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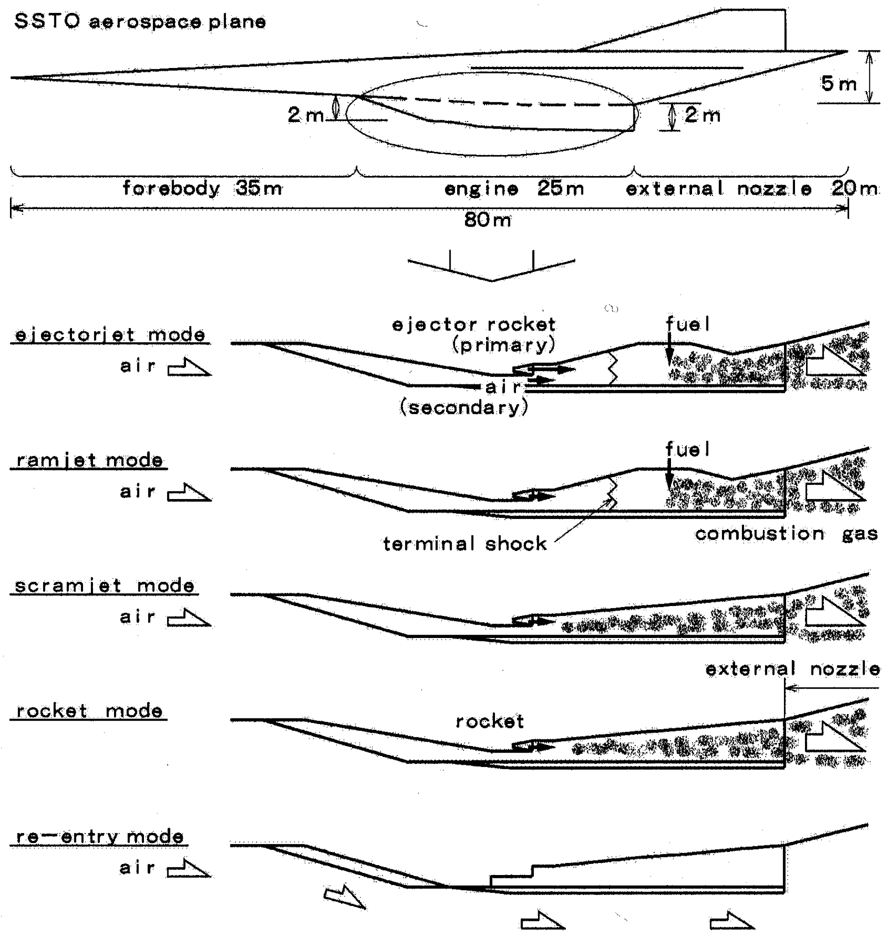
Technical report

Anders Hasselrot, Björn Montgomerie

An Overview of Propulsion Systems for Flying Vehicles



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Report title

An Overview of Propulsion Systems for Flying Vehicles

Abstract

This report presents an overview of propulsion systems for flying vehicles, based on literature studies on current and future technologies. This effort is motivated by the need of this kind of information for the studies of vehicle concepts, being pursued in order to evaluate their performance merits. General information on inlets and outlets are also presented, as these influence the geometry and performance of the vehicles. The overview of inlet types is given with respect to the requirements at various Mach number regimes that govern the design. The stealth aspects are also viewed.

In the report the available propulsion principles are presented category-wise, where most are based on combustion according to the Brayton cycle (heat addition under constant-pressure) and the other are based on combustion according to the Humphrey cycle (heat addition under constant-volume). The choice of propulsion system is mainly dictated by the requirements on the vehicle in terms of flight mission and maximum Mach number. Each propulsion system is described more or less in detail, depending on the applicability interests for the vehicle studies. Some information to be of help for judging performance is also given. The main focus is given to the following propulsion areas: turbojet/turbofan, turboramjet, air turbo ramjet, and pulse detonation engine.

Keywords

Propulsion system, turbojet, turbofan, ramjet, scramjet, turboramjet, air turbo ramjet, pulse detonation engine, inlet

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Rapportens titel (i översättning)
En översikt av framdrivningssystem för flygfarkoster

Sammanfattning (högst 200 ord)

Denna rapport presenterar en översikt över framdrivningssystem för flygfarkoster, baserad på litteraturstudier imom befintliga och framtida teknologier. Detta arbete motiveras av att sådan information behövs för studier av farkostkoncept, som för närvarande bedrivs för att evaluera flygprestanda. Generell information beträffande inlopp och utlopp presenteras också, eftersom de påverkar både geometri och prestanda hos farkosterna. Översikten över intagen ges med referens till de machtalsberoende krav som styr deras utformning. Smygegenskaper hos intagen beaktas också.

I rapporten redovisas förekommande framdrivningsprinciper i kategorier, där de flesta baseras på förbränning enligt Brayton-cykeln (värmetillförsel under konstant tryck) och de resterande baseras på förbränning enligt Humphrey-cykeln (värmetillförsel under konstant volym). Valet av framdrivningssystem är huvudsakligen bestämt av kraven på farkosten, i termer av uppdragsprofil och maxfart. Varje framdrivningssystem beskrivs mer eller mindre i detalj, beroende på graden av applikationsintresse för studerade farkoster. Viss information till hjälp vid bedömning av prestanda ges också. Tonvikten läggs på följande framdrivningsområden: Turbojet/turbofan, turboramjet, air turbo rocket samt pulsdetonationsmotor.

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List of Acronyms and Explanations

Acronyms

AADC	Allison Advanced Development Company
AAF	Air-to-Air Fighter
ARCC	Afterburner-Ramjet Combustion Chamber
ARPA	Advanced Research Projects Agency
ATR	Air Turbo Rocket/Ramjet
ATREX	Air TurboRamjet EXpander Cycle
BPR	ByPass Ratio
C-D	Convergent-Divergent (nozzle)
CFD	Computer Fluid Dynamics
DDT	Deflagration to Detonation Transition
DoD	Department of Defense
DOE	Department of Energy
IHPTET	Integrated High Performance Turbine Engine Technology
HST	HyperSonic Transport aircraft
LID	length-to-diameter (PDE ejectors)
MOBY	Modulating Bypass VCE Concept
NASA	National Air-Space Administration
PDE	Pulse Detonation Engine
PDRE	Pulse Detonation Rocket Engine
PDWE	Pulse Detonation Wave Engine = PDE
PR	Pressure Ratio
RATR	Regenerative ATR
RBCC	Rocket Based Combined Cycle
RJ	RamJet
SFC	Specific Fuel Consumption
SOFRAM	SOlid Fuel RAMjet
STOVL	Short Takeoff Vertical Landing
TACE	Turbo Augmented Cycle Engine
TBCC	Turbo Based Combined Cycle
TF	TurboFan
TFRJ	TurboFan RamJet
TJ	TurboJet
TR	Throttle Ratio = total temperature at exit of core burner, related to static sea-level condition
TRE	TurboRamjet Engine = TRJ
TRJ	TurboRamJet
TSTO	Two-Stage-To-Orbit system
UAV	Unmanned Air Vehicle
UCAV	Unmanned Combat Air Vehicle
UEET	Ultra-Efficient Engine Technology
USAF	U S Air Force
VAATE	Versatile, Affordable, Advanced Turbine Engines
VABI	Variable Area Bypass Injector
VAPCOM	VAriable Pumping COMpressor
VCE	Variable Cycle Engine

Explanations

Brayton Cycle	Combustion that takes place as a constant-pressure process with burning governed by temperature front (heat addition at constant pressure)
Humphrey Cycle	Combustion that takes place as a constant-volume process with burning governed by pressure front (heat addition at constant volume)
Stoichiometric Combustion	Combustion where all fuel and all oxygen in the mixture are used

1 Introduction

This report is intended to be a guide to available propulsion systems for use on air vehicles. General technical descriptions of these systems are outlined. The author's perspective is based on relative merits to facilitate judging/choosing alternative propulsion systems for different vehicle needs. Another aim has been to get the latest information on recent work within various propulsion technologies, but also to get relevant installation aspects for air vehicles, such as operational conditions for inlets and outlets, and low-signature effects. The information in this report is based on literature studies, which span from basic textbook material to bibliographic information obtained through searching report libraries and the web. As a result various conference proceedings emerge as the main source of information.

There are countless ideas of how to generate force, which are all founded on the Newtonian law of action – reaction, i.e. accelerating an air or gas mass to produce force. The energy needed for this process is supplied by a fuel, which can be in liquid, solid or gaseous state, burning with oxygen in the air or in a carried oxidizer, also in the mentioned states. The means of propulsion can be a propeller, fan, ejector, jet, or a combination thereof. In this report the Otto and Diesel engine principles are excluded.

The propulsion systems are either based on the Brayton cycle, which implies that the heat addition is performed at constant pressure, or on the Humphrey cycle, which implies that the heat addition is performed at constant volume. Each aspect is described in more detail later in the report.

Figure 1 presents a block graph of different jet propulsion concepts, based on the two primary combustion cycles.

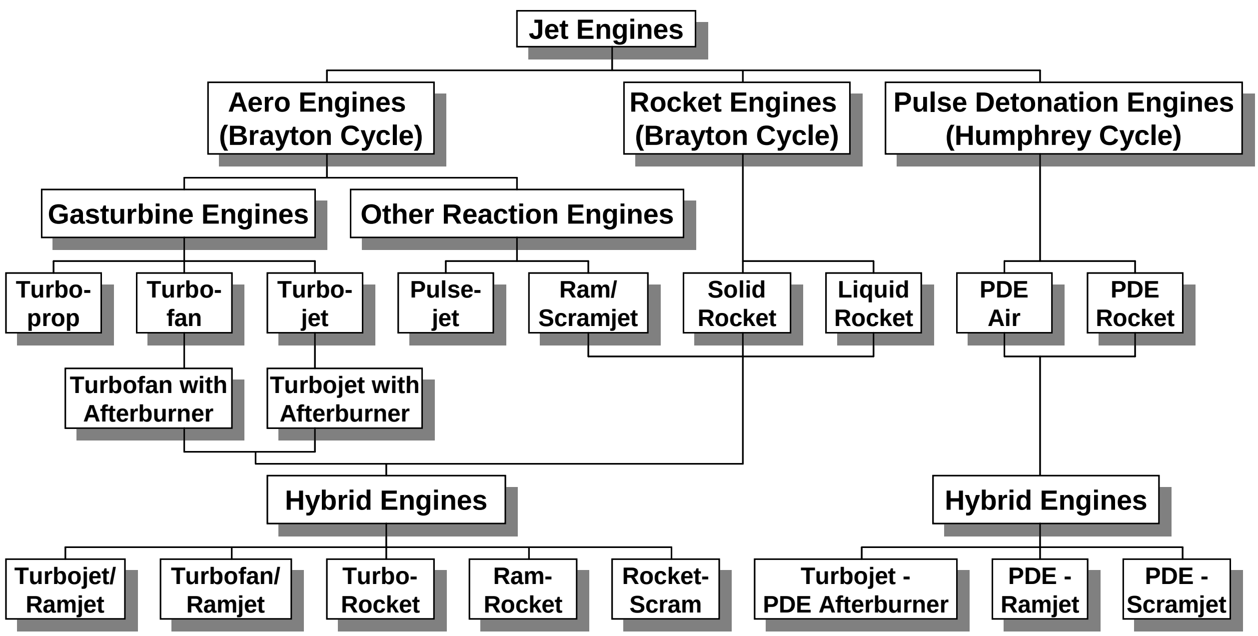


Figure 1. Classes of jet propulsion systems based on Brayton and Humphrey cycles. [the Brayton part fetched from Heitmeir et al., 1996].

Note that Figure 1 does not cover combined concepts with more than three cycles, which are currently studied for space applications. These come in two flavours: Turbine Based Combined Cycle (TBCC) and Rocket Based Combined Cycle (RBCC).

The differences in fuels, their states, burning processes, acceleration means, etc. lead to many combinations of propulsion systems. Probably most of them are not competitive, but many may have interesting properties. An attempt to categorize the many types of jet engines is shown in Figure 1, where the branches based on Brayton have been fetched from Heitmeir *et al.* (1996), while the Humphrey branch is ventured into by the authors, after going through the literature on the subject.

The technical descriptions of the various propulsion systems will be organized mainly according to the blocks in the figure. Depending on available information and on their merits for air vehicle propulsion, some concepts are explained in more detail.

The development of propulsion concepts has been driven mainly by the intention to get maximum thrust performance, reduce manufacturing costs and improve fuel economy. The performance can be expressed as thrust per propulsion system weight. The cost depends on the simplicity of the propulsion system and manufacturing procedure. The fuel economy relies on the fuels and the combustion process. These properties vary very much with the operating conditions, i.e. the altitude and the speed. The pressure and velocity that enter the engine inlet have to be conditioned to fit the engine needs. These are the main drivers for creating new propulsion concepts and inlet/outlet systems.

Most jet propulsion concepts are based on the so-called Brayton cycle, in which the combustion (heat addition) takes place as a constant-pressure process with burning governed by temperature front. Lately considerable interest has been focused on the Humphrey cycle, where the combustion (heat addition) takes place as a constant-volume process with burning governed by pressure front. The interest is based on the fact that this cycle is theoretically more efficient than the Brayton cycle. The fuel-oxidizer mixture has to be loaded and brought to detonation intermittently, and this introduces many problems that have to be resolved before operative propulsion systems for air vehicles can be created.

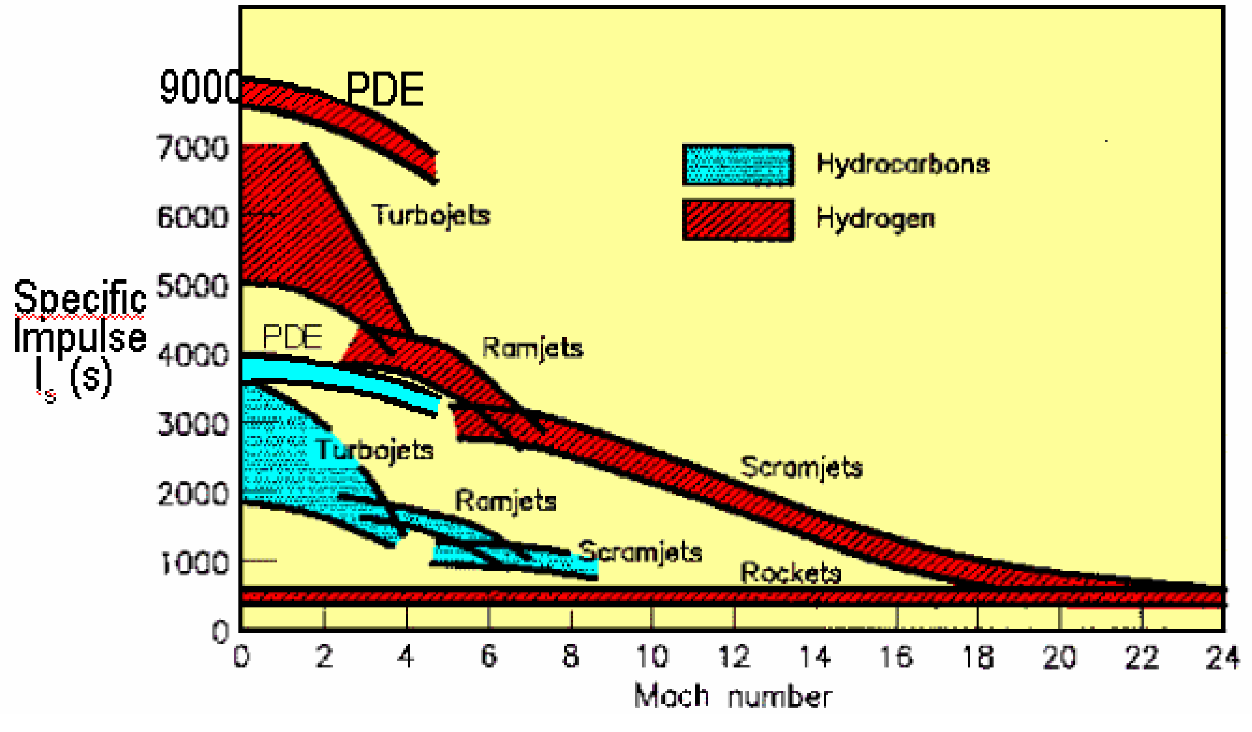


Figure 2. Specific Impulse for Proulsion Systems [from Kuentzmann and Falempin].

To navigate among the different propulsion systems from the air vehicle designer's point of view, Figure 2 may be of some help [Kuentzmann and Falemkin]. It shows the specific impulse, which is a measure of how much impulse (force per time) that can be extracted per unit weight of fuel (for rockets, also including oxidizer), for propulsion systems using either hydrocarbons (commonly used for jet engines) or hydrogen as fuel. The inversion of the specific impulse is the specific fuel consumption.

Not shown in the figure are turboprops and turbofans, and both these have higher specific impulse in the subsonic region than the PDE (pulse detonation) concepts.

Regarding the trends of propulsion choices, the very mature turboprop/turbofan/turbojet technologies remain the prime selections if subsonic and low supersonic speeds are of interest.

The theoretically more efficient PDE concepts are very attractive, mainly due to the simple build-up. These are also of great interest because they have a wide operating speed range, from zero to Mach numbers of about 5. However, many operative issues have to be resolved before this technology can be used for air vehicles.

Extreme speeds have been the focus of space flights, where rockets have been the dominant propulsion source up to now. In these days when the fleet of the US space shuttles is planned to be replaced, more economic ways of creating thrust at very high speeds are being studied. Among the ideas, using the oxygen of the atmosphere plays a great role in the studied propulsion concepts. Combined-cycle concepts are investigated, either as a system of parallel engine cycles, or an integrated one with variable mode operation. The reason for these studies is finding optimum fuel-efficient propulsion system for the whole speed regime. Here supersonic combustion in scramjets is of great interest.

For military flying vehicles it seems that presently an extreme speed is not a primary issue. A Mach number of 2 is sufficient for the near future. This conclusion comes from thoughts emerging in the ongoing studies where invasion defence is not a concern. In the context of European or United Nations joint peace keeping, or joint war deterrent, activities stealth is seen to replace speed as the means for protecting the vehicle.

For civil applications, mostly in the form of small flying vehicles, fuel efficient solutions appear to be most interesting, meaning mainly subsonic speeds.

We may also participate in aerospace work, for which we may extend our interest, for air breathing engines, to speeds up to around Mach 6 on the way toward orbit. The final acceleration to orbital velocity will have to be provided by rocket technology.

2 Propulsion Systems

The general theoretical aspects of different propulsion systems are well described in numerous textbooks. Even newer, not yet fully developed technologies, such as air turbo ramjets, scramjets, and pulse detonation engines, can be found there. The reader is encouraged to study these, guided by the reference list of this report. In this report we focus on the latest developments in the respective technology area.

An overview of air-breathing engines for air vehicles is given in Svensson (2004), where the engine categories, gas turbine components and their functions, and total performance are described. The procedure of computing performance is also presented. Finally, some future concepts are discussed.

Figure 3, based on information found Heitmeir *et al.* (1996) and Tanatsugu (1996), and other sources, presents an overview of possible propulsion systems for flying vehicles, where the upper operational speed for each system is shown. These engines are either based on single- or multi-cycles (e.g. PDE – Ramjet). The well-known turbojet/turbofan technology and the novel concepts PDE and ATR are the more promising for vehicle propulsion. All are based on the Brayton cycle, except the PDE engine which is based on the more efficient Humphrey cycle (see section 1 for further explanation).

We will put an emphasis on describing those propulsion systems that may have the greatest application interest for our studies of military air vehicles. The purpose of relating other systems is mainly completing the overall propulsion picture. The single cycle systems, of which PDE is one, are described first, starting with low-speed ones and progressing with the higher-speeds ones. Hereafter the multi-cycle (or combined cycle) systems, of which ATR is one, are examined.

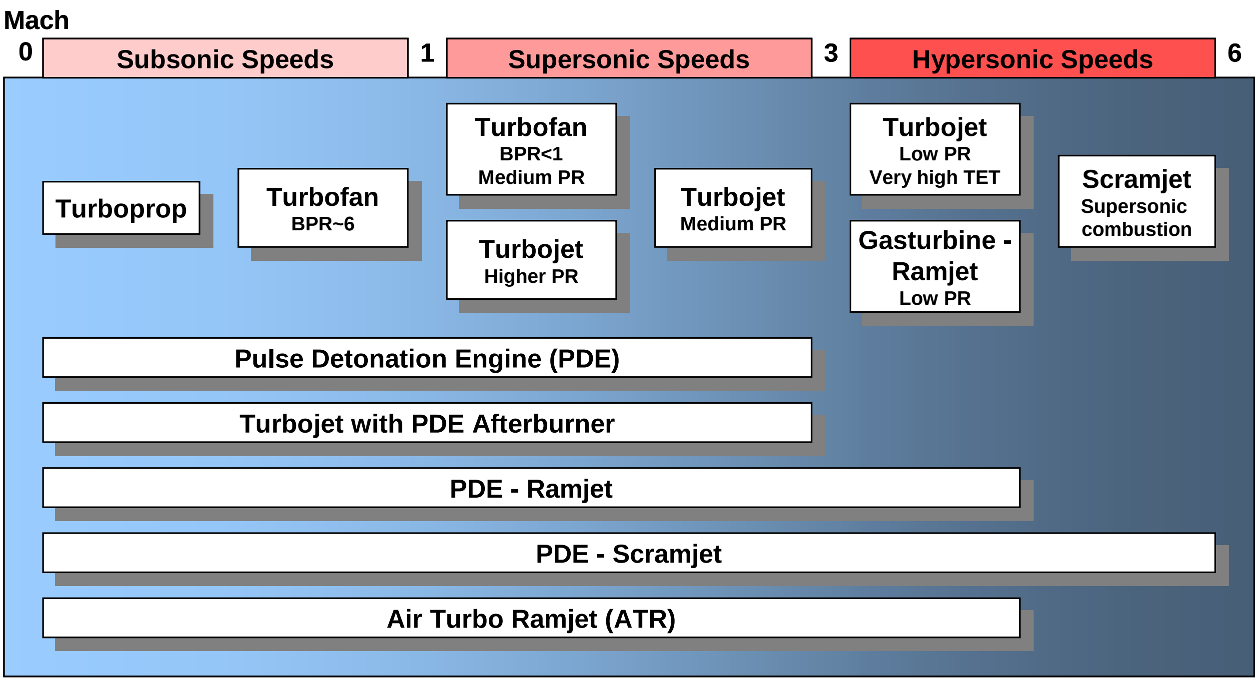


Figure 3. Classes of jet propulsion systems based on Brayton and Humphrey cycles.

2.1 Single Cycle Propulsion Systems

2.1.1 Turboprop

A gas generator that drives a propeller is a turboprop engine. For low speed flight and short field takeoff, the propeller has a performance advantage. When parts of the propeller approache the speed of sound, the compressibility effects become dominant and the propeller loses its aerodynamic efficiency. Most turboprop applications are designed for relatively low vehicle speeds, i.e. Mach numbers up to about 0.4. Higher speed can be attained with propellers designed specially in order to operate with propeller tips at speeds close to the Mach 1. Forward speeds in excess of Mach about 0.7 are then possible and very uneconomical. A specially motored WW2 era Grumman Bearcat obtained a speed of 540 m.p.h., or Mach 0.72.

