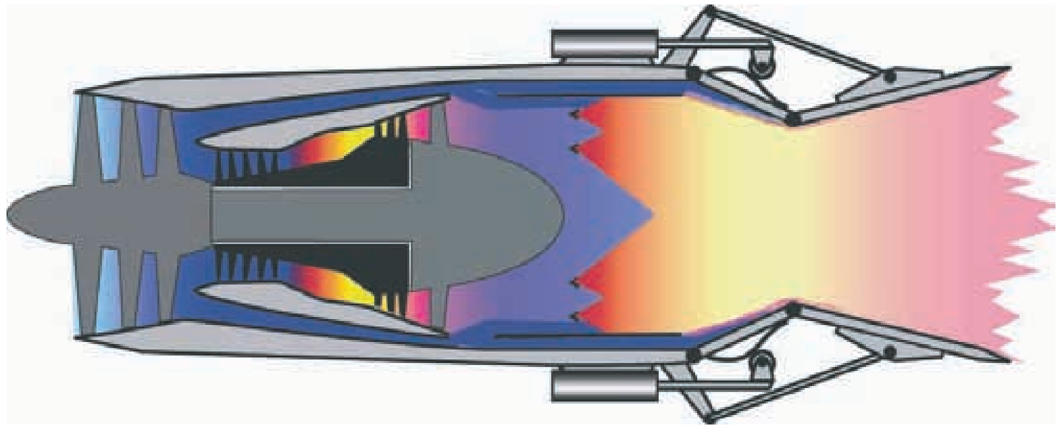
Design of a Turbofan Engine Cycle with Afterburner for a Conceptual UAV

BJÖRN MONTGOMERIE



Mixed flow turbofan schematic

FOI-R-- 1835 --SE Scientific report
ISSN 1650-1942 December 2005

Systems Technology

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FOI-R--1835--SE
ISSN 1650-1942
Scientific report
December 2005
Systems Technology

Issuing organization FOI – Swedish Defence Research Agency Systems Technology SE-164 90 Stockholm	**Report number, ISRN** FOI-R--1835--SE	**Report type** Scientific report
	Research area code 7. Mobility and space technology, incl materials	
	Month year December 2005	**Project no.** E830058
	Sub area code 71 Unmanned Vehicles	
	Sub area code 2	
Author/s (editor/s) Björn Montgomerie	**Project manager** Fredrik Haglind	
	Approved by Monica Dahlén	
	Sponsoring agency Swedish Defense Materiel Administration (FMV)	
	Scientifically and technically responsible Fredrik Haglind	

Report title

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Abstract

A study of two turbofan engine types has been carried out. The purpose was to find a suitable engine to serve as a propulsion unit for a conceptual UAV. The working tool for the preliminary design of the engines was a computer code called Gasturb10. The engines were equipped with afterburners with convergent/divergent nozzles. First a variable cycle engine was studied. This type of engine includes a technique called variable area bypass injection (VABI), whose purpose is to funnel a part of the air either as bypass flow or core engine flow. The simplified flying mission, used as a base for judging the suitability of the engine parameters, consisted of one cruise/loiter part with the afterburner unlit and an afterburner-on dash segment. The total fuel consumption served as an overall target function for optimization. After extensive testing of various settings of bypass ratios and compressor pressure ratios the emerging numbers for these ratios indicated that a standard engine would be equally useful for this particular purpose. This conclusion came as a surprise.

The continued effort was focused on a fixed cycle turbofan engine whose characteristics were similar to those of the more complicated VABI engine. The final result consisted of the thrust and fuel consumption tables for different Mach numbers and altitudes valid for the fixed cycle engine. Furthermore, some length dimensions, critical for installation, and weight are included.

