aircraft	1,2	'2001'	MW-54	Single spool, 1 x radl compr., 1 x axl turbine.	Wren Turbines UK
2	MTJ	REC EXP	Scale model aircraft	1,2	'2001'	T100 BB	Single spool, 1 x radl compr., 1 x axl turbine.	SWB Turbines
3	MTJ	REC EXP	Scale model aircraft	1,2	'2001'	SWB-35	Single spool, 1 x radl compr., 1 x axl turbine.	SWB Turbines
4	MTJ	REC EXP	Scale model aircraft	1,2	'2001'	SWB-45	Single spool, 1 x radl compr., 1 x axl turbine.	SWB Turbines
5	MTP MTS	REC EXP H	Scale model aircraft	1,2	'2002'	MW-54 TP	Twin spool, 1 x radl compr, 1 x axl turbine, 1 x free pwr tbine, Spider in shroud.	Wren Turbines UK
6	TS	GTS/ APU	F4- Phantom enigne starter unit	2		Plessey Solent	Rdl gas gen + axl free tbne 1 x radl compr. stage Annul rev flow brnr	Plessey Dynamics, UK (Everett Aero ?)
7	TS	GTS/ APU	AI-20/24 turboprops, helicopters	1	2001	AI-8	Rdl gas gen + axl free tbne	MotorSich
8	TS	GTS/ APU	Harrier engine starter unit/PTO	1		CR-201	Rdl gas gen + axl free tbne b2b twin radl compr. stage Annul rev flow, 5 x brnrs	Lucas Aerospace
9	TS	GST HVAC	Utility / Industrial power	N/A	1997	Capstone	Single spool. 1 x radl compr., 1 x axl turbine, Recuperator.	Capstone MicroTurbine
10	TS	GST HVAC	Utility / Industrial power	N/A	1997	Parallon 75	Single spool. 1 x radl compr., 1 x axl turbine, Recuperator.	ALliedSignal/Honeywell
11	SI	LA	Futuristic aircraft engine		'2000' approx.	Aerosix 155	4 Stroke, 120 deg V6, 24 valve, Quadcam, Twin spark plugs/cyl.	Sereti Aircraft Engines
12	SI	LA	Skyhawk 172 R, Piper PA-28-181, Piper PA-28R-200 Arrow	1		IO-360-A,C	4-cyl, Horiz. opp. Air cooled, Direct drive.	Lycoming
13	SI	LCA LA	Britten Norman BN-2A, Piper PA-23-250	2 1		O-540-A	6-cyl, Horiz. opp. Air cooled, Direct drive.	Lycoming
14	SI	LA	Mooney M20M Bravo	1		TIO-540-AF1B	6-cyl, Horiz. opp. Air cooled, Direct drive.	Lycoming
15	SI	LA	Piper PA-60-700P			(L)TIO-540-U	6-cyl, Horiz. opp. Air cooled, Direct drive.	Lycoming
16	TS	H H H	Bell Textron 206B III MD Helicopters MD500 Eurocopter BO-105	1 1 1		250-C20B/J	Multi-stge compr. 1 x gas gen. tbne, 1 x free pwr tbine.	Allison Gas Turbine Division/ Rolls-Royce UK
17	TS	H H H	Bell 206B MDH 500D/E A109C	1 1 1		250-C20R	Multi-stge compr. 1 x gas gen. tbne, 1 x free pwr tbine.	Allison Gas Turbine Division/General Motors Corp. USA
18	TS	H H H	Bell 206L-3 MDH 530 Sikorsky S76A	1 1 1		250-C30P	Multi-stge compr. 1 x gas gen. tbne, 1 x free pwr tbine.	Allison Gas Turbine Division/General Motors Corp. USA
19	TP	LA	PT6 derivative	1		ST6L-72	3 axl 1 radl compr. 1 x gas gen. tbne, 1 x free pwr tbine.	Pratt & Whitney USA
20	TP	LA	Pilatus PC-7 MK II Turbo Trainer	1		PT6A-25C	Axl gas gen + free tbne, 3 axl 1 radl compr. Annul rev flow brnr	Pratt & Whitney Ca.
21	TP	LA	PT6 derivative	1		ST6L-79	3 axl 1 radl compr. 1 x gas gen. tbne, 1 x free pwr tbine.	Pratt & Whitney USA
22	TP	LA	PT6A-40 derivative	1		ST6L-81	3 axl 1 radl compr. 1 x gas gen. tbne, 2 x free pwr tbines.	Pratt & Whitney USA

Engine Specifications:

No.	Mass	Weight (dry) g = 9.807		Thrust @ ISA		SP Max.		Equiv. Thrust	ESHP	T/W @ ISA	P/W	FF, FC		(E)SFC		AF		AFR (A/F)	TE
	[kg]	[N]	[lb]	[N]	[lb]	[kW]	[SHP]	[lb]		[ratio]	[ratio]	[gpm]	[lb/hr]	[mg/ sec.N]	[lbf/ lb.hr]	[kg/s]	[lb/s]		[%]
														kg/kW.hr					
1	0.75	7.4	1.653	53.4	12.0				3.33	7.26 7.26	2.02	210	27.8	65.6					
2	1.00	9.8	2.200	117.9	26.5				7.36	12.05 12.05	3.35	275	36.4	38.9					
3	3.40	33.3	7.496	154.9	34.8				9.67	4.64 4.64	1.29	373	49.3	40.1	1.473 *1.412*	0.30	0.66	48.2	
4	3.40	33.3	7.496	200.2	45.0				12.50	6.00 6.00	1.67	414	54.8	34.5	1.333 *1.214*	0.34	0.75	49.3	
5	1.60	15.7	3.527	155.7	35.0	5.3	7.1 = ESHPth	35.0	7.10	9.92 9.92	2.01	*358*	47.4	38.3	*1.348*				
6	30.0	294	66			52	70	252		3.81	1.06								
7	145	1,422	320			60	80	290		0.91	0.25	2,000	264.6	***25.9***	*3.288* *0.910*				
8	57.8	567.2	128			65	87 bhp	313		2.46	0.68								
9	490	4,805	1,080			28	38	136		0.13	0.04								24.9
10	1,295	12,700	2,855			75	101	362		0.13	0.04								28.5
11	93.0	912	205			82	110	396		1.93	0.54	253 (19L/hr)	33.5		0.304 lb/shp.hr				
12	132.9	1,303	293			149	200	720		2.46	0.68								
13	169.6	1,664	374			186	250	900		2.41	0.67								
14	212.7	2,086	469			201	270	972		2.07	0.58								
15	248.1	2,433	547			261	350	1,260		2.30	0.64								
16	71.7	703	158			313	420	1,512		9.57	2.66 2.70 @T.O.	2,079	275.0		0.655 0.650 lb/shp.hr	1.56	3.45	45.2	
17	78.5	770	173			336	450	1,620		9.36	2.60 2.86 @T.O.				0.608 lb/shp.hr	1.73	3.82		
18	111.1	1,090	245			485	650	2,340		9.55	2.65 2.65 @T.O.				0.592 lb/shp.hr	2.54	5.60		
19	104	1,020	229			508	681	2,452	681	10.70	2.97					3.00 exhst flow	6.6		23.4
20	156.9	1,539	346	1,512.4 Exhst thrust only	340.0	559	750	3,190	886	9.22	2.56				0.595 0.582 = ESFCth				
21	104	1,020	229			678	909	3,273	909	14.28	3.97					3.24 exhst flow	7.1		24.7
22	136	1,334	300			848	1,137	4,094	1,137	13.65	3.79					3.92 exhst flow	8.6		26.0

No.	Type:	Application:		Qty:	Date:	Model:	Lgth: [m]	Effect. dia.: [m]	GG Spd: [RPM]	PR/CR o/all: [ratio]	BPR [ratio]	EGT [K]	[Deg C]
1	MTJ	REC EXP	Scale model aircraft	1,2	'2001'	MW-54			160,000		N/A		
2	MTJ	REC EXP	Scale model aircraft	1,2	'2001'	T100 BB	0.249	0.114	120,000		N/A	823	550
3	MTJ	REC EXP	Scale model aircraft	1,2	'2001'	SWB-35	0.303	0.135	120,000	3.00 Cmpr.	N/A	1,067	794
4	MTJ	REC EXP	Scale model aircraft	1,2	'2001'	SWB-45	0.303	0.135	126,000	3.40 Cmpr.	N/A	1,053	779
5	MTP MTS	REC EXP H	Scale model aircraft	1,2	'2002'	MW-54 TP			155,000		N/A		
6	TS	GTS/ APU	F4- Phantom enigne starter unit	2		Plessey Solent	0.495	0.203	60,000		N/A		
7	TS	GTS/ APU	AI-20/24 turboprops, helicopters	1	2001	AI-8	0.917	wd,ht: 0.725 0.605	28,500		N/A	1,023 TET	750
8	TS	GTS/ APU	Harrier engine starter unit/PTO	1		CR-201	0.696	wd,ht: 0.466 0.508	77,000 (55kAPU)	3.65 Cmpr.	N/A		
9	TS	GST HVAC	Utility / Industrial power	N/A	1997	Capstone	1.344	wd,ht: 0.714 1.900	96,000		N/A		
10	TS	GST HVAC	Utility / Industrial power	N/A	1997	Parallon 75	2.334	wd,ht: 1.219 2.163			N/A		
11	SI	LA	Futuristic aircraft engine		'2000' approx.	Aerosix 155	0.517	wd,ht: 0.730 0.500	3,200 max.: 5,500		N/A		
12	SI	LA	Skyhawk 172 R, Piper PA-28-181, Piper PA-28R-200 Arrow	1		IO-360-A,C	0.757	wd,ht: 0.870 0.492	2,700	8.70	N/A		
13	SI	LCA LA	Britten Norman BN-2A, Piper PA-23-250	2 1		O-540-A	0.945	wd,ht: 0.848 0.624	2,575	8.50	N/A		
14	SI	LA	Mooney M20M Bravo	1		TIO-540-AF1B	1.022	wd,ht: 0.848 0.727	2,575	8.00	N/A		
15	SI	LA	Piper PA-60-700P			(L)TIO-540-U	1.319	wd,ht: 1.319 0.870	2,500	7.30	N/A		
16	TS	H H H	Bell Textron 206B III MD Helicopters MD500 Eurocopter BO-105	1 1 1		250-C20B/J	0.986	@ inlet: 0.140	50,970	7.10	N/A	961	688
17	TS	H H H	Bell 206B MDH 500D/E A109C	1 1 1		250-C20R	0.986	@ inlet: 0.163	50,970	7.90	N/A		
18	TS	H H H	Bell 206L-3 MDH 530 Sikorsky S76A	1 1 1		250-C30P	1.041	height: 0.638	51,000	8.60	N/A		
19	TP	LA	PT6 derivative	1		ST6L-72	1.230	wd,ht: 0.440 0.480	*50,000*	7.00	N/A	787	514
20	TP	LA	Pilatus PC-7 MK II Turbo Trainer	1		PT6A-25C			*50,000*		N/A		
21	TP	LA	PT6 derivative	1		ST6L-79	1.230	wd,ht: 0.440 0.480	*50,000*	7.00	N/A	862	589
22	TP	LA	PT6A-40 derivative	1		ST6L-81	1.280	wd,ht: 0.440 0.470	*50,000*	8.50	N/A	839	566

Table B1 CONTINUED

Engine Description:

No.	Type:	Application:		Qty:	Date:	Model:	Description:	Manufacturer:
23	TP	LA	Beech 1900	1		PT6A-65	Axl gas gen + free tbne,	Pratt & Whitney Ca.
		LTA	CASA C212-300	2			3 axl 1 radl compr.	
		LCA	Shorts SD-360	2			Annul rev flow brnr	
24	TP	LCA	SAAB 2000, IPTN N250	2	1992	GMA 2100	Multi-stge axl compr.	Allison Gas Turbines
		LTA	L100/C-130J	4			Axl & free pwr tbne, air cooled	
							Annular combustor	
25	TF	LCA	Cessna 526JPATS Citation Jet		1993	FJ44-1A	Twin-shaft turbofan	Williams International/
		LMA	Cessna CJ1		2000		1 fan, 1 LP 1 HP compr.	Rolls-Royce Bristol
							1 HP 2 LP stage tbne	
26	TF	LSA	RQ-4A Global Hawk	1	'2002'	AE3007H	Twin-shaft turbofan	Allison Gas Turbines
							1 fan, 14 axl compr.	
							3 HP 2 LP stage tbine	
27	TF	CA	Airbus A320-200	2	1989	V2500-A1	Twin-shaft turbofan	International Aero
							1 fan, 3 LP 10 HP axl compr.	Engines AG
							2 HP 5 LP stage tbne	
28	TF	CA	MD-90-30/50	2	1995	V2528-D5	Twin-shaft turbofan	International Aero
							1 fan, 4 LP 10 HP axl compr.	Engines AG
							2 HP 5 LP stage tbne	
29	TF	CA	Airbus A321-200	2	1997	V2533-A5	Twin-shaft turbofan	International Aero
							1 fan, 4 LP 10 HP axl compr.	Engines AG
							2 HP 5 LP stage tbne	
30	TF	MA	SAAB Gripen combat aircraft	1		RM12	Twin spl low bpr, annul brnr	Volvo/GE
	+ AB		(ie. JAS 39)				3 fan, 7 axl compr.	(GE: General Electric)
							1HP 1LP tbne, aftrbrnr, con-di	
31	TF	MA	Eurofighter Typhoon cmbt acrft	2	1994	EJ200	Twin spl low bpr, annul brnr	Eurojet Turbo GMBH
	+ AB						3 fan, 5 axl compr.	
							1HP 1LP tbne, aftrbrnr	
32	TF	MA	F15 C/D Eagle combat aircraft	2	1985	F100-PW-220	Twin spl low bpr, annul brnr	United Technologies
	+ AB	MA	F16 A/B/C/D Falcon cmbt acrft	1			3 fan, 10 axl compr.	Corp./Pratt&Whitney
							2HP 2LP tbne, aftrbrnr, con-di	
33	TJ	MA	F4 H-1F (F-4A) Phantom c.a.	2		J79-GE-2	Single spool, axial flow	General Electric
	+ AB	MA	A-5A Vigilante combat aircraft	2			17 axl compr.	
							3 stge tbne, aftrbrnr	
34	TJ	MA	F4-F Phantom combat aircraft	2		J79-MTU-17A	Single spool, axial flow	General Electric
	+ AB						17 axl compr.	
							3 stge tbne, aftrbrnr	
35	TJ	MA	SR-71 Blackbird	2	1962	J58	Single spool, axial flow	United Technologies
	+ AB					(JT11D-20A)	9 axl compr.	Corp./Pratt&Whitney
							2 stge tbne, aftrbrnr	
36	TJ	SST	Concorde	4	1969	Olympus 593	Twin spool, axial flow	Rolls-Royce/
	+ AB					Mk 610	7 LP 7 HP axl compr.	SNECMA
		GST	Utility / Industrial power	N/A			1 HP 1 LP axl tbne	
							Annul brnr, part'l aftrbrnr	

Engine Specifications:

No.	Mass	Weight (dry)		Thrust		SP		Equiv.	ESHP	**T/W**	**P/W**	FF, FC		**(E)SFC**		AF		**AFR**	**TE**
		g =	9.807	@ ISA		Max.		Thrust		@ ISA				[mg/	[lbf/			(A/F)	
	[kg]	[N]	[lb]	[N]	[lb]	[kW]	[SHP]	[lb]		[ratio]	[ratio]	[gpm]	[lb/hr]	sec.N]	lb.hr]	[kg/s]	[lb/s]		[%]
														kg/kW.hr					
23	218.2	2,140	481	800.7	180.0	895	1,200								0.575				
				Exhst thrust only				4,579	1,272	9.52	2.64				0.550	4.50	9.9		
									1,270						lb/shp.hr				
24	715.8	7,020	1,578			4,474	6,000	21,600	6,000	13.69	3.80			0.25	0.410				
														kg/kW.hr	lb/shp.hr				
25	203.2	1,993	448	8,452	1,900			8,452	528	4.24	1.18				0.486	28.7	63.3		
26	717.1	7,033	1,581	36,877	8,290			36,877	2,303	5.24	1.46				0.390	118	260		
27	3,357	32,918	7,400	111,210	25,000			111,210	6,944	3.38	0.94				0.543	359	792		
28	3,583	35,142	7,900	124,555	28,000			124,555	7,778	3.54	0.98				0.543	378	833		
29	3,402	33,363	7,500	146,797	33,000			146,797	9,167	4.40	1.22				0.543	396	872		
30	1,055	10,346	2,326	54,003	12,140			54,003	3,372	5.22	1.45	77,441	10,244	23.9	0.840	68.0	150	52.7	
	1,055	10,346	2,326	80,516	18,100			80,516	5,028	7.78	2.16	244,446	32,335	50.6	1.780	68.0	150	16.7	
31	1,035	10,150	2,282	60,053	13,500			60,053	3,750	5.92	1.64	82,873	10,962	23.0	0.770	77.0	170	55.7	
	1,035	10,150	2,282	90,080	20,250			90,080	5,625	8.87	2.47	264,834	35,032	49.0	1.700	77.0	170	17.4	
32	1,452	14,235	3,200	64,902	14,590			64,902	4,053	4.56	1.27				0.690				
	1,452	14,235	3,200	106,005	23,830			106,005	6,619	7.45	2.07				2.100	103	228		
33	3,620	35,501	7,981	46,041	10,350			10,350	2,875	1.30	0.36				0.870				
	3,620	35,501	7,981	71,841	16,150			16,150	4,486	2.02	0.56				2.000	75	166		
34	1,749	17,149	3,855	52,802	11,870			11,870	3,297	3.08	0.86				0.843				
	1,749	17,149	3,855	79,626	17,900			17,900	4,972	4.64	1.29				1.970	77	170		
35	2,869	28,140	6,326	91,192	20,500			20,500	5,694	3.24	0.90				0.000				
	2,869	28,140	6,326	153,469	34,500			34,500	9,583	5.45	1.51	**400,000**	52,911		5.521				
												approx.							
36	3,202	31,402	7,059	44,484	**10,000**			44,484	2,778	1.42	0.39	175,000	23,149		**8.333**	186	410	63.8	
	3,202	31,402	7,059	169,706	38,150			169,706	10,597	5.40	1.50	375,000	49,604		4.681	186	410	29.8	42.0
			5,814	(incl. 17% AB)													440 ?		

No.	Type:	Application:		Qty:	Date:	Model:	Lgth: [m]	Effect. dia.: [m]	GG Spd: [RPM]	PR/CR o/all: [ratio]	BPR [ratio]	EGT [K]	[Deg C]
23	TP	LA	Beech 1900	1		PT6A-65	1.900	0.480	36,300	9.20	N/A		
		LTA	CASA C212-300	2				@ inlet:		Cmpr.			
		LCA	Shorts SD-360	2				0.220					
24	TP	LCA	SAAB 2000, IPTN N250	2	1992	GMA 2100	2.734	1.152	15,375	16.60	N/A		
		LTA	L100/C-130J	4				@ inlet:	pwr tbne	Cmpr.			
								0.622					
25	TF	LCA	Cessna 526JPATS Citation Jet		1993	FJ44-1A	1.356	0.551		12.80	3.28		
		LMA	Cessna CJ1		2000								
26	TF	LSA	RQ-4A Global Hawk	1	'2002'	AE3007H	2.705	1.105			5.00		
27	TF	CA	Airbus A320-200	2	1989	V2500-A1	3.200	fan dia:		29.70	5.40		
								1.600		Clmb:			
										35.80			
28	TF	CA	MD-90-30/50	2	1995	V2528-D5	3.200	fan dia:		30.00	4.70		
								1.613		Clmb:			
										35.20			
29	TF	CA	Airbus A321-200	2	1997	V2533-A5	3.200	fan dia:		33.40	4.50		
								1.613		Clmb:			
										35.20			
30	TF	MA	SAAB Gripen combat aircraft	1		RM12	4.040	0.884		27.50	0.31		
	+ AB		(ie. JAS 39)										
31	TF	MA	Eurofighter Typhoon cmbt acrft	2	1994	EJ200	4.000	0.850		25.00	0.40		
	+ AB							@ inlet:					
								0.740					
32	TF	MA	F15 C/D Eagle combat aircraft	2	1985	F100-PW-220	4.850	1.180		25.00	0.60		
	+ AB	MA	F16 A/B/C/D Falcon cmbt acrft	1				@ inlet:			0.71		
								0.880					
33	TJ	MA	F4 H-1F (F-4A) Phantom c.a.	2		J79-GE-2	5.283	0.973		12.50	N/A		
	+ AB	MA	A-5A Vigilante combat aircraft	2									
34	TJ	MA	F4-F Phantom combat aircraft	2		J79-MTU-17A	5.301	0.993		13.50	N/A		
	+ AB												
35	**TJ**	MA	SR-71 Blackbird	2	1962	J58	5.377	1.407		8.80	N/A	672	399
	+ AB					(JT11D-20A)							
36	TJ	SST	Concorde	4	1969	Olympus 593	4.039	1.212		15.50	N/A		
	+ AB					Mk 610	7.112	fan dia:					
		GST	Utility / Industrial power	N/A			incl.	1.220					
							nzzle						

Table B2 Industrial Gas Turbine Comparison Sheet at ISA, SLS

No.	Type:	Application:		Qty:	Date:	Model:	Description:	Manufacturer:
37	TJ	LMA	Aermacchi MB 339C	1	1985	Viper 680	Annular flow brnr 8 axl compr. 2 stge tbne	Rolls-Royce
38	TJ/S	GST/ MDV	CHP Offshore power	N/A		601-K9	Twin spool, axial flow	Rolls-Royce Allison
39	TJ/S	GST/ MDV	Utility / Industrial power for: Compressors, elec. generators, pumps, etc.	N/A		LM1600		General Electric
40	TJ/S	MAP, GST/ MDV	COGES, CODAG Utility / Industrial power, CHP, combined cycle applications	N/A		LM2500		General Electric
41	TJ/S	MAP, GST/ MDV	COGES, CODAG Industrial power, CHP, Combined cycle applications	N/A	2000 (Military: 2003)	LM2500+		General Electric
42	TJ/S	MAP GST/ MDV	Potential Utility / Industrial power	N/A		LM6000		General Electric

Table B3 Main Battle Tank Gas Turbine Comparison Sheet at ISA, SLS

No.	Type:	Application:		Qty:	Date:	Model:	Description:	Manufacturer:
1	TS	MBT	M1/M1A1 Abrams	1		AGT-1500	Twin spool, radial gas gen	AVCO Lycoming Textron
2	TS	MBT	T-80U	1		GTD-1250	Triple spool, radial gas gen 1 HP compr. with 1 HP turbine 1 LP compr. with 1 LP turbine Free power tbne, annul rev flow	NPO im. Klimova Enterprise

TableB4 Main Battle Tank Performance Comparison Sheet

No.	Engine Type:	Application:		Qty:	Date:	Engine Model:	Engine Description:	Engine Manufacturers:
1	TS	MBT	M1/M1A1 Abrams	1		AGT-1500	Twin spool, radial gas gen	Lycoming Textron
2	TS APU	MBT	T-80U	1 1		GTD-1250 GTA-18 [18 kW]	Triple spool, radial gas gen 1 HP compr. with 1 HP turbine 1 LP compr. with 1 LP turbine Free power tbne, annul rev flow	NPO im. Klimova Enterprise
3	D, APU	MBT	T-90	1 1		V-92S2 AB-1-P28 [1kW]		Chelyabinsk

Table B2 continued:

No.	Mass:	Weight (dry): g = 9.807		Thrust @ ISA		SP Max.		Equiv. Thrust	ESHP	**T/W** @ ISA	**P/W**	FF, FC		**(E)SFC**		AF		**AFR** (A/F)	**TE**
	[kg]	[N]	[lb]	[N]	[lb]	[kW]	[SHP]	[lb]		[ratio]	[ratio]	[gpm]	[lb/hr]	[mg/ sec.N]	[lbf/ lb.hr]	[kg/s]	[lb/s]		[%]
37	379	3,719	836	19,395	4,360			4,360	1,211	5.22	1.45				0.980	28.8	63.5		
38	1,361	13,345	3,000			6,711	9,000	32,400	9,000	10.80	3.00	27,692	3,663		0.407	24.4	53.9	52.9	34.0
						6,449 kW BASE										exhst flow			
39	3,719	36,477	8,200			14,914	20,000	72,000	20,000	8.78	2.44	56,850	7,520		0.376	47	104		37.0
															lb/shp.hr	exhst flow			
40	4,672	45,818	10,300			25,056	33,600	120,960	33,600	11.74	3.26	94,746	12,533		0.373	70	155		37.0
															lb/shp.hr	exhst flow			
41	5,237	51,357	11,545			30,201	40,500	145,800	40,500	12.63	3.51	108,386	14,337		0.354	86	189		39.0
							39,000 bhp								lb/shp.hr	exhst flow			
42	8,169	80,115	18,010			42,751	57,330	206,388	57,330	11.46	3.18	142,591	18,862		0.329	124	273		42.0
															lb/shp.hr	exhst flow			

Table B3 continued:

No.	Mass:	Weight (dry): g = 9.807		Maximum torque:		SP Max.		Equiv. Thrust	ESHP	**T/W** @ ISA	**P/W**	FF, FC		**(E)SFC**		AF		**AFR** (A/F)	**TE**
	[kg]	[N]	[lb]	[Nm]	[lb.ft]	[kW]	[SHP]	[lb]		[ratio]	[ratio]	[gpm]	[lb/hr]	[mg/ sec.N]	[lbf/ lb.hr]	[kg/s]	[lb/s]		[%]
1			?		3,800	1,119	1,500	3,800	1,056			**1,893**	250						
						@ 3,000rpm					25hp/ton	0.625 gal/min							
2	1,050	10,297	2,315	4,395		932	1,250	4,500	1,250	1.94	0.54	2,882	381		0.305				
															g/kw.hr				

TableB4 continued:

No.	**MBT specs:** l:	w:	ht:	Weight:	Ground Press.:	Max Rd Speed:	X-cntry Speed:	Engine Power:	Power / Weight:	Range: (no ext.)	Fuel Load:
	[m]	[m]	[m]	[kg]	[kg/cm2]	[km/h]	[km/h]	[hp]	[hp/ton]	[km]	[L]
1	7.918	3.653	2.375	60,000	0.921	72	48	1,500	25.00	443	1,885
	(9.766)				(13.1 psi)				25.00		
2	6.982	3.582	2.202	46,000	0.925	70	48	1,250	27.17	335	1,770
	(9.651)		0.451						27.17		
			clrance								
3	6.860	3.780	2.226	50,000	0.910	65	45	1,000	20.00	650	
	(9.530)		0.492								
			clrance								

Table B2 continued:

No.	Type:	Application:	Qty:	Date:	Model:	Lgth: [m]	Effct. dia.: [m]	GG Speed [RPM]	PR/CR o/all [ratio]	BPR [ratio]	EGT [K]	[Deg C]
37	TJ	LMA Aermacchi MB 339C	1	1985	Viper 680	1.806	fan dia: 0.490		6.75	N/A		
38	TJ/S	GST/ CHP MDV Offshore power	N/A		601-K9	9.800	wd,ht: 2.700 3.300	11,500			788	515
39	TJ/S	GST/ Utility / Industrial power for: MDV Compressors, elec. generators, pumps, etc.	N/A		LM1600	4.240	height: 2.030	7,000 power turbine		N/A	783	510
40	TJ/S	MAP, COGES, CODAG GST/ Utility / Industrial power, CHP, MDV combined cycle applications	N/A		LM2500	6.520	height: 2.040	3,600 power turbine		N/A	839	566
41	TJ/S	MAP, COGES, CODAG GST/ Industrial power, CHP, Combined MDV cycle applications	N/A	2000 (Military: 2003)	LM2500+	6.700	height: 2.040	3,600 power turbine		N/A	792	518
42	TJ/S	MAP Potential GST/ Utility / Industrial power MDV	N/A		LM6000	7.300	height: 2.500	3,600 power turbine		N/A	729	456

Table B3 continued:

No.	Type:	Application:	Qty:	Date:	Model:	Lgth: [m]	Width, Hght: [m]	Spec. Power: [hp/m3]	PR/CR o/all [ratio]
1	TS	MBT M1/M1A1 Abrams	1		AGT-1500				
2	TS	MBT T-80U	1		GTD-1250	1.494	1.042 0.888	1,200	10.50

GRAPHICAL PERFORMANCE TABLES

Table C Tabular Representation of Graphical Performance Data

Miniature Gas Turbine Engines:

	1	2	3	4	5	6
ESHP	3.33	7.36	9.67	12.50		7.10
T/W	7.26	12.05	4.64	6.00		9.92
P/W	2.02	3.35	1.29	1.67		2.01
(E)SFC			1.47	1.33		1.35

Own, GTS and APU Gas Turbine Engines:

	1	2	3	4	5
ESHP	70.0	80.5	87.0	37.8	100.6
T/W	3.81	0.91	2.46	0.13	0.13
P/W	1.06	0.25	0.68	0.04	0.04
(E)SFC		**3.29**			

Reciprocating Spark Ignition Engines:

	1	2	3	4	5
ESHP	110	200	250	270	350
T/W	1.93	2.46	2.41	2.07	2.30
P/W	0.54	0.68	0.67	0.58	0.64
(E)SFC	0.30				

Helicopter Turboshaft Gas Turbine Engines:

	1	2	3
ESHP	420	450	650
T/W	9.57	9.36	9.55
P/W	2.66	2.60	2.65
(E)SFC	0.65	0.61	0.59

Turboprop Engines:

	1	2	3	4	5	6
ESHP	681	886	909	1,137	1,272	6,000
T/W	10.70	9.22	14.28	13.7	9.52	9.5
P/W	2.97	2.56	3.97	3.79	2.64	2.64
(E)SFC		0.58			0.58	0.55

Turbofan Engines without afterburners:

	1	2	3	4	5	7	9
ESHP	2,303	6,944	7,778	9,167	3,372	3,750	4,053
T/W	5.24	3.38	3.54	4.40	5.22	5.92	4.56
P/W	1.46	0.94	0.98	1.22	1.45	1.64	1.27
(E)SFC	0.39	0.54	0.54	0.54	0.84	0.77	0.69

Turbofan Engines afterburners only:

	6	8	10
ESHP	5,028	5,625	6,619
T/W	7.78	8.87	7.45
P/W	2.16	2.47	2.07
(E)SFC	1.78	1.70	2.10

Turbofan Engines without/ with afterburners:

	1	1ab	2	2ab	3	3ab
T/W	5.22		5.92		4.56	
T/Wab		7.78		8.87		7.45
P/W	1.45		1.64		1.27	
P/Wab		2.16		2.47		2.07
ESFC	0.84		0.77		0.69	
SFCab		1.78		1.70		2.10
ESHP	3,372		3,750		4,053	
SHPab		5,028		5,625		6,619

Turbojet Engines without afterburners:

	1	3	5	7
ESHP	2,875	3,297	5,694	2,778
T/W	1.30	3.08	3.24	1.42
P/W	0.36	0.86	0.90	0.39
(E)SFC	0.87	0.84		

Turbojet Engines afterburners only:

	2	4	6	8
ESHP	4,486	4,972	9,583	10,597
T/W	2.02	4.64	5.45	5.40
P/W	0.56	1.29	1.51	1.50
(E)SFC	2.00	1.97	5.52	4.68

Turbjet Engines without / with afterburners:

	1	1ab	2	2ab	3	3ab	4	4ab
T/W	1.30		3.08		3.24		1.42	
T/Wab		2.02		4.64		5.45		5.40
P/W	0.36		0.86		0.90		0.39	
P/Wab		0.56		1.29		1.51		1.50
ESFC	0.87		0.84		0.00		0.00	
SFCab		2.00		1.97		5.52		4.68
ESHP	2,875		3,297		5,694		2,778	
SHPab		4,486		4,972		9,583		10,597

Turbjet Engines for MAPs' and GSTs':

	1	2	3	4	5
ESHP	9,000	20,000	33,600	40,500	57,330
T/W	10.80	8.78	11.74	12.63	11.46
P/W	3.00	2.44	3.26	3.51	3.18
(E)SFC	0.407	0.376	0.373	0.354	0.329

ENGINE DESIGN DETAIL CALCULATION SHEETS

Table D1 Basic Design Point Calculations

THRUST REQUIREMENTS:

Fuel type:	Kerosene
LHV:	43,100

Flight condition: ***Steady flight***

Aircraft weight:	1,170	kg
Wing area:	15.23	m^2
Airframe max. equiv. airspeed:	260.3	kph
Lift/Drag ratio:	12.50 :	1

Cl - Lift Coefficient:	0.0182	
Cd - Drag Coefficient:	0.0015	
F drag:	918.29	N
Approx. Shaft Power:	61.220	kW

Power Requirement according to Aircraft Performance Chart:

Flight condition: ***Maximum Rate of Climb***

F drag:	1,658	N

Approx. Shaft Power:	110.552	kW
Calculated Shaft Power:	129.665	kW
Rated Brake Power:	**149.333**	**kW**

Ambient, Optimum Design Point Conditions:

At ISA:

Atmospheric gas:		Dry air	
Elevation above SL:		0	m 0.2882
Operational flight speed:	(ie. 260)	260.3	kph
		72.32	m/sec
	=	**260.3**	**km/h**
Flight Mach number:		0.212	

ISA conditions:		
T ambient:	**288.2**	**K**
=	15.00	°C
P ambient:	**101.3**	**kPa**
Air density:	1.225	kg/m^3
Relative air density:	1.000	
Cp:	1.003	kJ/kg K
Gamma:	1.401	
Speed of sound:	340.4	m/sec

Total temperature T01:	**290.76**	**K**
Total pressure P01:	**104.56**	**kPa**

At Operational Design Condition (ODC):

As at: 13/02/2003; TWR, Doornfontein - 18 km SW Jhb Int'l Airport

Elevation above SL:		2,073	m
Ambient Pressure Pamb:		**82.9**	**kPa**
Ambient Temperature Tamb:		**22.5**	**°C**
		295.65	**K**
		0.29565	TZ:
Operational flight speed:	(ie. 260)	260.3	km/h
		72.3	m/sec

Total temperature T01:	**298.26**	**K**
Total pressure P01:	**85.48**	**kPa**

PA, variables and relative values:		
Pressure Altitude wrt ISA:	1,661	m
Pressure Alt. wrt elevation:	3,734	m
ISA temperature:	263.9	K
	-9.27	°C
Air density:	0.977	kg/m^3
Relative air density:	0.798	
Cp:	1.004	kJ/kg K
Gamma:	1.401	
Speed of sound:	344.8	m/sec
Relative speed of sound:	1.013	
Flight mach number (M):	0.210	
True air speed (TAS):	140.6	kt
Equivalent air speed (EAS):	125.5	kt
Theta:	1.035	
Delta:	0.844	

Ballpark figures (below 11km ASL):		
Pressure Altitude:	3,758	m
ISA temperature:	-9.43	°C
Temp. deviation:	31.93	°C
PA-DA difference:	1,168	m
Design Density Altitude * :	**4,925.6**	**m**

Selection of the above (ISA, ODC): **odc**

Total temperature T01:	**298.26**	**K**
Total pressure P01:	**85.48**	**kPa**
Amb. speed of sound:	**344.76**	**m/sec**
Flight mach no.:	**0.20976**	

* ***Note:*** ***The DDA is an indication of the altitude an aircraft needs to be designed for wrt ISA reference condition for comparison.***

Static temperature T1:	**295.65 K**	**Gamma:**	
Static pressure P1:	**82.9 kPa**		**1.4006**

Water Vapour Content:

Relative humidity: 32 % 40%: 13/02/2003, S.A. Weather Service for Jhb International Airport

Table D2 General Engine Input Parameters

Intake:						
Intake air mass flow:	1.4644	kg/sec				
Intake pressure loss:	0.5	%				
Radial compressor:						
Pressure Ratio:	3.1 : 1		**Compressor input power:**	**194.8421**	**kW**	
Compressor polytropic efficiency:	88	%				
Compressor exit diffuser:						
Compressor exit diffuser pressure loss:	3.0	%				
Turbine disc cooling/rim sealing bleed:	0.5	%				
Bearing chamber sealing/chamber:	0.2	%				
Leakage from high - low pressure air system:	0.5	%				
Customer bleed extraction:	0.0	%				
Is a Recuperator employed? (Y/N):	y					
Recuperator air side:			**Pressure losses:**	**Lambda:**	**%tage:**	
Cooling flow offtake position (Inlet/Exit):	inlet		Compressor delivery to air inlet	0.7 - 1.5	3 - 6	
Recuperator air side inlet duct pressure loss:	3.0	%	Air outlet to combustor inlet	2.5 - 6	1 - 2.5	
Blade and disc cooling flow offtake:	8.0	%	Turbine outlet to gas inlet	0.4 - 1.2	2 - 6	
			Gas outlet	Exh. Duct	Calculation	
Recuperator:						
Recuperator effectiveness:	88	%				
Gas side inlet temperature 1st guess T6ini.:	900	K	**-> 1079.6 K**			
Air side total pressure loss:	3.0	%				
Recuperator inlet duct - Turbine:						
2nd … guess(es):	**1,078.1 K --->** 1,079.6	K	**T6 Error:**	**1.5**	**K**	
			% error:	**0.136**	**%**	
Recuperator exit duct:						
Recuperator exit duct pressure loss:	1.0	%				
Combustor:						
Turbine inlet temperature (SOT) T41:	1,400	K	**FAR:**	**0.0116**		
Combustor pressure loss:	3.0	%	**Fuel mass flow:**	**0.0155**	**kg/sec**	
Equivalent IC engine fuel flow:	0.01323	kg/sec	**Margin for improvement:**	**0.002216**	**kg/s**	
			Relat. fuel flow efficiency:	**85.65549**	**%**	
Fuel type:	Kerosene					
Combustion efficiency:	99.9	%				
Pseudo-station 415 mixing:						
Cooling air addition, doing work: (Max: 8%)	8.0	%				
Turbine:						
Recuperator gas side inlet duct pressure loss:	4.0	%	**Turbine output power:**	**403.3409**	**kW**	
Recuperator gas side pressure loss:	4.0	%	**Capacity WRTP, station 41:**	**0.2178**	**kg√K/s kPa**	
Jet pipe/Exhaust pressure loss:	1.0	%				
Turbine polytropic efficiency:	89	%				
Recuperator exhaust:						
Exhaust design mach number:	0.05		**P9/Ps9:**	**1.002**		
			Exhaust plane area A9:	**0.130**	**m^2**	
			Exhaust plane dia. D9:	**0.406**	**m**	
Final calculations:						
Mechanical efficiency: (0)	99.9	%	**Available shaft power:**	**208.3102**	**kW**	
(Derived: Comp dsgn)			**Spec. power/thrust:**	**142.25**	**Ns/kg**	
			Spec. fuel consumption:	**0.2670**	**kg/kWh**	
			S. P. thermal efficiency:	**31.282**	**%**	

Table D3 Component Input Parameters

General input parameters:

Type of Compressor:	**Centrifugal**	
Relative shroud inlet mach number:	**0.9**	
Relative shroud inlet angle:	**32**	°

Subsonic air-intake - Ram Recovery & Friction Losses:

Type of intake:	**Scroll/Pod mtg**
Lambda:	**0.1**

Type of intake:	**Lambda:**	**Mach no.:**
Pod mounting	0.05-0.1	Exit:0.4-0.6
Scroll	0.5 - 1	Inlt:0.1-0.25
Bypass duct	0.3 - 0.4	0.3 - 0.4
Comb entry diff - axl compr	0.2 - 0.3	0.3 - 0.4
Inter turb duct - axl turbs	0.05-0.2	0.3 - 0.55
Jet pipe - converg nozzle	0.05-0.1	Inlt:0.3- 0.5

Compressor Performance:

Mechanical compressor efficiency:		**97**	%
Number of impeller vanes:	(20 -30)	**30**	
Power Input Factor:	(1.02 -1.05)	**1.04**	
Exit Mach number:	(< 0.2)	**0.19**	
Reaction Ratio:	(+/- 50%)	**50**	%
Diffuser efficiency:		**88**	%
Impeller length:			
Impeller length parameter:	(1.1 -1.3)	**1.20**	
Vaneless space:			
Radius ratio:	(>= 1.05)	**1.05**	
Diffuser parameters:			
Radial diffuser exit to impeller tip radius ratio:		**1.40**	
Diffuser radial to axial (outer) bend:			
Bend parameter:	(0.4 -1.5)	**0.400**	
Diffuser axial walls:			
Swirl angle (bend & straighteners):	(< 10)	**9.0**	
Axial impeller exit depth:		**3.3**	**mm**

NB: Regulates RPM and impeller tip radius!!!

Ind. hub-tip ratio: 0.978525

Recuperator Iterations:

Recuperator gas side:

Gas side inlet temperature 1st guess T6ini.: **500** **K** **-> 1170 K**

Recuperator inlet duct:

2nd … guess(es): **1,169.8 K --->** 1,169.7 K

T6 Error:	**0.0**	**K**
% error:	**-0.003**	**%**

Combustor:

General input parameters:

Combustor pressure cold losses:

Cold loss factor: (Rig tests! [47]) **2.00**

FAR:	**0.0099**	
Fuel mass flow:	**0.0132**	**kg/sec**

Combustor volume:

Critical loading conditions:

a) @ S.L. Static Maximum Rating:

Initial combustor design loading:	(< 5 -10)	**4**	
		kg/sec atm$^{1.8}$ m^3	

b) Idling @ highest altitude, lowest flight Mach number, and coldest day:

Flight Mach number:	Lowest:	**0.2**	
Inlet gas temperature T3:	Lowest:	**200**	**K**
Inlet gas pressure P3:	Lowest:	**350**	**kPa**
Initial combustor design loading:	(< 50-75)	**50**	
		kg/sec atm$^{1.8}$ m^3	

c) When windmilling @ highest altitude, lowest flight Mach number:

Flight Mach number:	Lowest:	**0.2**	
Inlet gas temperature T3:	@ Alt.:	**200**	**K**
Inlet gas pressure P3:	@ Alt.:	**350**	**kPa**
Initial combustor design loading:	(< 300)	**300**	
		kg/sec atm$^{1.8}$ m^3	

Combustor intensity:

Primary zone - air flow and can area:

Equivalence ratio - primary zone:	(+/-1.02)	**1.02**	
Exit Mach number- primary zone:	(.02-.05)	**0.03**	
Exit temperature - primary zone:	(+/-2300)	**2,300**	**K**

Combustor radii:

Outer annuli Mach number:	(+/- 0.1)	**0.10**

Residence time:

Combustor average Mach number:	(+/-0.02)	**0.02**

2ndary air flow:

Equivalence ratio - 2ndary zone:	(+/- 0.6)	**0.60**

Tertiary air flow:

Combustor pressure hot losses:

Hot loss factor:	(Rig tests! [47])	**0.10**

Turbine:

General input parameters:				
Type of turbine:		**Axial**		
Mean inlet mach no. to NGV:	(< 0.2)	**0.190**	n/a?	
Nozzle guide vane-exit angle:	(65 - 73°)	**70**	**°**	
Maximum turbine rim speed:		**400**	**m/sec**	***Note: U max to be larger than 265.67 m/sec***
Diagram efficiency:		**90**	%	***Above 68.94 %***
Ave. Stage Aspect Ratio:	(2.5 -3.5+)	**3.25**		
Blade inlet hub rel. mach number:	(< 0.7)	**0.680**		
Mean inlet NGV mach number:	(< 0.2)	**0.195**		
Turbine-exit condition:				
Hade angle:	(< 15)	**14.5**	°	
Turbine loading:				
Approx. turbine exit mach number:	(+/-0.3)	**0.30**		

Mechanical losses:

General input parameters:			
Type of bearing:	**Ball bearing**		
Bearing race diameter PCD:	**65.0**	**mm**	
Type of lubricating oil:	**Medium mineral**		
Oil temperature:	**100**	**°C**	
Oil flow rate:	**0.0**	**l/hr**	

Table D4 COMPONENT DETAIL AND ENGINE DESIGN

General input parameters:

Total temperature T01:	**298.26**	**K**
Total pressure P01:	**85.48**	**kPa**

Note: Theta and Delta are wrt stagnation conditions for the aircraft velocity.

Theta:	**1.035**
Delta:	**0.844**

Water vapour content:

Ambient air temperature:	295.7 K			
Ambient air pressure:	82.90 kPa		Saturation pressure:	2.735 kPa
Relative humidity:	32 %		Specific humidity:	0.664
			Mass of vapour/kg dry air:	0.007 kg
Mass flow of intake air:		**1.464**	**Mass of vapour/second:**	**0.0097 kg/sec**
Shaft speed:		**12,500**	**Total mass flow/second:**	**1.4644**

Nacelle intake:

Nacelle intake mach number:	0.20976			
Static air pressure P1:		82.90 kPa		
Gamma @ pt.1:		1.401 (guess)	**T/Ts ratio:**	1.009
			P/Ps ratio:	1.031
			Q:	14.272
Mass flow of intake gas:	1.464	kg/sec		
Static air temperature T1:	0.2957	295.65 K	**A2:**	0.0213 m^2
			Nacelle intake diameter:	**0.1646 m**

Compressor:

Operating parameters:

Type of Compressor:	**Centrifugal**			
Name of intake gas:		Dry air		
Gas constant - dry air:		287.05 kJ/kg K		
Relative inlet shroud mach number:	**0.9**			
Relative inlet shroud angle:	**32**	°	**Inlet mach number:**	**0.4769 (0.4-0.6)**
Ambient air temperature:	295.65 K			
Specific heat Cp 2:		1.0037 kJ/kg K		
Heat Ratio @ pt.2:		1.4006	0.3091	**Total temperature T02: 309.1186 K**
				Total air pressure P02': 96.87282 kPa

Theta:	1.073
Delta':	0.956

Subsonic air intake - ram recovery & friction losses:

Type of intake:	**Scroll/Pod mtg**	
Lambda:	**0.1**	
Ambient air pressure:	82.90 kPa	
Intake air pressure P2':	96.87 kPa	**Total air pressure P02: 84.30 kPa**

Note: Theta and Delta are wrt stagnation conditions for the inducer intake.

Theta:	**1.073**
Delta:	**0.832**

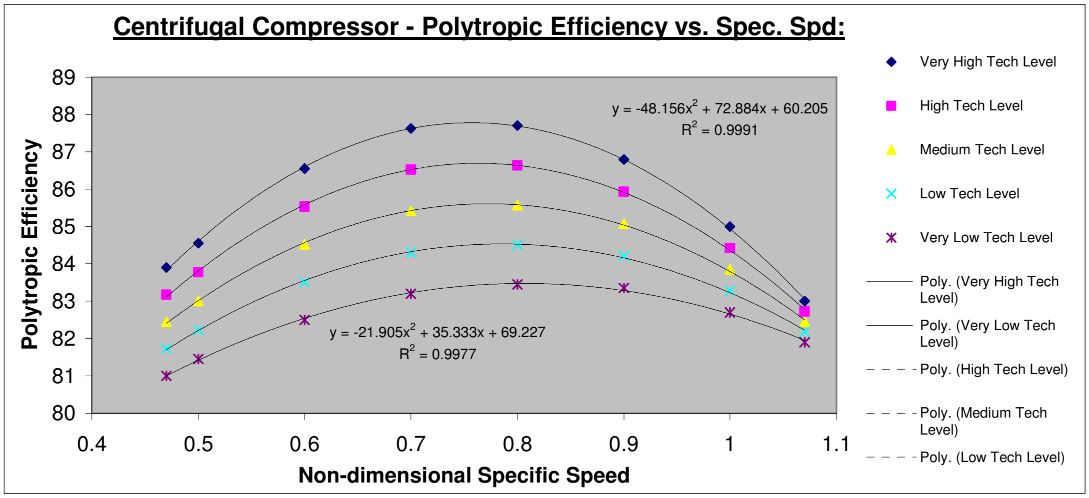
Compressor Performance:

Isentropic efficiency-determination and compressor power required:

Total temperature T02:		309.12	K
Specific heat Cp 2:		1.0037	kJ/kg K
Pressure Ratio (P03/P02):	(For Al <= 4.5)	3.1	
(Gamma - 1) / Gamma @ 2-3:		0.2840	
Gamma/(Gamma - 1) @ 2-3:		3.52109	
Isentropic efficiency (guess) - 01:		0.880	
Total air pressure P02:		84.30	kPa
Optimum gas turbine speed:		12,500	
Mass flow intake air+vapour:		1.464	kg/sec
Specific heat Cp 2-3:		1,010.7	J/kg K

Isentropic energy supply:	**117.57**	**kJ/kg**
Isentropic temp. T3(07)i-01:	**426.26**	**K**
Exit temperature T3(07)-01:	**442.23**	**K**
Specific speed - 01:	**0.2553**	

Isentropic efficiency estimates according to Chart and Specific Speed:



Compressor Tech Level (vl,l,m,h,vh): h

Specific speed: 0.2553

Polytropic Efficiencies wrt Technology and Specific speed:

Very High Tech Level:	vh	75.674	%
High Tech Level:	h	75.960	%
Medium Tech Level:	m	76.247	%
Low Tech Level:	l	76.533	%
Very Low Tech Level:	vl	76.820	%

Gamma/(Gamma - 1) @ 2-3: 3.5211

Polytropic efficiency:	**75.960**	**%**
Isentropic efficiency:	**71.968**	**%**
Error:	**-22.277**	**%**

Iteration-data table:

Iterations:	Factors: Isen Eff.	H.R.@3:	T307:	Spc.Spd:	Poly.Eff.
1	0.8800	1.3898	440.31	0.2583	75.96
2	0.7202	1.3923	470.27	0.2569	76.03
3	0.7208	1.3896	469.22	0.2580	76.07
4	0.7215	1.3897	469.09	0.2580	76.07
5	0.7215	1.3897	469.10	0.2580	76.07
6	0.7215	1.3897	469.10	#DIV/0!	#DIV/0!

Specific heat Cp 2-3:	1,009.9 J/kg K
Specific heat Cp 2-3:	1,011.7 J/kg K
Specific heat Cp 2-3:	1,011.6 J/kg K
Specific heat Cp 2-3:	1,011.6 J/kg K
Specific heat Cp 2-3:	1,011.6 J/kg K

	1	2	3	4	5
vh	75.751	75.805	75.803	75.803	#DIV/0!
h	76.028	76.075	76.073	76.073	#DIV/0!
m	76.305	76.346	76.344	76.344	#DIV/0!
l	76.581	76.615	76.614	76.614	#DIV/0!
vl	76.859	76.886	76.885	76.885	#DIV/0!