Turboprop/turboshaft engines have high power-to-weight ratio and high reliability compared to piston engines. They also run on standard jet fuels. A theoretical outline on the subject can be found in Mattingly *et al.* (2002).

We can note that the turboprop engine may be viewed as an unducted turbofan engine, having an extreme bypass ratio. Both the turboprop engine and the turbofan types have qualitatively the same component build-up, so the reader may get an understanding of the turboprop principle by going to the turbofan description. The main difference is that the turboprop engine utilises the power to provide torque rather than exhaust gas momentum. Most turboprop engines direct their exhaust backwards to gain maximum power effect, which results in a total equivalent horse-power measure. The turbofan technology, described in more detail below, thus covers all essential features of the turboprop. Figure 4 shows a typical layout of a turboprop engine, with a two-stage centrifugal compressor and a propeller gear.

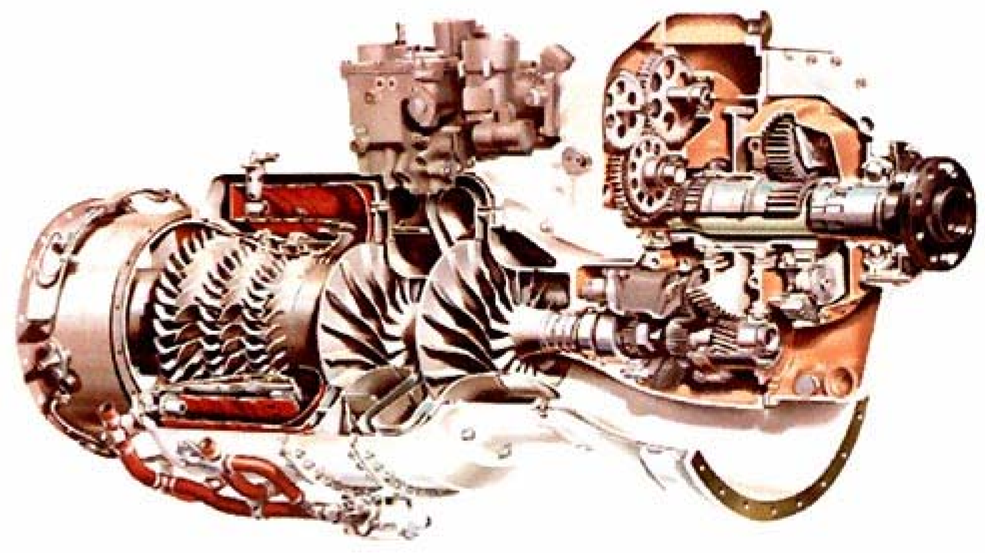


Figure 4. GE engine TPE-331-10T used e.g in the "Predator" UAV.

2.1.2 Turboshaft

A gas generator that drives machinery other than a propeller is usually referred to as a "turboshaft" engine. Typical applications are found in helicopters and military land vehicles. The component build-up of the core engine is similar to that of the turboprop engine. Figure 5 shows a turboshaft engine with a combined axial and radial compressor. Note that the exhaust nozzles are directed outwards, which means that they do not contribute to the forward thrust.

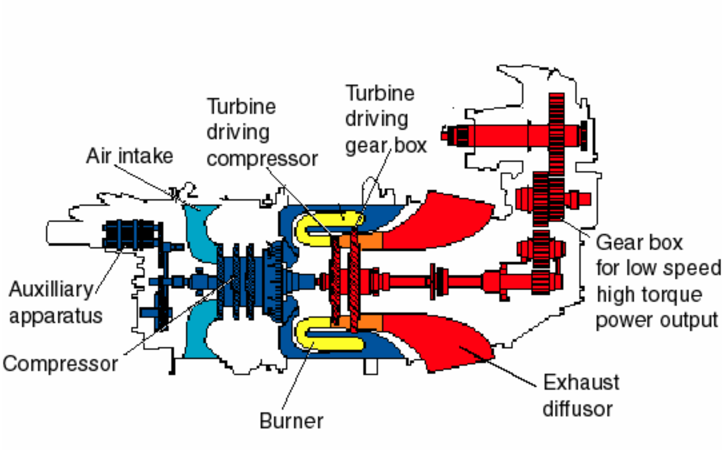


Figure 5. PW Canada PT6B.

The similarity between the turboshaft and turboprop engines, as depicted in Figure 5 and Figure 4, respectively, is striking although the engines are used for different purposes and coming from different manufacturers, suggesting a mature technology.

2.1.3 Turbojet/Turbofan

2.1.3.1 The Principle and Variants

A very good overview of the historical and future turbojet/turbofan developments is given in Johnson (1996), from which information and figures are fetched for the present report.

Turbojet and turbofan can be viewed as two different engine cycles, the reason being the former forms the basis for the latter, see Figure 6. In addition to the core engine, which is the same as a turbojet, the turbofan engine features a secondary nozzle into which part of the inlet flow is bypassed. Theoretically two types of the turbofan type can be considered, which differ in where the afterburning process takes place - after the core engine (Figure 6b) or within the bypass flow (Figure 6c).

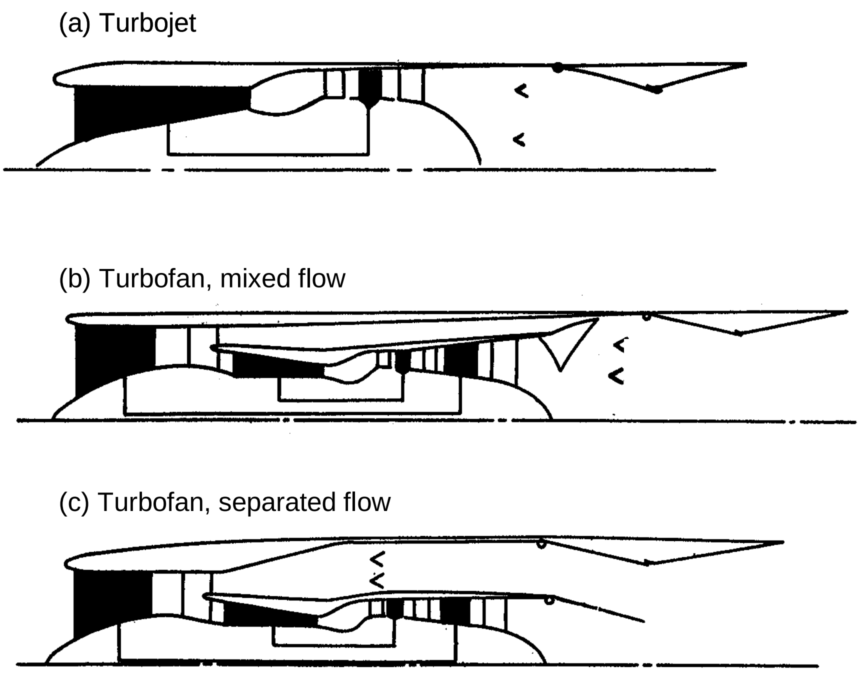


Figure 6. Afterburning turbojet schematic and augmented turbofan schematics [from Johnson, 1996].

The fan assembly is powered by a separate turbine behind the core engine, with an axle going straight through. To enable this extra power need, the core engine is designed for a relatively higher overall pressure ratio. The bypass ratio (BPR), which is defined as the ratio between the bypassed and core mass flows, is determined for the design conditions. As mentioned in Svensson (2004), the main purpose of the fan is improving fuel efficiency by involving external mass flow in the mixing with the core flow. The higher the BPR is, the better fuel economy – for subsonic flight speeds. However, the importance of having a large fan decreases when the flight speed approaches Mach 1, to be almost useless at supersonic speeds. Therefore civil airliners, which operate at about Mach 0.8, tend to have BPR levels of 5 or more, whereas military aircraft with supersonic capabilities have BPR in the region of 0.3-1.0.

Of the two turbofan cycles, the mixed flow concept has been the predominant choice for the last forty years, as well as in current and future propulsion systems under study. There are several reasons for this. One is maximizing the specific thrust. Another is that mixed flows offers significantly better dry subsonic SFC. Finally, only a single nozzle needs to be considered.

The three basic turbojet/turbofan concepts according to Figure 6 have different inherent properties, which determine their applicability depending on the requirements of the actual air vehicle. Table 1, where TF means turbofan, lists a number of viewpoints, of which some are described in more detail in the following text.

Table 1. Cycle Suitability for Mixed Mission Application [from Johnson,1996].

Parameter	Turbojet	Mixed flow TF	Separated flow TF
Supersonic SFC	1	2	3
Subsonic SFC	3	1	2
Specific thrust (sizing)	1	2	3
High Mach flow potential	1	3	2
Part power spillage drag	(none adequately address this problem)		
Thrust/weight at sizing point	1	2	3
Simplicity	1	2	3
Development and acquisition cost	1	2	3

Ranking 1 = best, 3 = worst.

2.1.3.2 Engine Selection

The turbojet/turbofan technology is very mature, with well-founded theory, established design procedures, experience with solutions ("know-how"), material usage, control-laws, etc. It is, however, still evolving, stimulated by research programs such as the US programs IHPTET and VAATE, see section 2.1.3.5.

The purpose of the engine selection is to provide an aircraft with enough thrust to fulfil a prescribed flight mission in all its elements, and do it with minimum fuel usage. Airframe factors must be regarded, where the total weight and the wing geometry are the most important. Low weight relative to the thrust enhances acceleration performance and larger wings increases the turning performance. Regarding the engine, the thrust that can be extracted per captured airflow (specific thrust) needs to be high for good performance, and at the same time, the fuel flow relative to this thrust (specific fuel consumption, SFC) needs to be low for the fuel economy. These properties are influenced by a number of engine parameters, such as compressor, fan and turbine pressure ratios, and combustion temperatures (in the core engine and the afterburner). Establishing an optimum set of all these parameters for a specific aircraft, satisfying its mission requirements, is the purpose of the engine design process.

A very good description of the engine design process is found in Mattingly *et al.* (2002). The process consists of a number of distinct phases, listed in Table 2. Each phase has a special focus, an appropriate study material/methodology, and an objective to prepare for the next phase. These design phases are described below.

Table 2. Preliminary propulsion design sequence. Parameters are described in the text.

Design phase	Focus	Studies	Objective
Constraint analysis	Forces on aircraft: lift, drag, thrust, total weight	Approximate aircraft drag polar, thrust characteristics, and weights	T_{SL}/W_{TO} vs W_{TO}/S due to constraints
Mission analysis	Determine fuel and thrust level through "flying" the mission	Approximate aircraft drag polar, thrust characteristics, and updated weights	Determine W_{TO} and T_{SL}
Parametric cycle analysis	Determine set of reference point engines (max throttle)	Ranges: BPR, compressor pressure ratio, fan pressure ratio, T_{4max}, T_{7max}, sensitivity analysis	Attainable F/mdot and SFC
Performance cycle analysis	Determine off-design characteristics (reduced throttle), optimizing engine configuration	Critical flight conditions and mission fuel consumption	Baseline engine: BPR, compressor pressure ratio, fan pressure ratio, T_{4max}, T_{7max}

Constraint Analysis

The design process begins by outlining an aircraft that is able to satisfy the formulated requirements of a mission. At this stage the absolute values for the static sea-level thrust (T_{SL}), the aircraft wing area (S), and the aircraft takeoff weight (W_{TO}) are unknown. Instead, the various aspects of the mission requirements are evaluated by using simple equations governed by ratios: T_{SL}/W_{TO} (thrust loading) and W_{TO}/S (wing loading). An example of a mission with various requirements for an air-to-air fighter (AAF) aircraft is shown in Figure 7.

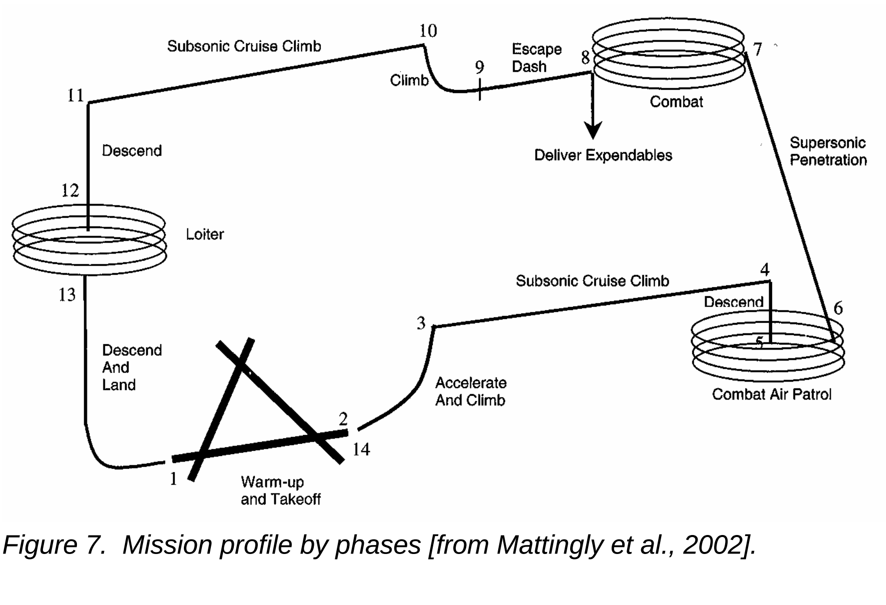


Figure 7. Mission profile by phases [from Mattingly et al., 2002].

The set of equations are derived from simple models for the aircraft aerodynamics and the installed propulsion characteristics. By plotting the lowest value of T_{SL}/W_{TO} that fulfils the mission requirement in view against W_{TO}/S, a border line is created, along which this requirement is exactly satisfied. Below this line no solution is possible, while above it the solutions are exceeding the requirement. Other border lines regarding other critical mission requirements can also be plotted in the same diagram, see for example Figure 8. The diagram now contains a set of border lines, which represent the different mission aspects. The overall solution that satisfies all mission requirements is of course a point that lies above all the border lines, like the point denoted by "AAF" in the figure, where the throttle ratio (TR) is at its maximum value.

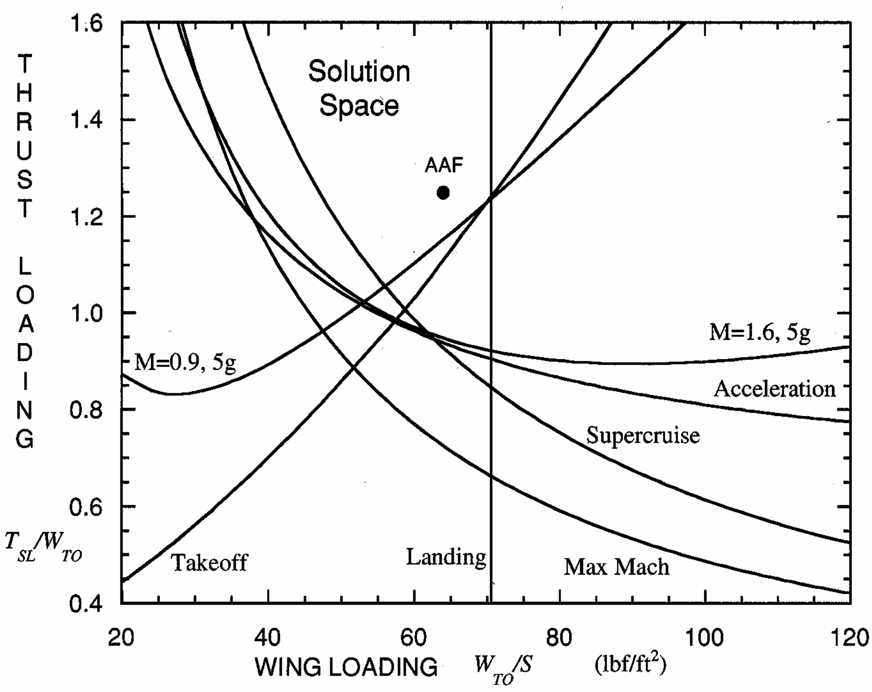


Figure 8. The complete preliminary AAF constraint diagram (TR=1.07) [from Mattingly et al., 2002].

Mission Analysis

This analysis aims to determine the scale of the aircraft and its engine. Here the focus is fuel accommodation and consumption. The set of equations is now completed with engine consumption characteristics. The W_{TO} is determined by "flying" through the required mission profile, i.e. by summing up the empty weight and fuel consumptions during the phases. The T_{SL} and the wing area are then given by the appropriate thrust loading and wing loading, respectively.

Parametric Cycle Analysis

The mission requirements usually place demands on both high specific thrust and low specific fuel consumption. This depends on what mission element is viewed. Therefore a number of reference point engine candidates (at maximum, uninstalled throttle level) need to be evaluated, i.e. different engine cycles need to be studied. Here performance critical parameters are viewed, such as the T_{4max} (max. turbine inlet temperature), T_{7max} (max. nozzle temperature), α (BPR), π_c (compressor pressure ratio), and π_f (fan pressure ratio). A computer program, containing the one-dimensional theoretical foundation, is needed for the calculations. Figure 9 shows how α and π_c influence the specific thrust and the specific fuel consumption for mixed flow turbofans, without and with afterburning, respectively. Mattingly *et al.* (2002) do not state the Mach number condition for the figure, but probably it is a high subsonic speed.

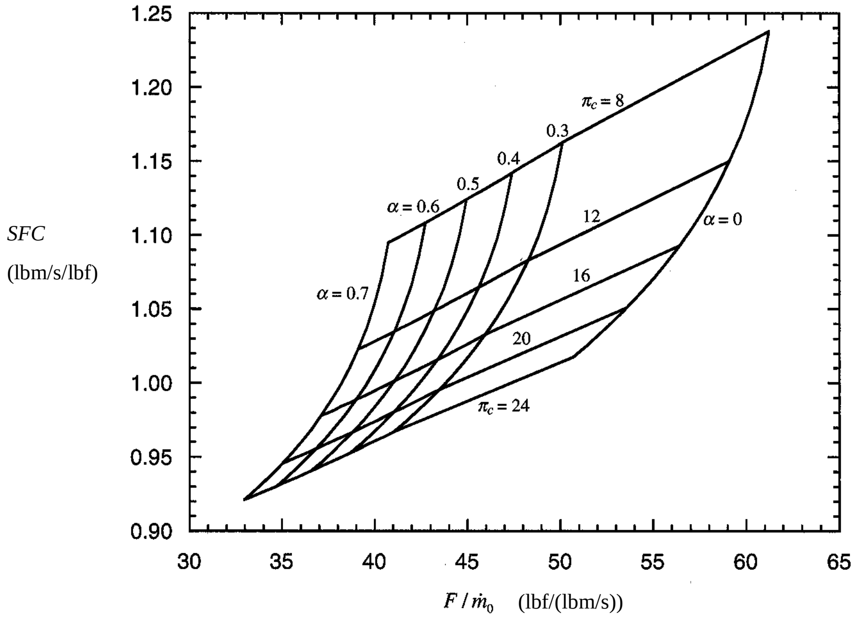


Figure 9. Parametric performance of mixed flow turbofans (no AB) [from Mattingly et al., 2002].

In the selection of reference point parameters for each critical flight condition, it is important to have some knowledge of realistic operation conditions of candidate engines, at full and at partial throttle levels. For example, if the reference point condition in view is high subsonic speed at very low altitude, then there is no point in choosing an extreme π_C value or this case. The reason is that the T_4 at the turbine (driving the compressor) is exceeding the maximum limit at high speed off-design conditions. This is illustrated by Figure 10, showing the useable maximum π_C for different Mach numbers (M_0) and altitudes. Here the critical M_0 is just below 0.6 at sea-level. By lowering the π_C a little, which is a result of burning at lower T4, a better reference point case is created. If, instead, the reference point altitude had been at 40 kft, then the highest π_C value would present no problems for Mach numbers up to over 1.4.

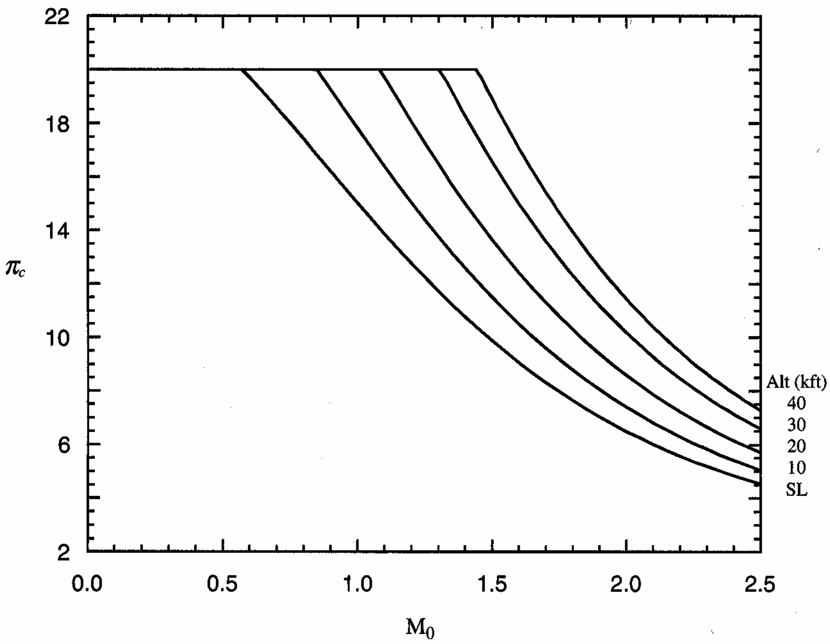


Figure 10. Compressor pressure ratio (π_c) performance characteristics for a low bypass ratio mixed flow turbofan with TR=1.065 [from Mattingly et al., 2002].

Performance Cycle Analysis

Having established the reference point engines that would be useful for each mission point of view, it is time to select an optimum compromise of a single propulsion system. The critical mission element(s) will decide size at maximum throttle ratio. All other elements will involve off-design operation, with the risk of reduced efficiency. This has to be evaluated (and maybe corrected through redesign), for which maps for the compressor, the fan and the turbine need to be created. A special computer tool is used for creating these. A map example is shown in Figure 11 for a typical compressor, where the “corrected mass flow” is the flow transformed to the equivalent sea-level condition and “$\%N_{c2}$” is engine fan speed.

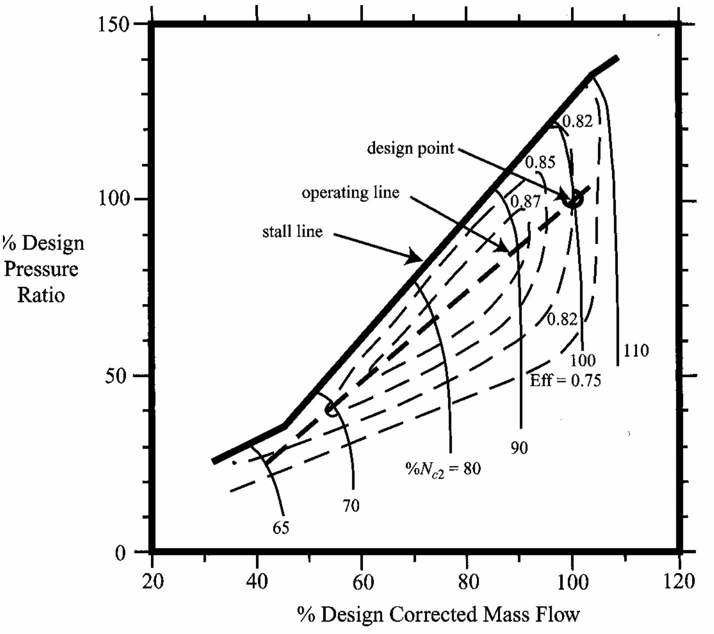


Figure 11. Typical compressor map [from Mattingly et al., 2002].