Keywords

turbofan, VABI, Gasturb, bypass

Further bibliographic information	**Language** English
ISSN 1650-1942	**Pages** 30 p.
	Price acc. to pricelist

Utgivare FOI - Totalförsvarets forskningsinstitut Systemteknik 164 90 Stockholm	**Rapportnummer, ISRN** FOI-R--1835--SE	**Klassificering** Vetenskaplig rapport
	Forskningsområde 7. Farkost- och rymdteknik, inkl material	
	Månad, år December 2005	**Projektnummer** E830058
	Delområde 71 Obemannade farkoster	
	Delområde 2	
Författare/redaktör Björn Montgomerie	**Projektledare** Fredrik Haglind	
	Godkänd av Monica Dahlén	
	Uppdragsgivare/kundbeteckning FMV	
	Tekniskt och/eller vetenskapligt ansvarig Fredrik Haglind	

Rapportens titel

Framtagning av en turbofläktmotor med efterbrännkammare för ett UAVkoncept

Sammanfattning

En studie av två turbofläktmotorer har utförts. Syftet var att finna en lämplig motor för en viss UAV. Arbetsredskapet för studien var datorprogrammet Gasturb10. Motorerna har efterbrännkammare och konvergent/divergent utloppsmunstycke.

Först studerades en motor med variabelt bypassförhållande. Till denna relativt avancerade teknik knöts förhoppningen att motorn skulle kunna fås att arbeta tämligen optimalt både i transportdelen och högfartsdelen av ett hypotetiskt uppdrag. Den för uppdraget totala bränsleåtgången användes som optimeringskriterium då de olika motorcykeluppsättningarna jämfördes. Det befanns att den optimala konfigurationen inte kunde sägas utnyttja de potentiella fördelarna hos denna mera komplicerade motor. Detta förvånande resultat ledde till att en fix turbofläktmotor fortsättningsvis studerades. Resultaten från beräkningarna av prestanda, längmått och vikt, för denna motor, utgör slutprodukten av studien.

Nyckelord

turbofan, VABI, Gasturb, bypass

Övriga bibliografiska uppgifter	**Språk** Engelska
ISSN 1650-1942	**Antal sidor:** 30 s.
Distribution enligt missiv	**Pris: Enligt prislista**

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Main text

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1. Introduction

This report was created in a project called “Framdrivning”, which is Swedish for propulsion. The project was launched to become a learning experience of the different air vehicle propulsion options that are known today. The present text, however, reports a studyaimed at the developmemt of an advanced engine for a UAV of a companion project acronymed “KoKoS”. The KoKoS configuration and characteristics are briefly touched upon in terms of their interface to the engine requirements. The tool used for the engine thermodynamic cycle evaluations has exclusively been the Gasturb10 computer program, see Ref. 6. Two types of bypass turbofan engines were studied. Other concerns of this study are those of sizing the engine and estimating its weight.

2. The Variable Cycle Engine with Afterburner

Experimentation with the variable cycle was carried out by the General Electric company from about 1985 to 1995. Two engines, YF119 and YF120 competed internally within GE for application partly in the JSF airplane. Further development took place with Rolls-Royce as a partner.

The Gasturb10 program offers many types of turbojet and turbofan engines. The variable cycle turbofan engine, seen below, was initially chosen for its assumed flexibility to adapt to widely varying flight envelope circumstances. The flexibility is implemented through the use of three openings, not present in most engines, which can close out or allow certain bypass streams. The engine can then operate in different modes, as determined by the state of the bypass doors, for best performance.