			Isentropic efficiency:	**72.150 %**
			Error:	**0.000 %**
			Isentropic temp. T3(07)i:	**426.26 K**
Total air temperature T02:	309.12 K		**Exit temperature T3(07):**	**471.48 K**
Pressure Ratio (P03/P02):	3.1			
(Gamma - 1)/Gamma @ 2-3:	0.2840		**Polytropic efficiency:**	**76.117 %**
			Polytropic eff. check:	**76.117 %**
Specific heat Cp 2-3:	1,011.6 J/kg K			
Total air pressure P02:	84.30 kPa			
Mass flow intake air+vapour:	1.464 kg/sec			
Optimum turbine speed:	12,500 rpm		**Specific speed:**	**0.2552** (SI Units)

Compressor input power:

Number of impeller vanes:	**30**		**Slip Factor:**	**0.9340**
Power Input Factor:	**1.040**			
Pressure Ratio (P03/P02):	3.1	*Static T.:*		
Total air temperature T02:	309.12 K	*295.65 K*		
Specific heat Cp 2-3:	1,010.7 J/kg K			
Compressor efficiency:	0.7215			
(Gamma - 1)/Gamma @ 2-3:	0.2840		**Impeller rim speed U3:**	**411.01 m/sec**
Slip Factor:	0.9340			
Total mass flow/second:	1.464 kg/sec		**Compressor hydr. power:**	**240.3 kW**
Mechanical compressor efficiency:	**97** %		**Compressor input power:**	**247.7 kW**
			Overall Compr. efficiency:	**69.985 %**

Compressor exit temperature-check:

Compressor hydr. power:	240.3 kW		
Total air temperature T02:	309.119 K		
Mass flow intake air+vapour:	1.464 kg/sec		
Specific heat Cp 2-3:	1,010.7 J/kg K	**Exit temperature T3(07):**	**471.48 K**

Compressor input power-check:

Overall Compressor efficiency:	0.69985	Modified exit temperature:	476.4968 K
Total air temperature T02:	309.12 K		
Mass flow intake air+vapour:	1.464 kg/sec		
Specific heat Cp 2-3:	1,010.7 J/kg K	**Compressor input power:**	**247.7 kW**

Exducer/Inducer:

Pressure Ratio (P03/P02):	3.1		
Total air pressure P02:	84.30 kPa	**Total air pressure P3':**	**261.32 kPa**
Heat Ratio @ pt.2:	1.4006		
Gas constant - dry air:	287.05 kJ/kg K		
Static inlet temperature:	295.65 K	**Local speed of sound:**	**344.76 m/sec**
Inlet mach number:	0.4769	**Inlet air velocity V2:**	**164.43 m/sec**
Gamma @ pt.3:	1.3898		
Gas constant - dry air:	287.05 kJ/kg K		
Exit temperature T3(07):	471.48 K	**Appr. local speed of sound:**	**433.70 m/sec**
Exit mach number:	0.190	**Exit air velocity V3:**	**82.40 m/sec**

Impeller exit conditions:

Total temperature T02:	309.12	K			
Inlet air velocity V2:	164.43	m/sec			
Specific heat Cp 2:	1,003.7	J/kg K	**Static T inlet:**	**295.7**	**K**
			Check:	**295.7**	**K**
Exit temperature T3(07):	471.48	K			
Specific heat Cp 2-3:	1,010.7	J/kgK			
Exit air velocity V3:	82.40	m/sec	**Static T exit - T inlet:**	**172.4**	**K**
Reaction Ratio:	50	%	**Static T imp. exit - T inlet:**	**86.2**	**K**
			Static T @ impeller exit:	**381.8**	**K**
			Static T @ impeller exit/1000:	0.3818	
			Specific heat @ impeller exit:	1.0107	kJ/kgK
			Approximate Heat Ratio:	1.3967	
			Impeller exit air velocity:	**425.68**	**m/sec**
Gas constant - dry air:	287.05	kJ/kg K	**Local speed of sound:**	**391.26**	**m/sec**
			Impeller exit mach no.:	**1.088**	
Slip Factor:	0.9340				
Impeller Rim Speed:	411.01	m/sec	**Whirl vel. @ impeller o/let:**	**383.9**	**m/sec**
Impeller exit air velocity:	425.68	m/sec	**Rad'l vel. @ impeller o/let:**	**183.91**	**m/sec**
Reaction Ratio:	50	%			
Static T exit - T inlet:	172.37	K	**Static T exit - T imp. exit:**	**86.2**	**K**
Static T @ impeller exit:	381.84	K	**Static exit temperature:**	**468.0**	**K**
			Average impeller to exit T:	**424.9**	**K**
			Average imp. to exit T/1000:	0.4249	K
			Specific heat @ impeller exit:	1.0164	kJ/kgK
			Approximate Heat Ratio:	1.3936	
Gamma/(Gamma - 1):	3.5409				
Diffuser efficiency:	88	%	**Exit to impeller o/let P.R.:**	**1.8994**	
Gamma/(Gamma - 1):	3.5409				
Compressor exit gas pressure P3':	261.322	kPa			
Static exit temperature:	468.02	K			
Exit temperature T3(07):	471.48	K	**Static exit pressure:**	**254.61**	**kPa**
Exit to impeller o/let P. R.:	1.8994		**Static imp. exit pressure:**	**134.05**	**kPa**
Gamma/(Gamma - 1):	3.5409				
Static T @ impeller exit:	381.84	K	**Impeller exit pressure:**	**282.84**	**kPa**

Impeller speed:

Static imp. exit pressure:	134.05	kPa			
Static T @ impeller exit:	381.84	K			
Gas constant - dry air:	287.05	kJ/kg K	**Impeller exit air density:**	**1.223**	**kg/m^3**
Axial impeller exit depth:	0.0033	m			
Rad'l vel. @ impeller o/let:	183.91	m/sec			
Impeller Rim Speed:	411.01	m/sec			
Mass flow intake air+vapour:	1.464	kg/sec	**Impeller wheel velocity:**	**12,500**	**rpm**
			Impeller tip radius:	**0.3140**	**m**

Impeller total-to-total efficiency:

Static T inlet:	295.65	K			
Static T @ impeller exit:	381.84	K	**Average intake to imp. T:**	**338.7**	**K**
			Average imp. to exit T/1000:	0.34	K
			Specific heat @ impeller exit:	1.0065	kJ/kgK
			Approximate Heat Ratio:	1.3990	
(Gamma - 1)/Gamma:	0.2852				
Total temperature T02:	309.12	K			
Exit temperature T3(07):	471.48	K			
Total air pressure P02:	84.30	kPa			
Impeller exit pressure:	282.84	kPa	**Impeller t-t efficiency:**	**78.51**	**%**

Impeller inlet area:

Relative inlet shroud mach number:	0.9		
Relative inlet shroud angle:	32 °		
Inlet air velocity V2:	164.43 m/sec	**Inducer max. tip speed U2:**	**102.74 m/sec**
Total air pressure P02:	84.30 kPa		
Total air temperature T02:	309.12 K		
Heat Ratio (Gamma) 2:	1.4006		
Static T inlet:	295.65 K	**Static imp. inlet pressure:**	**72.14 kPa**
Static T @ impeller inlet:	295.65 K		
Gas constant - dry air:	287.05 kJ/kg K	**Impeller entry air density:**	**0.8500 kg/m^3**
Mass flow of intake air:	1.464 kg/sec		
Inlet air velocity V2:	164.43 m/sec	**Impeller entry area:**	**0.0105 m^2**

Area from Q-curve formulae:

Max. inlet mach number:	(0.4-0.6)	0.4769		
Total air pressure P02:		84.30 kPa		
Gamma @ pt.2:		1.4006	**T/Ts ratio:**	1.046
			P/Ps ratio:	1.169
			Q:	29.151
Mass flow of intake air:		1.464 kg/sec		
Total air temperature T02:	0.3091	309.12 K	**Impeller entry area:**	**0.0105 m^2**
			Ave. impeller entry area:	**0.0105 m^2**

Inducer:

Impeller wheel velocity:	12,500 rpm			
Inducer tip speed U2:	102.74 m/sec		**Inducer tip radius:**	**0.0785 m**
Impeller tip radius:	0.3140 m		**Shroud-tip ratio:**	**0.2500**
Ave. impeller entry area:	0.0105 m^2	0.35 - 0.5, 0.7 max.)	**Inducer hub-tip ratio:**	**0.9785**
	0.0578 m	(Min: 0.05775 m)	**Inducer hub radius:**	**0.0768 m**
			Inducer hub speed U2:	**100.54 m/sec**

Impeller length:

Impeller length parameter:	(1.1 -1.3)	**1.20**		
Impeller tip radius:	0.3140 m			
Inducer tip radius:	0.0785 m			
Inducer hub radius:	0.0768 m		**Impeller length:**	**0.3758 m**

Varying relative inlet angles - hub to tip:

Inlet air velocity V2:	164.43 m/sec		**Relative inlet hub angle:**	**31.44 °**
		Check:	**Relative inlet tip angle:**	**32.00 °**
Impeller rim speed U3:	411.01 m/sec			
Rad'l vel. @ impeller o/let:	183.91 m/sec			
Whirl vel. @ impeller o/let:	383.90 m/sec		**Exducer vane backsweep:**	**8.39 °**
Impeller exit air velocity:	425.68 m/sec		**Diffuser vane inlet angle:**	**25.60 °**

Diffuser components:

Vaneless space:

Radius ratio:	(>= 1.05)	**1.05**		
Impeller tip radius:	0.3140 m		**Diffuser vane inlet radius:**	**0.3297 m**

Diffuser parameters:

Impeller tip radius:	0.3140 m			
Radial diffuser exit to impeller tip radius ratio:		**1.40**	**Diffuser vane exit radius:**	**0.4396 m**
Local speed of sound:	391.26 m/sec			
Rad'l vel. @ impeller o/let:	183.91 m/sec		**Radial exit mach no. compnt:**	**0.470 (+/- 0.25)**
Exit temperature T3(07):	471.48 K			
Gamma @ pt.3:		1.3898	**T/Ts ratio:**	1.043
			P/Ps ratio:	1.162
			Q:	28.743
Total air pressure 3':	261.322 kPa			
Mass flow intake air+vapour:	1.464 kg/sec		**Radial diffuser area:**	**0.0042 m^2**
			Diffuser vane height:	**0.0096 m**

Diffuser radial to axial (outer) bend:

Bend parameter:	(0.4 -1.5)	**0.400**	**Radial-axial outer diff. wall radius:**	**0.4434 m**

Diffuser axial walls:

Swirl angle (bend & straighteners):	(< 10)	**9.0**		
Exit Mach number:	(< 0.2)	**0.190**	**T/Ts ratio:**	1.007
			P/Ps ratio:	1.025
			Q:	12.940
Mass flow intake air+vapour:	1.464 kg/sec			
Exit temperature T3(07):	471.48 K			
Total air pressure 3':	261.322 kPa			
Outer diffuser wall radius:	0.4434 m		**Axial straightener area:**	**0.0094 m^2**
Diffuser vane exit radius:	0.4396 m		**Axial straightener outer wall radius:**	**0.8830 m**
			Axial straightener inner wall radius:	**0.8813 m**
			Radial-axial inner diffuser wall radius:	**0.4417 m**

Compressor exit diffuser:

Compressor exit gas pressure P3':	261.322 kPa		
Compressor exit diffuser pressure loss:	3.0 %	**Compr. exit diffuser pressure P3:**	**253.48 kPa**
Turbine disc cooling/rim sealing bleed:	0.5 %		
Bearing chamber sealing/chamber:	0.2 %		
Leakage from high - low pressure air system:	0.5 %		
Customer bleed extraction:	0.0 %	**Flow rate after offtake & leaks:**	**1.4468 kg/sec**

SINGLE TURBINE ARRANGEMENT

Recuperator:

Is a Recuperator employed? (Y/N):		y					
Recuperator inlet duct:							
Cooling flow offtake position (Inlet/Exit):		inlet			**T307 = T3:**	**471.5**	**K**
Recuperator air side inlet duct pressure loss:		3.0	%		**P307:**	**245.88**	**kPa**
Blade and disc cooling flow offtake:		8.0	%		**Mass flow W307:**	**1.330**	**kg/sec**
					As W308 = W307:	**1.330**	**kg/sec**
Recuperator:							
Recuperator effectiveness:		88	%				
Air side inlet temperature T307:	471.5	K					
Gas side inlet temperature 1st guess T6ini.:		500	K	-> 1170 K	**Exit temperature T308:**	**1,085.9**	**K**
Air side total pressure loss:		3.0	%		**Exit pressure P308:**	**238.5**	**kPa**
Recuperator exit duct:							
T31 = T308:	1,085.9	K					
Recuperator exit duct pressure loss:		1.0	%		**Exit pressure P31:**	**236.1**	**kPa**
					Flow rate after exit W31:	**1.3297**	**kg/sec**
Exit temperature T31:	1,085.9	K					
Impeller exit air velocity:	425.68	m/sec					
Specific Heat Cp @ T31:		1.1614	kJ/kg K		**Static exit temperature:**	**1,007.93**	**K**

Combustor:

General input parameters:

Type of fuel:	Kerosene			
LHV:	43,124 kJ/kg			
Air flow for combustion:	1.32967 kg/sec			
FAR:	**0.0099**	**Fuel flow rate:**	**0.0132**	**kg/sec**
		Exit flow W4(1):	**1.343**	**kg/sec**

Combustor pressure cold losses:

Cold loss factor:	**2.00**			
Mass flow of intake gas:	1.46439 kg/sec			
Inlet gas temperature T31 = T308:	1,085.9 K			
Inlet gas pressure P31:	236.116 kPa	**Combustor cold loss:**	**0.084**	**kPa**
		Inlet gas pressure P31':	**236.03**	**kPa**

Combustor volume:

Critical loading conditions:

a) @ S.L. Static Maximum Rating:

Mass flow of intake W31:	1.32967 kg/sec			
Inlet gas temperature T31:	1,085.9 K			
Inlet gas pressure P31':	236.033 kPa		Equally, inlet gas pressure:	2.3295 atm
Initial combustor design loading:	(< 5 -10)	**4** **kg/sec atm$^{1.8}$ m^3**	Approx. unrestr. comb. efficiency:	99.686 %
			a) Combustor volume:	7.3E-03 m^3

b) Idling @ highest altitude, lowest flight Mach number, and coldest day:

Flight Mach number:	Lowest:	**0.2**		
Inlet gas temperature T31:	Lowest:	**200 K**		
Inlet gas pressure P31':	Lowest:	**350 kPa**	Equally, inlet gas pressure:	3.4542 atm
			Density of intake gas @ opt.:	1.000 kg/m^3
			Density of intake gas @ alt.:	1.000 kg/m^3
			Relative density wrt. optimum:	1.000
Mass flow @ optimum:	1.32967 kg/sec		Mass flow of intake gas:	1.329665 kg/sec
Initial combustor design loading:	(< 50-75)	**50** **kg/sec atm$^{1.8}$ m^3**	Approx. unrestr. comb. efficiency:	99.361 %
			b) Combustor volume:	5.6E-03 m^3

c) When windmilling @ highest altitude, lowest flight Mach number:

Flight Mach number:	Lowest:	**0.2**		
Inlet gas temperature T31:	@ Alt.:	**200 K**		
Inlet gas pressure P31':	@ Alt.:	**350 kPa**	Equally, inlet gas pressure:	3.4542 atm
			Density of intake gas @ opt.:	1.000 kg/m^3
			Density of intake gas @ alt.:	1.000 kg/m^3
			Relative density wrt. optimum:	1.000
Mass flow @ optimum:	1.32967 kg/sec		Mass flow of intake gas:	1.329665 kg/sec
Initial combustor design loading:	(< 300)	**300** **kg/sec atm$^{1.8}$ m^3**	Approx. unrestr. comb. efficiency:	78.829 %
			c) Combustor volume:	9.3E-04 m^3
			Combustion efficiency:	**99.69**
			Combustor volume:	**7.3E-03 m^3**

Combustor intensity:

Fuel flow rate:	0.0132 kg/sec		
Combustion efficiency:	99.686 %		
LHV:	43,124 kJ/kg		
Inlet gas pressure P31':	236.033 kPa	Equally, inlet gas pressure:	2.3295 atm
Combustor volume:	7.3E-03 m^3	**Combustor intensity:**	**33.10** **MW/atm m^3**

Primary zone - air flow and can area:

FAR Stoichiometric:		0.0666		
Equivalence ratio - primary zone:	(+/-1.02)	**1.02**	FAR local:	0.0679
Fuel flow rate:	0.0132 kg/sec		**Mass flow - primary zone:**	**0.1940 kg/sec**
Exit Mach number- primary zone:	(.02-.05)	**0.03**		
Inlet gas pressure P31':	236.033 kPa			
Exit temperature - primary zone:	(+/-2300)	**2,300 K**		
Gamma @ pt.4:		1.3157	**T/Ts ratio:**	1.000
			P/Ps ratio:	1.001
			Q:	2.030
Mass flow of intake gas:	0.1940 kg/sec		**Overall can area:**	**0.0194 m^2**

Combustor radii:

Overall can area:	0.0194 m2		**Can radius:**	**0.0786 m**
Outer annuli Mach number:	(+/- 0.1)	**0.10**		
Inlet gas pressure P31':	236.033 kPa			
Isentropic gas temperature T3(07)i:	426.26 K			
Gamma @ pt.3:		1.3898	**T/Ts ratio:**	1.002
			P/Ps ratio:	1.007
			Q:	6.917
Mass flow of intake gas:	1.46439 kg/sec		**Outer annular area:**	**0.0185 m^2**
			Outer annular radius:	**0.1099 m^2**

Combustor length:

Can volume:	0.0073 m3		
Can area:	0.0194 m2	**Combustor can length:**	**0.3783 m**

Residence time:

Combustor average Mach number:	(+/-0.02)	**0.02**		
Exit temperature - primary zone:	2,300 K			
Fuel-Gas mixture constant:		287.05 J/kg K		
Average gamma @ 3-4:		1.3220	**Residence velocity:**	**18.685 m**
Combustor can length:	0.378 m		**Residence time:**	**20.2 msec**

2ndary air flow:

FAR Stoichiometric:	0.0666			
Equivalence ratio - 2ndary zone:	(+/- 0.6)	**0.60**	FAR local:	0.0400
Fuel flow rate:	0.0132 kg/sec		**Mass flow - 2ndary zone:**	**0.330 kg/sec**

Tertiary air flow: **Mass flow - tertiary zone: 0.941 kg/sec**

Combustor pressure hot losses:

Hot loss factor:	**0.10**		
Combustor exit temp. T4(1):	1,400.0 K		
Mass flow of intake gas:	1.32967 kg/sec		
Inlet gas temperature T31:	1,085.9 K		
Inlet gas pressure P31':	236.0 kPa	**Combustor hot loss:**	**0.235 kPa**
		Exit gas pressure P4:	**235.80 kPa**

Pseudo-station 415 mixing:

Cooling air addition, doing work:	(Max: 8%)	8.0 %		
Combustor exit flow W41:	1.343 kg/sec		**Mass flow W415:**	**1.460 kg/sec**
Temperature T41 = T4:	1,400.0 K			
Temperature T307, or 8:	471.5 K			
Specific Heat Cp @ T307, or 8:		1.0240 kJ/kg K		
Specific Heat Cp @ T4:		1.1962 kJ/kg K	**Specific Heat Cp @ T415:**	**1.1825 kJ/kg K**
			Approx. check Cp @ 415:	**1.1539 kJ/kg K**
			Turbine rotor inlet T415:	**1,335.4 K**
			Approx. check:	**1,367.2 K**

Turbine:

General input parameters:

Type of Turbine:	**Axial**
Absolute stator inlet/exit temperature:	1,400.0 K
Absolute rotor inlet temperature:	1,335.4 K

Note: a) Purely axial inlet and exit flows have been considered.
b) All calculations are at the mean diameter of the stage.

Turbine-stage calculations:

Maximum turbine rim speed:	**400.00**	**m/sec**		**Actual turbine rim speed:**	**380.91**	**m/sec**
Shaft speed:	12,500	rpm		**Shaft speed:**	**1,309.0**	**rad/sec**
				Rim-speed Correction Factor:	***0.0150***	(See: T-07)
				Rim-speed Correction:	0.038	***0.053***
Mean rotor blade speed:	**378.8**	**m/sec**		**Mean rotor blade speed:**	**378.80**	**m/sec**
				Mean blade diameter:	**0.5788**	**m**
Turbine output power:	**402**	**kW**				
Rotor inlet mass flow rate:	1.460	kg/sec		**Specific Engy requirement:**	**275.49**	**kJ/kg**

Velocity diagrams:

Diagram efficiency:	**90.0**	**%**				
Specific Engy requirement:	275.49	kJ/kg	C_1	**Absolute rotor inlet velocity:**	**782.43**	**m/sec**
				Check - Euler:	**782.43**	**m/sec**
Absolute rotor inlet temperature:	1,335.4	K				
Absolute rotor inlet velocity:	782.43	m/sec				
Specific heat Cp 415:	1,153.9	J/kg K		**Static rotor inlet temp.:**	**1,070.2**	**K**
Mean rotor blade speed:	378.80	m/sec				
Specific Engy requirement:	275.49	kJ/kg	C_{x1}	**Rtor inlet abs whirl cmp'nt:**	**727.26**	**m/sec**
Rtor inlet abs whirl cmp'nt:	727.26	m/sec				
Mean rotor blade speed:	378.80	m/sec	W_{x1}	**Rtor inlet rel whirl cmp'nt:**	**348.46**	**m/sec**
Rtor inlet abs whirl cmp'nt:	727.26	m/sec				
Absolute rotor inlet velocity:	782.43	m/sec	C_a	**Absl. axl rtor inlet velocity:**	**288.59**	**m/sec**
Absl. axl rtor inlet velocity:	288.59	m/sec				
Mean rotor blade speed:	378.80	m/sec		**Flow coefficient:**	**0.762**	(Best: 0.8)
Rtor inlet abs whirl cmp'nt:	727.26	m/sec				
Absolute stator exit vel.:	782.43	m/sec		**Absolute stator exit angle:**	**68.36**	°
Rtor inlet rel whirl cmp'nt:	348.46	m/sec				
Absl. axl rtor exit velocity: Density!	288.59	m/sec		**Stator exit blade angle:**	**50.37**	°
Flow coefficient inverse:	1.3126	m/sec		**Rtor relat. exit blade angle:**	**52.70**	°
Mean rotor blade speed:	378.80	m/sec				
Absolute axial rotor exit velocity:	288.59	m/sec	W_2	**Rtor exit rel whirl velocity:**	**476.21**	**m/sec**
NGV exit blade angle:	50.37	°		**Check:**	**476.21**	**m/sec**
Rtor relat. exit blade angle:	52.70	°				
Flow coefficient:	0.762			**Blade Load Coefficient:**	**1.920**	
			R	**Reaction Ratio:**	**0.040**	**4.01%**
						(Best: 0.5, > 0.3)

Turbine efficiency-loss coefficients:

STATOR:

Absolute stator inlet angle:	0.00 °		
Absolute stator exit angle:	68.36 °	**Deflection angle - stator:**	**68.36 °**
		Nom. loss coeff.@ AR 3, Re 10^5 :	**0.0680**
Aspect Ratio:	3.250		
		Nom. loss coeff. @ Re 10^5 :	**0.0675**
Abs. stator exit temperature T4(1):	1,400.0 K		
Absolute stator exit velocity:	782.43 m/sec		
Specific heat Cp 415-6:	1,159.0 J/kg K	**Static stator exit temp.:**	**1,135.9 K**
Nominal loss coefficient-S:	**0.0283**	**Static isentropic exit temp.:**	**1,128.4 K**
Absolute stator inlet pressure:	235.80 kPa		
Absolute stator inlet temperature:	1,400.0 K		
Heat Ratio (Gamma) @ 415-6:	1.3292	**Static stator exit pressure:**	**98.72 kPa**
		Absolute stator exit press.:	**229.61 kPa**
		Absolute stator blade P loss:	**2.63 %**
Gas constant - dry air:	287.05 kJ/kg K	**Stator exit air density:**	**0.3028 kg/m^3**
Static stator exit temp.:	862.74 °C	**Stator exit gas viscosity:**	**4.2E-05 kg/m.sec**
		Stator exit kinematic visc.:	**1.4E-04 m^2/sec**
Absolute stator exit velocity:	782.43 m/sec		
Mean blade diameter:	0.5788 m	**Reynold's coefficient:**	**3.3E+06**
		Nominal loss coefficient-S:	**0.0283**

ROTOR:

Total ambient pressure (P09 = P01):	85.48 kPa		
Recuperator gas side inlet duct pressure loss:	4.0 %		
Recuperator gas side pressure loss:	4.0 %		
Jet pipe/Exhaust pressure loss:	1.0 %	**Exit pressure P416:**	**93.69 kPa**
Stator exit blade angle:	50.37 °		
Rtor relat. exit blade angle:	52.70 °	**Deflection angle - rotor:**	**103.07 °**
		Nom. loss coeff.@ AR 3, Re 10^5 :	**0.1037**
Aspect Ratio:	3.250		
		Nom. loss coeff. @ Re 10^5 :	**0.1031**
Absolute rotor exit temperature:	1,166.6 K		
Absolute (axial) rotor exit velocity:	288.59 m/sec		
Specific heat Cp 416:	1,164.0 J/kg K	**Static rotor exit temp.:**	**1,130.8 K**
Absolute rotor exit pressure:	93.69 kPa		
Heat Ratio (Gamma) @ 416:	1.3273	**Static rotor exit pressure:**	**82.57 kPa**
Gas constant - dry air:	287.05 kJ/kg K	**Rotor exit air density:**	**0.2544 kg/m^3**
Static rotor exit temp.:	857.6 °C	**Rotor exit gas viscosity:**	**4.2E-05 kg/m.sec**
		Rotor exit kinematic visc.:	**1.7E-04 m^2/sec**
Absolute (axial) rotor exit velocity:	288.59 m/sec		
Mean blade diameter:	0.5788 m	**Reynold's coefficient:**	**1.0E+06**
		Nominal loss coefficient-R:	**0.0579**

Total turbine efficiencies:

Nominal loss coefficient-Stator:	0.0283		
Nominal loss coefficient-Rotor:	0.0579		
Specific Engy requirement:	275.49 kJ/kg		
Absolute rotor inlet velocity:	782.43 m/sec		
Rotor exit rel. whirl velocity:	476.21 m/sec		
Static stator exit temp.:	1,135.9 K		
Static rotor exit temperature:	1,130.8 K		
Absolute axial rotor velocity: Avg:	288.59 m/sec	**Total - static turb. efficiency:**	**0.829**
		Total - total turb. efficiency:	**0.948**

Turbine-stage detail parameters:

Absolute stator inlet velocity: $C_1 = C_a$	288.59	m/sec				
Combustor exit temp. T4(1):	1,400.0	K				
Specific Heat Cp @ 4(1):	1.1962	kJ/kg K	**Static combustor exit temp.:**	**1,365.2**	**K**	
						Ideal <
Gas constant - dry air:	287.05	kJ/kg K				
Heat Ratio (Gamma) @ 41:	1.31572		**Mean inlet NGV mach no.:**	**0.402**		*0.195*

STATOR INLET ANNULUS:

Static stator inlet temperature:	**1,365.2**	**K**	**Static stator inlet temp.:**	**1,365.2**	**K**	
Abs. stator inlet temperature T4:	1,400.0	K				
Stator inlet pressure P4:	235.80	kPa				
Heat Ratio (Gamma) @ 41:	1.31572		**Static stator inlet pressure:**	**212.31**	**kPa**	
Gas constant - dry air:	287.05	kJ/kg K				
Static stator inlet pressure:	212.31	kPa	**Stator inlet air density:**	**0.5418**	**kg/m^3**	
Absolute stator inlet velocity:	288.59	m/sec				
Stator inlet mass flow rate:	1.330	kg/sec	**Stator annulus inlet area:**	**0.0085**	**m^2**	
Stator inlet (= rotor inlet) root radius:	0.2878	m	**Stator inlet tip radius:**	**0.2924**	**m**	
			Stator inlet blade height:	**0.0047**	**m**	
			Stator inlet hub-tip ratio:	**0.984**		(> 0.5,< 0.85)

STATOR EXIT ANNULUS:

Static stator exit temperature:	1,135.9	K				
Gas constant - dry air:	287.05	kJ/kg K				
Static stator exit pressure:	98.72	kPa	**Stator exit air density:**	**0.3028**	**kg/m^3**	
Absolute stator exit velocity:	782.43	m/sec **(> 288.59)**				
Stator exit mass flow rate:	1.330	kg/sec	**Stator annulus exit area:**	**0.0056**	**m^2**	
Stator exit (= rotor inlet) root radius:	0.2878	m	**Stator exit tip radius:**	**0.2909**	**m**	
			Stator exit blade height:	**0.0031**	**m**	
			Stator exit hub-tip ratio:	**0.989**		(> 0.5,< 0.85)

ROTOR INLET ANNULUS:

Absolute rotor inlet temperature:	1,335.4	K				
Absolute rotor inlet pressure:	229.61	kPa				
Static rotor inlet pressure:	98.72	kPa				
Heat Ratio (Gamma) @ 415:	1.3311		**Static rotor inlet temp.:**	**1,082.5**	**K**	
			Compare to:	**1,070.2**	**K**	
Static rotor inlet temperature:	1,082.5	K				
Gas constant - dry air:	287.05	kJ/kg K				
Static rotor inlet pressure:	98.72	kPa	**Rotor inlet air density:**	**0.3177**	**kg/m^3**	
Absolute rotor inlet velocity:	782.43	m/sec				
Rotor inlet mass flow rate:	1.460	kg/sec	**Rotor inlet annulus area:**	**0.0059**	**m^2**	
Rotor inlet annulus area:	0.0059	m^2				
Mean rotor blade speed:	378.80	m/sec				
Shaft speed:	12,500	rpm	**Rotor inlet blade height:**	**0.0032**	**m**	
Mean blade diameter:	0.5788	m	**Rotor inlet root radius:**	**0.2878**	**m**	
			Rotor inlet tip radius:	**0.2910**	**m**	
			Rotor inlet hub-tip ratio:	**0.989**		(> 0.5,< 0.85)
Shaft speed:	12,500	rpm				
Rotor inlet tip diameter:	0.5820	m	**Rotor inlet tip velocity:**	**380.91**	**m/sec**	

ROTOR EXIT ANNULUS:

Rotor exit temperature T416:	1,166.6	K				
Static rotor exit temperature:	1,130.8	K				
Rotor exit pressure P416:	93.69	kPa				
Heat Ratio (Gamma) @ 416:	1.32732		**Static rotor exit pressure:**	**82.57**	**kPa**	
Gas constant - dry air:	287.05	kJ/kg K	**Rotor exit air density:**	**0.2544**	**kg/m^3**	
Absolute rotor exit velocity:	288.59	m/sec				
Rotor exit mass flow rate:	1.460	kg/sec	**Rotor annulus exit area:**	**0.0199**	**m^2**	
Rotor exit (= rotor inlet) root radius:	0.2878	m	**Rotor exit tip radius:**	**0.2986**	**m**	
			Rotor exit blade height:	**0.0108**	**m**	
			Rotor exit hub-tip ratio:	**0.964**		(> 0.5,< 0.85)

Turbine exit conditions:

Exit pressure P416:	93.69 kPa		
Absolute stator inlet pressure P4:	235.80 kPa	**Turbine Expansion Ratio:**	**2.517**
Heat Ratio (Gamma) @ 4 - 416 average:	1.3215	Gamma/(Gamma -1) @ 4-416:	4.110
		Isentropic t-s efficiency:	**0.829**
Stator inlet temperature T4(1):	1,400.0 K	**Exit temperature T416:**	**1,166.6 K**
Rotor exit temperature T416:	1,166.6 K		
(Gamma -1)/Gamma @ 4 - 416:	0.2433		
Turbine Expansion Ratio:	2.5167	**Actual, polytropic efficiency:**	**0.674** **67.40%**

Check:

Blade Load Coefficient:	1.9199		
Static stator inlet temp.:	1,070.2 K		
Mean rotor blade speed:	378.8 m/sec		
Cp 4 - 416 average:	1,180.1 J/kg K	**Static rotor exit temperature:**	**836.7 K**
Turbine Expansion Ratio:	2.5167		
Turbine polytropic efficiency:	67.404 %	**Isentropic efficiency:**	**0.698** **69.84%**

Turbine power output:

C_p 4 - 416 average:	1.1801 kJ/kg K	
Mass flow through turbine:	1.460 kg/sec	
Inlet temperature T4(1):	1,400.0 K	
Exit temperature T416:	1,166.6 K	**402.21 kW**

Approximation-check:

Mass flow through turbine:	1.460 kg/sec		
Static stator inlet temperature:	1,365.2 K		
Static rotor exit temperature:	1,130.8 K	**'Static' Tb output power:**	**403.9 kW**
Inlet temperature T4(1):	1,400.0 K		
Specific Heat Cp @ T4(1):	1.1962 kJ/kg K	**Absolute stator inlet vel.:**	**288.59 m/sec**
Absolute stator inlet velocity:	288.59 m/sec	**Temperature equivalent:**	**34.8 K**
		Absolute stator inlet temp.:	**1,400.0 K**
Absl. axl rtor exit velocity:	288.59 m/sec		
Specific heat Cp 416:	1.1640 kJ/kg K	**Temperature equivalent:**	**35.8 K**
		Absolute rotor exit temp.:	**1,166.6 K**
		Change in temperature equ.:	**-1.0 K**
		Turbine output power:	**402.2 kW**

Cooling air downstream of turbine - Mixing:

Mass flow rate W416 = W415:	1.460 kg/sec		
Cooling air % addition at 5:	0.0 %	**Cooling air flow addition @ 5:**	**0.000 kg/sec**
		Mass flow W5:	**1.460 kg/sec**
Exit temperature T416:	1,166.6 K		
Specific Heat Cp @ T416:	1.1640		
Temperature T307, or 8:	471.5 K		
Specific Heat Cp @ T307, or 8:	1.0240 kJ/kg K		
Specific Heat Cp @ T5:	1.1608 kJ/kg K	**Specific Heat Cp @ T5:**	**1.1608 kJ/kg K**
		Approx. check Cp @ 5:	**1.1608 kJ/kg K**
		Gas temperature T5:	**1,169.8 K**
Recuperator inlet duct:			
Gas temperature T6 = T5:	1,169.8 K		
Gas temperature T6ini. guessed:	500.0 K		
2nd … guess(es):	**1,169.7 K**	**T6 error:**	**0.0 K**
		% error:	**-0.003**
Turbine exit pressure P5 = P416:	93.69 kPa		
Recuperator inlet duct pressure loss:	0.040	**Inlet pressure P6:**	**89.94 kPa**
		Mass flow W6 = W5:	**1.460 kg/sec**
Recuperator gas side:			
Mass flow W307:	1.330 kg/sec		
Specific Heat Cp @ T3(07):	1.0234 kJ/kg K		
Compr. exit temp. T307 = T3:	471.5 K		
Recup. exit temp. T308:	1,085.9 K		
Mass flow W6 = W5:	1.45999 kg/sec	**Heat-transfer relationship:**	
Gas temperature T6 = T5:	1169.7 K		
Specific Heat Cp @ T5:	1.1608 kJ/kg K	**Temperature T601:**	**1,169.7 K**
Inlet pressure P6:	89.94 kPa		
Recuperator gas side pressure loss:	0.040	**Pressure P601:**	**86.35 kPa**
		Mass flow W601 = W6:	**1.460 kg/sec**
Exhaust:			
Specific Heat Cp @ 601:	1.1701 kJ/kg K		
Heat Ratio (Gamma) @ 601:	1.3251	Gamma/(Gamma -1) @ 415-6:	4.076
Exhaust design mach number:	0.05	**P9/Ps9:**	**1.0017**
Static pressure Ps9 = Pamb:	85.48 kPa	**P9:**	**85.62 kPa**
		Q:	**3.392 kg√K/sm²kPa**
Mass flow W9 = W5:	1.45999 kg/sec		
Gas temperature T9 = T601:	1,169.7 K	**Exhaust plane area A9:**	**0.172 m^2**
		Exhaust plane dia. D9:	**0.468 m**

Mechanical losses:

General input parameters:

Type of bearing:	**Ball bearing**			
Bearing race diameter PCD:	(< 575.5)	**65.0 mm**	0.065	
Rotational speed:	12,500 rpm		**Bearing DN number:**	**812500**

Bearing friction losses: [47]

Type of lubricating oil:	**Medium mineral**		
Oil temperature:	**100 °C**	**A:**	**-0.1240**
	373.15 K	**Kinematic viscosity:**	**9.6688**
		Oil density:	**789.6 kg/m^3**
		Dynamic viscosity:	**0.0076 kg/m s**
Power losses:			
Oil flow rate:	**0.00 l/hr**		
Dynamic viscosity:	0.0076 kg/m sec	**Ball bearing**	**0.4 kW**
		Roller bearing	**0.2 kW**
		Hydrodynamic radial brng	**1.9 kW**
		Hydrodynamic thrust brng	**4.4 kW**

Disc windage losses:

Compressor:

Static exit pressure:	254.61 kPa		
Static exit temperature:	468.02 K	**Compressor exit air density:**	**1.895 kg/m^3**
Impeller tip diameter:	0.6280 m		
Impeller rim speed:	411.01 m/sec	**Disc windage power:**	**0.222 W**

Turbine:

Stator inlet air density:	0.5418		
Blade tip diameter:	0.5820 m		
Turbine rim speed:	380.91 m/sec	**Disc windage power:**	**0.0434 W**
		O/all disc windage power:	**0.2655 W**

Shaft mechanical efficiency & output power available:

Bearing friction losses:	0.36 kW		
Disc windage power loss:	0.0003 kW	**Overall mechan. losses:**	**0.36 kW**
Turbine power:	402.2 kW	**Overall mech. efficiency:**	**99.910 %**
Mechanical compressor efficiency:	**97 %**	**Turbine mech. efficiency:**	**99.000 %**
		Useful turbine power:	**398.2 kW**
Compressor input power:	247.7 kW	**Available shaft power:**	**150.4 kW**
Intake air mass flow:	1.464 kg/sec	**Spec. power/thrust:**	**102.74 Ns/kg**
Fuel flow:	0.013 kg/sec	**Spec. fuel consumption:**	**0.315 kg/kWh**
LHV:	43,124 kJ/kg	**S. P. thermal efficiency:**	**26.482 %**

Minimum allowable diagram-efficiency iteration procedure:

Note: U max to be larger than 265.67 m/sec

Mean inlet mach no. to NGV: (< 0.2)	0.402		
Heat Ratio (Gamma) @ 41:	1.31572	**T/Ts ratio:**	**1.025**
Absolute stator inlet/exit temperature:	1,400.0 K	**NGV static inlet temp.:**	**1,365.2 K**
Gas constant - dry air:	287.05 kJ/kg K		
Heat Ratio (Gamma) @ 415:	1.3311	**Local speed of sound:**	**722.25 m/sec**
Mean inlet mach no. to NGV:	0.402	C_a **Stator inlet axial gas vel.:**	**290.28 m/sec**
		Check:	**288.59 m/sec**
(Iteration procedure:	**68.94 %)**	**Initial min. diag. efficiency:**	**68.95 %**
Compressor input shaft power:	247.7 kW		
Rotor mass flow rate:	1.460 kg/sec	**Specific Engy requirement:**	**169.68 kJ/kg**
		Relative inlet gas velocity:	**701.6 m/sec**
Stator inlet axial gas vel.:	290.28 m/sec	**Rtor inlet abs whirl cmp'nt:**	**638.7 m/sec**

As the flow out of the rotor stage is purely axial, whirl component out is zero:

Compressor input shaft power:	247.7 kW		
Rtor inlet abs whirl cmp'nt:	638.7 m/sec		
Rotor mass flow rate:	1.460 kg/sec	**Mean rotor blade speed:**	**265.67 m/sec**

Limiting value of Diagram efficiency for a purely 100% impulse device (ie. o/let whirl = mean tip speed):

Specific energy requirement:	169.68 kJ/kg		
Stator mass flow rate:	1.343 kg/sec		
Mean rotor tip speed:	265.67 m/sec		
Stator inlet axial gas vel.:	290.28 m/sec	**Min. Diag. (Hydr.) efficiency:**	**68.94 %**
		Av'g. min. diag. efficiency:	**68.94 %**

Table D5 ENERGY CALCULATIONS

Compression Process:

Input Parameters:

Name of intake gas:	Dry air				
Static air temperature T2s:	295.7	0.2957	Spec. enthalpy H2 isentr.:	0.7183	MJ/kg
			FT2:	5.6738	
Specific heat Cp 2:	1.0037	kJ/kg K			
Heat ratio (Gamma) 2:	1.4006				
Gas constant R:	287.05	J/kg K			
Polytropic Efficiency of Compressor-stage:				1.38984	
Pressure Ratio (P3/P2):	3.1		Isentropic integral Cp/TdT:	0.3248	kJ/kg K
Static temperatureT3s:	468.0	0.4680	Specific heat Cp 3:	1.0234	
			Heat ratio (Gamma) 3:	1.3898	
			Total average temperature:	381.84	K
			Cp 2-3:	1,010.7	J/kg K
			Gamma 2-3:	1.3967	
			(Gamma - 1) / Gamma @ 2-3:	0.2840	
Isentropic efficiency:	0.7215		**Polytropic efficiency:**	**76.12%**	

TURBINE

Combustion Process:

Specific Heat (Cp) and Enthalpy for Products of Combustion of Kerosene or Diesel in Dry Air:

Fuel:	Kerosene				
F/A Ratio:	0.0099		R:	287.05	J/kg K
Static inlet temp. T31s:	1,007.9	1.0079	Specific enthalpy, H31:	1.4889	MJ/kg
				1,488.9	kJ/kg
			FT31:	6.9632	kJ/kg K
Specific Heat Cp @ T31:	1.16141	kJ/kg K			
Heat Ratio (Gamma):	1.32830				
Static comb. exit temp. T4(1)s:	1,365.2	1.3652	Specific enthalpy, H4(1):	1.9149	MJ/kg
				1,914.9	kJ/kg
			FT4(1):	7.3182	kJ/kg K
Specific Heat Cp @ T4(1):	1.19625	kJ/kg K			
Heat Ratio (Gamma):	1.31572		**Isentr. Integral Cp/TdT:**	**0.3550**	**kJ/kg K**
			Required dH:	**425.9**	**kJ/kg**
			Average Heat Ratio:	**1.3220**	
Combustion Process: Air Fuel Ratio:					
Combustion efficiency:	99.69	%			
LHV:	43,124	kJ/kg	**FAR:**	**0.0099**	
			A/F:	**100.93**	

Combustion Process: Air Fuel Ratio:

Type of fuel:	Kerosene		LHV:	43,124	kJ/kg
Combustion efficiency:	99.69	%			
Inlet temperature T31s:	1,007.9	K	FAR1:	0.0950	
Exit temperature T4(1)s:	1,365.2	K	FAR2:	-0.0019	
			FAR3:	0.0000	
			FAR:	**0.0104**	
			A/F:	**96.367**	

Pseudo-station 415 mixing:

Temperature T307, or 8:	471.5	0.4715	Specific Heat Cp @ T307, or 8:	1.0240	kJ/kg K

Turbine Expansion Process:

Specific Heat (Cp) and Enthalpy for single turbine stage inlet and exit conditions:

Static inlet temp. T41s:	1,365.2	1.3652	Specific enthalpy, H41:	1.8960	MJ/kg
				1,896.0	kJ/kg
			FT41:	7.3182	kJ/kg K
Specific Heat Cp @ T41:	1.19625	kJ/kg K			
Heat Ratio (Gamma):	1.31572				
Static inlet temp. T415s:	1,070.2	1.0702	Specific enthalpy, H415:	1.5489	MJ/kg
				1,548.9	kJ/kg
			FT415:	7.0320	kJ/kg K
Specific Heat Cp @ T415:	1.15392	kJ/kg K			
Heat Ratio (Gamma):	1.33114				
Static exit temp. T416s:	1,130.8	1.1308	Specific enthalpy, H416:	1.6192	MJ/kg
				1,619.2	kJ/kg
			FT416:	7.0958	kJ/kg K
Specific Heat Cp @ T416:	1.16401	kJ/kg K			
Heat Ratio (Gamma):	1.32732		**Isentr. Integral Cp/TdT:**	**0.0639**	**kJ/kg K**
			Required dH:	**70.3**	**kJ/kg**
			Cp 415-6:	**1.1590**	**kJ/kg K**
			Gamma 415-6:	**1.3292**	

Cooling air downstream of turbine - mixing:

Mixture temperature T5:	1,169.8	1.1111	Specific Heat Cp @ T5:	1.1608	kJ/kg K
Mixture temperature T601:	1,169.7	1.1697	Specific Heat Cp @ T601:	1.1701	kJ/kg K
			Heat Ratio (Gamma) @ T601:	1.3251	

SELECTED DESIGN FORMULAE

Basic Design Point Calculations:

THRUST REQUIREMENTS:

$$\text{LiftCoefficient} = \frac{\text{AircraftWeight} \cdot 9.807}{0.5 \cdot 1.2248 \cdot \text{MaxAirspeed}^2 \cdot \text{WingArea}}$$

$$\text{DragCoefficient} = \frac{\text{LiftCoefficient}}{\text{LiftDragRatio}}$$

AircraftWeight = 'Acrftperf chrts'! (1170 kg)

WingArea = 'Acrftperf chrts'! (15 m^2)

$$\text{DragForce} = 0.5 \cdot 1.2248 \cdot \text{MaxAirspeed}^2 \cdot \text{DragCoefficient} \cdot \text{WingArea}$$

LiftDragRatio = 'Acrft perf chrts'!BZ6 · 100

$$\text{ApproxShaftPwrStdy} = \frac{\text{DragForce}}{15}$$

$$\text{ApproxShaftPowerClimb} = \frac{\text{DragForceClimb}}{15}$$

$$\text{AmbTempDegC} = \text{AmbTempK} - 273.15$$

Ambient, Optimum Design Point Conditions:

At ISA:

$$\text{AirDens} = \left[\frac{\text{AmbPress} \cdot 1000}{287.05 \cdot \text{AmbTempK}}\right]$$

$$\text{T\_01} = \text{AmbTempK} \cdot \left[1 + \frac{\text{HeatRatio} - 1}{2} \cdot \text{E60}^2\right]$$

$$\text{P\_01} = \text{AmbPress} \cdot \left[\frac{\text{TempT01}}{\text{AmbTempK}}\right]^{\left[\frac{\text{HeatRatio}}{\text{HeatRatio} - 1}\right]}$$

Where:

E60 = Operational flight speed

At Operational Design Condition (ODC):

$$PressAlt = E51 - (E54 - 101.325) \cdot 91.44$$

$$ISATemp = (PressAlt \cdot -0.0065) + 15$$

$$TempDeviat = E55 - ISATemp$$

$$PAtoDADiff = TempDeviat \cdot 36.58$$

Where:

E51 = Elevation above SL
E54 = Ambient Pressure Pamb
E55 = Ambient Temperature Tamb
K69 = Pressure Altitude
K70 = ISA temperature
K72 = PA-DA difference

Fuel Consumption Comparisons:

Approximate, Average Fuel Consumption:

$$FC\_kg\_per\_sec = \frac{E32 \cdot SFCs!\$J\$23}{3600000}$$

$$FC\_L\_per\_hr = 0.000011 \cdot J43^3 - 0.007887 \cdot J43^2 + 2.040465 \cdot J43 - 124.350061$$

Where:

E32 = Lycoming O-320-D2A Fuel Consumption (L/hr)
J23 = Fuel density
J43 = Extrapolated F.C.

SINGLE TURBINE ARRANGEMENT

Recuperator:

$$T\_3 = Exit\_Temperature\_T307$$

$$P\_307 = CompressorExit\_DiffiserPressure\_P3 \cdot \left[1 - \frac{E440}{100}\right]$$

$$Air\_FlowRate\_W\_307 = ?\ IF\left[E439 = "Inlet", J423 - \left[\frac{D83 \cdot E441}{100}\right], IF\,(E439 = "Exit", J423, J423)\right]$$

$$W\_308 = Air\_FlowRate\_W\_307$$

$$T\_308 = IF\left[E436 = "n", J439, ?\ IF\left[E910 = 0, \left[D445 + \left[\frac{E444}{100} \cdot (E446 - D445)\right]\right], \left[D445 + \left[\frac{E444}{100} \cdot (E910 - D445)\right]\right]\right]\right]$$

$$P\_308 = P\_307 \cdot \left[1 - \frac{E447}{100}\right]$$

$$P\_31 = P\_308 \cdot \left[1 - \frac{E451}{100}\right]$$

$$Air\_FlowRate\_W31 = ?\ IF\left[E439 = "Exit", J441 - \left[\frac{D83 \cdot E441}{100}\right], IF\,(E439 = "Inlet", J441, J442)\right]$$

$$T\_ExitStatic = D453 - \frac{D454^2}{2000 \cdot E455}$$

Where:

D83 = Mass flow intake air+vapour
D445 = Air side inlet temperature T307
D450 = T31 = T308
D453 = Exit temperature T31
D454 = Impeller exit air velocity
E436 = Is a Recuperator employed? (Y/N)
E439 = Cooling flow offtake position (Inlet/Exit)
E440 = Recuperator air side inlet duct pressure loss
E441 = Blade and disc cooling flow offtake
E444 = Recuperator effectiveness
E446 = Gas side inlet temperature 1st guess T6ini.
E447 = Air side total pressure loss
E451 = Recuperator exit duct pressure loss
E455 = Specific Heat Cp @ T31
E910 = 2nd … guess(es): Recuperator inlet duct T
J423 = Flow rate after offtake & leaks

Turbine efficiency-loss coefficients:

STATOR:

$$Stator\_DeflectionAngle = D687 + D688$$

$$Nom\_LossCoeff\_at\_AR3 = 0.04 + 0.06 \cdot \left[\frac{Stator\_DeflectionAngle}{100}\right]^2$$

$$Nom\_LossCoeff\_10pwr5 = (1 + Nom\_LossCoeff\_at\_AR3) \cdot \left[0.993 + 0.021 \cdot E691^{-1}\right] - 1$$

$$Static\_StatorExit\_T = D693 - \frac{D694^2}{2 \cdot E695}$$

$$Static\_IsentrExit\_T = Static\_StatorExit\_T - \frac{E698 \cdot D694^2}{2 \cdot E695}$$

$$Static\_StatorExit\_P = D700 \cdot \left[\frac{Static\_IsentrExit\_T}{D701}\right]^{\left[\frac{E702}{E702 - 1}\right]}$$

$$Absol\_StatorExit\_P = Static\_StatorExit\_P \cdot \left[\frac{D701}{Static\_StatorExit\_T}\right]^{\left[\frac{E702}{E702 - 1}\right]}$$

$$Absol\_StatorBlade\_P\_Loss = 100 - \frac{100 \cdot Absol\_StatorExit\_P}{D700}$$

$$StatorExit\_AirDensity = \frac{Static\_StatorExit\_P \cdot 1000}{Static\_StatorExit\_T \cdot D706}$$

$$StatorExit\_GasViscos = \left[-0.0021 \cdot D707^2 + 4.7172 \cdot D707 + 1710.6\right] \cdot 10^{-8}$$

$$StatorExit\_KinematViscos = \frac{StatorExit\_GasViscos}{StatorExit\_AirDensity}$$

$$Re\_coefficient = \frac{D709 \cdot D710}{StatorExit\_KinematViscos}$$

$$Nominal\_Loss\_coeff\_S = Nom\_LossCoeff\_10pwr5 \cdot \left[\frac{10^5}{Re\_coefficient}\right]^{0.25}$$

Where:

D687 = Absolute stator inlet angle
D688 = Absolute stator exit angle
D693 = Abs. stator exit temperature T4(1)
D694 = Absolute stator exit velocity
D700 = Absolute stator inlet pressure
D701 = Absolute stator inlet temperature

D706 = Gas constant - dry air
D707 = Static stator exit temp.
D709 = Absolute stator exit velocity
D710 = Mean blade diameter
E691 = Aspect Ratio
E695 = Specific heat Cp 415-6
E698 = Nominal loss coefficient-S
E702 = Heat Ratio (Gamma) @ 415-6

ROTOR:

$$\text{Exit\_P416} = \frac{D715}{\left[1 - \frac{E716}{100}\right] \cdot \left[1 - \frac{E717}{100}\right] \cdot \left[1 - \frac{E718}{100}\right]}$$

$$\text{Rotor\_DeflectAngle} = D720 + D721$$

$$\text{Nom\_LossCoeff\_AR3} = 0.04 + 0.06 \cdot \left[\frac{\text{Rotor\_DeflectAngle}}{100}\right]^2$$

$$\text{Nom\_LossCoeff\_10pwr5\_} = (1 + \text{Nom\_LossCoeff\_AR3}) \cdot \left[0.993 + 0.021 \cdot D724^{-1}\right] - 1$$

$$\text{Static\_RotorExit\_T} = D726 - \frac{D727^2}{2 \cdot E728}$$

$$\text{Absol\_RotorExit\_P} = D729 \cdot \left[\frac{\text{Static\_RotorExit\_T}}{D726}\right]^{\left[\frac{E730}{E730 - 1}\right]}$$

$$\text{RotorExit\_AirDensity} = \frac{\text{Absol\_RotorExit\_P} \cdot 1000}{\text{Static\_RotorExit\_T} \cdot D732}$$

$$\text{RotorExit\_GasViscos} = \left[-0.0021 \cdot D733^2 + 4.7172 \cdot D733 + 1710.6\right] \cdot 10^{-8}$$

$$\text{RotorExit\_KinematViscos} = \frac{\text{RotorExit\_GasViscos}}{\text{RotorExit\_AirDensity}}$$

$$\text{Re\_coeff} = \frac{D735 \cdot D736}{\text{RotorExit\_KinematViscos}}$$

$$\text{Nominal\_Loss\_coeff\_R} = \text{Nom\_LossCoeff\_10pwr5\_} \cdot \left[\frac{10^5}{\text{Re\_coeff}}\right]^{0.25}$$

Where:

D715 = Total ambient pressure (P09 = P01)
D720 = Stator exit blade angle
D721 = Rtor relat. exit blade angle
D724 = Aspect Ratio
D726 = Absolute rotor exit temperature
D727 = Absolute (axial) rotor exit velocity
D729 = Absolute rotor exit pressure

D732 = Gas constant - dry air
D733 = Static rotor exit temp.
D735 = Absolute (axial) rotor exit velocity
D736 = Mean blade diameter
E716 = Recuperator gas side inlet duct pressure loss
E717 = Recuperator gas side pressure loss
E718 = Jet pipe/Exhaust pressure loss
E728 = Specific heat Cp 416
E730 = Heat Ratio (Gamma) @ 416

Total turbine efficiencies:

$$\text{Ttl\_Static\_TrbnEff} = \left[1 + \frac{D744 \cdot D747^2 + D743 \cdot D746^2 \cdot \left[\frac{D749}{D748}\right] + D750^2}{2 \cdot D745 \cdot 1000}\right]^{-1}$$

$$\text{Ttl\_Ttl\_TrbnEff} = \left[1 + \frac{D744 \cdot D747^2 + D743 \cdot D746^2 \cdot \left[\frac{D749}{D748}\right]}{2 \cdot D745 \cdot 1000}\right]^{-1}$$

Where:

D743 = Nominal loss coefficient-Stator
D744 = Nominal loss coefficient-Rotor
D745 = Specific Engy requirement
D746 = Absolute rotor inlet velocity
D747 = Rotor exit rel. whirl velocity
D748 = Static stator exit temp.
D749 = Static rotor exit temperature
D750 = Absolute axial rotor velocity:

Turbine-stage detail parameters:

$$\text{Static\_CombusorExit\_T} = D759 - \frac{D758^2}{2000 \cdot E760}$$

$$\text{MeanInlet\_NGV\_MachNo} = \frac{D758}{\sqrt{E763 \cdot D762 \cdot \text{Static\_CombusorExit\_T}}}$$

Where:

D758 = Absolute stator inlet velocity: $C_1 = C_a$
D759 = Combustor exit temp. T4(1)
D762 = Gas constant - dry air
E760 = Specific Heat Cp @ 4(1)
E763 = Heat Ratio (Gamma) @ 41

Mechanical losses:

General input parameters:

$$BearingDN\_no = D947 \cdot E946$$

Bearing friction losses:

$$A = 76.14233 - 86.75707 \cdot LOG(D953\ ,10) + LOG(D953\ ,10) \cdot (34.35917 \cdot LOG(D953\ ,10) + LOG(D953\ ,10) \cdot (-4.726616 \cdot LOG(D953\ ,10)))$$

$$Kinematic\_visc = 6.82 \cdot \left[10^{\left[10^{A} - 0.6 \right]} \right]$$

$$Dynamic\_visc = \frac{Kinematic\_visc \cdot OilDensity}{1000000}$$

$$BallBearingLoss = 0.00845 \cdot G946^{3.95} \cdot D947^{1.75} \cdot D960^{0.4} + 0.001358 \cdot D947 \cdot G946 \cdot E959$$

$$RollerBearingLoss = 0.0036 \cdot G946^{3.95} \cdot D947^{1.75} \cdot D960^{0.4} + 0.0006613 \cdot D947 \cdot G946 \cdot E959$$

$$Hydrodyn\_RadlBrngLoss = 0.0432 \cdot G946^{3.95} \cdot D947^{1.75} \cdot D960^{0.4} + 0.00793 \cdot D947 \cdot G946 \cdot E959$$

$$Hydrodyn\_ThrustBrngLoss = 0.1014 \cdot G946^{3.95} \cdot D947^{1.75} \cdot D960^{0.4} + 0.0163 \cdot D947 \cdot G946 \cdot E959$$