In the figure the design condition is placed at the 100 percent positions of both axes, which correspond to the maximum mass flow that can be achieved with compressor efficiency of 0.82. At part load conditions, higher efficiency values can be found. Note that this characteristic depends on the compressor design. This reasoning leads to construction of an optimum operation line, as shown in the figure. This example, where the most favoured conditions are in the mass flow region of 60 to 85 percent, can be seen as a good off-design compressor characteristic for range and loiter flying.

Is should be noted that the compressor has operational limits due to special flow phenomena: stall (surge) and choke [Svensson, 2004]. The stall condition is indicated in Figure 11 as a line. The choke condition occurs when the corrected flow cannot be increased past a certain limit, determined by the sonic flow condition at a critical throat location.

Also, a combustor map has to be established, where the most important parameters to the engine overall performance are the pressure loss through the combustor and the combustion efficiency. Having these maps at hand for the baseline engine, each of the critical flight conditions can be evaluated at the appropriate throttle positions and with the airframe properties involved. A number of additional parameters, such as M_6 and M_{16} (Mach numbers in core exhaust flow and in bypass flow, respectively, just before the afterburner flame holders), can be adjusted in order to improve both on-design and off-design conditions.

Sizing for Installed Performance

When all the relevant characteristics of the selected engine have been established, the installation effects need to be evaluated. For this purpose it is necessary to choose an inlet and outlet system and perform computations for these and other installation effects, before the engine can be scaled to its appropriate size. Here the preliminary design process is finished. What follows hereupon is a detail design process.

2.1.3.3 Potentials and Problems

Turbojet Cycle

The afterburning turbojet cycle is simple in its layout, as shown in Figure 6a. It is capable of producing high specific thrust levels in non-afterburning mode. The afterburner was introduced as a means of producing high supersonic thrust, as a smaller and a more weight-saving alternate to scaling up the basic engine, having the same thrust. This was in fact the first variable cycle feature used in a jet engine. According to Johnson (1996), the General Electric Company paved way for this type of afterburning engine by creating, with the J47 as the first afterburning turbojet (late 40s), and the J79 as the first Mach 2 engine. The company then developed the Mach 3 class augmented turbojets J93 and GE4.

A problem, associated with simple single rotor systems of turbojets, is that the compressor and the turbine are linked together (through an axle) and these have to cope with large internal airflow variations, from takeoff to high supersonic speeds. Here a major advancement was introduced in the J79: a variable stator in the front of the compressor, which allowed higher flow rates at higher pressure ratio than before. Also, compressor stalls could be avoided. The variable stator idea was further refined in the J93 and GE4 compressors: a variable stator in the rear part.

An old study, Gabriel *et al*.(1953), about design and operating variables for afterburning turbojets for aircraft in high-speed missions, indicated the prime importance of the thrust-to-weight issue of the power plant. In the case of afterburner or larger engine at high speed, it is a question of trading between engine weight and consumed fuel. The other design variables, such as compressor pressure ratio, turbine inlet temperature, afterburner temperature, and air-handling capability, interact in a complicated manner.

The turbine entry temperature is the main factor for improving the engine overall efficiency and the maximum thrust level, see Figure 12. This requires that the afterburner and nozzle system can utilize the extra performance, i.e. new materials and the cooling techniques must be developed.

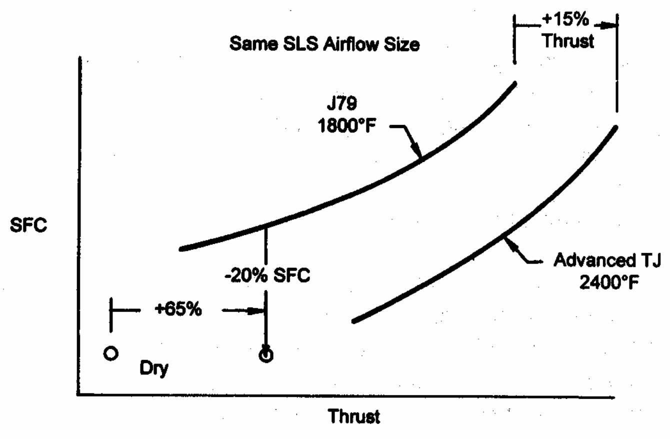


Figure 12. Advanced turbojet performance potentials at M2.0/50000 ft [from Johnson, 1996]

As with all jet engines with turbo-compressors, the turbojet has a main deficiency: the problem of maintaining high flows at high supersonic speeds, where the turbine temperature is a limiting factor. This has been mentioned in conjunction with Figure 10 of section 2.1.3.2. Usually high-speed flight is associated with compressor pressure ratios (PR) lower than the maximum design value, i.e. off-design conditions, and therefore the useable flight speed is higher than the one indicated by the breakpoint in the figure for the appropriate altitude. There are a number of ways of extending the off-design potential, and these are shown in Figure 13. In order to increase the operational Mach limit by lowering the PR level is easily understood when viewing Figure 10. Having variable rear stator and raising the design compressor inlet temperature (T_2), the off-design flow potential at high Mach numbers is increased considerably, see Figure 13b. The variable stator can also be used in conjunction with an increase in the compressor rotational speed in the tailoring to match the maximum demands of a supersonic inlet system.

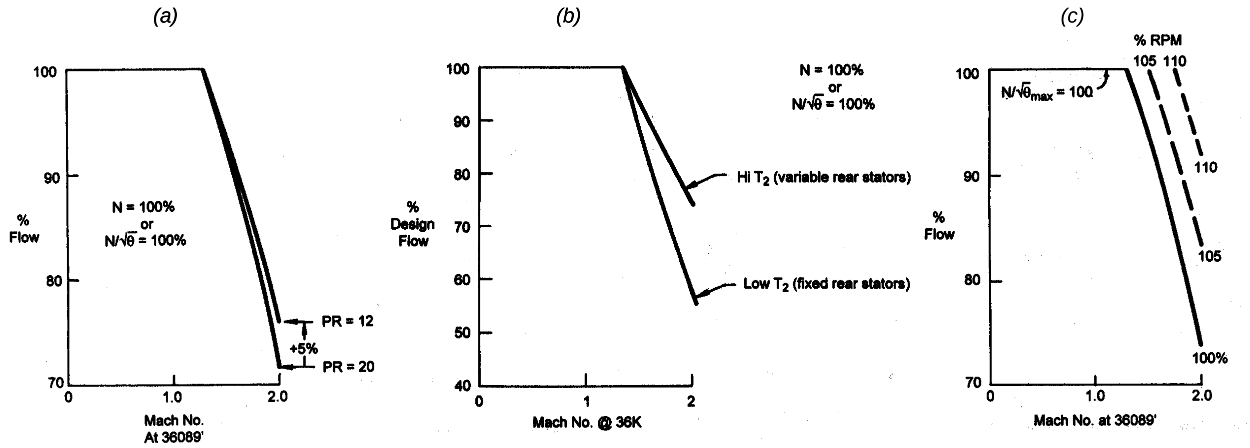


Figure 13. Turbojet flow trends; effects on off-design flow potential: design PR (a), rear variable stator (b), and increased RPM (c) [from Johnson,1996].

In an engine selection for an aircraft to perform high supersonic flight in the regime of Mach 1.5-3.5, the turbojet cycle remains the best choice over all other candidate systems [Johnson, 1996]. Probably this is of interest only in specialized missions, dominated by supersonic performance.

Turbofan Cycle

In mixed missions, there are often requirements for high thrust when needed in certain mission elements, and for good fuel economy in other elements. For such applications the afterburning turbofan is a more competitive propulsion system. This trend started during the 1960s and early 1970s, primarily to improve the subsonic performance. Newer technology, where higher turbine temperatures were involved, allowed a raised overall pressure ratio in excess of 20, and thus achieved high thrust/weight potential and enough power left to drive the bypass system [Johnson, 1996]. The bypass ratio can be used to trade between good subsonic fuel economy and good supersonic thrust characteristics. Typically, the bypass ratio has been in the range of 0.3 to 1.0 for later mixed mission fighter and attack aircraft. The process for establishing mission optimized bypass ratio and other cycle parameter is described in section 2.1.3.2.

As mentioned initially in section 2.1.3.1, there are two main types of turbofans: the mixed flow engine, and the separate flow duct burning turbofan. Of these the former is almost exclusively the preferred type in current aircraft and in projects being studied. There are several reasons for this fact [Johnson, 1996]: one is better specific thrust, since all air from the core and bypass flow can be burned in a single afterburner. Another reason is a significant subsonic SFC improvement potential. Also, the mixed flow principle involves a single nozzle arrangement (lighter system).

Regarding the operation properties of the mixed flow engine, there are several potential problems [Johnson, 1996]. First, due to the matching requirements in the exhaust system of the static pressure of the cold-duct and hot-core flows, the maximum flow capability at high supersonic speed will fall below that of the pure afterburning turbojet, see Figure 14. The matching of flow conditions also changes with the effective bypass ratio as the speed goes up supersonically, as shown in Figure 15. The reason is that the bypass flow increases more rapidly than the core flow, which is basically due to aerodynamic characteristics of the fan and the compressor. Finally, there are limits in the form of core rotational speed, compressor pressure discharge, and turbine inlet temperature, which lead to inability to maintain static pressure balance at full fan speed [Johnson, 1996].

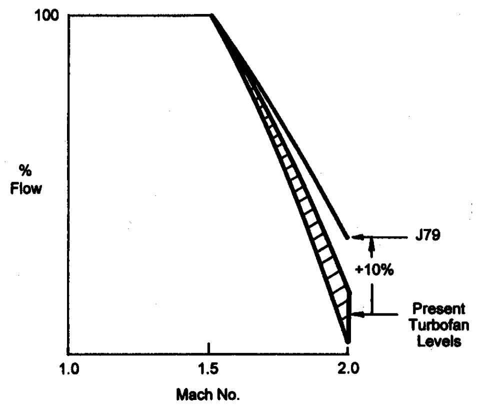


Figure 14. Flow – Mach number comparisons: turbojets – turbofan [from Johnson, 1996].

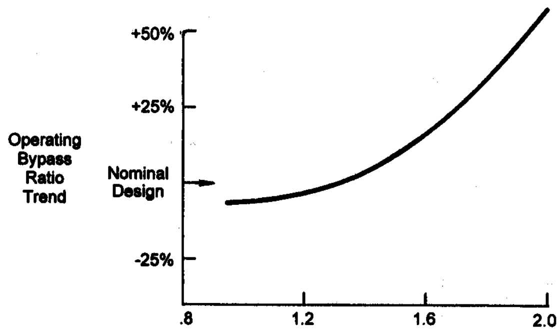


Figure 15. Typical operating bypass ratio increase with flight Mach number [from Johnson, 1996].

Turbofans, when compared to turbojets in the same thrust category, tend to have larger inlet and outlet areas, as determined by flow needs at extreme conditions. Also, the need of high turbine temperature to improve supersonic performance leads to difficulty in throttling back without drastically reducing the airflow. This means higher potential drag problems due to larger stream tube variations and flow mismatch at part thrust. An important parameter for both the inlet and the outlet is the relation between the actual stream tube area at part thrust (A_0 or $A_{g.1}$, respectively) and the maximum area (A_c or A_{max}, respectively), see Figure 16. In the inlet case, the air that lies between these two areas is forced to go outside of the cowl, causing the so-called spillage drag. Nozzles for supersonic aircraft usually have adjustable exit geometries, i.e. the area can be set to an appropriate value. However, while adapting the area the external shape can cause a drag increase, the so-called boat-tail drag, due to some degree of flow separation. The effect of the drag increments for typical inlets and nozzles, respectively, due to the stream tube area ratio is shown in Figure 16. Note the points denoted by “conventional cycle” refer to the mixed flow turbofan cycle, while VCE means variable cycle engine (described in section 2.1.3.4).

Figure 17 shows typical trends expressed as specific fuel consumption due to inlet and outlet losses as functions of the non-afterburning thrust setting, where the minimum characteristics are the results of design considerations.

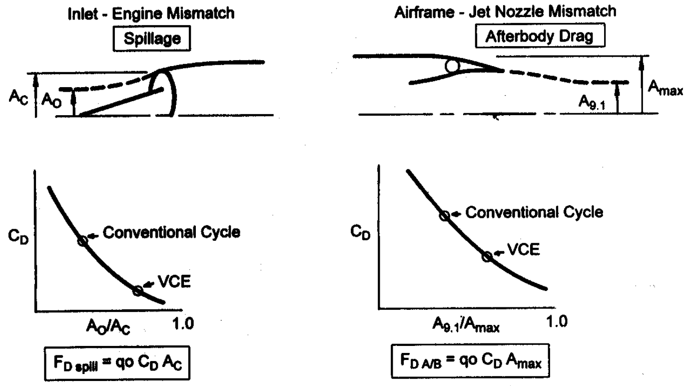


Figure 16. Part power installation losses [from Johnson, 1996].

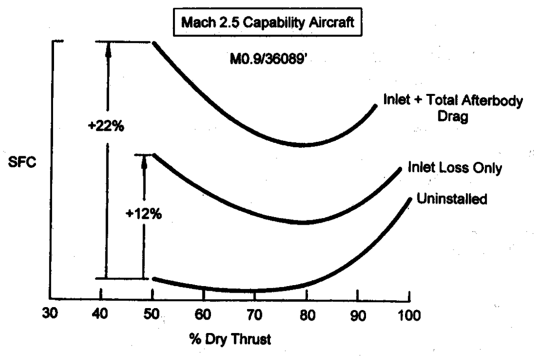


Figure 17. Installation losses for mixed flow turbofan [from Johnson, 1996].

2.1.3.4 Variable Cycle Engines (VCE)

Finding an engine cycle that operates optimally in all aspects of a mixed mission is impossible. Traditionally, it has been a question of choosing a best compromise, where the cycle parameters have been selected carefully to satisfy the most important performance aspects of the mission. Roughly it is a tradeoff between good supersonic performance (turbojet property) and low fuel consumption (possible with turbofans). This process is described in section 2.1.3.2.

An obvious way to avoid compromising is to switch between different engine cycles, i.e. to operate at the most optimal cycle at each mission phase. The afterburner is in fact an early example of this. Instead of having separate engine components, where each is brought into use by directing flows (and thereby leaving temporarily the other components as dead mass), variable components can be used and be shared by the different cycles. The latter principle promises optimum performance at low weight.

By using VCE, the problems with increased drag levels at part power for turbofans due to variability of the stream tube, can be solved, see Figure 16. The idea is to vary the thrust without changing the flow, i.e. to preserve the optimum flow design conditions for the inlets and outlets.

Variable Cycle Concepts

Variable cycle concepts have been studied and developed during the last 30 years by the General Electric Company in conjunction with the US Air Force, Navy, and NASA [Johnson, 1996]. Many ideas have been pursued in the search of a cost-effective concept that combines the attributes of turbojets, i.e. high specific thrust (for given air flow), and those of turbofans, i.e. low specific fuel consumption (for given thrust).

Also sought were concepts that allowed reduced drag increments associated with part power operation. The main idea to keep this drag problem at a low level is adapting the incoming airflow to a level that satisfies the design conditions for the inlets, i.e. letting the engine swallow both used and unused air by varying the effective bypass ratio.

Technically, the idea can be interpreted as modulating the engine thrust while keeping the airflow at a constant level (at design flow conditions). Having two spools with their fan/compressor-turbine assemblies, it may be possible create mixed flow or separated flow engines with high supersonic and high subsonic performance at maximum operating conditions, and this implies that these spools are used optimally together with the set of variable compressor and turbine vanes. In order to regulate the bypass flow for the optimum part thrust requirements, an additional controlling device is needed. When studying the concept descriptions in Johnson (1996), the controlling means seem to fall into three main groups: variable exhaust nozzle systems, variable area systems, and additional fan-turbine systems. The parameters associated with each of the systems are varied independently of the basic engine operation.

In the following subsections, a number of basic ideas of switching between engine operation modes (turbojet and turbofan) are presented: the VAPCOM, the flex cycle, the TACE, and the MOBY. The MOBY represents a solution for a separated flow engine, with very good part thrust characteristics. The last subsection discusses solutions for the mixed flow engine with good part thrust characteristics. This development path finally led to a flyable hardware engines, where the GE f120 engine marked a milestone.

Variable Pumping Compressor (VAPCOM)

This concept was conceived around 1960 by the Aero Propulsion Laboratory at U.S. Air Force Wright Field. It consists of an afterburning turbojet (core) with a parallel duct, where a fan is placed well in front of the core and duct entries, and an extra turbine driving the fan is placed in the core engine. Figure 18 shows the arrangement. By having doors at the nozzle and regulating them, a variable bypass ratio is achieved. For maximum power and supersonic cruise operation, the turbojet mode is set by closing the doors. With these open, the engine operates as a separated flow turbofan. To operate satisfactorily, the turbines have to have variable geometry. The idea was discarded because of performance losses associated with the core flow modulation, which negated much of the operating bypass benefits.

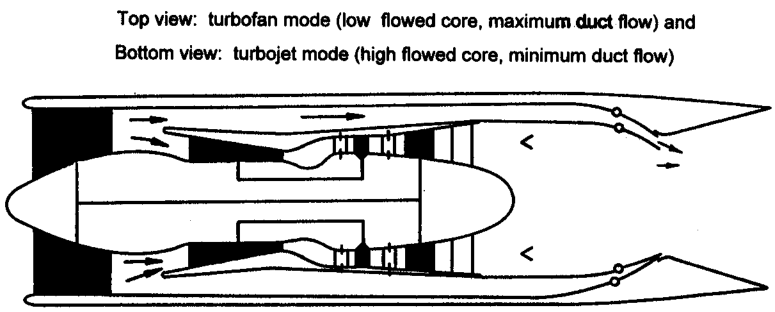


Figure 18. Variable pumping compressor (VAPCOM) [from Johnson, 1996]. Note that the two views, representing two modes of operation, are separated by the symmetry axis.

Flex Cycle

The flex cycle principle, conceived by GE in the 1960s, is based on a concentric arrangement of two engines: a turbojet as a core, surrounded by a fan and turbine of an outer engine. The core engine is always operating, whereas the bypass duct of the outer engine has two operation modes: no burning or burning. The mode effects of the whole engine concept are turbofan and turbojet (in fact two in parallel). In high supersonic operation the core engine is slowed down to avoid high compressor discharge temperatures, leaving the outer engine in full speed. In the dry mode (the duct burner off), the workload is on the core engine, but the fan assembly still requires high energy to keep it working properly. In this mode the problems with the flex cycle arose: radical swings in turbine energy, corrected speed, and flow function [Johnson, 1996]. This has reduced the apparent subsonic advantage of the system. Neither did its supersonic specific thrust prove to be competitive enough.

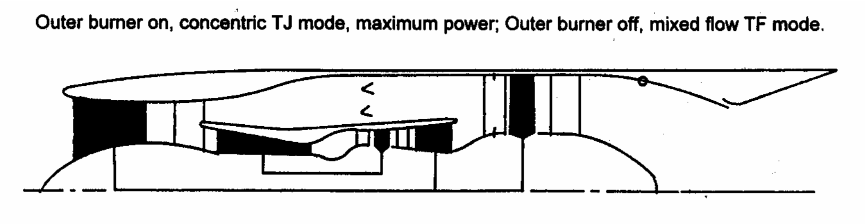


Figure 19. Flex cycle schematic [from Johnson, 1996].

Turbo Augmented Cycle Design (TACE)

The TACE principle is based on a marriage of turbofan and turbojet engines, in series as shown in Figure 20, with flow switching for the different operation modes. The forward part consists of a complete turbofan for subsonic usage. In this mode the bypass flow is led past (outside) of the rear turbojet engine, where also the exhaust flow from the turbofan is led and mixed with the bypass flow. In the supersonic mode, the turbofan bypass stream is fed into the rear engine. Since this stream is being used for burning, the turbofan is operating as a turbojet, in all two such engines are working. This engine principle leads to a complicated and heavy system. The supersonic performance may be good, whereas the subsonic properties suffer from the dead mass of the unused rear engine. To conclude, the main drawbacks are prohibitive overall weight, complexity and cost.

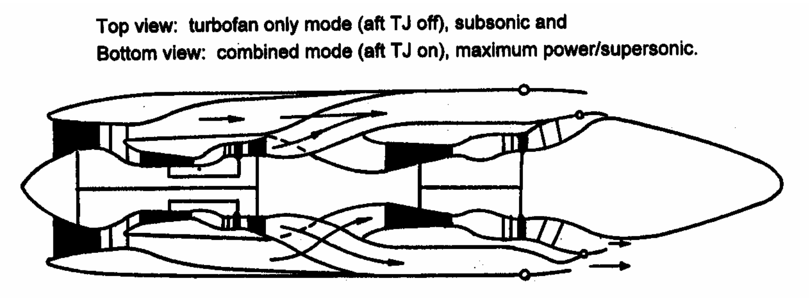


Figure 20. Turbo augmented cycle engine schematic [from Johnson, 1996]. Note that the two views, representing two modes of operation, are separated by the symmetry axis.

Modulating Bypass VCE Concept (MOBY)

This concept, having a modulating bypass system, was conceived by the General Electric Company in 1973 as a response to an US Air Force request to address problems of throttle-dependent installation losses [Johnson, 1996]. The layout arrangement, as shown in Figure 21, consists of a separated flow turbofan (compare Figure 6c), surrounded by an extra fan/bypass system. It has three spools, three variable area stators, three variable area nozzles, and two variable stator fan systems. The compressor in the core engine has only a variable inlet guide vane.

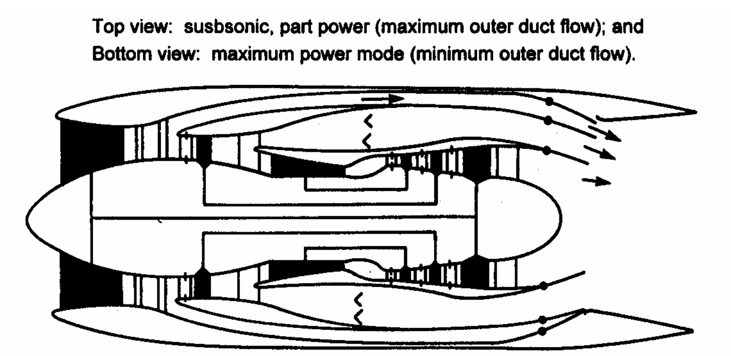


Figure 21. MOBY schematic, three spool double bypass [from Johnson, 1996]. Note that the two views, representing two modes of operation, are separated by the symmetry axis.