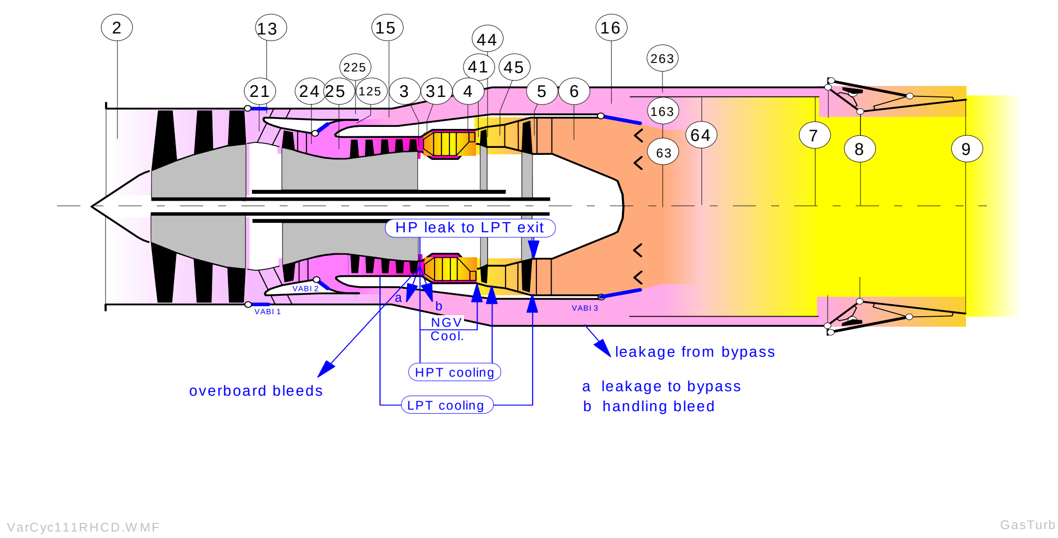

Figure 1 - Engine principles and the station definition numbering system

The engine station numbering system of Fig. 1 is applied throughout the text. The schematic view of the engine, copied from the Gasturb10 program, also shows the principles of the variable cycle. There are two shafts, one inside the other. The outer shaft (= a tube) carries the “high pressure” components, compressor and turbine, where the turbine is directly subjected to the hot flow from the burner. The heat of the flow, remaining after passage of the high pressure turbine, is then partially used by the low pressure turbine to drive the fan package (inner shaft). The heat remaining is sufficient to accelerate the still hot gases out through the nozzle. This acceleration can further be boosted by fuel injection in the afterburner. This increases the engine thrust considerably at a heavy expense of increased fuel consumption. The afterburner occupies the engine between stations 63 and 7.

In this particular engine there is a closable outlet for bypass air at station 13 and at station 125. The closable opening for the former is referred to as VABI 1, the latter is associated with the notation VABI 2, where VABI is short for Variable Area Bypass Injection. These VABIs can either be closed or fully open. There are presently no intermediate VABI 1 or 2 positions available in the program. The rear part of the core engine closes with VABI 3 allowing variable mixing of the hot flow and the "cold" flow.

The outlet modeling includes the possibility to use a convergent-divergent nozzle (as in Fig. 1). The cross section area A_8 , at station 8, determines the mass flow, provided the Mach number at that section equals one. Therefore A_8 is mostly (except possibly for idling situations) uniquely defined by the flow equations in the program. The outlet area A_9 is, however, available to the user's experimentation. The supersonic expansion to the ambient pressure is a goal since this corresponds to maximizing the thrust.

As it turns out the VABIs 1 and 3 were best left open in all situations while VABI 2 mostly, but not always, gave the best thrust and fuel consumption if it was left closed in dash and open in cruise.

Numerous runs with parametric variations of the miscellaneous input variables were carried out in this project. The variable cycle engine type was used in these. The Gasturb10 optimization feature of the design mode was relied upon extensively. The output surprisingly showed a discouraging recurring feature, which did not speak in distinct favor of this type of engine. The reason was that the dash requirement, se below, gave approximately the same maximum thrust whether VABI2 was open or closed, although mostly with a small favor given to a closed state. Under no circumstances was it advantageous to close the other VABIs. The cruise requirement always favored VABI2 being open. Moreover, the intermediate compressor (between stations 21 and 24) was found to be best dimensioned with a pressure ratio just barely above 1.0. This is equivalent with having no compressor at all at that position. All of these observed circumstances raised the question why the variable cycle engine should be used. The mixed flow turbofan engine with afterburner was in fact mimicked by the variable cycle engine. Therefore the later part of the study was devoted to runs with the latter engine.

3.The Mixed Flow Turbofan Engine with Afterburner

The figure below shows a schematic of the mixed flow engine. Its features are identical to those of the previously discussed engine if all VABIs are closed and the intermediate compressor is eliminated. The computer runs did, as expected, return results that were very similar to those of the more complicated variable cycle engine. In the appendix, describing the numerical output, only one configuration is presented. Its main characteristics, resulting from the design optimization, can be captured in a few key variables as follows. The numerical values are those that were arrived at in the optimization.

Typical variables for optimization:

Inner fan pressure ratio	2.6
Outer fan pressure ratio	2.85
High pressure compressor pressure ratio	9
Bypass ratio	1.4

The maximum temperatures applied were

Max burner temperature	1900°K
Max afterburner temperature	2100°K

A power off-take of 100kW in all calculated points was assumed in order to approach a modest degree of realism although in reality this figure is probably highly variable.