Where:

E946 = Bearing race diameter PCD
D947 = Rotational speed
E951 = Type of lubricating oil: **Medium mineral**
D953 = Oil temperature

Power losses:

E959 = Oil flow rate
D960 = Dynamic viscosity

Disc windage losses:

Compressor:

$$ComprExit\_AirDensity = \left[\frac{D972 \cdot 1000}{287.05 \cdot D973} \right]$$

$$ComprDiskWindage\_PwrLoss = 0.00000000428 \cdot ComprExit\_AirDensity \cdot D974^{2} \cdot D975^{3}$$

Where:

D972 = Static exit pressure
D973 = Static exit temperature
D974 = Impeller tip diameter
D975 = Impeller rim speed

Turbine:

$$TbneDiskWndg\_PwrLoss = 0.00000000428 \cdot D979 \cdot D980^{2} \cdot D981^{3}$$

$$DiskWndge\_Pwr = ComprDiskWindage\_PwrLoss + TbneDiskWndg\_PwrLoss$$

Where:

D979 = Stator inlet air density
D980 = Blade tip diameter
D981 = Turbine rim speed

Shaft mechanical efficiency & output power available:

$$Overall\_MechLosses = D987 + D989$$

$$Overall\_MechEff = 100 \cdot \left[\frac{D991 - Overall\_MechLosses}{D991} \right]$$

$$Turbine\_MechEff = ???\ IF\left[\frac{100 \cdot Overall\_MechEff}{E993} < 99 , \frac{100 \cdot Overall\_MechEff}{E993} , \text{ç}\right.$$

$$Useful\_TbnePwr = \frac{D991 \cdot Turbine\_MechEff}{100}$$

$$Avlble\_ShftPwr = Useful\_TbnePwr - D997$$

$$Specific\_Pwr\_to\_Thrust\_Ratio = \frac{Avlble\_ShftPwr}{D999}$$

$$Spec\_FuelConsmp = \frac{3600 \cdot D1001}{Avlble\_ShftPwr}$$

$$SpecPwr\_ThermalEff = \frac{100 \cdot Avlble\_ShftPwr}{D1001 \cdot D1003}$$

Where:

D987 = Bearing friction losses
D989 = Disc windage power loss
D991 = Turbine power
D997 = Compressor input power

D999 = Intake air mass flow
D1001 = Fuel flow
D1003 = LHV
E993 = Mechanical compressor efficiency

Minimum allowable diagram-efficiency iteration procedure:

$$NGV\_StaticInlet\_T = \frac{D1012}{J1010}$$

$$LocalSpdOfSnd = \sqrt{NGV\_StaticInlet\_T \cdot D1014 \cdot E1015}$$

$$StatorInlet\_Axl\_GasVel = D1017 \cdot LocalSpdOfSnd$$

$$Checking = Absol\_Axl\_RotorInlet\_vel$$

$$Initial\_MinDiagrEff = D1021 + 0.005$$

$$SpecEngyRequmnt = \frac{D1022}{D1023}$$

$$RelatInlet\_GasVel = \sqrt{\frac{2 \cdot SpecEngyRequmnt \cdot 1000}{\frac{Initial\_MinDiagrEff}{100}}}$$

$$RotorInlet\_AbsWhrlCmpnt = \sqrt{RelatInlet\_GasVel^2 - D1027^2}$$

Where:

D1009 = Mean inlet mach no. to NGV: (< 0.2)
D1012 = Absolute stator inlet/exit temperature
D1014 = Gas constant - dry air
D1017 = Mean inlet mach no. to NGV
D1021 = **Iteration procedure**
D1022 = Compressor input shaft power
D1023 = Rotor mass flow rate
D1027 = Stator inlet axial gas vel.
E1010 = Heat Ratio (Gamma) @ 41
E1015 = Heat Ratio (Gamma) @ 415

As the flow out of the rotor stage is purely axial, whirl component out is zero:

$$Mean\_RotorBladeSpeed = \frac{D1031 \cdot 1000}{D1032 \cdot D1033}$$

Where:

D1031 = Compressor input shaft power
D1032 = Rtor inlet abs whirl cmp'nt
D1033 = Rotor mass flow rate

Limiting value of Diagram efficiency for a purely 100% impulse device (ie. o/let whirl = mean tip speed):

$$MinDiagrEff = ROUND\left[\frac{100 \cdot 2 \cdot D1037 \cdot 1000}{D1038 \cdot \left[4 \cdot D1039^2 + D1040^2\right]}, 4\right]$$

$$AvgMinDiagEff = AVERAGE(MinDiagrEff\ ,\ Initial\_MinDiagrEff)$$

Where:

D1037 = Specific energy requirement
D1038 = Stator mass flow rate
D1039 = Mean rotor tip speed
D1040 = Stator inlet axial gas vel.

COMPRESSOR / FREE POWER TURBINE ARRANGEMENT

Recuperator:

Recuperator inlet duct:

$$Temp\_T307 = Exit\_Temperature\_T307$$

$$Press\_P307 = CompressorExit\_DiffiserPressure\_P3 \cdot \left[1 - \frac{AH440}{100}\right]$$

$$MassFlow\_W307 = ?\ IF\left[AH439 = \text{"Inlet"}, J423 - \left[\frac{D83 \cdot AH441}{100}\right], ?\ IF\ (AH439 = \text{"Exit"}\ , J423\ , J423)\right]$$

$$MassFlow\_W308 = MassFlow\_W307$$

Recuperator:

$$Exit\_T308 = ?\ IF\left[AH436 = \text{"n"}\ , AM439\ , ?\ IF\left[AH948 = 0, \left[AH445 + \left[\frac{AH444}{100} \cdot (AH446 - AH445)\right]\right], \left[AH445 + \left[\frac{AH444}{100} \cdot (AH948 - AH445)\right]\right]\right]\right]$$

$$Exit\_P308 = Press\_P307 \cdot \left[1 - \frac{AH447}{100}\right]$$

Recuperator exit duct:

$$Exit\_P31 = Exit\_P308 \cdot \left[1 - \frac{AH451}{100}\right]$$

$$FlowRate\_W31 = ?\ IF\left[AH439 = \text{"Exit"}, AM441 - \left[\frac{R83 \cdot AH441}{100}\right], ?\ IF\ (AH439 = \text{"Inlet"}\ , AM441\ , AM442)\right]$$

$$StaticExit\_T = AH453 - \frac{AH454^2}{2000 \cdot AH455}$$

Where:

D83 = Mass flow intake air+vapour
AH436 = Is a Recuperator employed? (Y/N)
AH439 = Cooling flow offtake position (Inlet/Exit)
AH440 = Recuperator air side inlet duct pressure loss
AH441 = Blade and disc cooling flow offtake

AH444 = Recuperator effectiveness
AH445 = Air side inlet temperature T307
AH446 = Gas side inlet temperature 1st guess T6ini.
AH447 = Air side total pressure loss
AH450 = T31 = T308
AH451 = Recuperator exit duct pressure loss
AH453 = Exit temperature T31
AH454 = Impeller exit air velocity
AH455 = Specific Heat Cp @ T31

Combustor:

General input parameters:

$$FuelFlow = AG465 \cdot AH468$$

$$ExitFlow\_W41 = FuelFlow + AG465$$

Where:

AG463 = LHV
AG465 = Air flow for combustion
AH468 = FAR

Combustor pressure cold losses:

$$Combust\_ColdLoss = AH475 \cdot \left[\frac{AH476 \cdot \sqrt{AH477}}{AH479}\right]^2$$

$$InletGas\_P31 = AH479 - Combust\_ColdLoss$$

Where:

AH475 = Cold loss factor: (Rig tests!)
AH476 = Mass flow of intake gas
AH477 = Inlet gas temperature T31 = T308
AH479 = Inlet gas pressure P31

Pseudo-station 415 mixing:

$$W415\_MassFlow = AG610 + \frac{AG609}{100} \cdot \$D\$83$$

$$Cp\_at\_T415 = \frac{100 - AG609}{100} \cdot AH614 + \frac{AG609}{100} \cdot AH613$$

$$TbneRotor\_Inlet\_T415 = \frac{AG610 \cdot AH614 \cdot AG611 + \frac{AG609}{100} \cdot \$D\$83 \cdot AG612 \cdot AH613}{W415\_MassFlow \cdot Cp\_at\_T415}$$

Where:

D83 = Mass flow intake air+vapour
AG609 = Cooling air addition, doing work
AG610 = Combustor exit flow W41
AG611 = Temperature T41 = T4
AG612 = Temperature T307, or 8
AH613 = Specific Heat Cp @ T307, or 8
AH614 = Specific Heat Cp @ T4

Free Power Turbine:

General input parameters:

$$FreeTbne\_Inlet\_T = \left[1 - \frac{BJ631}{100}\right] \cdot BI630$$

$$FreeTbne\_Inlet\_P = \left[1 - \frac{BJ634}{100}\right] \cdot BI633$$

$$FreeTbne\_Exit\_P = \frac{BI638}{1 - \frac{BJ639}{100}}$$

Where:

BI630 = Compressor turbine exit T416
BI633 = Compressor turbine exit P416
BI638 = Total ambient pressure (P9 = P1)
BJ631 = Inter turbine duct heat loss
BJ634 = Inter turbine duct pressure loss (0.5 - 2.5%)
BJ639 = Jet pipe/Exhaust pressure loss

Turbine-stage calculations:

$$ActlTbneRim\_vel = BO828$$

$$Shaft\_vel = \frac{2 \cdot \pi \cdot BI647}{60}$$

$$Rim\_vel\_CorrectFactor = ROUND\left[-2.23634 \cdot 10^{-16} \cdot \left[BI647^{3}\right] + 4.268047 \cdot 10^{-11} \cdot \left[BI647^{2}\right] + 4.794814 \cdot 10^{-7} \cdot BI647 + 3.069754 \cdot 10^{-3}, 3\right]$$

$$MeanRotorBldeSpeed = BJ652$$

$$MeanBldeDia = \frac{2 \cdot MeanRotorBldeSpeed}{Shaft\_vel}$$

$$SpecEnergyRequmnt = \frac{BJ656}{BI658}$$

Where:

BI647 = Required shaft speed
BI658 = Rotor inlet mass flow rate
BJ652 = Mean rotor blade speed
BJ656 = Turbine output power
BJ645 = Maximum turbine rim speed
BO828 = Rotor inlet tip velocity

Turbine exit conditions:

$$\text{Trbne\_ExpansionRatio} = \frac{BI851}{BI850}$$

$$\text{Exit\_T\_416} = BI858 - BP856 \cdot BI858 \cdot \left[1 - \text{Trbne\_ExpansionRatio}^{\left[\frac{-1}{BO853}\right]} \right]$$

$$\text{PolytropEff} = \left[\frac{LN\left[\frac{BI858}{BI860}\right]}{LN(BI862)} \right]^{BI861}$$

Where:

BI850 = Exit pressure P416
BI851 = Absolute stator inlet pressure P4
BI858 = Stator inlet temperature T4(1)
BI860 = Rotor exit temperature T416
BI861 = (Gamma -1)/Gamma @ 4 - 416
BI862 = Turbine Expansion Ratio
BJ853 = Heat Ratio (Gamma) @ 4 - 416 average

APPENDIX F

RECORDED DATA FOR ENGINE OPTIMISATION

I Single spool GT at a pressure ratio of 4.5:1

Our first approach was to utilise given engine limits such as the maximum attainable compressor pressure ratio of 4.5 and to optimise such data with respect to the SFC levels. For best performance levels, the fuel flow rate was varied via the axial impeller exit depth width. The axial impeller depth width indirectly varies the air flow rate through the engine. Thus:

a. No recuperator employed

Cycle condition:	**OPTIMISED** *
Compressor-Turbine configuration:	**SINGLE SPOOL**
Recuperator:	**OFF**

Compressor PR:	**4.5**

* Optimal at parameter (s.a. PR) settings.

Table I a1 No recuperator employed

Axl impell. exit depth:	[mm]	**12**	**14**	**16**	**18**	**20**	**23**	**26**	**29**	**32**	**35**	**40**	**45**
Shft Pwr:	[kW]				2,223	2,318	2,411	2,476	2,518	2,539	2,519	2,444	
GT speed:	[RPM]				6,245	6,952	7,984	9,024	10,060	11,095	12,103	13,799	
Air intake:	[kg/sec]				23.79	23.72	23.71	23.69	23.68	23.68	23.74	23.82	
SFC:	[kg/kW.hr]				0.987	0.946	0.912	0.888	0.874	0.868	0.877	0.906	
Tip vel. a):	[m/sec]				391.3	393.6	396.7	400.8	405.4	410.5	415.9	426.6	

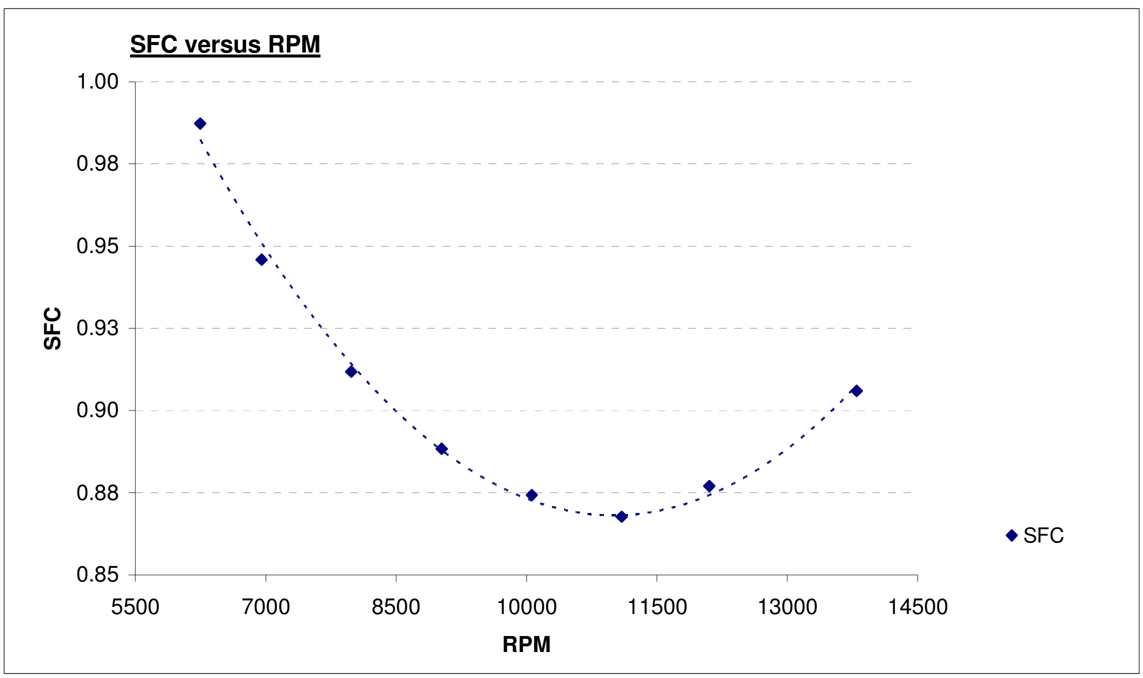


Figure I a1.1 SFC versus RPM

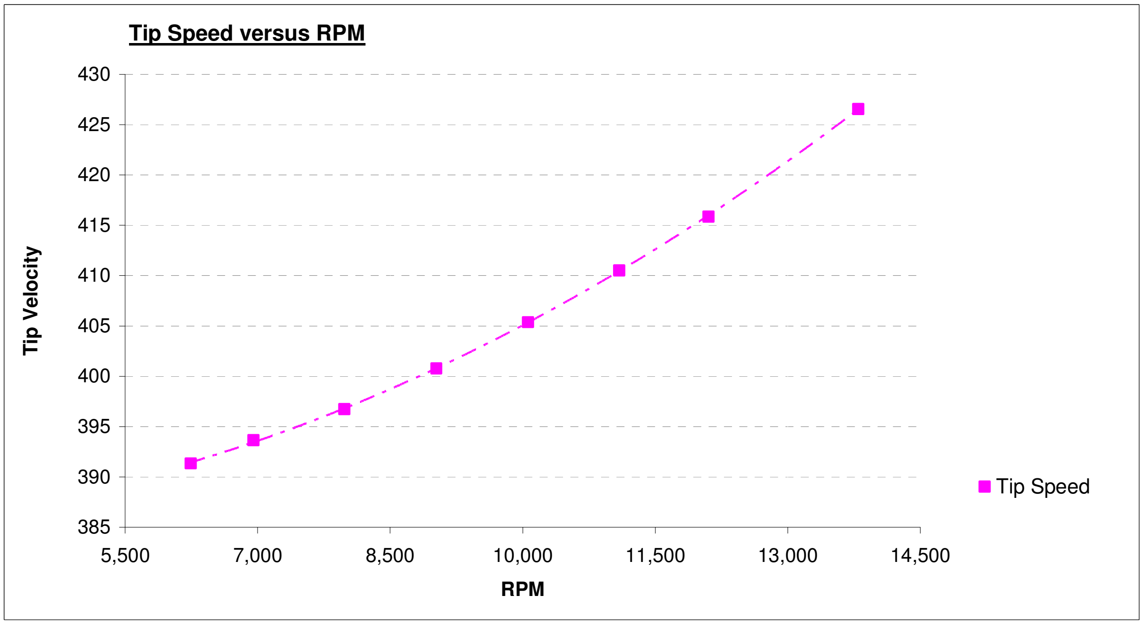


Figure I a1.2 Tip Speed versus RPM

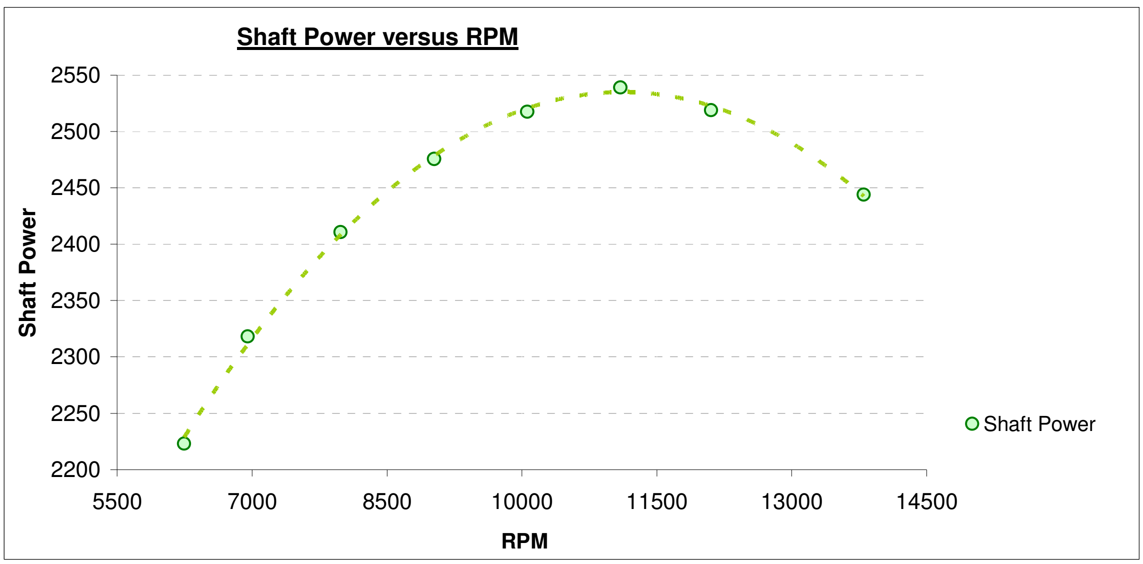


Figure I a1.3 Shaft Power versus RPM

Note:

Although the previous graphs represent a somewhat reasonable SFC of 0.87, a generated power of 2,500 kW is oversized. In order to achieve a lower maximum power output of 150 kW, the calculation was repeated yielding the following results:

Cycle condition:	**REQUIRED** *
Compressor-Turbine configuration:	**SINGLE SPOOL**
Recuperator:	**OFF**

Compressor PR:	**4.5**

Table I a2 Single spool GT at 150 kW

* Strictly evaluated for power rating requirement.

Axl impell. exit depth:	[mm]	**10**	**12**	**15**	**18**	**20**	**23**	**26**	**29**	**32**	**35**	**40**	**45**
Shft Pwr:	[kW]		149	149.7	150.3	151.6	148.7	149.8	150.4	150.1	150.2	150.3	
GT speed:	[RPM]		3,909	4,850	5,777	6,390	7,306	8,222	9,140	10,063	10,995	12,575	
Air intake:	[kg/sec]		25.43	25.58	25.72	25.81	25.93	26.01	26.07	26.11	26.13	26.12	
SFC:	[kg/kW.hr]		14.90	14.98	15.02	14.96	15.34	15.29	15.28	15.35	15.36	15.34	
Tip vel. a):	[m/sec]		386.6	389.0	391.5	393.4	396.9	400.9	405.5	410.7	416.4	426.9	

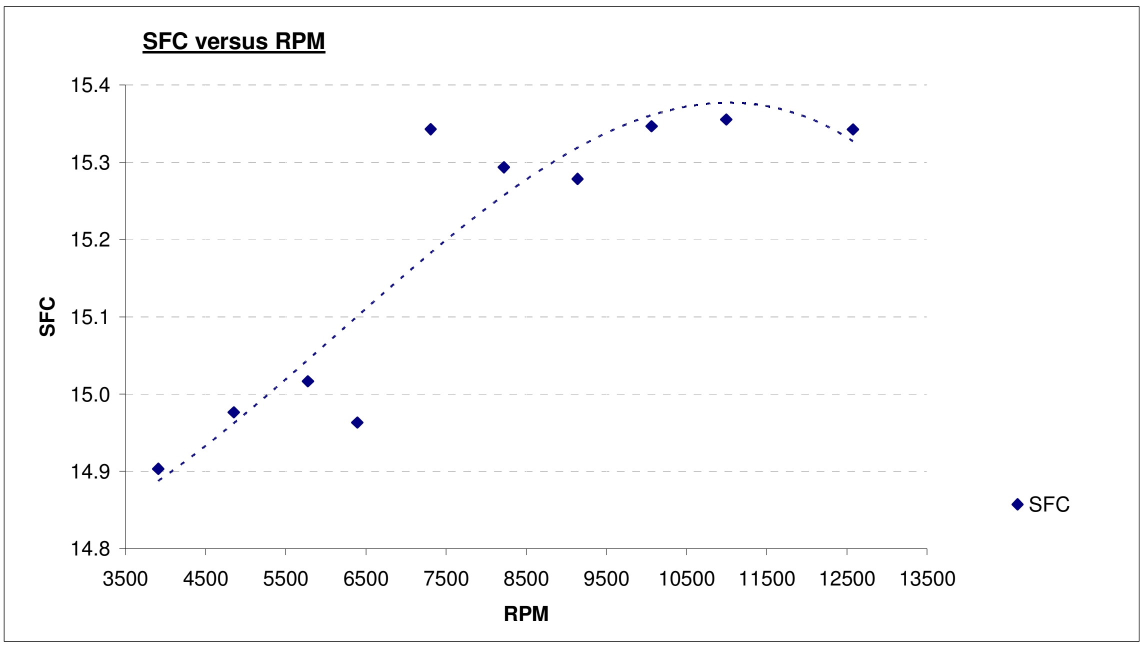


Figure I a2.1 SFC versus RPM

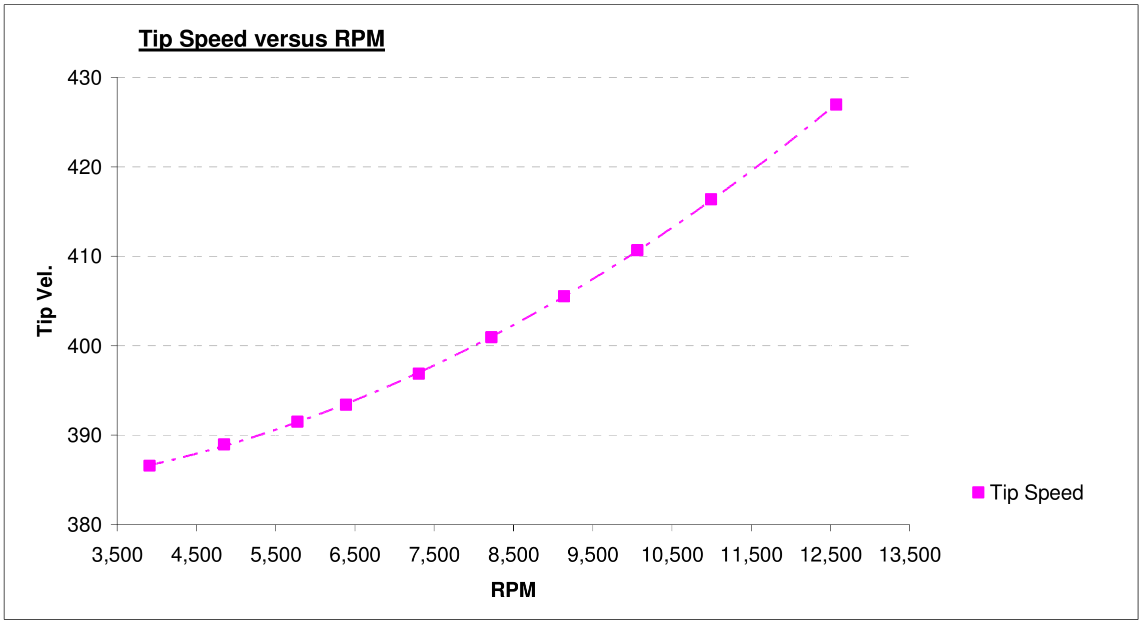


Figure I a2.2 Tip Speed versus RPM

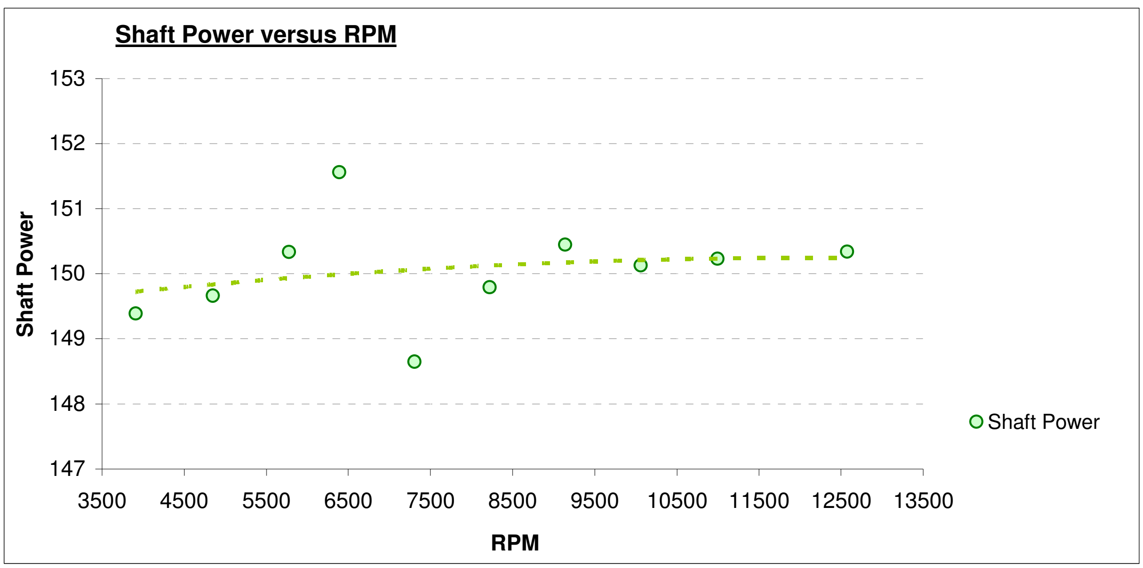


Figure I a2.3 Shaft Power versus RPM

REMARK:
From the above it is evident, that although a 150 kW engine is approachable the SFCs have become unacceptably high, and the effect of a recuperator employed would in this instance be beneficial and worth of further investigation.