In the mode for high power and supersonic flight, all of the front fan air is passed directly to the second fan of the separated flow turbofan, where the fuel is burned in both the core and the bypass flow. For the part power conditions, the front fan speed is set to match the inlet flow requirements, which are set to minimize the part-thrust drag problems. The idea is then to keep this favourable flow condition, while the thrust is modulated. This is done by reducing the speeds of both core and the intermediate spools, while going down with the fuel flow. The matched inlet flow can be maintained down to 50 percent dry thrust without any sacrifice in installed SFC [Johnson, 1996]. However, the overall complexity, both in the hardware and in the handling of the many operation parameters, detracts the concept to be used practically. In addition, the resultant specific thrust potential not as high as that of competitive mixed flow fan systems.

Further Developments for the Mixed Flow Engine

The four concept ideas described in the preceding sections did not appear to demonstrate an overall improvement large enough to replace the mixed flow turbofan as the best mixed mission propulsion system [Johnson, 1996]. This conclusion could be drawn when also complexity, risk, and cost were weighed in.

The General Electric Company invented in 1974-1975 two new types of VCE that retained the best features of the mixed flow cycle, such as maximum augmented thrust and good subsonic SFCs: single bypass VCE and double bypass VCE, respectively. These two ideas are based on controlling the duct flow by changing the area where the flow joins with the core flow. In this way the complexity of having an extra bypass-fan-turbine assembly was avoided. The principle is called variable area bypass injection (VABI). However, a VABI system is not as flexible as the MOBY system, due to the requirements on the static pressure of the duct and core flows in the mixing region.

The single bypass VCE has two independently variable nozzle area systems that allow better control of compressor speed and turbine inlet temperature for fine tuning of the maximum dry thrust and part power fan airflow scheduling. This implies that the speeds of the fan and the core can be operated independently. For supersonic flight the fan speed is kept at higher levels to maximize thrust and improve supersonic inlet matching. Figure 22 shows an initial version of the engine principle.

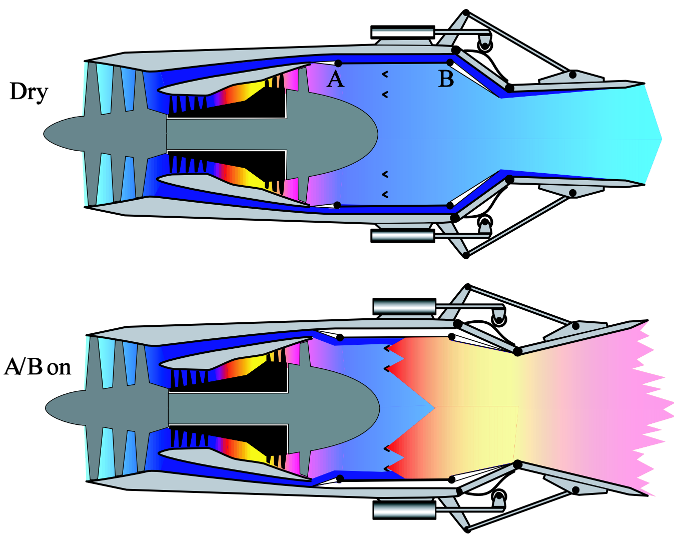


Figure 22. Principle of initial Dual Cycle Single Bypass VCE (1974-1976) [interpretation from Johnson (1996)].

In the upper part of the figure it is seen how the dry configuration conducts the bypass air through the full length of the engine for good subsonic fuel economy. The valves at A are closed, whereas those at B are modulated in conjunction with the fan speed for the part thrust variation at high flow conditions. The lower part of the figure shows the operation condition for maximum thrust performance at high supersonic speed. Here the valves are open at A and closed at B in order to lead the bypass air into the afterburner, where the air is joined with the exhaust gases from the core engine for further burning. Thus all oxygen of the incoming air can be used.

The double bypass VCE is similar as the single bypass variant but incorporates an additional flow modulating capability, where air flow at some mid station of the multistage fan is lead past an extra duct to a yet another variable area nozzle system, thus three in all for the engine. This feature allows higher bypass ratios to be attained, while also extending the constant airflow - thrust modulation potentials down to 50 percent of maximum dry thrust [Johnson, 1996]. The improvement of the SFC is also considerable.

Figure 23 and Figure 24 illustrate the airflow gains that can be attained by using these VCE concepts, at maximum thrust and part power conditions, respectively.

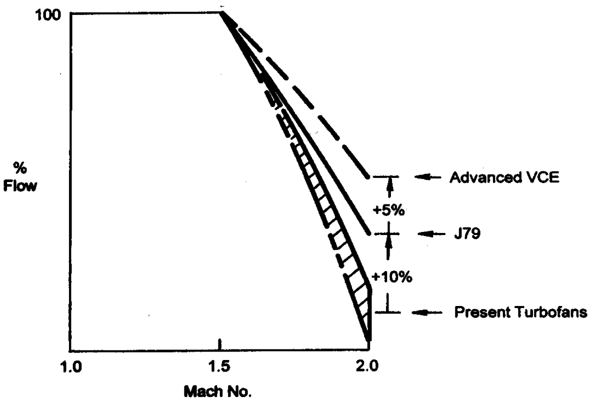


Figure 23. Improved maximum power airflow potentials for advanced VCEs [from Johnson, 1996].

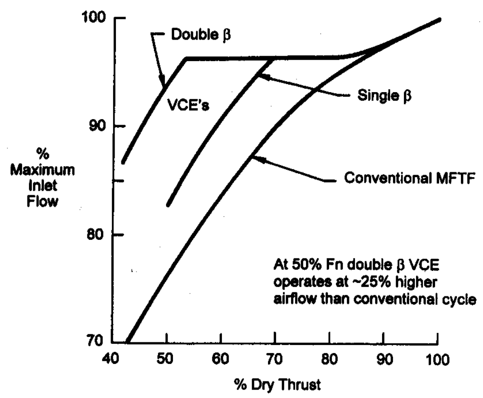


Figure 24. VCE part power flow matching potentials; same cycle parameters [from Johnson, 1996].

A summary of the main characteristics of the single and double bypass VCE concepts is given in Table 3.

Table 3. Current VCE Concepts (1976 Reference) [from Johnson, 1996].

Engine configuration:
- Single or double bypass
- Only two rotors
- Fan flow modulated with IGV change
- Static pressure balance control system, simplified exhaust
- One variable jet nozzle

Results:
- Retains best features of mixed flow TF
- Improves supersonic flow potentials
- Eliminates part power inlet drag
- Improves aft end closure drag
- Produces low noise thrust for second generation SST aircraft

Johnson (1996) presents the background and overview for the development of variable mode engines. It describes a large number of VCE designs, several of them based on the mentioned single and double concepts. Here we are only listing them, but the source describes them in detail and presents principle sketches:

1. VAPCOM – The original Variable cycle (1960-1965)
2. Composite Cycle: Turbojet and turbofan modes (1965-1970)
3. Combined Turbofan and turbojet revisited (1973-1974)
4. Series/Parallel Mode VCE Concepts (1973-1975)
5. Three Spool Modulating Bypass Ratio VCE-MOBY (1973-1974)
6. Initial Dual Cycle Single Bypass VCE (1974-1976)
7. Initial Double Bypass VCE Configurations with Three Stream Exhaust (1974-1976)
8. Single and Double Bypass VCE Simplification (1976-1978)
9. Core Fan Architecture for Additional Simplicity (1975-1981)

Some VCE demonstrators were developed, after gaining support from General Electric management and the US government technology centers. They were based on the YJ101 engine, initially produced for the YF17 entry in the lightweight fighter competition in the 1970s. These demonstrator concepts, which are described in detail in Johnson (1996), are listed here:

1. 1*2 Split Fan VCE (Air Force sponsored, 1975-1976)
2. 2*1 Split Fan + Rear VABI (Navy sponsored, 1976-1977)
3. Front VABI and Acoustic Nozzle (NASA sponsored, 1977-1978)
4. Core Fan Concept Demonstrator (NASA sponsored, 1979-1981)

The experiences gained through the demonstrators were incorporated in the VCE concepts for the aircraft and mission studies in the 1978-1982 time period. Figure 25 shows a typical cross section of one of these double-bypass VCE study engines.

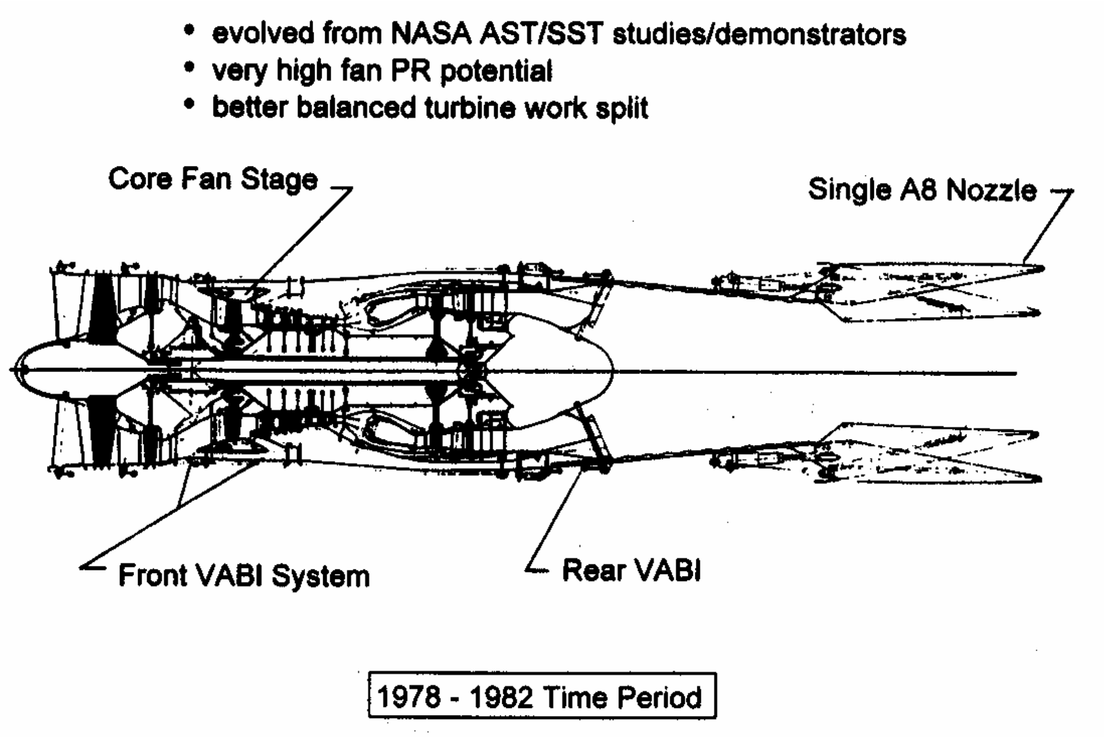


Figure 25. Double-bypass VCE with core fan architecture [from Johnson (1996)].

These study engines evolved to a final architecture that was selected as the propulsion contender (the other being the Pratt & Whitney F119) in the competition for the advanced tactical fighter (ATF), in the shape of XF120 (for bench testing), YF120 (for proof-of-concept flight testing in the YF22 and YF23 aircraft), and F120 (the proposed final engine). Figure 26 presents the schematic layout of this engine family, where, however, the labels “double bypass mode” and “single bypass mode” erroneously have been switched.

As the F120 is the only publicly described flyable VCE engine, it may be worthwhile to present its qualitative characteristics [Johnson, 1996]:

- Two-stage front fan system and five-stage core compressor system, each of which driven by a single-stage turbine (shaft in shaft)
- Stage-one of the core compressor equipped with an extended tip, acting as the third stage for the fan system
- Greatly simplified design of the low pressure turbine system, due to counter rotating spools plus the optimum turbine energy split made possible by the core fan architecture

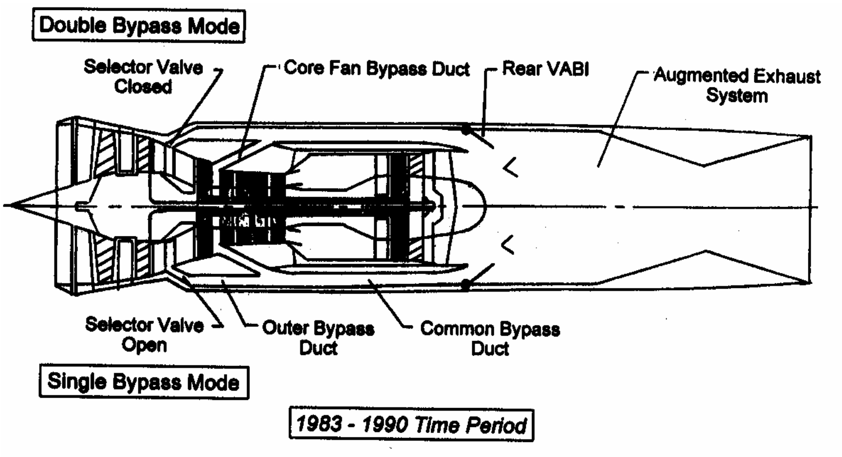


Figure 26. XF, YF, and F120 variable cycle configuration schematic [from Johnson (1996)].

This engine retained the single- and double-bypass modes that were initially tested on the YJ101 VCE concept demonstrators. They worked as predicted during the flight tests with the two contender aircraft. These were able to attain their design maximum super-cruise speeds (with the afterburner off).

Johnson (1996) further states that more advanced versions of the basic YF120 VCE were defined and tested during several programs in the time period 1984-1991. Also, a new type of VCE was defined 1991-1992 by General Electric and jointly pursued with Allison Advanced Development Company. The performance and architectural simplicity will allow the full phase 2 of IHPTET (US multi-year research program) goals to be achieved by 1998.

2.1.3.5 Research Programs

The IHPTET Program

The Integrated High Performance Turbine Engine Technology program is a US technology development plan pursued aggressively and in close coalition between Army, Navy, Air Force, NASA, ARPA, Allison Advanced Development Company (AADC) and other gas turbine companies [IHPTET – brochure].

The IHPTET program was established in 1987 in order to double aircraft propulsion capability by 2005. Its activities are subdivided into three clearly defined phases, where each has explicit performance goals. These are expressed as improvement percentages relative to 1987 technology. Table 4 shows an overview of the expected technology improvement for the three phases [IHPTET – phases].

The motivation for pursuing the IHPTET program can be summarized [IHPTET - introduction]:

- Ensure continued US aircraft superiority and cost effectivenes
- Nearly every technology developed within IHPTET can be used to vastly improve the performance, reliability, life, and operational cost characteristics of commercial turbine engines
- Sustain the positive balance of aerospace trade and maintain US market share

Supercruise (supersonic flight without afterburner) and advanced STOVL (Short Takeoff Vertical Landing) have been made possible by investments in IHPTET technologies, and these technologies are transitioning today to the F-22 and F-35 [IHPTET – brochure]. The interested reader is advised to study this source to see how the improvement program, in the many technology fields, is being pursued.

Table 4 The IHPTET program phases and performance goals (relative to 1987 technology).

	Figures in %	**Phase I**	**Phase II**	**Phase III**
Fighter / Attack eg. F-22 & JSF	**Thrust/weight**	+30	+60	+100
	Fuel Burn	-20	-30	-40
	Schedule	Completed	1997	2005
Missile / UAV eg. Tomahawk Cruise Missile	**Thrust/Air Flow**	+35	+70	+100
	SFC	-20	-30	-40
	Cost	-30	-45	-60
	Schedule	Completed	1997	2005
Subsonic Patrol Rotary Wing, eg. AH-64 Apache	**Power/Weight**	+40	+80	+120
	SFC	-20	-30	-40
	Schedule	Completed	1997	2005

The phases II and III of IHPTET (2x capability by 2005) will lead to possible improvements of existing engines, such as the F100/F110 Engines (F-15, F-16):

- Up to 50% Life Increase
- Up to 25% Thrust Increase
- Up to 7% Range Increase
- Up to $1.3B Disk Replacement Cost Avoidance
- Potential $2.5B Life Cycle Cost Savings with Technology Insertion

The VAATE Program

For the future, the DoD, DOE, NASA, and industry program known as Versatile, Affordable, Advanced Turbine Engines (VAATE) will assure further dramatic improvements in turbine engine affordability, not only for military applications such as aircraft, rotorcraft, missiles, and Unmanned Air Vehicles (UAVs), but also for America's domestic applications. VAATE will develop technologies that enable affordable growth to legacy systems and provide propulsion and power for future air, land, and sea applications [VAATE – brochure]. The VAATE program consists of two phases, which are stretched to 2010 and 2017, respectively. An overview of the goals in the phases is shown in Table 5.

Table 5. The VAATE program phases and performance goals [from VAATE – brochure].

VAATE GOAL FACTORS				
Large Turbofan/Jet – (Government Example)				
Capability	**Baseline***	**IHPTET**	**VAATE I**	**VAATE II**
T/W (@ Max Power)	6.4	12 (1.9X)	16 (2.5X)	20 (3X)
SFC (SLS, Mil Power)	0.860	0.740 (-14%)	0.688 (-20%)	0.645 (-25%)
Cost (FY00 $'s)***	Base	-30%	-50%	-60%
Development	$1.85B	–	$0.92B	$0.63B
Production	$230/lb Fn	$152/lb Fn	$115/lb Fn	$92/lb Fn
Maintenance****	$1,300/EFH	$845/EFH	$650/EFH	$520/EFH
Capability/Cost Index (CCI)	**Base**	**3.1X**	**6X**	**10X**

* Baseline 2000 State-of-the-Art ** <20,000 lbs Thrust

*** Total Cost Based on Weighting Factors as a Function of Total Life Cycle Cost and Yearly Expenditures
Cost = 0.1(Development $) + 0.5(Production $) + 0.4(Maintenance $)

**** Includes Depot Costs but not Fuel Costs

The VAATE I phase (6x affordability by 2010) is expected to give performance improvement of current (legacy) turbofan engines:

F119 Engine (F-22)

- 10% Thrust Increase
- 5% Range Increase
- Potential $1.25B Life Cycle Cost Savings with Technology Insertion

F414 Engine (F/A-18E/F)

- 55 Longer Range
- 20% Increased Thrust or $1B-$2B Total Ownership Cost Savings
- Increased Time on Wing
- Improved Readiness

F135/F136 Engines (F-35 JSF)

- Greater Than 244 Pounds Weight Avoidance
- Greater Than 10% Thrust Growth
- Greater Than $315K Production Cost Avoidance Per Engine
- Greater Than $6B Life Cycle Cost Avoidance

The VAATE II phase (10x affordability by 2017) is expected to give performance improvement of future jet engines:

Advanced Supersonic Cruise Engine (Long Range Strike)

- Mach 2-4 Cruise Capability
- 30% Increased Mission Radius
- Potential $9.0B Life Cycle Cost Savings with Technology Insertion
- 3X Increased Sortie Generation Rate
- Fast Response to Time Critical Targets

Advanced UCAV Engine (USAF UCAV, Supersonic UCAV, Navy UCAV)

- Subsonic Vehicle
    - 150% Radius Increase or 3X Loiter Time
    - Potential $1.3B Life Cycle Cost Savings with Technology Insertion
- Supersonic Vehicle
    - Enables Supersonic Cruise Capability
    - 2-3X Sortie Generation rate Potential
    - Fast Response to Time Critical Targets

Supersonic Missile

- 2X Radius
- Mach 3.5 Cruise Capability
- Fast Response to Time Critical Targets

The UEET Program

The information that presented are more or less taken directly from the NASA site [UEET – Welcome…].

The Ultra-Efficient Engine Technology (UEET) Program includes seven key projects that work with industry to develop and hand off revolutionary propulsion technologies that will enable future-generation vehicles over a wide range of flight speeds.

A new program office, the Ultra-Efficient Engine Technology (UEET) Program Office, was formed at the NASA Glenn Research Center to manage an important National propulsion program for NASA. The Glenn-managed UEET Program, which began on October 1, 1999, includes participation from three other NASA centers (Ames, Goddard, and Langley), as well as five engine companies (GE Aircraft Engines, Pratt & Whitney, Honeywell, Allison/Rolls Royce, and Williams International) and two airplane manufacturers (the Boeing Company and Lockheed Martin Corporation). This 6-year and nearly $300 million program will address local air-quality concerns by developing technologies to significantly reduce nitrogen oxide (NO_x) emissions. In addition, it will provide critical propulsion technologies to dramatically increase performance as measured in fuel burn reduction that will enable reductions of carbon dioxide (CO_2) emissions. This is necessary to address the potential climate impact of long-term aviation growth.

According to initial plans the seven projects that were to make up the UEET Program were:

- Propulsion Systems Integration and Assessment
- Emissions Reduction
- Highly-Loaded Turbomachinery
- Engine Materials and Structures for High Performance

- Propulsion-Airframe Integration
- Intelligent Propulsion Controls
- Integrated Component Technology Demonstrations

Later (May 2004) the web site http://www.ueet.nasa.gov/ reports a redesign and extension of the project to 2007:
The redesigned Ultra-Efficient Engine Project has a strengthened strategic focus which will promote environmentally friendly, clean burning engines by developing innovative technologies to enable intelligent turbine engines that significantly reduce harmful emissions while maintaining high performance and increasing reliability.

The new project will provide cross-cutting technology development & maturation focused on the Subsonic, Supersonic and Rotorcraft:

- Subsonic, Reduced NOx and Improved TSFC & T/W
- Supersonic, High Performance Inlets
- Rotorcraft, Advanced Drive Systems

*The new sub-project*s are:

- Low Emissions Combustor
- Systems Integration and Demonstration
- Highly Loaded Light Weight Compressor and Turbines
- Highly Integrated Inlet
- Advanced Drive Systems for Heavy-Lift Rotorcraft
- Intelligent System Foundation Technologies
- The technology areas will be updated as new information becomes available.

Examples of achievements are:
A reduction of NOx by 67% was achieved in the NASA Glenn combustion rig facility. This holds out the hope that a 70% reduction during landing and takeoff (relative to the 1996 ICAO standards) will be achieved.

Limited test data indicates promising fuel burn reduction. But, the source, the annual report of 2002 is not explicit enough to consolidate a number.

Intelligent propulsion control implementation in a laboratory has convinced the researchers that wireless control communication is necessary. Comparison between goals and actual benefits achieved are said to be promising. The statement from the annual report reads: Impact of Technology: High-temperature wireless data communication was identified as one enabling technology to achieve several objectives since the elimination of wires in the engine will result in increased reliability and affordability as well as reduced engine weight and complexity.

2.1.4 Ramjet

When the speed is sufficiently high the ram pressure gained by decelerating the incoming air is high enough to sustain subsonic combustion. The ramjet principle reaches competitive performance at speeds from about Mach 2, although the engine works from Mach 0.5 with a high fuel consumption. The ramjet engine is built around this principle.

This engine is very simple in its buildup, with no involved compressor such as a turbo-machinery. There is no requirement for its cross-section to be circular, except possibly for structure reasons. After the air is compressed by means of the ram pressure, the fuel is injected through orifices in the combustion chamber and burned subsonically (according to the Brayton cycle), and finally, the exhaust gases are accelerated to supersonic condition by a convergent-divergent nozzle. In order to minimize the pressure losses, the inlet air is compressed by one or more oblique shock waves.