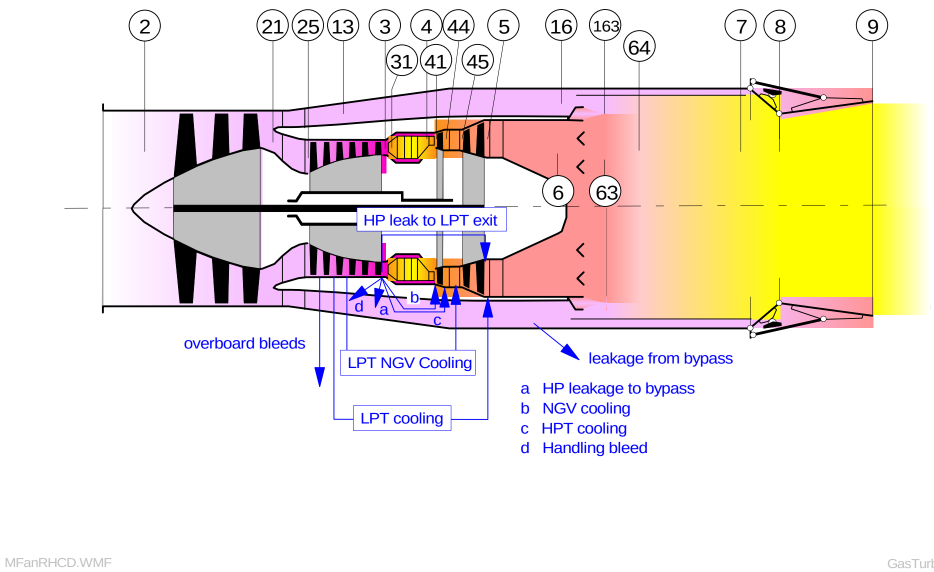

Figure 2 – The mixed flow turbofan model; a more realistic picture appears on the cover

4. Connection with the Airframe of the "KoKoS" project

A UAV preliminary design effort is conducted in another project called KoKoS in parallel with the propulsion project. It was seen as most appropriate to link the propulsion studies to those of the KoKoS program. In order to adapt an engine description to a preliminary design of an airplane certain key requirements on the airplane become the determining factors also for the choice of engine characteristics. Such requirements are typically take-off distance, landing distance, climb rate, a G acceleration limitation and economy at cruising altitude. For a military airplane also an afterburner dash capability and a turning capability, might add to the list.

The KoKoS mission is seen in the figure below.

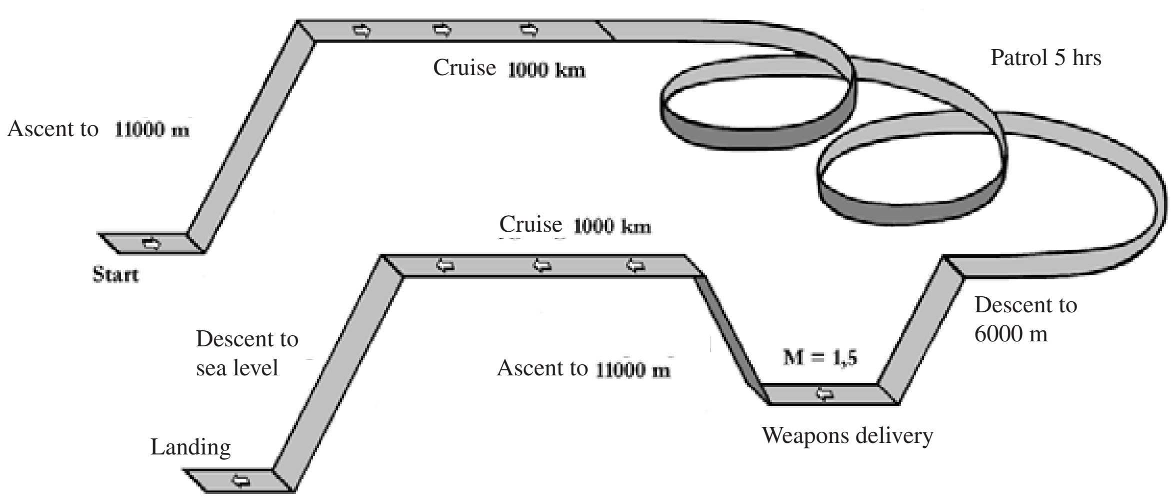

Figure 3 – Mission definition for the KoKoS project

In the present KoKoS UAV case a subset of these requirements were to be met by the engine. They were

1. Cruise and loiter at H=11km, M=0.7 during 381 minutes requiring 16kN of thrust
2. Dash with full afterburner at H=6km, M=1.5 during 5 minutes at 300kN of thrust
3. The engine must be operational at all "reasonable" combinations of altitude and Mach numbers. This includes, most prominently, the take-off conditions.