b. Recuperator employed

Cycle condition:	**REQUIRED**
Compressor-Turbine configuration:	**SINGLE SPOOL**
Recuperator:	**ON**

Compressor PR:	**4.5**

Table I b Single spool GT with recuperator at PR 4.5 and 150 kW

Axl impell. exit depth:	[mm]	**10**	**12**	**15**	**18**	**20**	**23**	**26**	**29**	**32**	**35**	**40**	**45**
Shft Pwr:	[kW]		147.3	147.4	150.0	148.8	150.0	150.8	149.7	150.3	150.7	150.2	149.8
GT speed:	[RPM]		2,882	3,531	4,157	4,569	5,175	5,780	6,376	6,980	7,582	8,608	9,673
Air intake:	[kg/sec]		34.52	35.18	35.81	36.16	36.67	37.07	37.44	37.70	37.93	38.15	38.19
SFC:	[kg/kW.hr]		7.428	7.233	6.974	6.951	6.804	6.689	6.681	6.605	6.560	6.549	6.571
Tip vel. a):	[m/sec]		385.4	386.8	388.1	389.4	391.2	393.7	396.1	399.2	402.2	408.2	415.1

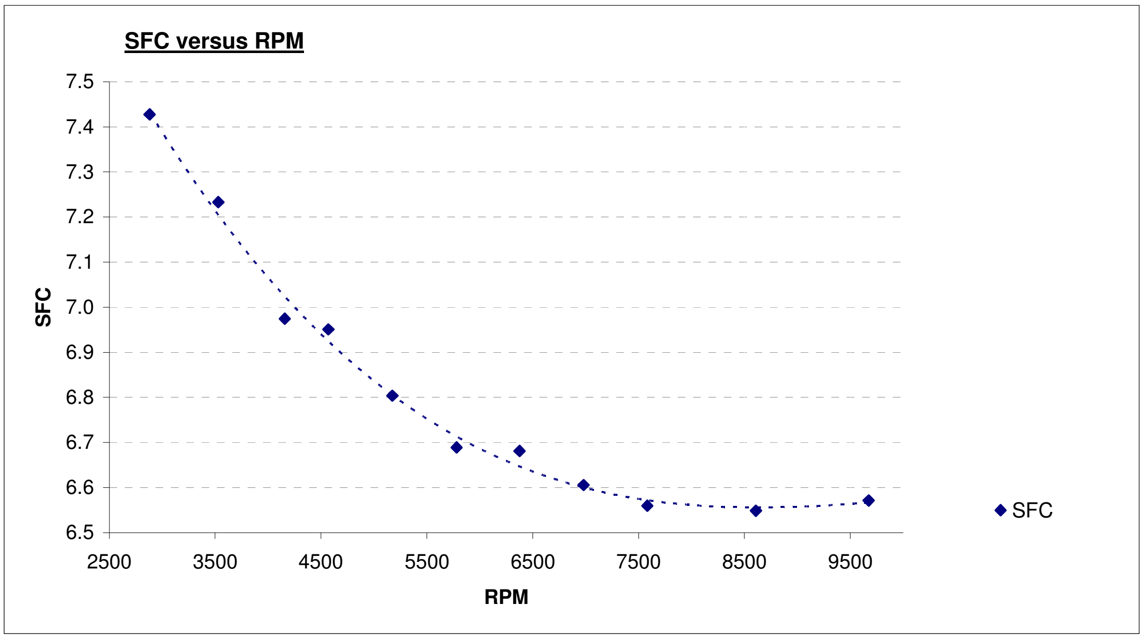


Figure I b1 SFC versus RPM

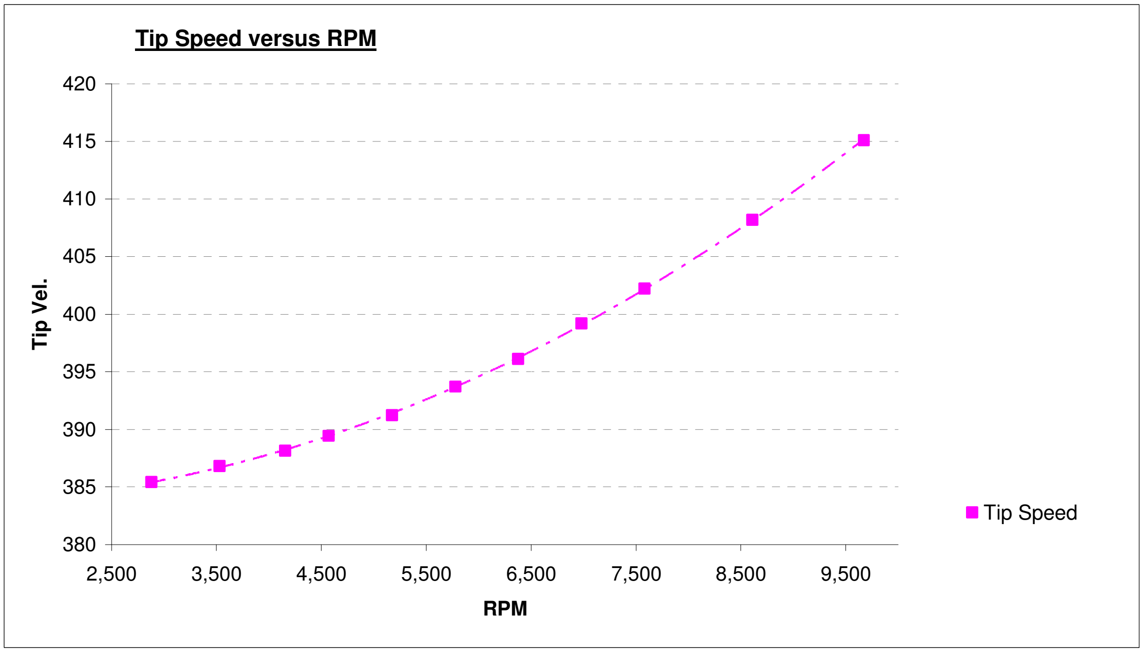


Figure I b2 Tip Speed versus RPM

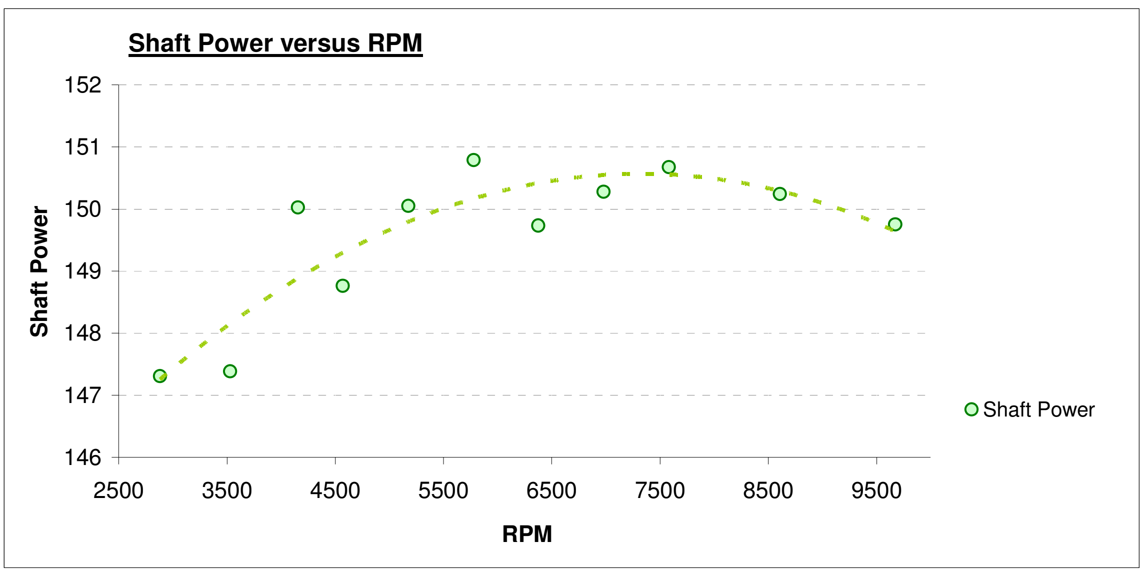


Figure I b3 Shaft Power versus RPM

REMARK:
Although the recuperator almost halves the fuel consumption, a value of about 6.5 for SFC appears still to be inhibitively high. In order to reduce the SFC but still be able to design for an optimum performance machine operating at 150 kW, the next step in the iterative design procedure was to reduce the compressor pressure ratio delivery.

II Single spool GT at varying pressure ratios

a. No recuperator employed

Note:
A 2.5 compressor pressure ratio was found to be optimal for our 150 kW power requirement; a higher value generated too much power. The right **P** rating was produced inefficiently while a lower PR did not attain to the required shaft power levels. An axial impeller exit depth setting of 8 was also found to be optimal for the chosen PR, as above 8 mm the power produced would be in excess of required and also at the expense of fuel efficiency.Therefore, all that needed to be done was to choose the 150 kW-results out of a set of input data with the engine RPM as the only possible variable.

Gas turbine configuration:	**SINGLE SPOOL**
Cycle condition:	**OPTIMAL**
Recuperator:	**OFF**
Compressor PR:	**2.5**
Ax'l impeller exit depth:	**8 mm**

Table II a No recuperator employed

Reading:		**1**	**2**	**3**	**4**	**5**	**6**	**7**	**8**	**9**	**10**	**11**	**12**
Shft Pwr:	[kW]	183.9	171.6	161.3	152.5	150.0	145.9	140.5	136.1	132.7	130.6	131.8	133.1
GT speed:	[RPM]	20,000	22,000	24,000	26,000	26,750	28,000	30,000	32,000	34,000	35,999	38,001	40,000
Air intake:	[kg/sec]	1.678	1.524	1.395	1.286	1.250	1.194	1.113	1.043	0.981	0.926	0.890	0.833
SFC:	[kg/kW.hr]	0.848	0.827	0.808	0.790	0.781	0.769	0.747	0.726	0.703	0.679	0.650	0.618
Reaction Ratio a):		0.253	0.238	0.224	0.209	0.201	0.189	0.168	0.145	0.119	0.089	0.065	0.008
Tip vel. a):	[m/sec]	381.4	380.6	379.8	379.0	378.5	377.8	376.7	375.6	374.5	373.4	372.0	371.0

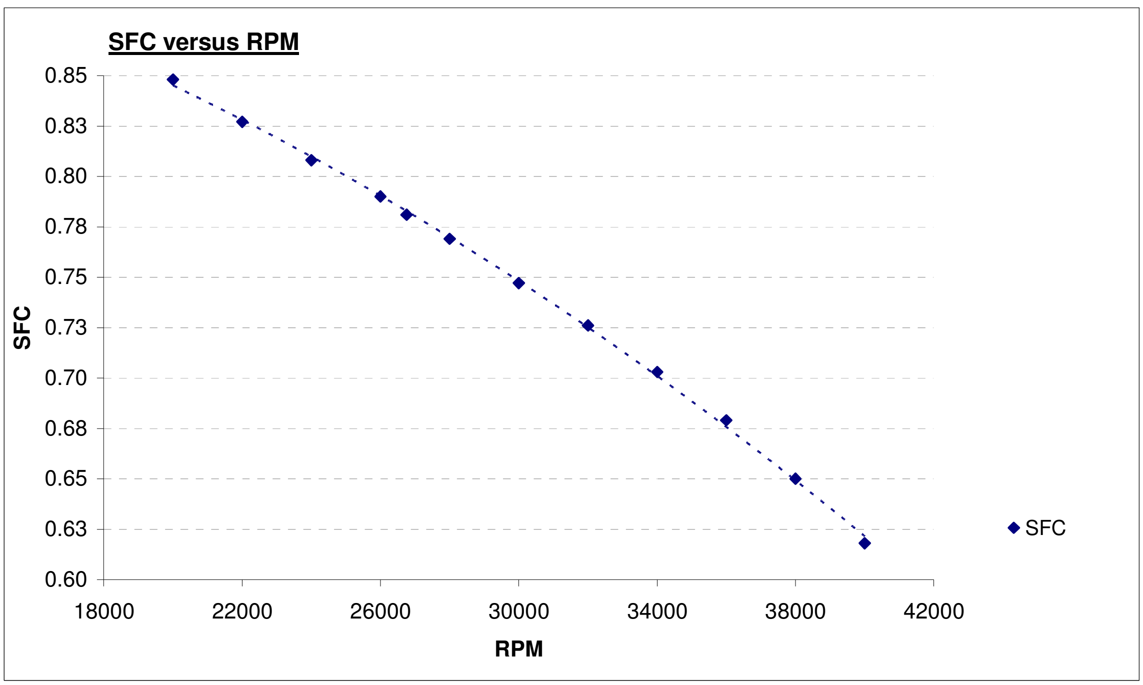


Figure II a1 SFC versus RPM

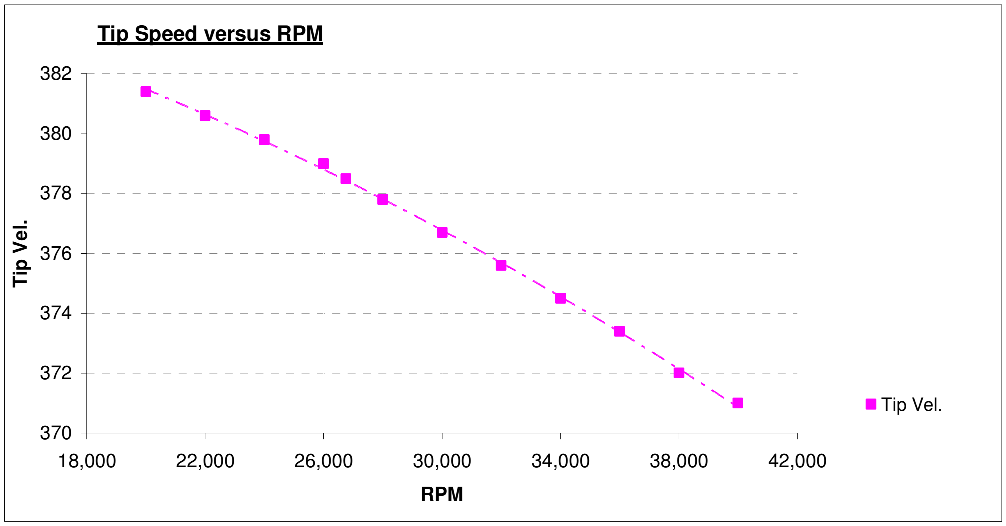


Figure II a2 Tip Speed versus RPM

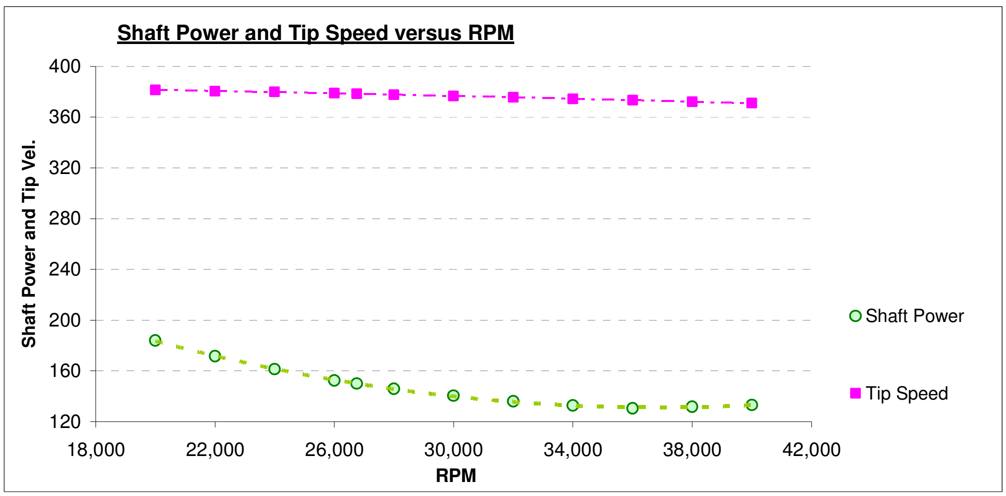


Figure II a3 Shaft Power and Tip Speed versus RPM

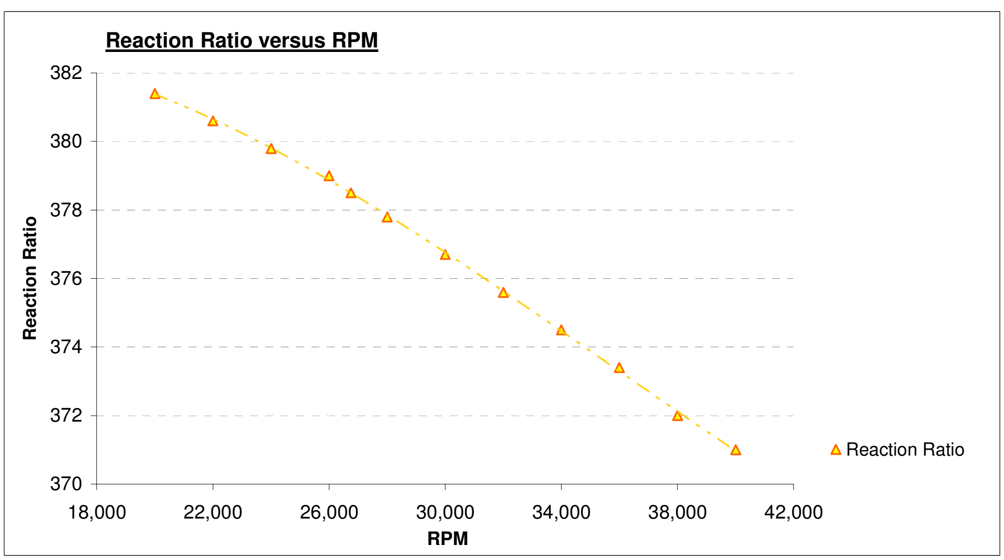


Figure II a4 Reaction Ratio versus RPM

b. Recuperator employed

Note:

By enabling the recuperator, higher axial impeller exit depths had to be chosen as not enough air flow was available for sufficient power generation. This is as a result of the preheated air entering the combustor reducing the fuel flow rate and thereby the power output. In choosing larger exit depths though, the SFC became too inefficiently high thereby requiring a higher PR compressor which explains the following data progression.

At an impeller exit depth of 11 and a PR of 2.9, the limiting lower power produced was just above 152 kW; ideal for our engine, and optimised to its lowest SFC limit.

Gas turbine configuration:	**SINGLE SPOOL**
Cycle condition:	**OPTIMAL**
Recuperator:	**ON**
Compressor PR:	**VARIES**
Ax'l impeller exit depth:	**VARIES**

Table II b Recuperator employed

Pressure Ratio:		**2.5**			**2.8**				**2.9**		
Axl' imp. exit depth:		**9**	**10**	**11**	**8**	**9**	**10**	**11**	**8**	**8.5**	**9**
Shft Pwr:	[kW]	150.2	150.2	150.2	150.1	150.0	150.0	152.4	150.1	159.3	169.0
GT speed:	[RPM]	6,060	5,750	5,733	18,683	22,773	27,860	36,186	28,924	30,045	32,141
Air intake:	[kg/sec]	6.346	7.484	8.179	2.070	1.903	1.724	1.459	1.388	1.418	1.403
SFC:	[kg/kW.hr]	0.777	0.879	0.971	0.342	0.317	0.289	0.256	0.258	0.252	0.242
Reaction Ratio a):		0.474	0.488	0.498	0.251	0.226	0.182	0.076	0.074	0.055	0.016
Tip vel. a):	[m/sec]	384.7	384.8	385.2	382.0	381.5	380.5	378.6	377.7	377.7	377.6

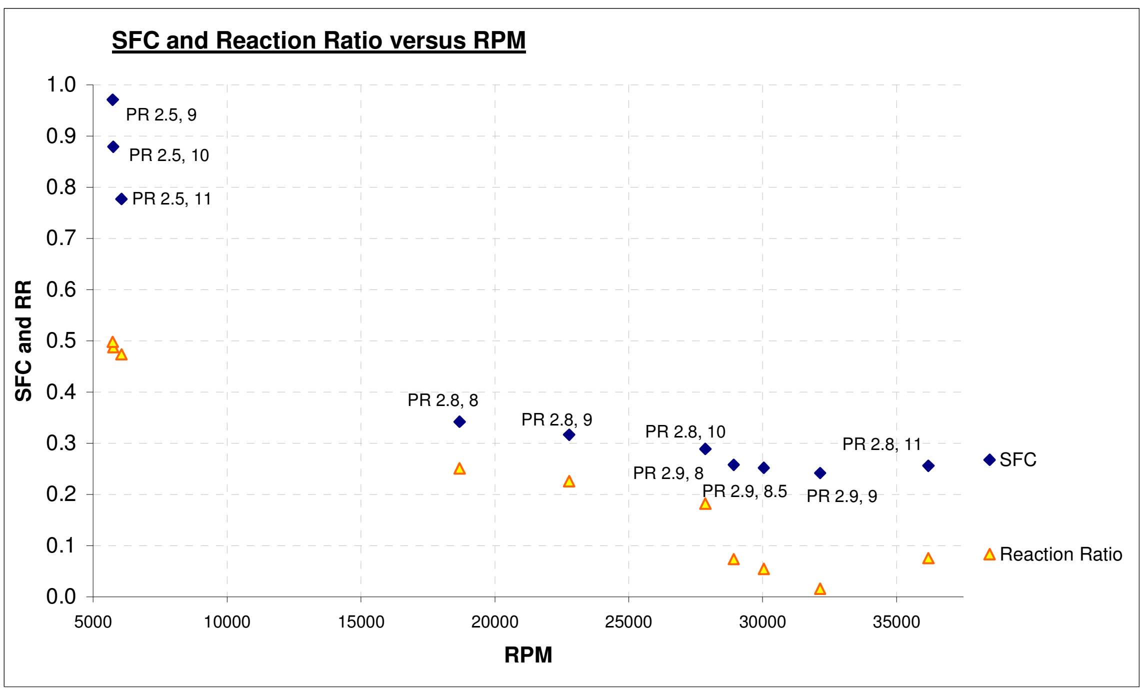


Figure II b1 SFC and Reaction Ratio versus RPM

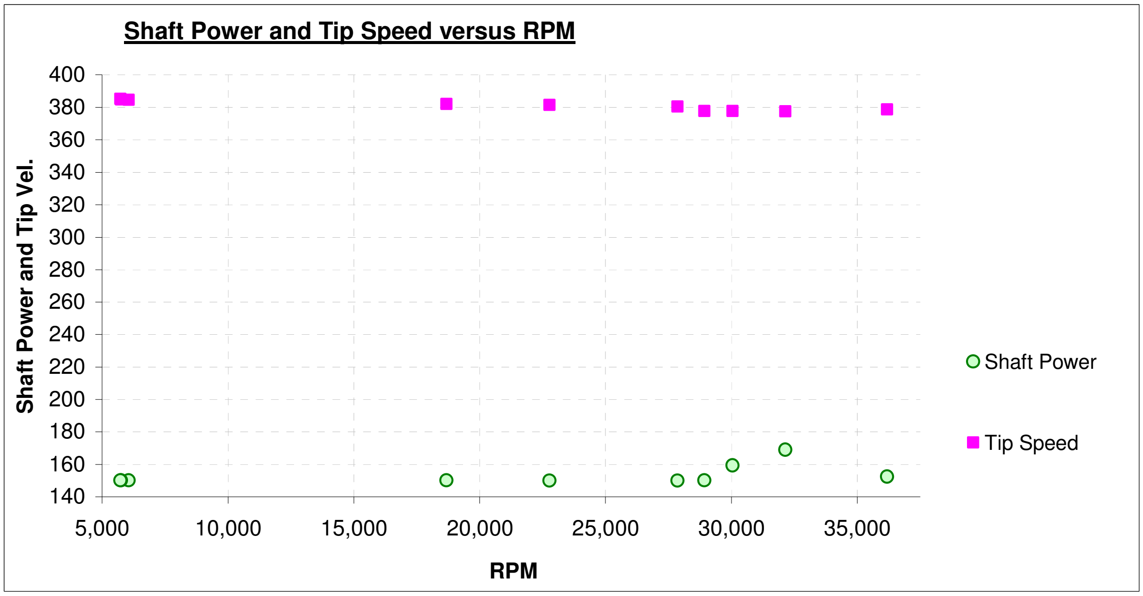


Figure II b2 Shaft Power and Tip Speed versus RPM

III Single spool GT at a fixed speed of 15,000 rpm

Seeing that this far no satisfactory solution to the single-spool problem has been found, the next aproach was to control the speed parameter spool velocity in that the most optimal SFC design lacked in sufficiently low external gear ratio requireme
By setting our speed to 12,500 rpm for optimal gearing, a satisfactory compromise could be found between the other settin

a. No recuperator employed

Cycle condition:	**REQUIRED**
Compressor-Turbine configuration:	**SINGLE SPOOL**
Recuperator:	**OFF**

Compressor PR:	**VARIABLE**

Table III a No recuperator employed

Pressure Ratio:		**2.4**	**2.5**	**2.6**	**2.7**
Axl' imp. exit depth:		5.5	4.6	3.9	3.2
Shft Pwr:	[kW]	150.5	150.1	152.5	152.7
Air intake:	[kg/sec]	1.777	1.571	1.402	1.208
SFC:	[kg/kW.hr]	1.085	0.963	0.849	0.737
Tip vel. a):	[m/sec]	382.1	381.6	381.2	380.8
Reaction Ratio a):		0.348	0.285	0.215	0.129