FOI has studied ramjet engines with solid fuel (SOFRAM) for many years, since the end of the 70s and, also, during 1995-1999 in collaboration with TNO in Holland [Andersson, 2001]. The object of the research was to study fuels and their burning properties and to judge the usefulness of the SOFRAM technology, with special view on cannon-launched projectiles. Models were also created for studying the performance of these.

Liquid fuels have a number of advantages over solid fuels [Sandén, 2001]:

- Higher specific impulse (solid fuels need oxidisers for gasifying the propellant)
- Possible to adjust the fuel rate
- Liquid hydrocarbons give no smoky combustion products
- Liquid fuels allows easy optimising of the available space

These fuels can be classified into three main groups: hydrocarbons (from petroleum and coal), synthetic (JP-10), and slurry fuels. Slurry fuels contain addition of very fine energetic particles in suspension, of which boron is an attractive additive, but carbon, aluminium, and magnesium can be also used [Sandén, 2001].

The ramjet reaches material limitations when the speed approaches hypersonic level, around Mach 5 or 6. The ram compression itself may lead to critical structural limitations. Also, the temperature rises to so high a level that there is not much to gain by burning a fuel mixture (no temperature rise) due to material property limitation.

The problem with ramjet propulsion is that this must be brought to over Mach 2 by other means. As mentioned, for projectiles this can be performed through cannon-launch. For missiles this is usually done with a rocket. For an advanced high-speed air vehicle, other methods are probably used, such as turbojet systems. There are several combined cycle concepts where the ramjet principle is included, such as the turboramjet and the air turbo ramjet, which are described below.

2.1.5 Scramjet

The scramjet engine, which is as simple as the ramjet, is based on supersonic combustion, which is a way of circumventing the pressure and temperature problems of the ramjets at hypersonic speeds. In scramjets the deceleration of the freestream air is decreased to high supersonic speed (around Mach 3.5) for the combustion, not further to subsonic combustion as in the ramjet, thereby allowing a margin for both temperature and pressure increase caused by the combustion. In addition the dissociation of the gases associated with high temperature is postponed to higher speeds, when compared to the operational conditions for ramjets.

Numerical studies of scramjet flows have been performed by Berglund and Fureby (2003), using simplified large eddy simulations. The theoretical approach is reported and the results were compared to German experimental values.

An example of an air inlet system with supersonic fuel injection for the supersonic combustion is shown in
Figure 52, see the section 3.5.

According to Peregrine (2000), at hypersonic speeds, two or three weak oblique shocks will efficiently increase the pressure up to two orders of magnitude with only a small drop of velocity. A consequence of supersonic combustion, where the fuel is injected supersonically, is that the time available for mixing and combustion is very short. It is determined by the ratio between the length of the combustor and the flight velocity. The combustion often results in only partial combustion according to laboratory experiments. This source also says that mixing is a major concern and turbulent eddies must be created in the flow to distribute the fuel using struts and other injection mechanism. Finally it states that no scramjet has yet demonstrated a positive net thrust.

On the 30th March 2004, however, a successful test was performed with NASA's research hypersonic vehicle X-43 equipped with a scramjet engine. After being boosted by a rocket, it demonstrated a sustained air-breathing flight at Mach 7 [Cambridge, 2004].

In order to produce enough thrust, the scramjet engine needs help with capturing the appropriate amount of air: the whole fore body of the vehicle may be used for supersonic compression [Heitmeir *et al.*, 1996]. Also, the aft body of the same is used for the thrust augmentation by expanding the exhaust gases to the free stream pressure level. Therefore it is important to design the scramjet and the vehicle as an integrated concept.

As with the ramjet the scramjet needs help to get into the proper speed conditions, where the speed range is higher than for the ramjet. It is thus usually studied as a component in a combined cycle propulsion system, parallel or integrated.

2.1.6 Pulse Detonation Engine

2.1.6.1 Introduction

A presentation of the workings of the pulse detonation engine (PDE) is given in Svensson (2004). Around the world this concept has been of great interest, mainly because of expected low cost (simple build-up) and the potential of achieving high efficiency (burning according to the Humphrey cycle), as a propulsion system for air vehicles. Particularly during the last decades it has been studied intensely. A variant of the PDE is the pulse detonation rocket engine (PDRE), the difference being the way of supplying the oxidizer: a separate source rather than taking it from the air.

The PDE technology is an evolving one. Although the principle is attractive, there are many practical issues in many areas that need to be developed, before competitive PDE systems for aero-space applications can be created.

As shown in Figure 27, the PDE operates in cycles with intermittent phases: injection of fuel and air, detonation initiation, detonation propagation, and high pressure gas production followed by exit of the detonation wave and the gases. It is only the last phase that results in thrust, and hence the PDE concept is characterised by pulsing thrust.

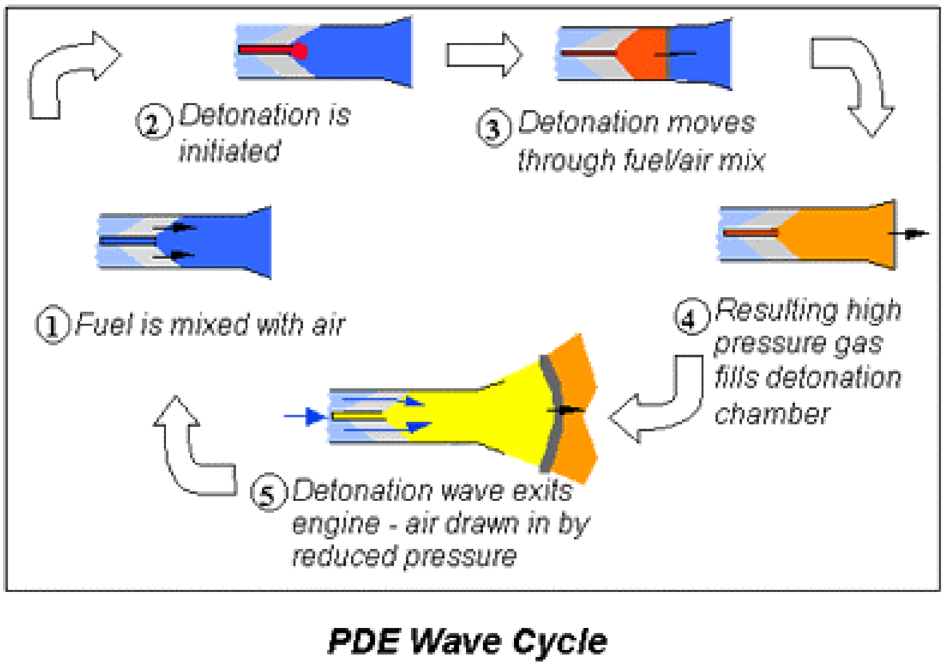


Figure 27. Working principle of the PDE engine [from NASA Glenn information].

2.1.6.2 Overview of PDE Technology

Excellent overviews of the research in various areas of the PDE technology are given in the following references: Kailasanath (a reference), covering the research in the field, Bussing and Pappas (1996), regarding theory and concepts, and Heiser and Pratt (2002), thermodynamic cycle analysis. An introduction to modelling analysis of PDE concepts is given by Lynch and Edelman (1996), where also chemical properties of fuels are discussed, including detonation ignition. When going through computational studies of pulse detonation engines, there is a wide variation in the predicted performance of even an idealized system. Plausible reasons for the observed differences are discussed in Kailasanath and Patnaik (2000).

2.1.6.3 Detonation Tube and Initialization

Experimental research both at FOI and elsewhere has shown that it is difficult to initiate the detonation. The available initiation methods can be roughly ordered into two categories [Tegnér, 1999]:

- Direct initiation. This method requires unrealistically large energy amounts (hundreds of Joules for acetylene-air mixtures).
- Deflagration to detonation transition (DDT). This method requires little energy, but it is slow.

The research studies in this field at FOI are based on the latter principle, which is the one that has the best development potential [Tegnér, 1999].

The DDT takes time and requires a certain length of the combustion chamber, which is too long for practical applications, unless a very detonable fuel is used. An interesting reading on the subject is found in Cooper *et al.* (2000).

One way of enhancing the DDT process is creating turbulence, for example through using the so-called Shchelkin spiral. Another way is applying a pre-detonator, which is a small tube that uses detonation-prone fuel. FOI has performed several studies on mechanical DDT enhancing devices and pre-detonators [Tegner, 2001; and Roman et al., 2002]. The purpose of the pre-detonator is to transmit the initial detonation into the main tube, which may use less detonable (and more practical) fuel. The common way of igniting the detonation is by using spark plugs and a pure oxidizer. To achieve the detonation a certain amount of power is required, depending on the fuel used.

The interested reader is referred to Bussing and Pappas (1996) and Murray *et al.* (2000) for more detailed information on ignition methods and DDT transitions. The reader may also read about an experimental engine at NASA [Litchford and Hutt , 2001].

An interesting ignition method using a pre-detonator with shock focusing (wave implosion) is reported by Jackson *et al.* (2003). The idea is that the detonation is initiated along a ring (planar front) in the cylindrical wall of the detonation chamber, and letting the continuing toroidical wave front wander to the centre of the tube, thus enabling a very high pressure to build up and initiate the detonation of the fuel-air mixture in the main chamber. This technology leads to much smaller pre-detonators, requiring less fuel and lower spark energy. The required energy is on the order of 100 times less than the usual spherical initiation, when considering propane-air mixtures. According Jackson *et al.* (2003) this technique is promising to ignite less detonation-prone fuels such as the jet fuel JP-10.

2.1.6.4 Fuels and Mixing

The properties of the fuel are critical to the detonation process. According to Bussing and Pappas (1996) the fuel has to be delivered as a gas, liquid, and/or solid of sufficiently small droplets/particles in order to create a stable detonation. The particles and the droplets must be no larger than several tens of microns. Only then the break-up time will be compatible with the detonation time scale. The various experimental setups at research institutes mostly use fuels such as hydrogen, acetylene, ethylene, propane. Occasionally variants of jet propulsion fuel, which is not easy to bring to detonation, have been used. At FOI studies on fuel injection methods are being performed on an experimental PDE, initially using hydrogen, with the objective of later using more practical fuels [Tegnér and Olsson, 2003].

The effects of the fuel mixing, the detonation process, and the chemical reactions are issues that need to be understood before PDE performance can be estimated. The mixing can be non-uniform (potential problems with non-detonated areas), and this is commonly ignored when calculating the detonation tube performance [Perkins, 2002]. This source indicates that good fuel-air mixing is not a prerequisite for optimal detonation tube performance. The PDE performance is affected by chemical dissociation and recombination of gases [Povinelli, 2001]. Performance effects of dissociation and recombination are reported in a number of sources [Povinelli and Yungster, 2002; Povinelli and Yungster, 2003; and Povinelli, 2003], and it was found that the amount of sensible heat, recovered through recombination in the PDE chamber and exhaust process, is significant. The results also clearly eludicate the competitive regime of PDE application relative to ramjets and gas turbines.

2.1.6.5 Shaping Detonation Tubes

Usually the studied detonation tubes are shaped as cylinders with a constant cross section area. However, a computer study is reported in Paxson (2003), where effects of the cross-sectional area variation on idealized PDE engine performance have been studied. Two parameters have been varied: the overall exit-to-inlet area ratio, and the distance along the tube at which continuous transition from inlet to exit area begins. These two parameters are varied over flight conditions (defined by inlet total temperature, inlet total pressure and ambient static pressure) and the performance is compared to a straight tube. It is shown that compared to straight tubes, increases of 20 to 35 percent in specific impulse and specific thrust are obtained with tubes of relatively modest area change.

The hardware of the PDE engine requires robust design of the engine components that are capable of enduring harsh detonation environments. The potential problems lies in the fatigue phenomenon associated with the intermittent operation of the combustor, when cracks form and propagate. A rig for testing such issues is described by Zhu *et al.* (2002). The test results indicated that oxidation and creep-enhanced fatigue at the oxide scale/alloy interface was an important mechanism for the surface crack initiation and propagation under the simulated PDE condition.

2.1.6.6 Shaping Inlets

Designing inlets for air vehicles aims at decelerating the air to subsonic combustor environment and also adapting them for intermittent operation of the PDE engine. According to Bussing and Pappas (1996) unsteady inlets can be used, either with mechanical valves as in the WW2 V1 pulse jet engine or a valve-less design where the inlet/combustor geometry is shaped to minimize unsteady perturbations within the inlet (aerodynamic rectifiers). The former of these approaches incurs significant external flowfield losses, while the latter offers promise but tends to be leaky and have poor off-design performance. A third approach is using large manifolds to dampen out pressure transients to a quasi-steady-state at the inlet lip. This tends to increase the total volume of the engine. Bussing and Pappas (1996) finally states that there are at least two approaches that can be used to operate a PDE inlet in a steady-state mode. One involves bleeding (dumping away) excess air between detonation cycles in a way to mitigate any inlet or performance losses. Another involves using clusters of combustors which operate cyclically one after the other and share the same inlet (and outlet).

Pegg and Hunter (1996) report a mixed-compression inlet design concept for potential pulse-detonation engine (PDE) to power supersonic aircraft. The analysis consisted in time-dependent CFD computations of the inlet flowfield, and the objective was to estimate the installed PDE cycle performance.

2.1.6.7 Shaping Nozzles

Nozzles for PDRE rocket engines have been studied. No sources on the subject for PDE have been found, but the results should apply for PDE too. Morris (2004) describes CFD computations where several PDRE combinations of a detonation tube and various nozzles. The studies covered a wide range of blowdown pressure ratios (1-1000). The results indicate that a convergent-divergent (C-D) nozzle is generally more effective than a straight extension in improving the PDRE performance, particularly at higher pressure ratios. Also, the results show that the blowdown process of the C-D nozzle systems could be beneficially cut off well before the pressure at the end-wall reaches the ambient value. For PDE in high speed flight as of 2002 the nozzle issue remained to be studied [Povinelli, 2002].

The energy due to the strong shock wave that exits from the detonation tube can be utilised by adding an ejector. Thrust augmentation effects of such a device have been computed by Yungster and Perkins (2002) in multiple cycle simulation of a PDE detonation tube with an ejector. Figure 28 shows the basic ejector characteristics, where the parameter values have been varied in the simulations. Impulse augmentation ratios of up to 1.9 have been achieved.

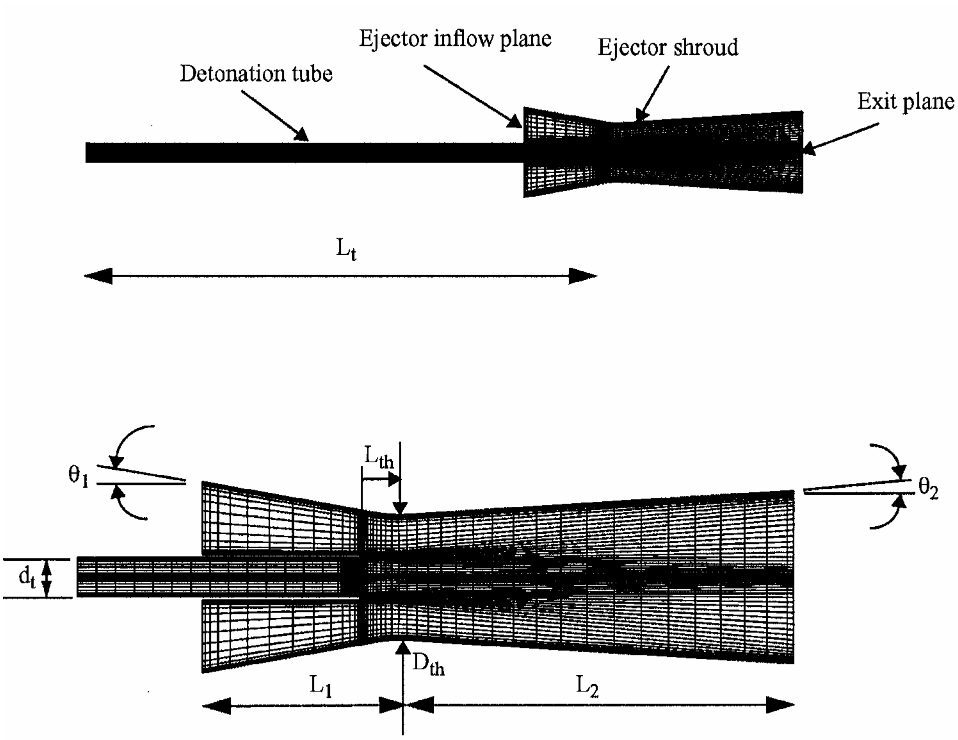


Figure 28. Computer model of a PDE ejector [from Yungster and Perkins, 2002]. The parameters define the ejector characteristics and are varied in the simulations.

Experimentally, thrust augmentation of ejectors have also been investigated [Fernandez *et al.*, 2003], where wind tunnel data and computer predictions were compared for a PDE entombed in a duct. The ejector primary flow is a pulsed, under-expanded, supersonic nozzle simulating the supersonic waves emanating from a PDE, while the ejector secondary flow is the 1 x 1 foot supersonic wind tunnel test section operated at subsonic Mach numbers. Pulsed ejectors have been shown to result in a 3 to 1 improvement in LID (length-to-diameter) and a near 2 to 1 improvement in thrust augmentation over a steady ejector. Another research program [Santoro and Pal, 2003] is studying PDE driven ejectors applicable to a hybrid Pulse Detonation/Turbofan Engine. The objective of the study is to characterize the PDE-ejector thrust augmentation. This experimental research involves the design of a PDE-ejector system, with a detonation tube and modular ejector assembly, to provide critical experimental data.

2.1.6.8 General Characteristics of the PDE

In the propulsion overview from Peregrine (2000) the following characteristics can be extracted:

- PDE engine is effective at both low and high speeds, due to the high detonation wave velocity
- The inlet is the limiting factor for the PDE at high speeds. The flow must be decelerated to subsonic speeds to allow the detonation, which is sensitive to the velocity of the unburned gases.
- The inlet is closed during the actual detonation, and the flowpath must be diverted during the closed portion of the cycle to avoid high ram losses.
- The constant-volume combustion process (Humphrey cycle) of the PDE has a 30-50% higher thermal efficiency than the constant-pressure process (Brayton cycle), when the compression ratio is about six. If translated successfully into propulsion efficiency, the specific fuel consumption would be decreased by a similar amount, leading to a specific thrust in the order of 3600 s. This is comparable to a turbojet at low speeds but twice the fuel efficiency of a ramjet at high Mach numbers.
- The PDE concept is relatively undeveloped and several significant technical challenges remain.
- Detonation wave is difficult to generate in any form, and for it to be stable from the inlet of the chamber to the exit is even difficult to predict.
- For sufficient vehicle thrust, cycle frequencies 100-200 Hz are required.

2.1.6.9 Noise

The intermittent operation of the PDE engine naturally causes noise. There may be an interest to be able to estimate its characteristics. The report by He and Karagozian (2002) gives hints for this.

2.1.6.10 Performance Estimation

For estimation of very basic performance, using the theory for the Humphrey cycle, Tegner (2000) can be used. This source covers only the combustion process. Another theoretical source can be found in Bussing and Pappas (1996). However, the results have to be adjusted for inlet losses, real gas effects including chemistry, and nozzle augmentation/losses.

An interesting way to estimate the performance is given by Paxson (2001). His scheme is based on an ideal, air breathing pulse detonation engine described in a manner that is useful for application studies, since the engine can be characterized by an averaged total pressure ratio. This ratio, in turn, is a unique function of the inlet temperature, the fraction of the inlet flow containing a reacting mixture, and the stoichiometry of the mixture. Total pressure ratios are derived from thrust calculations obtained from an experimentally validated, reactive Euler code.

2.2 Multi-Cycle Propulsion Systems

In space flight rocket propulsion has been the dominating power source, mainly because such systems can be operated over the whole speed range. They are simple and reliable, but consume enormous amounts of fuel, partly because they need to carry also the oxidizer. Therefore, in order to reduce this fuel consumption, the launch systems have consisted of several stages, which each were to be thrown away when emptied. Next step in the development of the launch principle was replacing the last stage by a reuseable one, namely the so-called shuttle vehicle with rockets (used in the ascent). The first stage rocket containers were also reused.

Now, since the shuttle fleet has been in service for a long time and needs to be replaced, the space agencies are studying new concepts of launching space vehicles. These concepts share the idea that the air in the atmosphere should be used optimally. Then the oxidizer need not be carried from the ground launch. There are many engine cycles that could be used, but most of these individual cycles have only a limited operative speed interval. Choosing a set of cycles with overlapping speed regimes (in all, from Mach 0 to about 14), in the form of separate engines, would lead to heavy systems (having to carry the extra masses of the engines currently not in use). Therefore the focus of interest has been directed to combined-cycle systems, i.e. systems with mode operating through redirecting flows in order to switch between the appropriate engine cycles. Even the rocket mode may be included. Such systems would lead to considerably lower weights.

There are two main ways to devise a combined-cycle flowpath for space flight applications, namely Rocket-Based Combined Cycle (RBCC) and Turbine-Based Combined Cycle (TBCC) [Clough, Lewis, 2004]. In the former category rockets are used for lower Mach number flight, then ramjets, scramjets, and finally rockets to reach orbit. The latter category begins with turbojets. Examples in this category are the turboramjet and the air turbo ramjet, which are described separately below.

2.2.1 Rocket Based Combined Cycle

How an integrated RBCC would operate in low-speed and high-speed is described in Billig (1996), with appropriate theoretical foundation for the various modes. The discussion is based on two modellings of rocket-ejector system with ramjet/scramjet, see Figure 29.

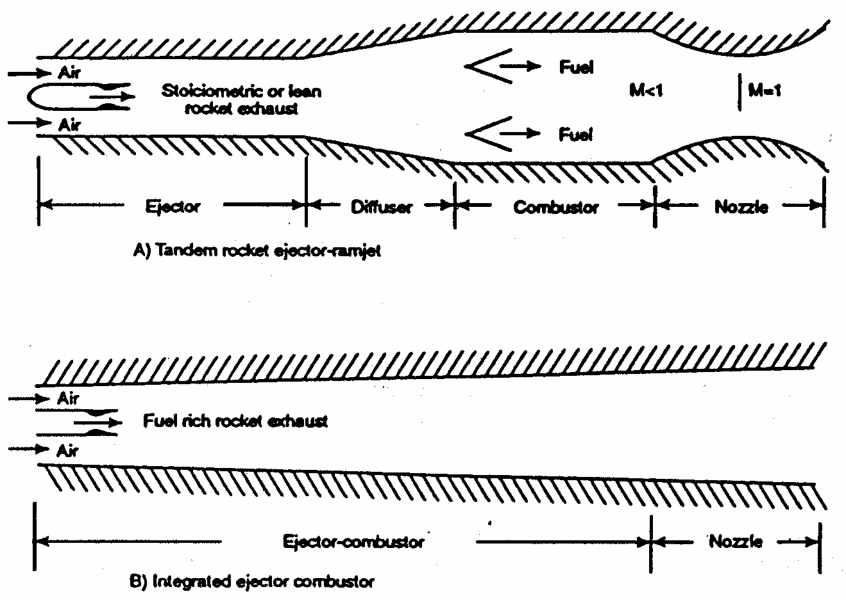


Figure 29. Schematic illustrations of ejector ramjets [from Billig, 1996].