In the simplified mission adopted for designing the thermodynamic cycle of the engine requirement, Nr 1 consumes about 2/3 of the mission fuel and Nr 2, the dash part, consumes the rest. The sum of the two, i.e. the total mission fuel consumption, constitutes the target function to be optimized. This optimization can be carried out as repeated runs using different sets of input variables, in the Gasturb program, trying to find a few successful candidate engines. Then the mission fuel consumptions from all of these varieties are compared and the one with the lowest consumption becomes the "winner". Presented in this report is, however, only one such winner. Any subsequent runs of the Gasturb10 program for this purpose will probably essentially belong in the same category of studies, which this report describes. Therefore, the final choice of engine cycle delivered to the KoKoS program may be different from the one presented here. This report will hopefully serve as an overall description even of such future studies.

It is pointed out that a better preliminary design method is to co-optimize the airframe and the engine. As an example the design iteration process includes drag adjustments between iteration steps. The drag is to be overcome by the engine thrust. The drag figure would then vary around 300kN for the dash case. The main reason for this variation is that the engine diameter would vary and the hull of the UAV would cause a corresponding variation of the airframe cross sectional area as a consequence. This would affect the wave drag noticeably. For practical reasons, however, a division of tasks between projects was adopted as the practical work method, which is adequate for this preliminary design purpose. Thus, the simplified mission definition, seen above, was not changed during the engine cycle development.

5. Optimization Technique

The Gasturb program has an optimization feature where a maximum of 7 variables may be included in the same run. A wide choice of target functions to be maximized or minimized exists. Running the program in design/optimization mode allows the user to view the progress in the target value and there is an animated view also of the tested optimization variables. It is emphasized that it is within the capability of the program to optimize for a one-point design only. The real problem is, however, typically more complex. In the calculations carried out in this study two design points consisting of required Mach number, altitude and thrust were the base for the optimization. In order to accommodate this double requirement in the one-point optimization capability of Gasturb some inventiveness was called for. Two schemes were devised and tested.

5.1 Technique #1

First a one-point optimization design was run for requirement number 1. In this case it was necessary to apply the afterburner. This was done using a very small amount of afterburner temperature rise. Then a full power optimization run for the number 2 requirement was carried out. This yields two quite different engines. One looks like a typical passenger airplane engine with a big fan with a very high pressure increase over the core engine compressor. The other looked more like a typical military engine with a small (negligible) bypass ratio and a relatively low pressure rise over the core compressor. Then an intermediate engine could be created by interpolation such that each participating optimization variable is interpolated. To explain the details the reader can think of e.g. the bypass ratio. It can be denoted b_1 from the first point optimization (the cruise case) and b_2 would be the bypass result from the 2^{nd} (the dash) optimization. Using an interpolation number called α, in the interval $0 < \alpha < 1$ the interpolated value for the bypass ratio is achieved through: $b = \alpha * b_1 + (1- \alpha) * b_2$. By variation of α, subject to the interval restriction, any intermediate value between b_1 and b_2 can be calculated. This was done for the whole set of optimization variables. The result from a sample calculation using the variable cycle engine is seen in the following table.