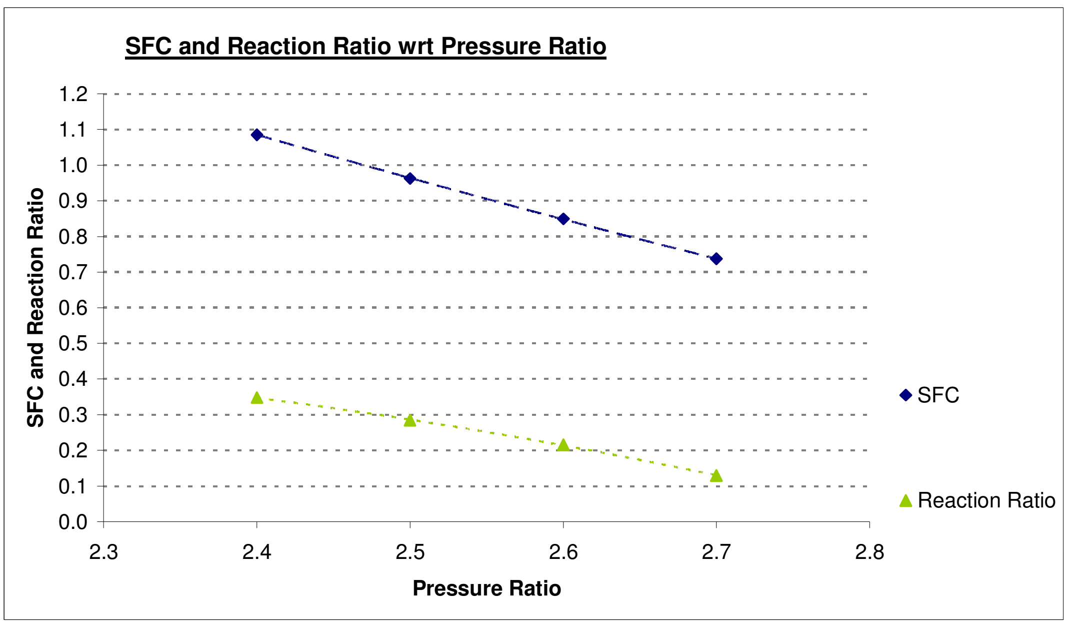


Figure III a1 SFC and Reaction Ratio versus PR

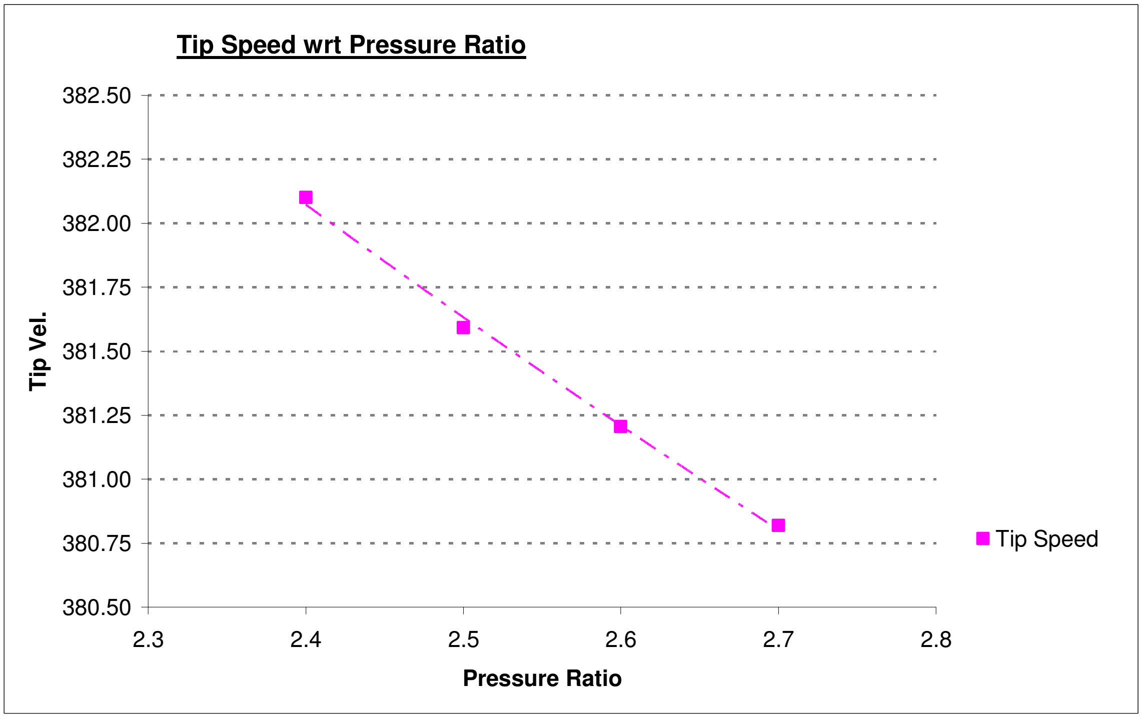


Figure III a2 Tip Speed versus PR

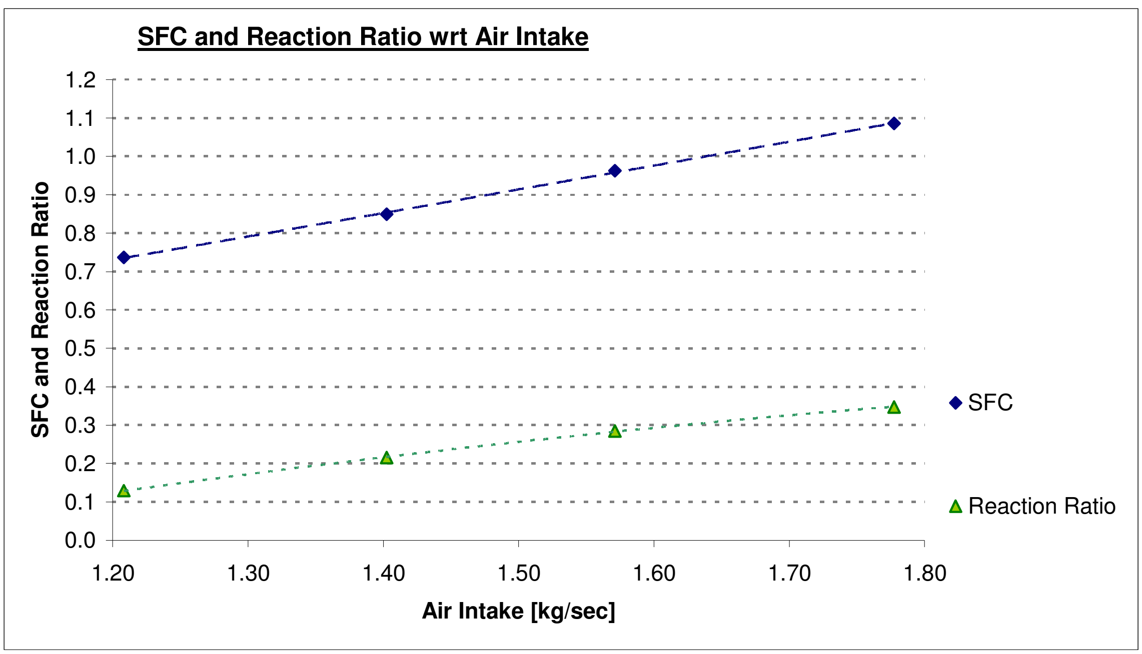


Figure III a3 SFC and Reaction Ratio versus Air Intake

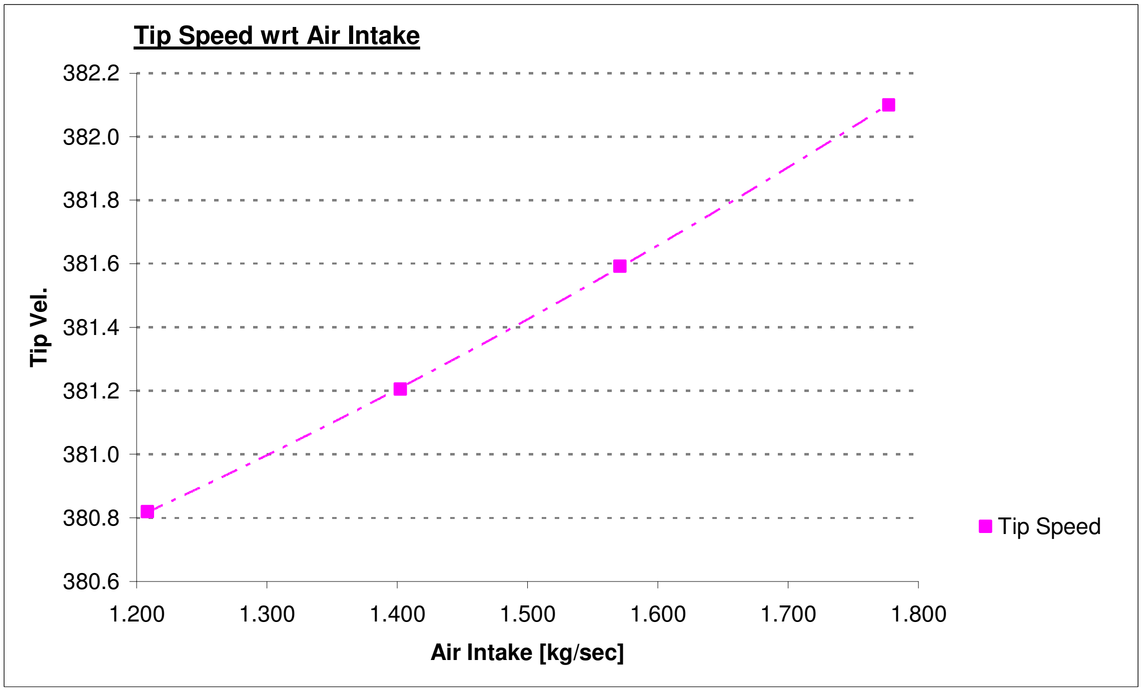


Figure III a4 Tip Speed versus Air Intake

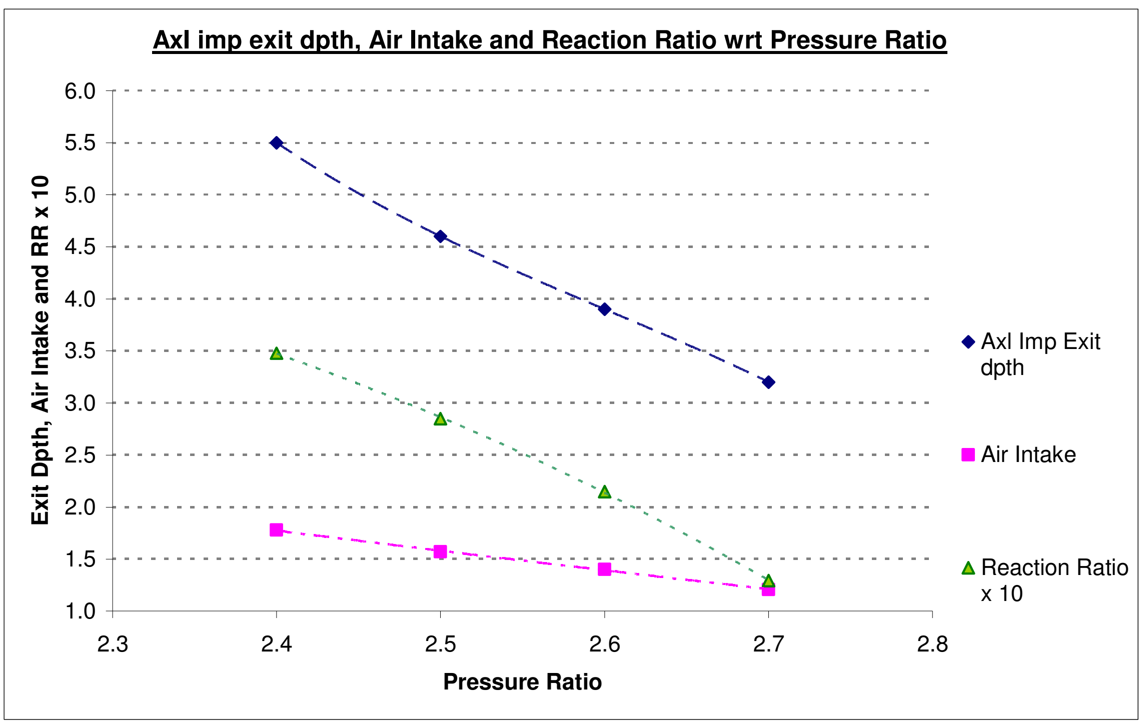


Figure III a5 Axial Impeller Exit Depth, Air Intake and Reaction Ratio w.r.t. Pressure Ratio

b. Recuperator employed

Cycle condition:	REQUIRED
Compressor-Turbine configuration:	SINGLE SPOOL
Recuperator:	ON

Compressor PR:	VARIABLE

Table III b Single spool GT with recuperator at variable PR and 150 kW

Pressure Ratio:			2.6	2.7	2.8	2.9	3	3.1
Axl' imp. exit depth:			9.1	7.5	6.3	5.2	4.3	3.3
Shft Pwr:	[kW]		151.2	150.0	151.2	150.2	152.5	150.4
Air intake:	[kg/sec]		3.227	2.796	2.461	2.124	1.832	1.464
SFC:	[kg/kW.hr]		0.432	0.407	0.386	0.362	0.340	0.314
Tip vel. a):	[m/sec]		384.2	383.3	382.6	382.0	381.5	380.9
Reaction Ratio a):			0.389	0.336	0.280	0.217	0.143	0.040

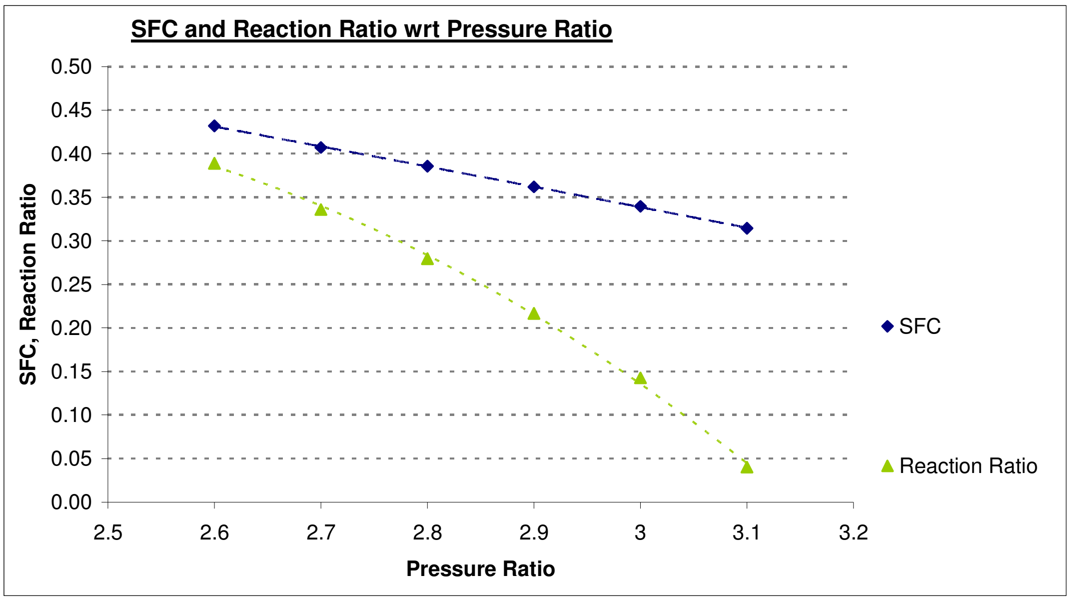


Figure III b1 SFC and Reaction Ratio versus Pressure Ratio

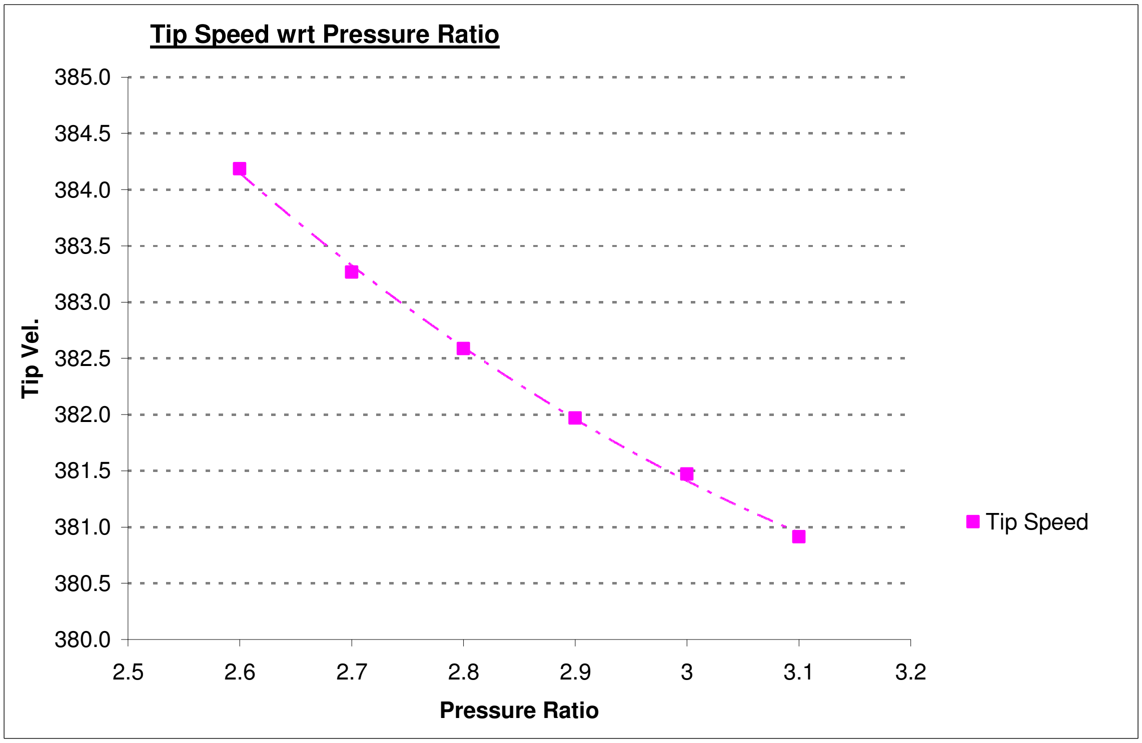


Figure III b2 Tip Speed versus Pressure Ratio

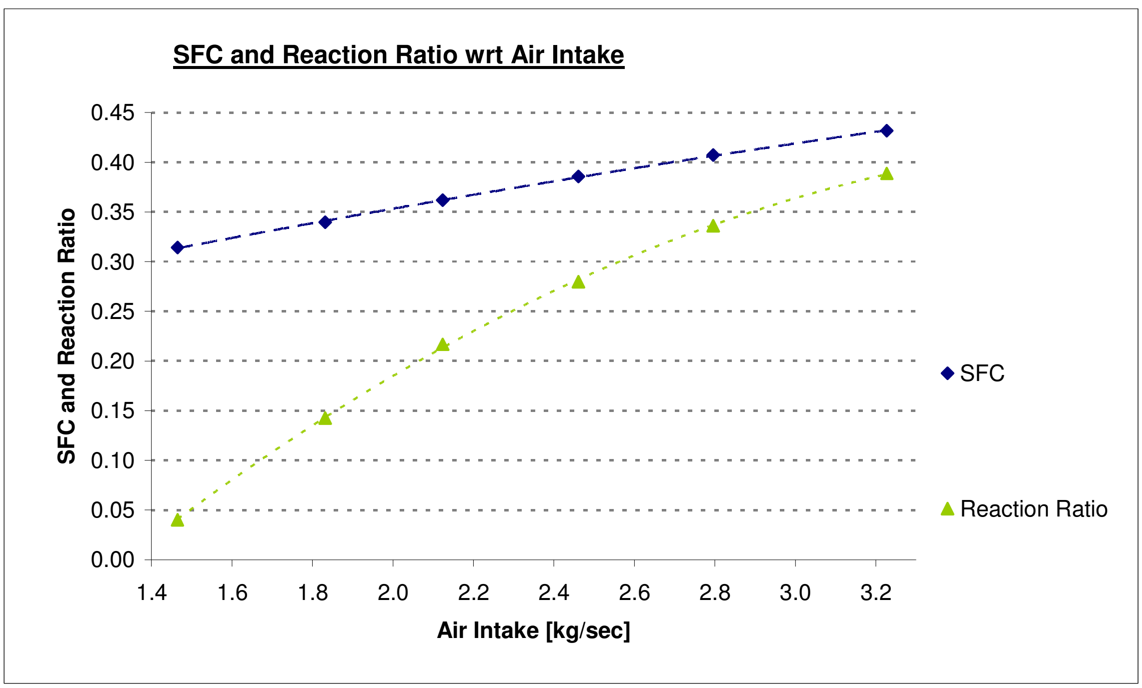


Figure III b3 SFC and Reaction Ratio versus Air Intake

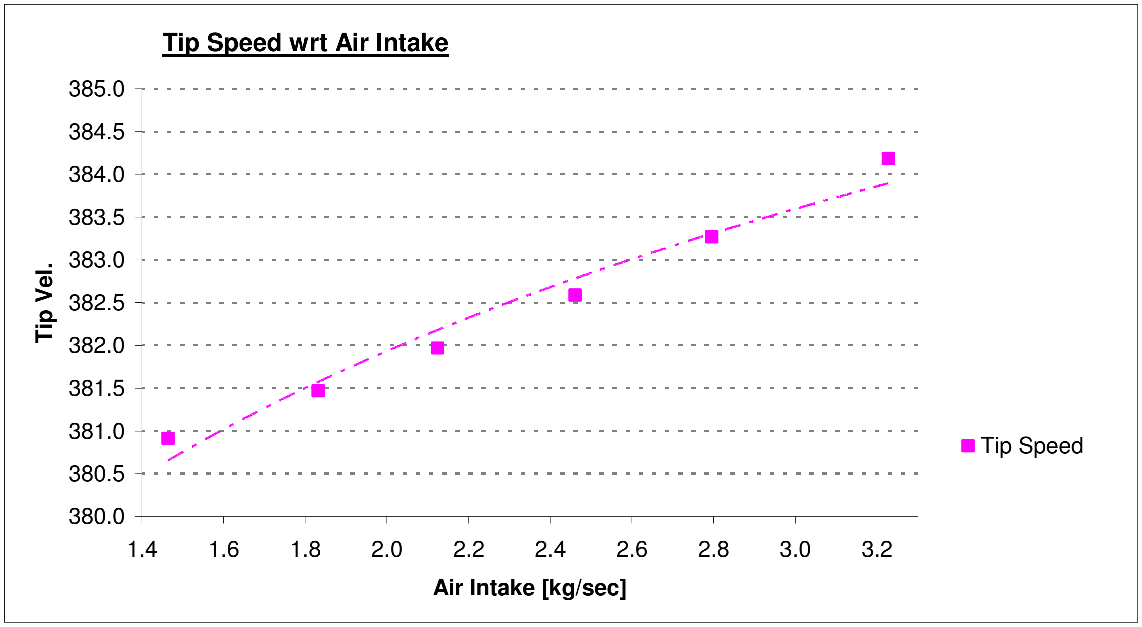


Figure III b4 Tip Speed versus Air Intake

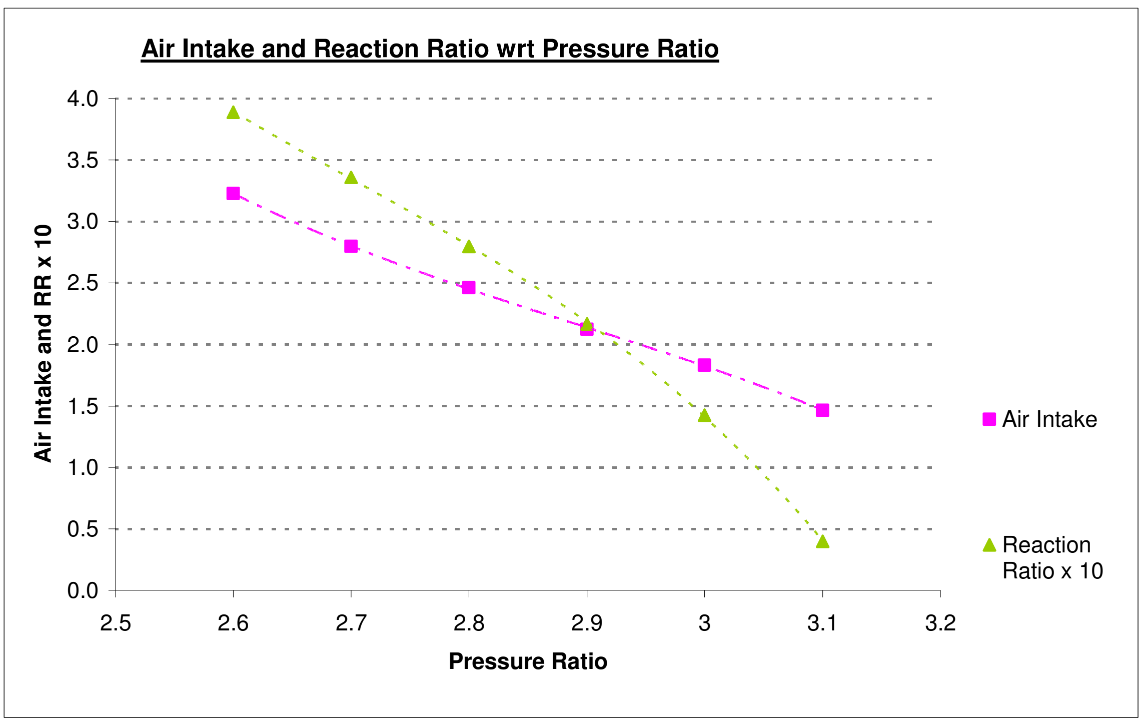


Figure III b5 Air Intake and Reaction Ratio wrt Pressure Ratio

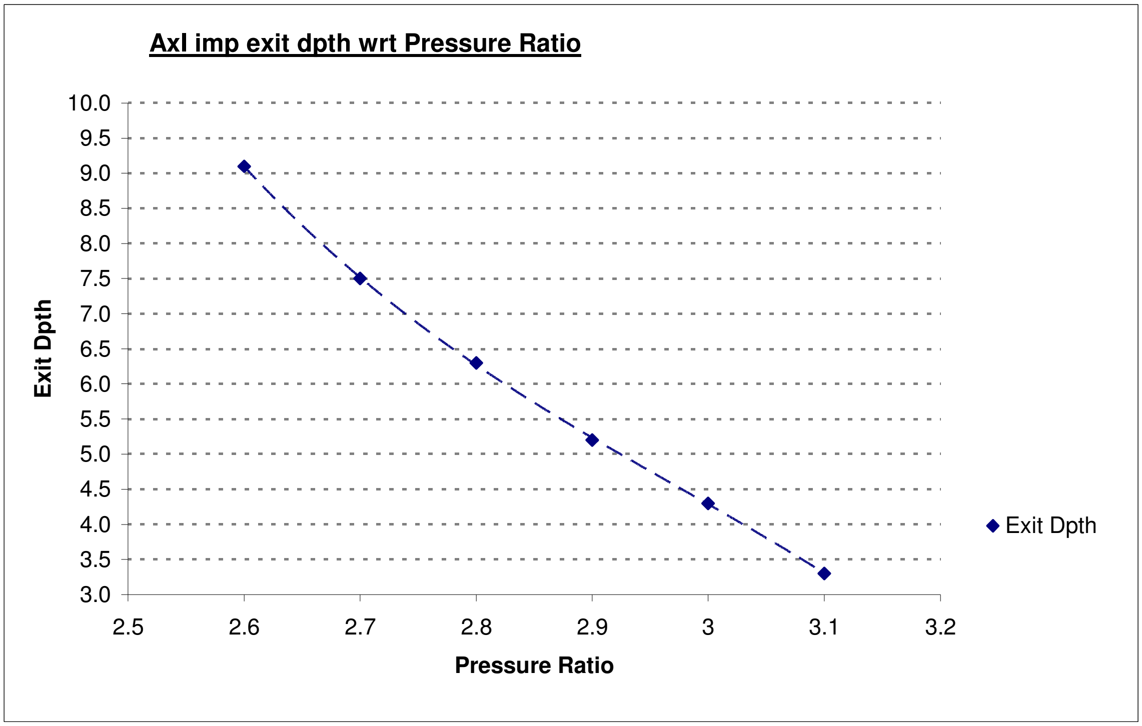


Figure III b6 Axial Impeller Exit Depth wrt Pressure Ratio

IV Twin spool GT without recuperation

a. At various impeller exit depths

Note: The approach adopted here is identical to section 5.3.1 at 150 kW: The optimised approach has been ignored simply because of excessively high power ratings, and again effort has been made to satisfy the 150 kW engine size requirement. An external gear ratio of 4:1 has been employed here thereby yielding slightly more optimistic values than if 5:1 was used.