The upper part of the figure shows a lean-burning rocket with an ejector and ramjet with diffuser and injectors. Lean-burning with ejector effect ensures good low-speed operation (near maximum cycle performance), while extra performance is achieved by injecting additional fuel for the ramjet operation. The diffuser is required for subsonic combustion and the nozzle for accelerating the exit gases to supersonic speeds. The arrangement with diffuser and nozzle adds to friction and weight of the overall system. The other model, depicted in the lower half of Figure 29, shows a rich-burning rocket with an ejector and scramjet. Fuel-rich (over-stochiometric) burning is not as efficient as lean-burning, but the excess fuel is further burned as supersonic combustion in the widening combustor (scramjet function). No deceleration to subsonic condition with the following acceleration to supersonic state, as in the ramjet principle, is needed. The benefits are higher operating speed without temperature problems, lower weight and lower internal friction.

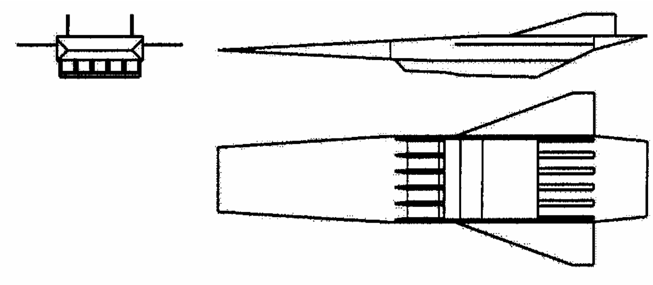


Figure 30. Schematic image of Single-Stage-to-Orbit aerospace plane [from Kanda and Kudo, 2002]

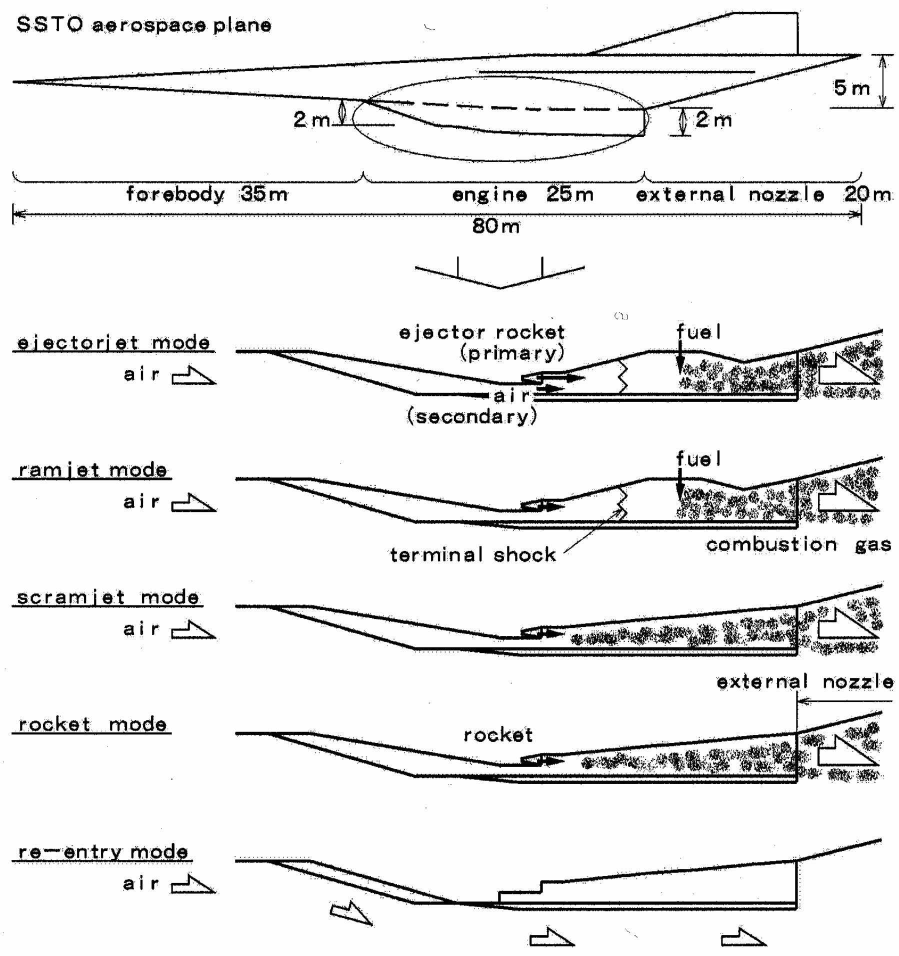


Figure 31. Schematic diagram of operationg conditions of the combined cycle engine [from Kanda and Kudo, 2002].

Both of these types of ejector ramjet principles have been studied in a combined cycle engine for a space plane at the National Aerospace Laboratory of Japan [Kojima and Tanatsugu, 2003]. This engine was composed of an ejector jet mode, a ramjet mode, a scramjet mode and a rocket mode. Propellants were liquid hydrogen and liquid oxygen. A schematic configuration of the aerospace plane was studied [Kanda and Kudo, 2002], with external air compression and external exhaust expansion for the thrust augmentation, see Figure 30. This is an example of a concept, which was designed together with the propulsion system and where the inlet ramp is sized for the hypersonic flight condition. A passive thermal protection system is used for the vehicle, in the form of ceramic tiles covering the body of the plane. However, an active cooling system was needed for the propulsion system. The operation modes of this combined cycle engine are shown in Figure 31.

Sample calculations results of the engine performances are presented in Kanda and Kudo (2002), where also the critical balancing of the pitching moment, due to pressure effects on the inlet ramp, the nozzle expansion surface, and the centre of gravity, is discussed.

2.2.2 Example of a Turbine Based Combined Cycle Propulsion

We choose to present an example of a TBCC concept, namely the recent study found in Peregrine (2000), which is an answer to a request for proposal from the US military authorities.

Our motivation for including this example is to study the arrangement and discussions of a chosen propulsion system for such a concept, the reason being a closer bearing to the kind of vehicles we may study at FOI in the future.

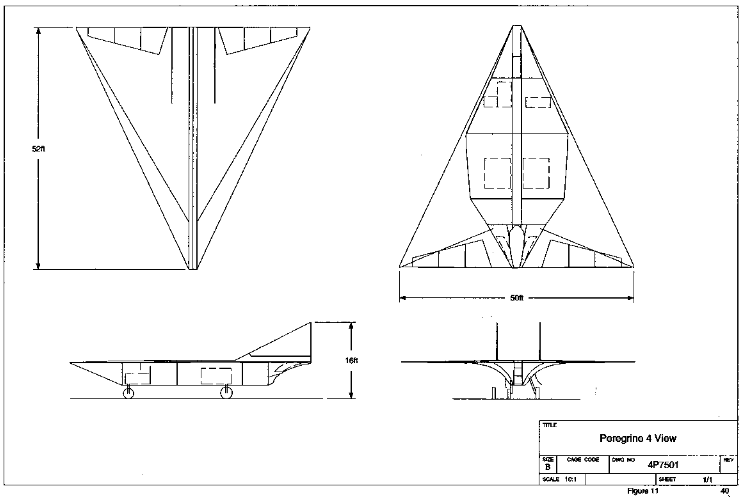
After an extensive literature search, where developments within “every” technology area - such as propulsion, fuels, aerodynamics, structure, etc - has been reviewed, the project team has proposed a configuration concept of a hypersonic strike fighter, which the team named the *Peregrine*.

The team evaluated many jet engine technologies: turbojet/turbofan, pulse detonation engine, rocket, ducted rocket, air turbo rocket, ramjet, and scramjet. The questions it asked regarding the suitability of the various propulsion arrangements are summarized as:

- The primary mission of a hypersonic strike platform is the use of high speeds to achieve short strike and mission times and to improve survivability.
- The benefit of decreasing strike time as compared to the increase in complexity of the propulsion system achieved by increasing the cruise speed to Mach 8 from Mach 4 may offset each other.
- Given that ramjet propulsion is generally considered limited to Mach numbers no greater than six, a vehicle that flies at Mach 6 or greater would either have to incorporate a scramjet or a rocket based propulsion system.
- The scramjet would have to be integrated with a lower speed propulsion arrangement and would promote a three-cycle design (turbofan-ramjet-scramjet).
- The rocket could provide propulsion throughout all flight regimes, but with low specific impulse. Thus the fuel weight for the vehicle would increase, and with it the vehicle size, determined by a carousel (multiplication) factor (>1) to account for the added structure to carry the fuel.

Fuel aspects (kinds and properties) and specific impulse were also weighed in, before the team arrived at the final propulsion solution: a turbofan for low-speed in parallel with a ramjet for high-speed. The targeted cruise speed of the *Peregrine* is Mach 4 at 91000 feet. The reason for choosing the parallel (over/under) layout over axial configuration is the vast differences in the size of the ramjet combustor area as compared to the afterburner area of the turbofan.

The *Peregrine* characteristics are shown in Figure 32. Note the low wing thickness, which is a measure to keep the low drag level needed for matching the thrust capacity of the propulsion system. The length of the mission profile is 750 nm. The internal component layout is illustrated by Figure 33.



Peregrine Vehicle Characteristics

Characteristic		Value
Takeoff Gross Weight	=	47,000 lbs.
Fuel Weight	=	15,500 lbs.
Empty Weight	=	31,500 lbs.
Zero Fuel Weight	=	27,000 lbs.
Structural Weight	=	16,500 lbs.
Length	=	52 ft.
Wing Span	=	50 ft.
Wing Area	=	1275 ft^2
Aspect Ratio	=	2
Taper Ratio	=	0
Airfoil	=	2% Biconvex
Vertical Tail Area	=	284 ft^2
Vertical Tail Height	=	16 ft.
Vertical Tail Airfoil	=	2% Biconvex
Landing Gear		
Nose Gear	=	1 Wheel
Main Gear	=	1 Wheel / Strut
Turbofan Sea Level Thrust	=	32,000 lbs.
Ramjet Thrust @ Cruise	=	5,500 lbs.

Figure 32. Four-view drawing and vehicle characteristics of the Peregrine [from Peregrine, 2000].

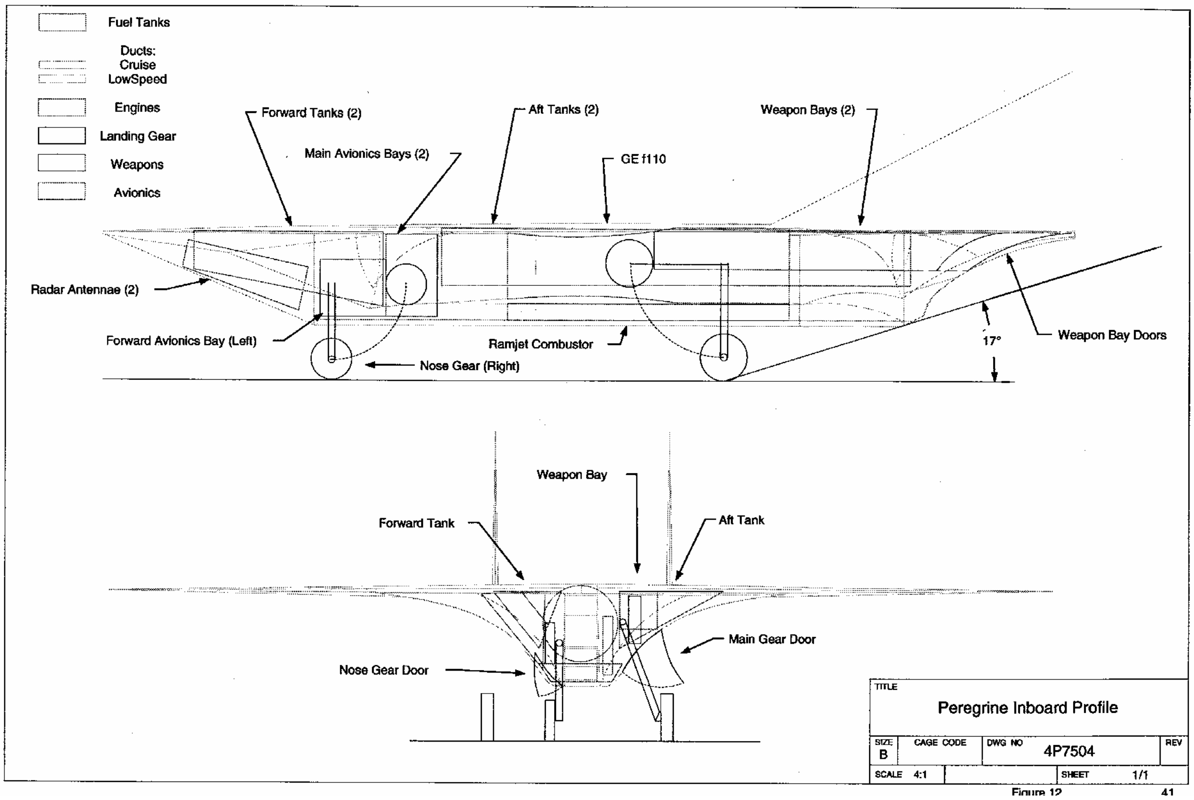


Figure 33. Inboard profile of the Peregrine [from Peregrine, 2000].

Regarding the propulsion characteristics, Figure 34 and Figure 35 show absolute thrust and specific impulse, respectively, as functions of Mach number and altitude.

The inlet design was also aimed at optimum efficiency over all flight regimes. The *Peregrine* is equipped with a five shock, mixed compression, variable geometry 2-D inlet, with pressure recovery of 71.9 percent at cruise conditions. The cowl lip is an actively cooled structure using a small portion of the fuel that is later combusted in the ramjet.

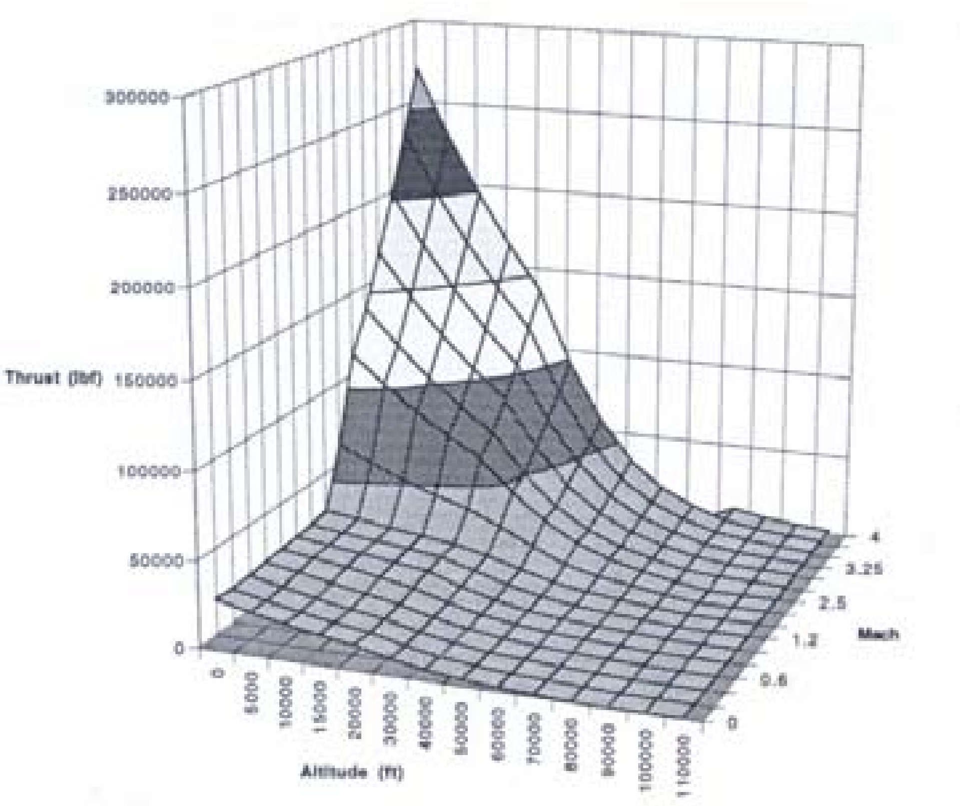


Figure 34. Thrust map for the Peregrine propulsion system [from Peregrine, 2000].

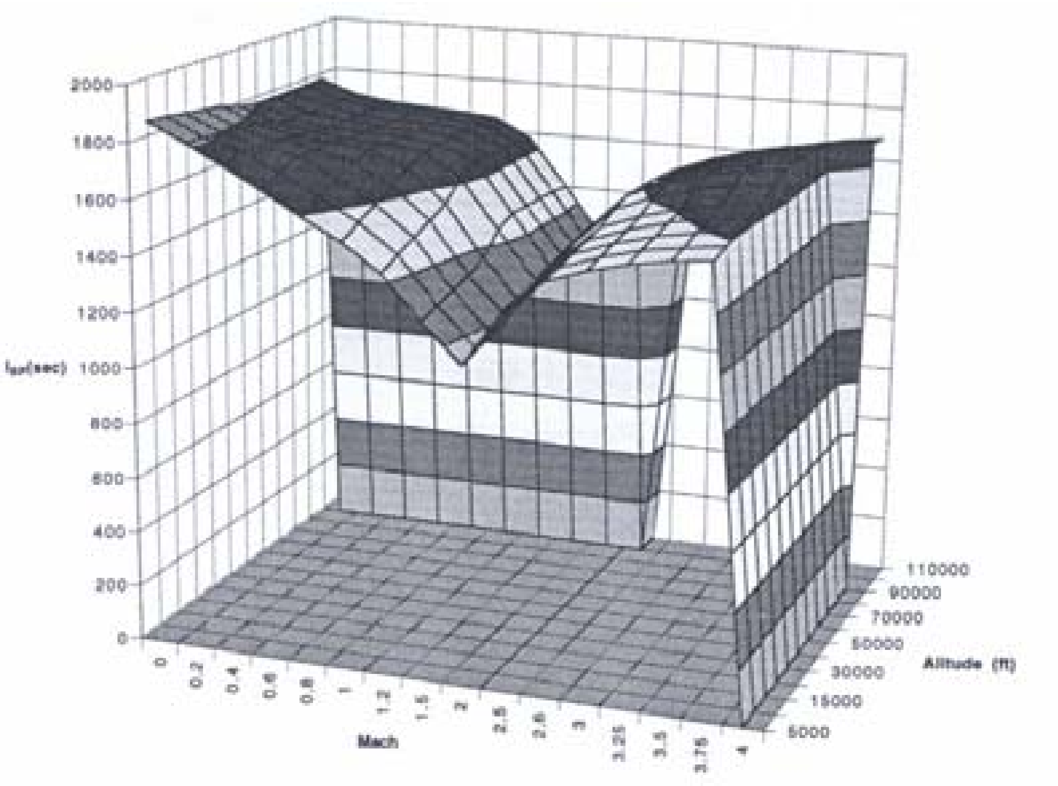


Figure 35. I_{sp} map for the Peregrine propulsion system [from Peregrine, 2000].

2.2.3 Turbo Ramjet Engines

Turboramjets may be seen as early examples of TBCC engines. An example of such engines that were used operationally is the Pratt & Whitney J-58 for the SR-71, which was designed for speeds over Mach 3. This engine started its life as an afterburning turbojet, but during the evolutional development the Lockheed team encountered problems with airflows and thrust. The solution became a partial (switchable) bypass of compressed air (from the middle of the compressor through 6 large tubes) past the turbine to the afterburner. This engine may be seen as a kind of turboramjet.

In the former USSR, the research in this field started in the late fifties, when there was a significant interest in creating an advanced high-supersonic and hypersonic flight technology [Sosounov *et al.*, 1996]. The Central Institute of Aviation Motors (CIAM) in Moscow began fundamental studies of combined air-breathing engines, which were based essentially on evolution of the concept of a bypass jet engine with reference to flight conditions at high-supersonic speeds. These studies led to a variety of turboramjet engines (TRE) based on a turbojet engine (TJ) and a turbofan engine (TF) as a core. Methods were developed for engine configuration selection, which is essential in the TRE case for optimum matching of gas turbine and ramjet parts. Also, a theory of TRE operation was developed: for switching from one operational mode to another, and for operation at the ramjet mode at maximum speed of flight under conditions of high kinetic heating, including the modes with the turbofan windmilling.

The investigations went further into real operating conditions and performance of the TRE, and several full-scale demonstrators were studied. An experimental TRE was tested under simulated flight conditions at Mach number 3-4 for hours of time. This work was intended for the development of a cruise mission flight vehicle using kerosene. According to Sosounov *et al.* (1996), the results from these developments and the Russian experience with the use of liquid hydrogen open the possibilities for development of TRE for a flight vehicle using liquid as fuel.

When developing the supersonic or hypersonic propulsion system, the main factor defining the configuration is the wide range of the operating conditions. For cruise flight the main interest is finding optimum thermodynamic parameters (fuel efficiency), which change considerably with flight speed and altitude. Requirements for high thrust and acceleration set the demands on high mass flow of the engine. The typical lowering of mass flow characteristics of the TJ/TF sets the operating speed limit. The ramjet (RJ) has no deficiency in this respect. At flight Mach numbers above 3-3.5 the specific thrust of the ramjet engine becomes higher than that for the turbojet engine with afterburning, see Figure 36. Combining into the same hardware a turbojet and a ramjet give the possibility of obtaining an engine system that can operate efficiently from low-speed to Mach 4-5.

The combination of TJ/TF and RJ can be done in several ways, as shown in Figure 37, where ARCC stands for afterburner-ramjet combustion chamber. The other notations need no further presentation. The alternatives for the separate RJ part combustion chamber represent solutions where TJ and RJ can be operated more or less simultaneously. The have, however, the drawback of being large, complex and heavy, which are due to the parallel arrangement of the two combustion chambers. Of these, the most simple, compact, and most advanced principle is the TRJ with the separate gasturbine and ramjet ducts (TRJ_S), which offers the use of fuel efficient (stoichiometric) TF function. The common ARCC are alternatives where the TJ and RJ parts are arranged in tandem, followed by a common afterburner. Low weight is gained, but the simultaneous operation of TJ and RJ at low and moderate speeds is impossible for pressure matching reasons. Sosounov *et al.* (1996) discuss all the alternatives in detail.

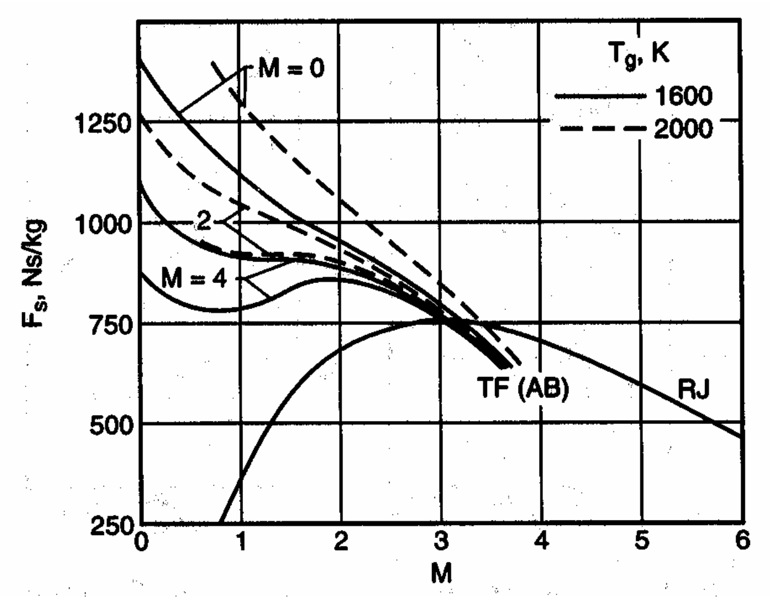


Figure 36. Specific thrust comparison of both turbofan and ramjet engines [from Sosounov et al. (1996)].