Table 1 – Example of interpolations to find the optimum cycle combination

Design	H=11km	H=6	Interpolated values using weight factors α					
conditions >	M=0.7	M=1.5	0.6	0.55	0.5	0.45	0.4	0.3
Altitude	11000	6000	9000	8750	8500	8250	8000	7500
Mach	0.7	1.5	1.02	1.06	1.1	1.14	1.18	1.26
Inner fan PR	1.63473	3.459	2.364438	2.4556515	2.546865	2.63808	2.72929	2.912
Outer fan PR	1.88025	3.4766	2.51879	2.5986075	2.678425	2.75824	2.83806	2.998
IP Compressor PR	1.20116	1.0003	1.120824	1.110782	1.10074	1.0907	1.08066	1.061
HP Compr PR	20.5657	6.0613	14.763956	14.038738	13.31352	12.5883	11.8631	10.41
Des BPR W13/W21	0.955833	0.0163	0.5800014	0.53302245	0.4860435	0.43906	0.39209	0.298
Des BPR W15/W25	7.52312	0.3326	4.646902	4.28737475	3.9278475	3.56832	3.20879	2.49

The two design criteria are seen in the left part of the table. The prominent differences are seen in the high pressure compressor ratios and in the bypass ratios. The intermediate "engines" appear in the right part of the table. These were run in Gasturb in off-design mode as inputs. They were not all successful in the sense that they would be flyable at dash conditions. The leftmost intermediate varieties did not function for lack of thermodynamic cycle convergence. Those to the right, i.e. the 0.3 to 0.5 columns gave answers from the program. The most fuel economic cycle, after mission fuel consumption calculation, was for $\alpha = 0.5$. It turned out to have a very large diameter, which has no consequence in the pure engine investigation but it punishes the KoKoS vehicle with bulk. Bulk translates to wave drag.

As was to be found later, the mission fuel of the best variety in the table was inferior in comparison with subsequent cycle combinations of optimization efforts.

5.2 Technique #2

The learning from the first exploration of the optimization possibilities lead to the next. Instead of optimizing for the two utterly disparate requirements an intermediate design point was tested. Thus the burner and afterburner temperatures were chosen to be intermediate (between idle and full power). The design values of altitude and Mach number were interpolated between the corresponding values of the KoKoS requirements. Typically numbers around H = 9000 m of altitude and Mach = 1 were used as the program design base for cycle variable optimization. This turned out to be successful in the sense that both conditions for cruise and dash (see Sect. 3) could be satisfied in the off-design mode of the program. The study, which included several variations of the engine temperatures, gave rise to an engine cycle that was better than the one resulting from the first optimization. These studies were carried out still employing the variable cycle engine. Thus the good version could satisfy all three KoKoS requirements and had smaller dimensions than that of the first optimization effort.

A scrutiny of the input variables, which were optimized in the described manner, revealed what is described in the last paragraph of Section 2. I.e. they could also be produced by a simpler engine without the VABIs. The continued optimization effort was equal to the one just described. The key variable values were presented in Sect. 3. The engine performance numbers from these calculations became the final output delivered to the KoKoS project. Some numerical results, from the output set, are presented in Section 8.

6. Engine Dimensions

It is assumed that the airframe configuration air intake has brought the air velocity down to a Mach number less than one, at station 2. The intake cross section area, perpendicular to the oncoming flow, is A_1 and A is the compressor face area. The flow through A is a concern of the engine analysis while the flow before and at A_1 is not. From an engine design standpoint the flow is given at station 2. If the flight Mach number is greater than 1.0 there is a formula in the program calculating an approximation for the total pressure at station 2. The user may also consider using a diffusion loss factor on total pressure. Thus it is up to the intake designer to allow the mass flow, specified by the engine designer, to enter through area A_1.

The KoKoS mission includes requirements, which directly affect the engine design variables. One such variable is the required thrust (to balance the KoKoS drag) at specified altitude and Mach number. The engine input includes the total air mass flow. It can not be known in advance before the calculation has taken place. Instead the program standard input file default number of 50 kg/s is retained initially and the resulting thrust is assumed to be directly proportional to the air flow. After the calculation the required thrust is divided by the output thrust. This quotient constitutes a scale factor on the engine. In the next run the input air mass flow is scaled using this factor. A small adjustment of the air flow may be required in a subsequent run before the target thrust requirement is hit.

The mass flow is available in the Gasturb10 output. For the variable cycle engine, and later for the mixed flow fix cycle turbofan, the area *A* was found not to be available in the output. This section is therefore partly devoted to the deduction of the size of this area.

If the flow *W*(kg/s) and the state of the air at station 2 are known the cross section flow area at the engine entry (*A*) can be calculated. From the study of several engines from the Jane's reference, an overall estimate of the external measures can be found by means of scaling. The intake area diameter thus has an important role in the estimation of the engine dimensions.