Cycle condition:	**REQUIRED**
Compressor-Turbine configuration:	**TWIN SPOOL**
Recuperator:	**OFF**

Compressor PR:	**4.5**

Table IV a Variable axial impeller exit depth approach

Axl impell. exit depth:	[mm]	**12**	**14**	**16**	**18**	**20**	**23**	**26**	**29**	**32**	**35**	**40**	**45**
Shft Pwr:	[kW]	157.7	206.1	268.1	338.5	420.4	564.7	718.3	906.1	1,141.7	1,444.0		
GT speed:	[RPM]	100,000	88,500	77,500	69,000	62,000	53,000	47,000	41,500	36,500	31,500		
Air intake:	[kg/sec]	1.005	1.326	1.732	2.189	2.704	3.610	4.642	5.861	7.345	9.297		
SFC:	[kg/kW.hr]	0.535	0.534	0.532	0.533	0.534	0.534	0.533	0.534	0.536	0.538		
Tip vel. a):	[m/sec]	488.0	489.8	489.7	489.8	489.9	489.5	489.3	488.8	487.8	485.7		
Tip vel. b):	[m/sec]	381.3	381.8	382.3	382.9	383.6	384.8	386.1	387.8	389.8	392.5		

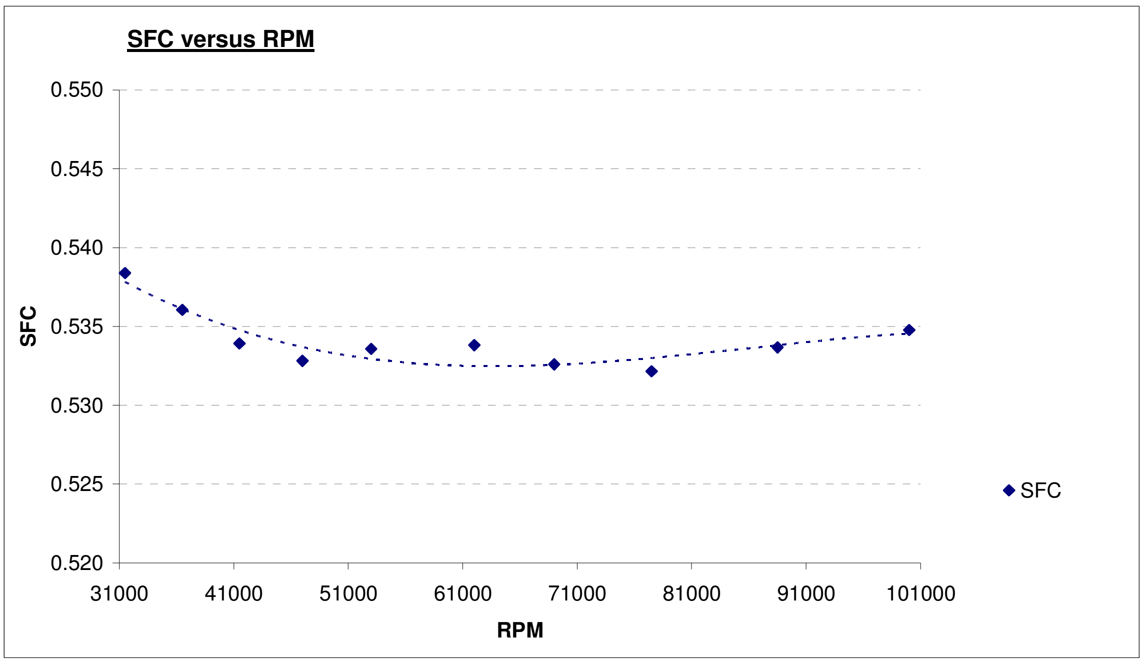


Figure IV a1 SFC versus RPM

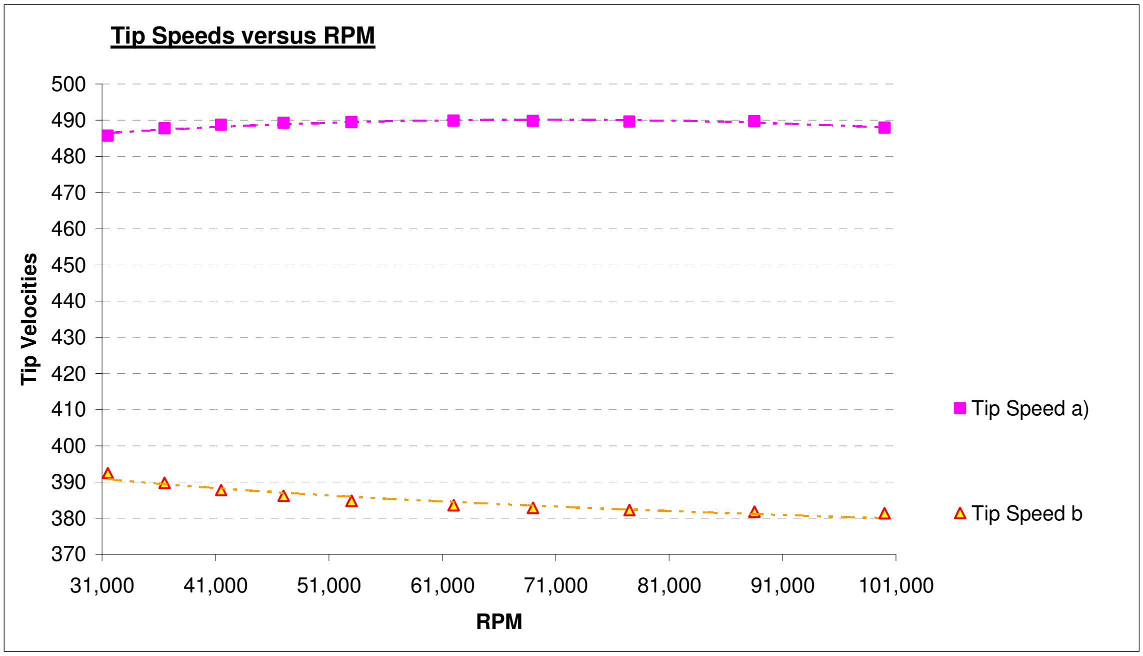


Figure IV a2 Tip Speeds versus RPM

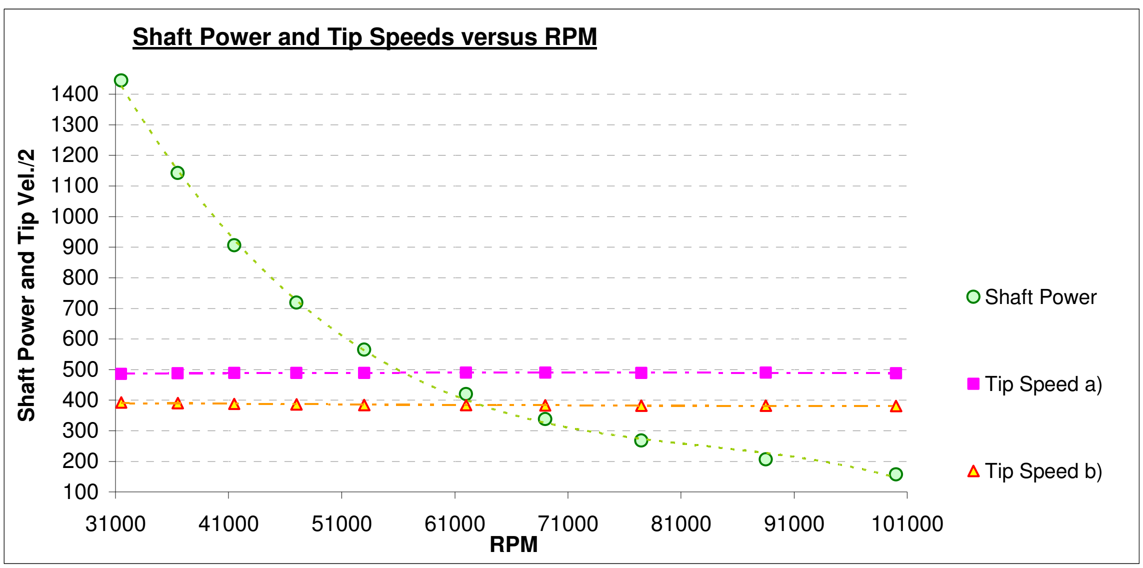


Figure IV a3 Shaft Power and Tip Speeds versus RPM

REMARK:

It should be apparent from the above, that the lowest power setting of 157.7 kW is the only reasonable approach for our engin
Utilising a recuperator for this kind of parameter setting yields excessively high engine power and is therefore ignored.

b. At various pressure ratios

Approaching the twin spool design from a variable pressure ratio perspective yielded the following set of results:

Gas turbine configuration:	**TWIN SPOOL**
Cycle condition:	**OPTIMAL**
Recuperator:	**OFF**
Compressor PR:	**VARIABLE**
Ax'l impeller exit depth:	**11 mm**

Table IV b Variable pressure ratio approach

Pressure Ratio:		**N/A 3.8**	**3.9**	**4.0**	**4.1**	**4.2**	**4.3**	**4.4**	**4.5**
Shft Pwr:	[kW]	132.8	150.2	150.2	150.9	150.8	150.2	150.4	149.8
GT speed:	[RPM]		83,302	86,261	88,835	91,618	94,494	96,929	99,538
Air intake:	[kg/sec]	0.871	0.937	0.931	0.928	0.937	0.919	0.918	0.915
SFC:	[kg/kW.hr]		0.535	0.529	0.523	0.519	0.517	0.517	0.513
Reaction Ratio a):		0.497	0.499	0.481	0.463	0.445	0.426	0.409	0.390
Reaction Ratio b):		0.511	0.486	0.483	0.479	0.484	0.476	0.474	0.475
Tip vel. a):	[m/sec]	476.4	474.4	476.0	477.4	478.9	480.5	481.9	483.4
Tip vel. b):	[m/sec]	381.2	381.3	381.3	381.2	381.2	381.2	381.2	381.2

Remark:
Similarly to the single-spool design approach with recuperation disengaged, an exit depth of 11 mm proved to be optimal. What was not certain was the optimal pressure ratio applicable to the twin-spool design, and thus the most efficient engine was derived by recording a set of 150 kW engines with their PR's varying from the lowest limiting value of 3.8 below which too little power was available to an upper value of 4.5 which is the compressor limiting pressure. In all the above values though, the compressor turbine tip velocity has been exceeded, and account has to be made in a possible design for excessive radial stresses because of this. In general though a PR of 4.5 would appear most fuel economical.

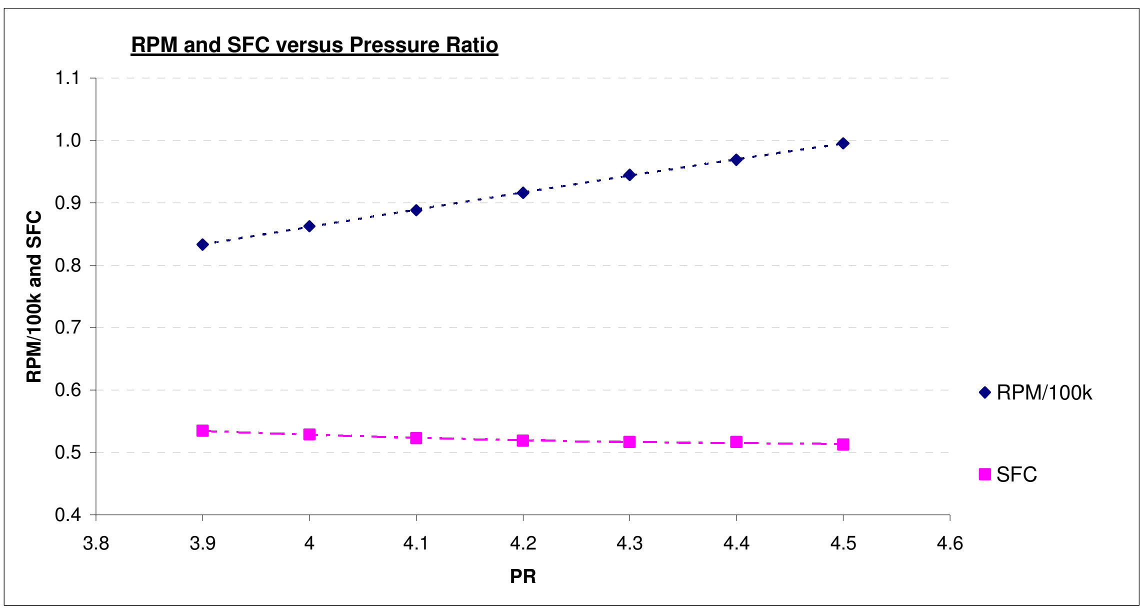


Figure IV b1 RPM and SFC versus Pressure Ratio

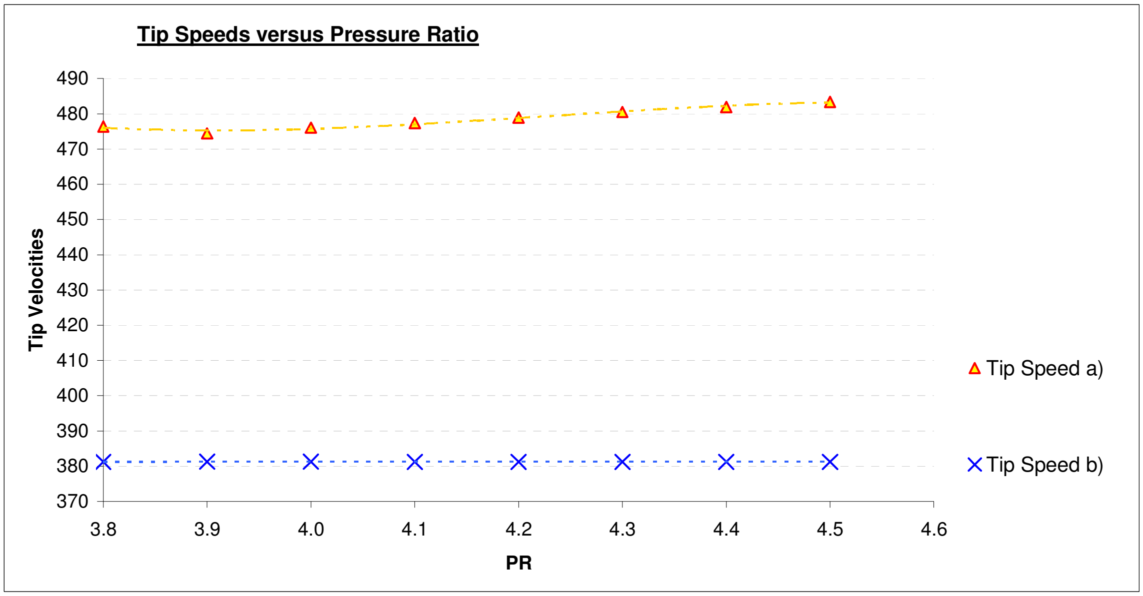


Figure IV b2 Tip Speeds versus Pressure Ratio

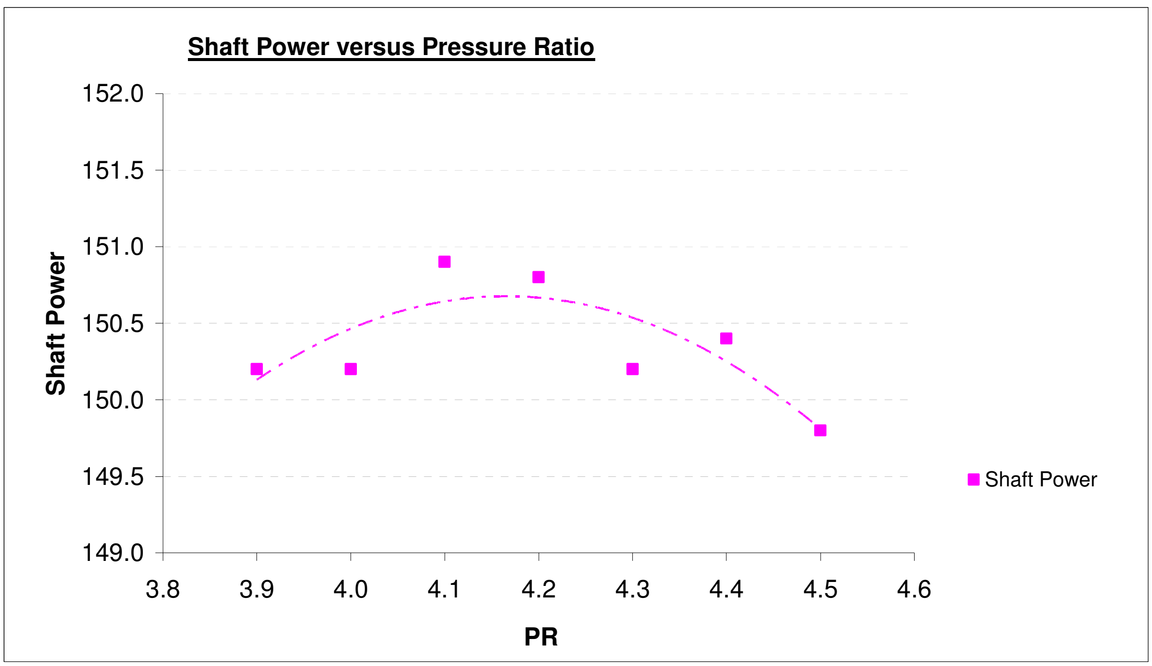


Figure IV b3 Shaft Power versus Pressure Ratio

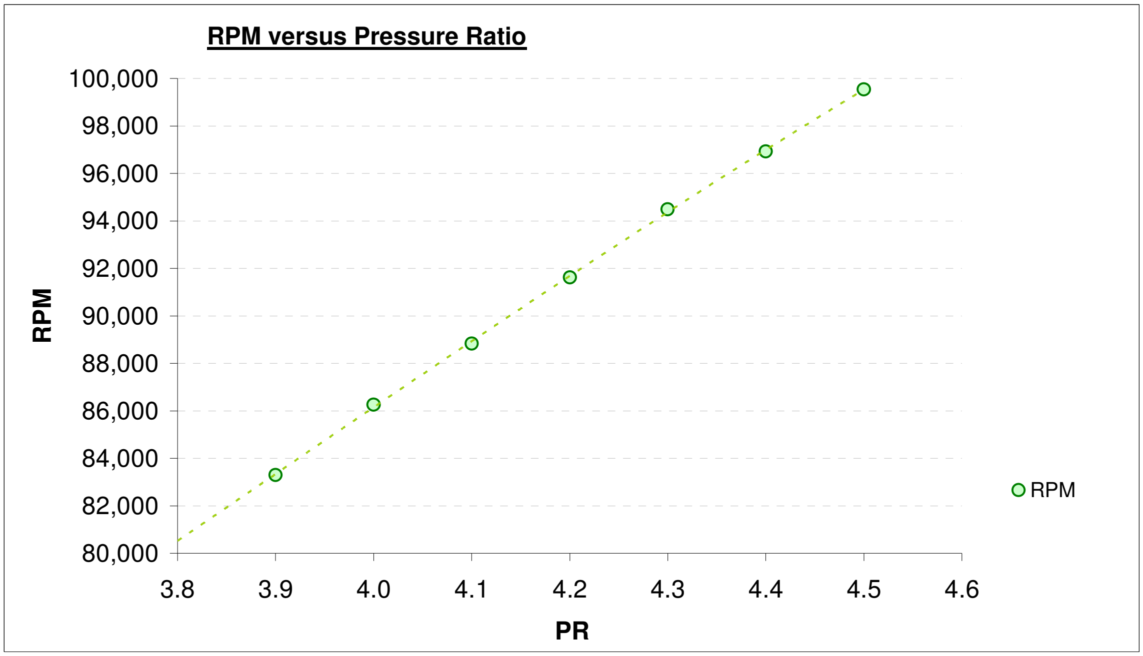
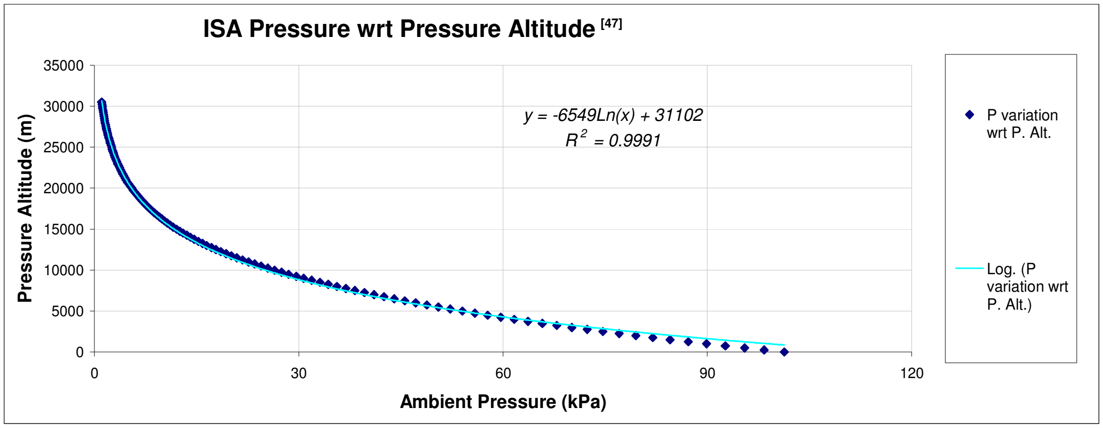
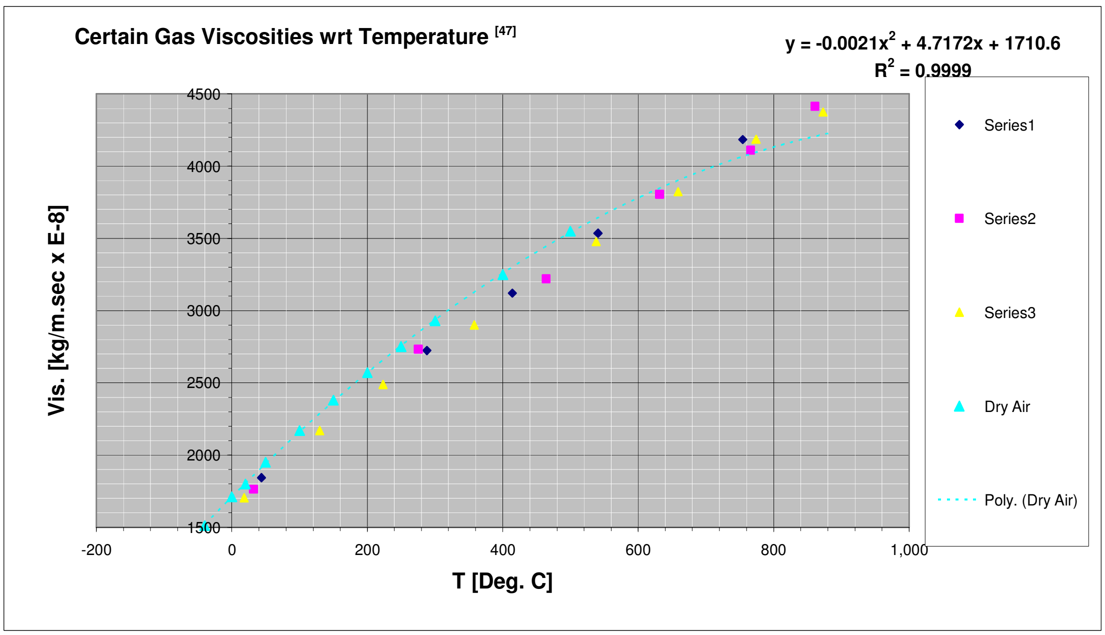


Figure IV b4 RPM versus Pressure Ratio





Thermodynamic Properties [47]:

Name:	Chem. abbrev.:	Molecular weight:	Gas constant: [J/kg K]
Dry air	See T-3	28.964	287.05
Oxygen	O2	31.999	259.83
Water	H2O	18.015	461.51
Carbon dioxide	CO2	44.01	188.92
Nitrogen	N2	28.013	296.80
Carbon	C	12.01	
Argon	Ar	39.948	208.13
Hydrogen	H2	2.016	4,124.16
Neon	Ne	20.183	411.95
Helium	He	4.003	2,077.02
Carbon monoxide	CO		
Water vapour	H2O,vap		
Nitric oxide	NO		
Nitrous oxide	N2O		
Nitrogen dioxide	NO2		
Ammonia	NH3		
Sulphur	S2	64.132	
Sulphur dioxide	SO2		
Sulphur trioxide	SO3		
Methanol	CH4O		
Ethanol	C2H6O		
Hydrogen chloride	HCl		

Spec. Heat (Cp) for key Gases:

A0	A1	A2	A3	A4	A5	A6	A7	A8	A9	A10
0.99231	0.23669	-1.85215	6.08315	-8.89393	7.09711	-3.23473	0.79457	-0.08187	0.42218	0.00105
1.00645	-1.04787	3.72956	-4.93417	3.28415	-1.0952	0.14574	0	0	0.36979	0.00049
1.93704	-0.96792	3.33891	-3.65212	2.33247	-0.81945	0.11878	0	0	2.86077	-0.00022
0.40809	2.0272	-2.40555	2.03917	-1.16309	0.38136	-0.05276	0	0	0.36674	0.00174
1.07513	-0.2523	0.34186	0.52394	-0.88898	0.44262	-0.07479	0	0	0.44304	0.00174

Chem. Comp.- Dry Air:

RUNIVERSAL: [J/molK]

% by mol/vol:	% by mass:
20.95	23.14
0.03	0.05
78.08	75.52
0.93	1.28
0.002	0.001
Overall % compos.:	
99.992	99.991
Molar wt:	R [J/kg K]:
29.0	287.1