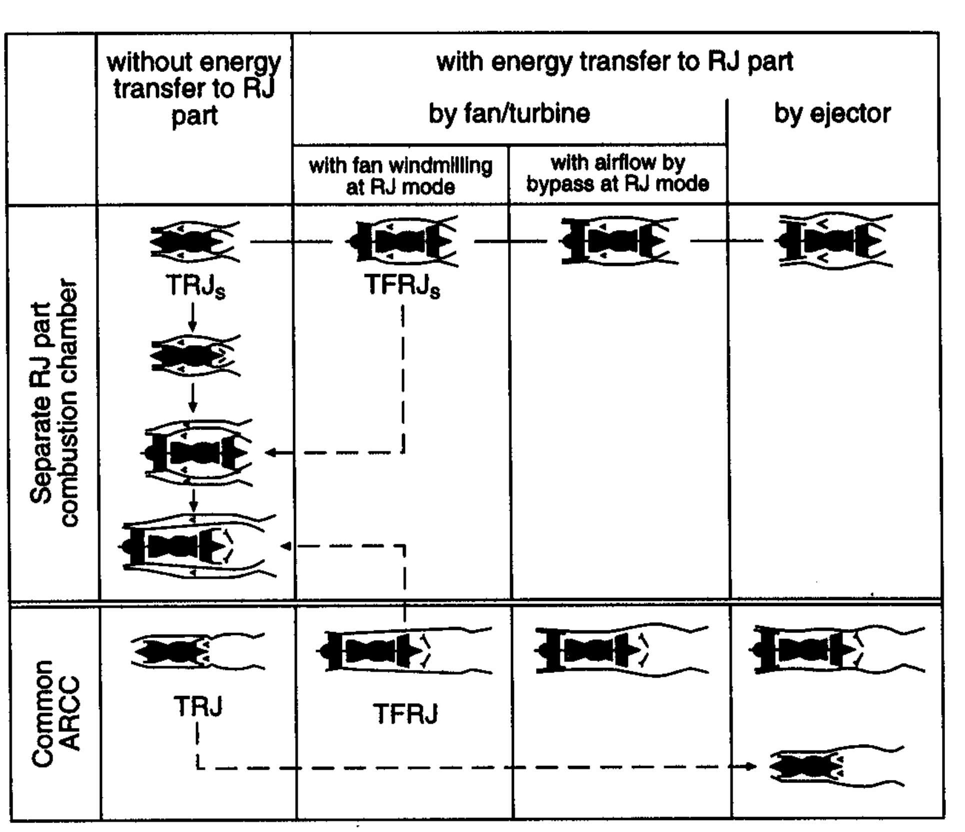


Figure 37. Classification of TRE [from Sosounov et al., 1996].

Some of these engine concepts were analyzed in a couple of hypersonic aircraft studies (space application): a two-stage-to-orbit system (TSTO) with separation at Mach 6, and a hypersonic aircraft (HST) with cruise speed at Mach=6. In both cases liquid hydrogen was used as fuel. See Figure 38.

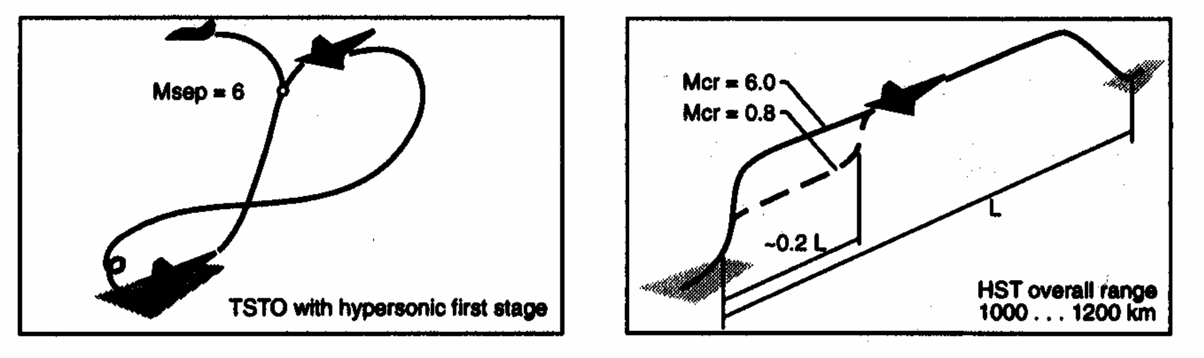


Figure 38. The flight profiles of TSTO and HST [from Sosounov et al., 1996].

Four engine concepts were studied, two turboramjet alternatives and two air turbo ramjet (ATR) alternatives. A description of the ATR principle is given in section 2.2.4. The alternatives are depicted in Figure 39, category-wise as upper and lower figure halves, respectively. The studied ramjets, based on a turbojet (TJ) and a turbofan (TF), respectively, are both of the type shared ARCC according to Figure 37, but the variant TRJ_S (see Figure 37), having separate RJ combustion chamber, was also considered. The studied ATR are representatives of the two main ATR principles, see terms used in section 2.2.4: Expander cycle, here called regenerative (RATR), and gas generator cycle, here called ATR. The regenerative (=expander cycle) engine has a heat exchanger in the combustion chamber for converting the liquid hydrogen into gas state. The hydrogen is burnt with air. The gas generator principle uses oxygen in the fuel.

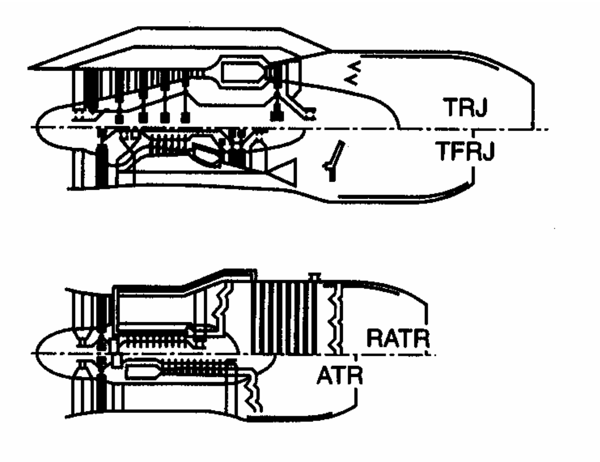
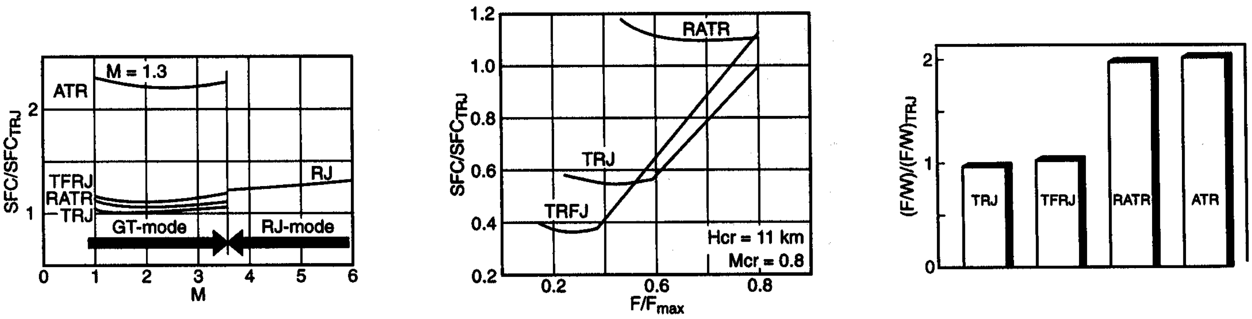


Figure 39. The engine concepts [from Sosounov et al., 1996].

It is interesting to study comparative results of different propulsion systems, not only between different variants of turboramjets, but also between different propulsion categories. The compared category is here the ATR cycles. One of the objectives in this report is to present relative merits of engine categories. When operating in cruise, the TRJ, TFRJ, and RATR alternatives show very similar SFC levels for the nearly same selected pressure ratio, whereas the ATR process, as expected, results in a much higher level due to the fact that the onboard oxygen is carried. However, in the ramjet mode, which is effective from Mach number about 3.5, all alternatives use incoming air, hence the lower SFC level compared to that of the ATR, see Figure 40a. Looking at Figure 40b, subsonic operation at part thrust conditions is much better for the two turboramjet alternatives than the RATR. Figure 40c shows that the thrust-to-weight ratio of the ATR variants, of which the RATR is the serious alternative due to the low SFC, is about twice that for the TRJ and TRFJ engines. It should be noted that the range of the RATR throttling performance is limited due to the heat exchanger structure overheating.



(a) Relative SFC (b) Subsonic relative SFC (c) Relative thrust-to-weight ratio

Figure 40. Engine performance comparison (fuel: LH_2) [from Sosounov et al., 1996].

2.2.4 Air Turbo Rocket/Ramjet (ATR)

The ATR belongs to the flow path category TBCC (turbo-based combined cycle), and is being developed in various forms for space applications. It uses two cycles for fuel-efficient propulsion, with turbine-driven air compression, at low speeds and high thrust ramjet/rocket-like operation at high speeds. The speed regime is between 0 and Mach about 6 [Tanatsugu, 1996]. In general the ATR offers high thrust-to weight ratio at low speeds and good performance at high Mach numbers [Kerrebrock, 1996].

The acronym ATR stands for both Air Turbo Rocket and Air Turbo Ramjet, and they refer to the two primary cycles described below: gasgenerator (=rocket) and expander (=ramjet) cycles.

Figure 41 shows the two main variants of the ATR engine principle. The gas generator cycle ATR, see the upper half of the figure, resembles a turbojet engine with an afterburner in its general layout, where the main difference is the way the turbo-compressor receives its energy. This version, also called the bipropellant the ATR, uses two liquid propellants, a fuel and an oxidizer, which are pumped to high pressure and burned in a rocket-like combustion chamber. The combustion products are then expanded through a turbine, which drives the air compressor and are then lead to the main flow path to be burned further in the afterburner. As opposed to the turbojet, the turbine will not be imposed by a temperature limit due to the speed. In this way the operative Mach regime is extended considerably.

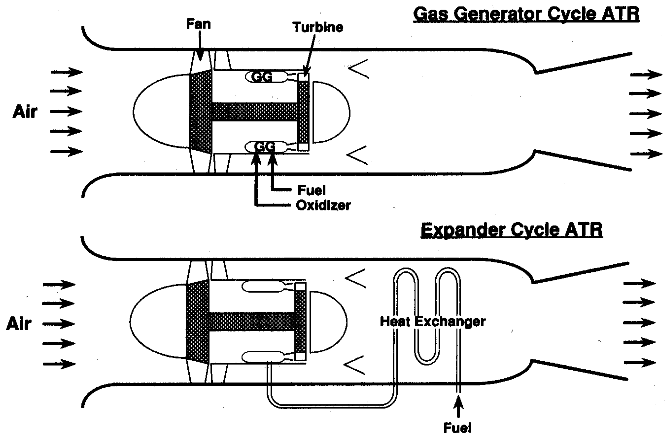


Figure 41. Two Primary Cycles of ATR [from Tanatsugu, 1996].

In the expander cycle ATR, see the lower part of Figure 41, liquid hydrogen is pumped to high pressure, vaporized and heated by exchange with combustion products, burned and expanded through a turbine, which drives the compressor.

Following Clough and Lewis (2004), the ATR cycle can be characterized by its bypass ratio (BPR), which is the mass through the engine (the air) divided by the mass that the rocket generates. Note that this BPR can range from zero to infinity. At high BPR the rocket gas generator will act as a fuel injector to an air-supplied turbojet process. On the other hand, very low BPR means that the majority of the mass flow comes from the gas generator, thus depending on the rocket behaviour. The main advantage of the ATR cycle over other cycles is that it can provide higher thrust, when needed, than a traditional turbojet. This can be executed simply by changing the fuel rate.

The theoretical aspects of both ATR cycle versions are given by Kerrebrock (1996). He explains that from fuel efficiency point of view, the expander cycle is more efficient, because the oxygen is taken from the incoming air, as opposed to being carried onboard.

Practically, there are several implications of integrating turbomachinery into the ATR cycle [Clough and Lewis, 2004]:

- Where to place the turbine and how to operate it. One way is to place it inline and route the compressor flow around the turbine. Another way is to package the gas generator outside of the main flow, thus removing the need of internal ducting. The advantage of the latter is that the turbine dimension can be optimized (smaller).
- How to transmit power to the compressor, which is layout-dependent. In the inline layout a common axis can be used, but as the bypass characteristics varies with the operation conditions, a transmission may be needed. With the separate gas generator package the transmission cannot be avoided.
- An overlooked issue to date, using analytical model, is mixing the gas generator and core engine air flows, which must be performed with streams at essentially equal pressure. For studying this, an advanced component matching model is needed.
- The engine performance is highly dependent on the inlet design, the reason being a wider operative Mach number range. A variable geometry inlet could maintain higher compression at low Mach numbers, while also having the capability to keep the inlet flow within the compressor temperature limits at high Mach numbers.

The material in Clough and Lewis (2004) is based on work supported by National Sciences Foundation Graduate Research Fellowship, with additional support provided under the Reusable Launch Vehicle Institute by NASA and the Dept. of Defense.

Extensive research work in the ATR field has been done since 1986 in Japan due to the interest of the country to commercialize and popularize space flight, to realize cost reduction, improve reliability, safety, and usability of the future transportation system. Several reports, see for example Kojima and Tanatsugu (2003), cover the developments of the “ATREX” (Air TurboRamjet EXpander Cycle) engine, which describes concept studies, demonstrators and nozzle work. Tanatsugu (1996) presents the detailed comparative analysis of two gas generator cycle and two expander cycle ATR concepts, aimed at choosing one for further experimental investigation. These concepts are shown in Figure 42.

Note that for all of the concepts, the turbine is mounted along the periphery of the first fan stage, which saves weight but also is causing vibration problems (which were cured). A detail view is shown in Figure 43.

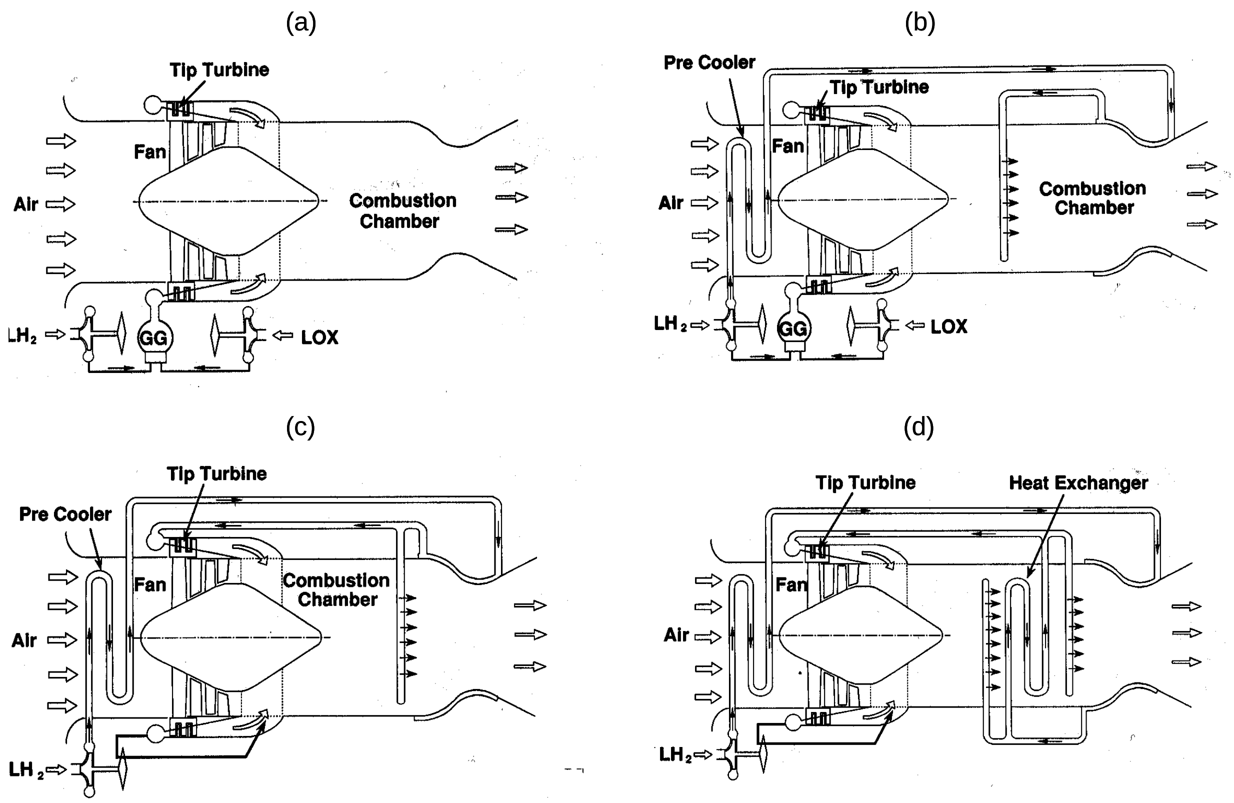


Figure 42. Four ATR engine designs at Institute of Space and Astronautical Science, Japan: (a) Basic gas-generator cycle without precooler (only for reference purpose), (b) Precooler gas-generator cycle, (c) Precooler expander cycle, and (d) Precooler expander cycle with heat exchanger [from Tanatsugu,1996].

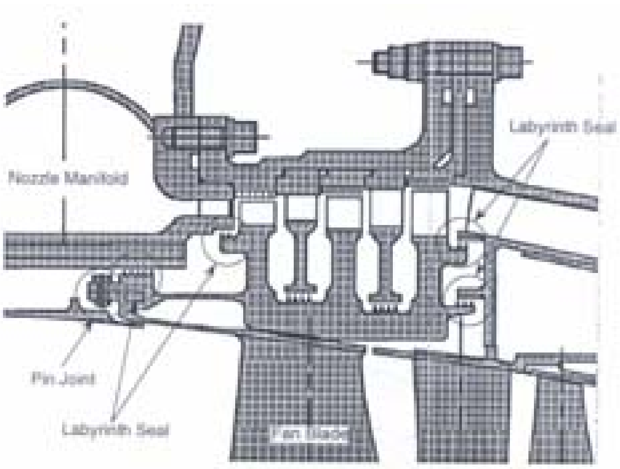


Figure 43. Tip turbine structure [Tanatsugu et al.].

The common characteristics of the engines are: A two-stage fan with a pressure ratio of 2.5 and adiabatic efficiency of 0.85 at static sea-level conditions and a multistage tip turbine having expansion ratio of 6 and adiabatic efficiency of 0.5 at the design point. The fan is driven by combustion just in front of the linked tip turbine. The turbine is assumed to be made of carbon-carbon composite. A pre-cooler in front of the fan is installed for three of the engine types, with the dual purpose of cooling the incoming air to protect the fan blades and increasing the airflow (without overheating the structure) into the engine to improve the maximum thrust. The two expander engines use the heat of the burner and the nozzle to expand the liquid hydrogen fuel to be burned at the tip turbine and in the main burner. Finally, one of these engines has a heat exchanger in the exhaust system, placed in between two fuel injectors. In this way the gas temperature around the heat exchanger can be controlled within design limits for the fuel and the metal. The use of carbon-carbon composite, ceramics, or refractory materials are considered for this exchanger.

These concepts were analysed extensively using assumptions of variable intake and nozzle and following design constraints: Temperature less than 1270 K for the fan inlet and precooler and 1470 K for the turbine inlet. These limits represent the near-term technology levels. The fuel is fed through the tip turbine and further through one or two injectors inside of the main burner located after the fan. Stoichiometric combustion in the main burner was assumed, whenever possible. The studied flight conditions are: Altitudes between 0 and 40 km and Mach numbers from 0 to 7.

According to Tanatsugu (1996), pre-cooling improves the absolute thrust level for all altitudes and speeds, most pronounced for speeds above Mach 4. When viewing specific thrust, the performance improves (higher values) over the whole operating range, except at the highest altitudes and the highest speeds. Heat exchanging in the burner has little effect on the absolute thrust, while for the specific thrust the curves are radically changed. In general, for altitude below 20 km and at practically all speeds the effect is tremendous, in many places the performance is doubled. Another observation is that the heat exchanger seems to improve the operating conditions above around Mach 6, when compared to cycles featuring only pre-cooling.

Of these four ATR concepts presented in Figure 44, the pre-cooled expander cycle engine with the heat exchanger (E-2 engine) has been chosen for the further research. Therefore we reproduce the performance graphs for this engine variant:

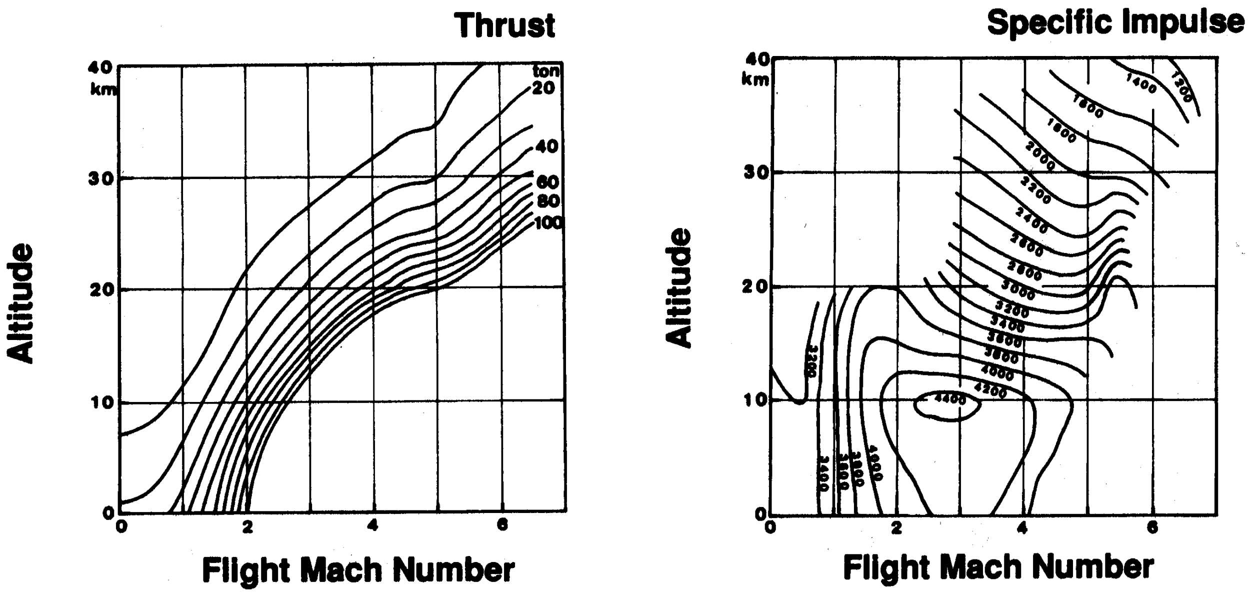


Figure 44. Maximum thrust contour and specific impulse contour for the precooled expander cycle engine with a heat exchanger in the main burner (E-2 engine) [from Tanatsugu, 1996].