The station 2 velocity or Mach number, however, is not directly available in the output because it is not needed in the engine equations, which should surprise the newcomer to the technology. The engine entry Mach number is a function of the flight Mach number and the diffusion in the air intake to the engine. The design of the intake constitutes no part of the engine thermodynamic cycles, which explains why the creator of the Gasturb program does not furnish the entry velocity. Instead an educated estimate of the Mach number at station 2, i.e. *M,* is used for design circumstances as elaborated below. When the engine operates off-design considerable deviations from the design Mach number may occur. The off-design is of no algorithmic consequence for the dimensioning of the engine, however. The engine diameter that matches the air mass flow through station 2 can be calculated as follows.

From the equation of continuity

$$A = \frac{W}{\rho V} \tag{1}$$

The general gas law relates density (ρ) to pressure and temperature (°K) as follows.

$$P = \rho RT \tag{2}$$

The speed of sound is generally given by

$$a^2 = \gamma RT \tag{3}$$

Substitution, using the definition for Mach number ($M=V/a$) and inserting (2) and (3) in (1) gives

$$A = \sqrt{\frac{R}{\gamma}}\frac{W\sqrt{T}}{P\,M} \tag{4}$$

The output from Gasturb10, however, offers the total temperature and the total pressure only at station 2. Expression (4) therefore needs to be converted to include the totals. The following isentropic relationships are therefore used.

$$T = T_0\left(1+\frac{\gamma-1}{2}M^2\right)^{-1} \qquad (5)$$

$$P = P_0\left(1+\frac{\gamma-1}{2}M^2\right)^{-\frac{\gamma}{\gamma-1}} \qquad (6)$$

Inserting (5) and (6) in (4) and using the standard atmospheric values for dry air R = 287.1 [J/(kg°K)] and γ = 1.4 gives

$$A = 14.32\frac{W\ \sqrt{T_0}\left(1+0.2M^2\right)^3}{P_0 M} \qquad (7)$$

Eq. (7) can now be numerically evaluated from the output results from Gasturb10 requiring also an assumption of the numerical value of M, normally appearing in the interval $0.4 < M < 0.6$ according to Walsh and Fletcher (Ref. 7). The value chosen below is thus somewhat arbitrary.

Application of the formula for the chosen engine includes using the output from the thermodynamic calculations. The *required* output (notably the maximum thrust) for the dimensioning case, i.e. the *dash requirement*, is presented below together with the definition of notations.

Notations pertaining to station 2 and dash requirement numbers:

A = Air entry cross sectional area, m^2.
W = 295.4 kg/s; Mass flow entering the air intake
a = Speed of sound
M = 0.55; Subsonic Mach number (after all the shocks have been passed)
γ = Cp/Cv - ratio between specific heats for const pressure and const volume respectively
m = The molar weight of air
R = The gas constant per mol
T = Static temperature, °K
T_0 = 361.16; Stagnation temperature, °K
ρ = Density of the air, kg/m^3
P = Static pressure
P_0 = 166.462; Stagnation pressure

Using these numbers in Eq. (7) gives A = 1.047 m^2 corresponding to a diameter = 1.15 m. A general impression, scrutinizing the Jane's photographs of military engines, this internal diameter should be multiplied by a factor in the interval 1.12 < factor < 1.26 to allow for a powerful afterburner including its nozzle mechanism. The latter tends to have the largest outer diameter. In summary it can be speculated that a factor of 1.2 can be used, which would give the engine a maximum external diameter of 1.2*1.15 = 1.39m.

Also to be noted is that the auxiliary equipment, typically occupying the lower part of the engine (the engine "belly"), has a vertical extent of about 70% of the engine compressor face radius. The orientation, of the auxiliaries to the belly of the engine, can of course be changed at the design stage to a lateral placement where spare volume might be more generously available.

Finally the length of the engine can be estimated from the Jane's photographs where a slenderness ratio seems to be between 4 and 5. This suggests that the KoKoS vehicle should allow an internal engine length of 4.5*1.39 = 6.2m.