3 Inlets

3.1 Inlet Problems for Different Regions on the Mach Number Scale

All aircraft design involves compromises trying to obtain minimum weight while satisfying a set of requirements. The inlet is no exception to this rule. The most basic requirement is to match the air mass flow to the needed engine power under variable flight conditions. A multitude of design details must be considered such as avoiding separation in the inlet where mostly a positive (adverse) pressure gradient is present.

In many designs there is a need for movable parts to allow a variable inlet area or high pressure relief through bleed channels etc. Further description of such devices can be had from the references. In the present text only an overview of the total intake design complex is presented. This only permits the presentation of some typical problems that a designer of air inlets must deal with.

3.2 Intakes for Subsonic Flight (Mach<1.0)

The basic task of the intake is to allow a suitable mass flow to the engine. Design problems typically also involve decelerating the flow to below Mach 0.4 - 0.5 under cruise or high speed circumstances. The inlet is typically geometrically optimized for cruise conditions. For low speed operation the flow may be forced to round the lips of the intake, which might include a very rapid pressure increase, thereby inducing separation near the lips and inside the duct. Under cruise conditions the design must include consideration of the pressure increase that accompanies the deceleration to Mach 0.4.

The positive pressure gradient is detrimental to good flow behavior, near the inner surface of the inlet, and the flow might separate if the duct is too short. However, as short a duct as possible, benefits low weight of the inlet. Diffusion/compression must be carried out with a moderate pressure gradient to avoid excessive separation. This, together with the required entry speed into the compressor, dictates the length of the diffuser. Typically an internal angle of about 10 degrees should not be exceeded. A slightly shorter inlet (shorter than optimal) might deteriorate the pressure recovery somewhat and allow an overall lower weight of the vehicle, which might benefit the design as a whole.

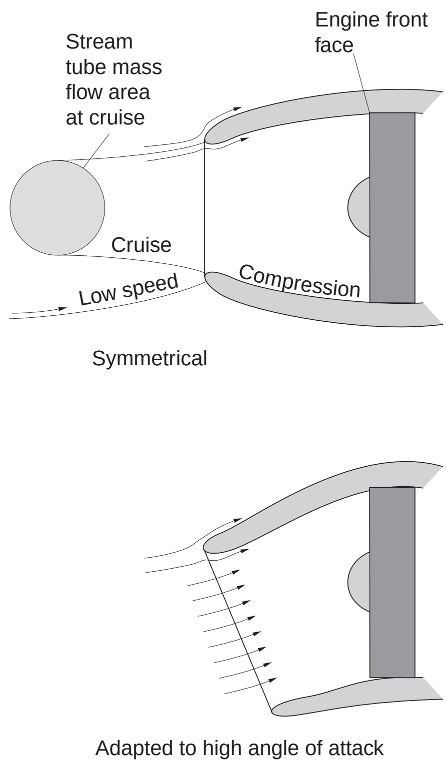


Figure 45. Subsonic inlets

Another design feature might be to adapt the intake for high angles of attack. This involves a compromise between optimality in cruise versus optimality at high angles of attack. The recurring problem of separation avoidance frequently also requires the intake to be sufficiently separated from the fuselage of the vehicle. Many airplanes therefore have a plate near the fuselage diverting the fuselage boundary layer, thereby letting only “fresh air” to enter into the inlet duct.

Stealth technology requires the radar signature of an air vehicle to be minimized. Areas of particular concern are those where two flat surfaces come together such as at the junction of a wing and the fuselage. The angle between the surfaces is simply avoided in a way which is characteristic of modern low signature airplanes. More specifically, the compressor is a powerful reflector for radar energy. It must, at the cost of aerodynamic efficiency, be concealed. One immediate solution is seen in Figure 46.

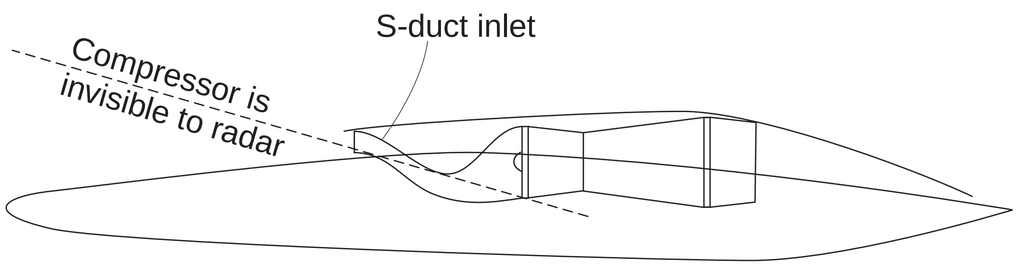


Figure 46. Stealth adapted inlet

The sketch in Figure 46 indicates the difficulties in maintaining a homogeneous “one-dimensional” deceleration in the S duct. The longitudinal curvature invites adverse pressure gradients in more than one spot. It requires careful wind tunnel development with CFD support. Such work was reported e.g. by Le Moigne (2004). A possible improvement, of the S duct flow quality, can be accomplished by the introduction of an inlet “guide” vane as explained in Figure 47.

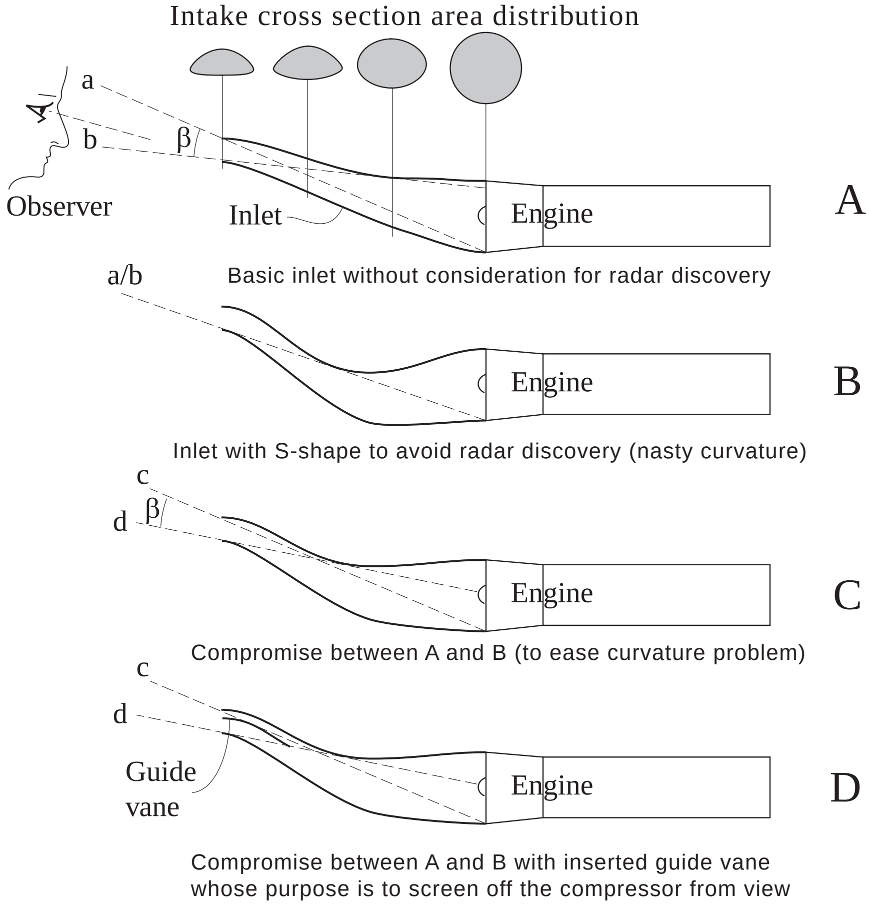


Figure 47. Possible improvement of a stealthy S-duct inlet.

The imaginary observer can easily see the compressor as long as the angle β is greater than zero. What immediately comes to mind as a remedy is to bend the intake duct as sketched in B, whose shape is a reflection of tests that have recently been carried out at the FOI Systems Division aeronautical laboratories – formerly the Aeronautical Research Institute of Sweden (FFA). Ongoing discussions have also touched upon the guide vane idea. This gives the inlet designer another design parameter, to be included in the optimization of the inlet geometry. The vane allows the intake shape to have a less pronounced curvature, which may be assumed to be like an interpolated mix between A and B. Even this allows a view of the compressor, although at a lesser angle, which represents an improvement in this respect, see Figure 47.

The completion of this improvement consists of the insertion of the guide vane as seen in D. It totally blocks the compressor from view. With respect to the shape of B, which is heavily prone to induce separation, the lesser curvature of D (or C) is obviously more aerodynamically benevolent in this respect. However, the vane will generate a boundary layer leading to a pressure loss in the duct flow, which is detrimental to good intake performance. The design thus should consist of a balanced medium between the vane geometry and the S shape of the duct. A close-up of the intake according to Figure 47D is seen below

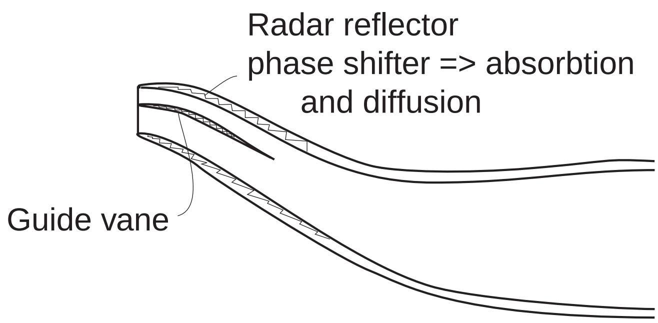


Figure 48. Intake with radar radar reflection treatment

The inlet aerodynamic wall surface can be made translucent to radar wave lengths. This opens up possibilities to insert arrangements for reflection, absorbtion and diffusion of the radar radiation energy. The overall logics of the radar reflection problem can be stated as follows:

There is no certainty that a geometry blocking the compressor from vision leads to complete radar absorbtion. The reverse can be stated more boldly, i.e. that if the compressor is not hidden from view then it certainly is not protected from radar reflection. Thus, visual unavailability is necessary but not sufficient for avoidance of radar detection.

The engine implications of the radar avoidance problem is that the stagnation pressure losses are increased significantly. An estimate of the size of these losses may therefore become an important issue in the study of military air vehicles.

It should be mentioned that the intake problems touched upon above may also be part of the design concerns for moderately supersonically flying vehicles. This is, however, not specifically treated in the sfollowing section.

3.3 Supersonic Inlets (Mach 1 -3)

The turbojet engine receives subsonic air into its compressor. It is the task of the inlet to provide the subsonic flow, regardless of the speed of the vehicle which it propels. The compressor typically requires an entry Mach number of 0.4 to 0.5. For best efficiency the thermodynamic cycle requires that the stagnation pressure be as high as possible for best efficiency of the inlet-engine combination. Additionally there is the general requirement, valid for all speeds, that the air entering the compressor face is evenly distributed in terms of speed and pressure.

In supersonic flow a direct entry into an inlet opening gives rise to a shock nearly normal to the flow. A single normal shock represents the worst solution if maximum efficiency is sought. It can be shown that an oblique shock followed by a final normal shock is more efficient, thus providing a higher pressure behind the final normal shock. A higher (total) pressure recovery is obtained with such an arrangement.

In Figure 42 **b** the design Mach number case is sketched. The stream tube swallowed by the intake is determined by the intake lip radius. For this flow case there is practically no pressure gradient from the apex of the cone to the inlet.

If the same inlet, as in **b**, see subfigure Figure 42 **c**, is subjected to a Mach number lower than the design value, the oncoming flow “gets a warning” before entering the inlet. This leads to the flow turning outward. Therefore the full design stream tube is not captured.

Depending on the back pressure from the jet engine compressor operation, the final normal shock may occur outside of the inlet, see Figure 42 **d**. This indicates the intimacy of the co-operation between the parameters of the intake and the engine.

Assuming isentropic compression, using a continuous curvature ramp or cone arrangement will give rise to nearly full pressure recovery for a very narrow Mach number interval, see **f**. Because of real life viscosity, however, the arrangement is very sensitive to off design conditions in which the basic functioning breaks down. The build-up of the boundary layer tends to create unplanned shocks, thereby destroying the intended pressure recovery. The isentropic inlet should be compared with the two-shock inlet, which has no pressure gradient on the cone surface. That is benign with respect to the ability to resist boundary layer build-up and shock separation.

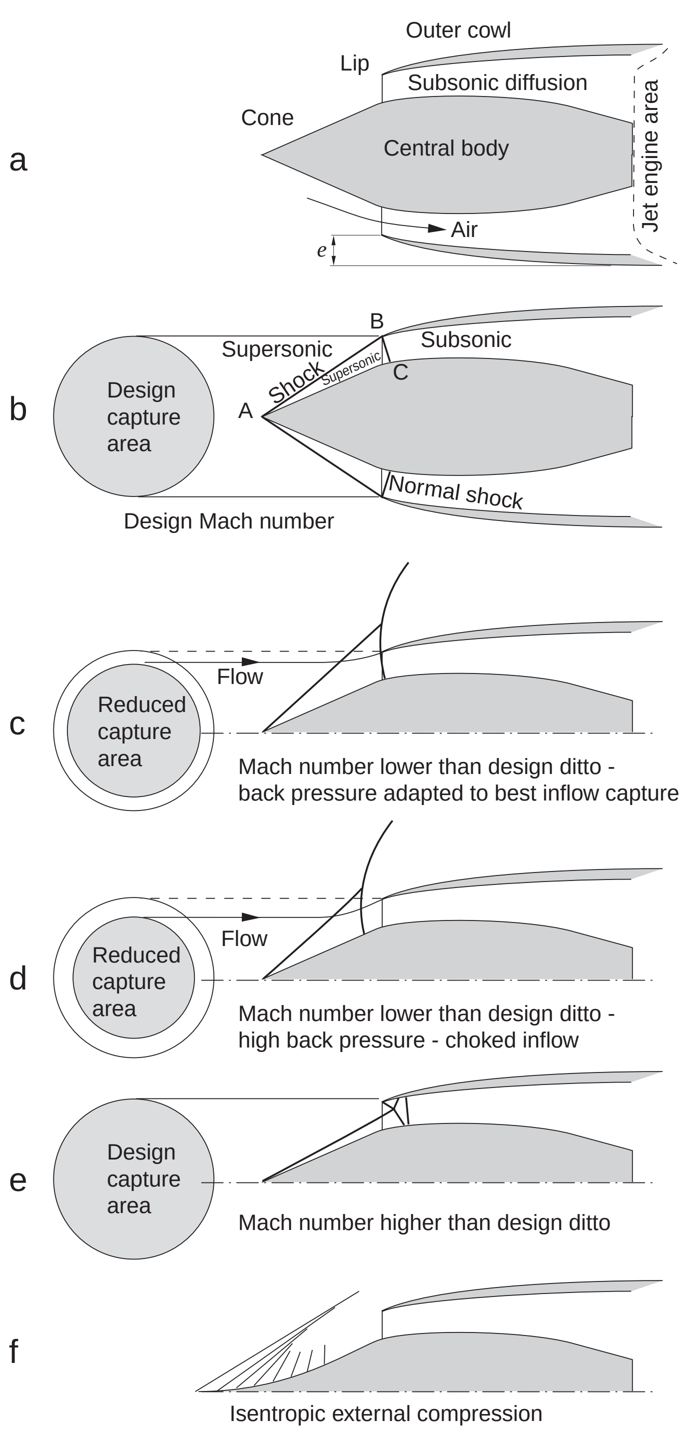


Figure 49. Supersonic inlets with external compression

The cone, sometimes referred to as a “spike”, gives rise to external compression. This is followed by an internal compression normal shock.

There are, however, a vast amount of design parameters and combinations of design parameters giving a multitude of design possibilities (and difficulties). One possibility is to let a larger part of the compression take place inside the intake, as seen in Figure 50. In comparison with the previous intake, the internal compression allows capture of more air for a given lip diameter.

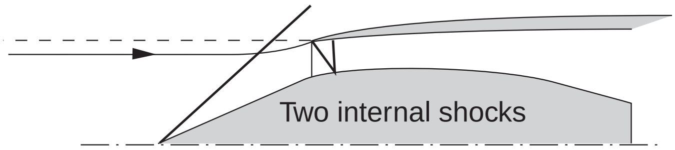


Figure 50. Internal compression.

The outer shape of the cowl must be designed for minimum wave drag. The measure *e*, seen in Figure 49 **a**, should be a compromise between zero, which would give zero form drag but unwanted internal compression shock placement and some angle, like that in the sketch, which might result in good internal flow and excessive external pressure drag.

Also flow control to the engine may include internal bleed-off of air and/or extra intakes of air perhaps representing the needs during takeoff.

An excellent overview treatment of the topic is found in Raymer (1999).

Although the discussion above is focused around the polar symmetric spike inlet, the very same principles are valid for so called two-dimensional (2D) inlets. They can have external compression, internal compression etc, as described above. One example is seen in Figure 51 on the experimental strategic bomber XB-70 Valkyrie, which, in the 60s, obtained Mach 3 at 70000 feet of altitude. The 2D intake compression ramps are seen to split the intake flow to the left and to the right, providing airflow to six General Electric turbojet engines (J93-GE 3) housed in the pod. The inlet does not deserve the denotation “2D” when the aircraft flies subsonically or at low supersonic speeds. At subsonic speeds all points influence all other points in the flow regime of interest and at low subsonic speeds the Mach cone is wide enough to engulf most of the intake area.

Figure 51. XB-70 Valkyrie with “2D” inlets.

3.4 Ramjet Inlet (Mach 2.5 – 7)

The ramjet propulsion solution works best in the approximate speed range from Mach 2.5 –7.0. Its air inlet principles do not differ much from those of the supersonic turbojet engine. The detailed geometry of the inlet is dictated essentially by the design mass flow and the Mach number. Inlet requirements for subsonic flows are non-existent since the ramjet combustion can hardly take place at subsonic speeds. Acceleration through the subsonic speed regime must therefore be accomplished by other means – such as using a rocket booster, which may be integrated in the ramjet combustion chamber. The ramjet air inlet can therefore be designed with the purpose of optimizing the intake for a certain Mach number. Thus, no particular compromising side glances at needs for subsonic flight speeds should bother the designer.

3.5 Scramjet Inlet (Mach 5 – 20)

Operation at high Mach numbers, say in excess of 5.0, causes very high temperatures when the air, after shock compressions is brought to subsonic velocity, reaches the combustion chamber. Fuel combustion will further increase the temperature, thus risking temperature levels where the combustion chamber walls would eventually melt in a standard design. The solution therefore includes features to suppress the incoming air temperature increase, such as avoiding compression to subsonic speed. The combustion then takes place in supersonic flow. The consequence is that the time from entry to exit, in the combustion chamber, tends to be very short. This will drive the combustion chamber design length in order to allow complete combustion before the flow reaches the exit.

Fuels that tend to burn more rapidly, than most, are favored by the designer. This points to the use of hydrogen rather than a liquid fuel. The latter must first break up its droplets into near molecular size before efficient mixing with air and combustion can take place. Such a process obviously takes time and combustion length and it therefore requires a longer combustor. In several studies in the literature hydrogen is proposed as fuel. Hydrogen is most easily available totally gasified once it enters from its injection nozzles. Therefore, and for reasons of inherent characteristics of hydrogen, mixing with the air is rapid allowing a relatively short combustion chamber.

Inside the combustion chamber the addition of fuel must occur as far upstream as possible to allow as long a mixing length as possible. The deceleration of the incoming air flow is accomplished through one or several oblique shocks. Good mixing and combustion efficiency, over a large speed range, is not to be expected. Rather the fixed air intake can function relatively well only within a narrow speed range. This speed range must be attained by other means than the scramjet engine itself. A typical solution is to employ an initial rocket for acceleration. For an extended speed range, and/or variable angles of attack, it may be necessary to use variable intake geometry arrangements.

The problem of having to allow the fuel to mix with the air was taken to the extreme by the inventive authors in Sislian *et al*. (2004). The fuel is injected at an upstream position on the shock inducing ramp well ahead of the combustion chamber as seen in Figure 52, which is not scale accurate.

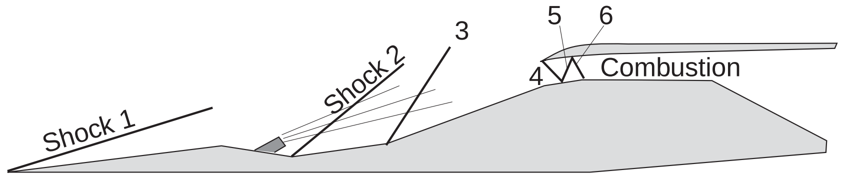


Figure 52. Supersonic combustion with fuel/air mixing in the external compression region. Several shocks occur externally and a few internally. The fuel injection is seen to happen ahead of shock 2.

As the flow is going through the chock decelerations the fuel mixes with the air and the last chock increases the temperature such that the air-fuel mixture ignites in the very beginning of the combustion chamber. The study referred to is a theoretical study where conclusions are drawn from results generated using CFD technique. The concept certainly requires testing before inclusion in the realm of reality.

Curran *et al*. (1999) includes several papers, which convincingly report that positive thrust was not generally obtained in laboratory testing. In one exception a small positive thrust was obtained. This attests to the difficulty to make the scramjet operational, which is partly due to the inflow, which must fit the purpose of the intake geometry. Thus, the shocks must behave as intended, meaning that the velocity, the angle of attack and the boundary layer interaction with the compression must all fit within narrow limits. Only then can the combustion take place.

The most successful test so far seems to be the Boeing X-43 scramjet vehicle, which was attached to the nose of a Pegasus rocket for initial acceleration to operational conditions. The combined rocket-X43 was airlifted and dropped from a B52 bomber modified to allow the attachment of the rocket-X-43 combination.

Upon ascent and acceleration the X-43 eventually reached Mach 9.8 (a world record speed) at about 100000 ft, being driven by its scramjet engine for ten seconds. In the series of tests that were carried out hydrogen was used as a fuel, while the conclusion was that the handling of the hydrogen was too complicated. Recommendations for the future were to use hydrocarbon fuels. The program was, however, discontinued before any such proposals were implemented.

Figure 53. X-43.

Figure 54. Wind tunnel model of X 43 showing inlet, combustion chamber and exit.

The fore-body of the X-43 serves as a “semi-two-dimensional” air inlet. In fact the whole vehicle and its engine are co-designed to provide lift and thrust in a most integrated way. Very few details of the design was made public. But, photographs are available. In the picture the whole vehicle is seen mounted on the rocket and the landing gear of the B52 is visible.

A better view of the intake is available from the wind tunnel model (same size as the actual vehicle) where the simple entry ramp appears before the combustion chamber. It is also imaginable how the pressure on all surfaces must be taken into account for moment balance of the vehicle. This includes the need to carefully design all external surfaces after the intake geometry has been defined.

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