7.Weight

The weight of an engine can be calculated using a formula given by Torenbeek, see Ref. 2. It appears below in a slightly different form adapted to the metric system units.

$$m_E = \frac{10 \cdot OPR^{0.25} \cdot \dot{m}}{1 + BPR} + 0.12\frac{F}{g}\left(1 - \frac{1}{\sqrt{1 + 0.75BPR}}\right) \tag{8}$$

where

m_E = Engine dry mass, kg
OPR = Overall pressure ratio
BPR = Bypass ratio
$\dot{m}$ = mass flow, kg/sec
F = Maximum dry thrust, N
g = Acceleration of gravity = 9.81 m/s^2

I can be noted that no spatial dimensions are needed in the weight formula.

One source of engine information was Jane's, see Ref. 4. Among 17 engines investigated by Svensson, see Ref. 1, and by this author the following statistical observations could be made.

The comparison, concerning the commercial engines, between the weight numbers given in the reference and those calculated, using the formula, by average and standard deviation, underestimates the weight by 12%. All engines, except for the last four military engines, were designed for civil aviation. It seems that the weight formula grossly overestimates the military engine weights. Further investigation is apparently necessary to improve the quality of the conclusions from the statistics for the military engines. The formula could e.g. be expanded to include an overall factor reflecting the military status. For details of the comparison refer to the spread sheet presentation of the engines in Appendix 2. From this Appendix it can be concluded that the excess from the commercial engine weight estimate varies from 1.6% to 57%. This corresponds to a factor on the calculated weight from 0.98 to 0.64. The confidence in any guesstimate between the two numbers is of course very low. But, the average deviation is 28%, which corresponds to a factor on the formula weight of 0.78.

Since the weight formula requires sea level standard condition and zero speed at maximum dry thrust (afterburner off) the off-design result for this condition is reflected in the following numbers from Gasturb10.

Max dry thrust =142000N
Mass flow = 222kg/s
OPR = 28
BPR = 1.4

These values inserted in Eq. (8) gives the engine mass = 2127.8 + 523.8 = 2652 kg. Considering the reduction factor of 0.78 for military engines the final weight estimate is 0.78*2652 = 2068kg.

8. Thermodynamic Calculations Using Gasturb10

8.1 Preparations

An effort was made to find a set of input variables where the bypass ratios were to be as large as possible for mission fuel consumption reasons. The variable cycle was studied initially. As it turned out only modest bypass ratios were found to function throughout the mission flight regime envelope. The key characteristics, listed below, were obtained in the design mode of the program, by means of an automatic optimization feature of the program. They were the output in response to an *intermediate* requirement between cruise and dash because both cruise and dash can not be used as an optimization basis. If the cruise circumstances are taken to be the basis for the engine key variable optimization the dash requirement can not be satisfied. This is frequently made known to the user of the program by short messages that say things similar to "Your data combination is no good" or "the mixer Mach number exceeds 1". Generally there is little support to lead the user toward a direction to solve the problem. The user must either guess or be quite educated in the engine technology field in order to find an alternate set of key input values, which will satisfy the requirements. Once a combination of input variables has been found the optimization run can be started and be brought to a conclusive end.

After lengthy manipulation of the input key variables, an engine cycle resulted, which satisfied both cruise and dash requirements. Moreover, it could operate at sea level. Its capability to operate in a reasonably large Mach/altitude envelope was by no means guaranteed at this stage. It must be tested point by point. (A *point* is here defined as the calculation determined by a pair of input variables consisting of Mach number and altitude for the given engine.)

The most important input key variable values, suitable to be subjected to optimization, were found to be:

Inner Fan Pressure Ratio	6.00	HP Compressor Pressure Ratio	6.7755
Outer Fan Pressure Ratio	5.865	Design Bypass Ratio W13/W21	0.1
IP Compressor Pressure Ratio	1.0679	Design Bypass Ratio W15/W25	0.3

They were subject to the fixed input values for burner exit temperature (T_4=1900°K) and an afterburner low temperature of 1300°K. Altitude and Mach number were chosen to be 10km and 1.0 respectively. This intermediency between requirements should be viewed in the light of the two design point criteria as elaborated below. It is pointed out that these numbers are the result from a sub-optimization the ultimate optimization being determined by the mission fuel consumption carried out in an outer manual loop.

The discouraging low bypass ratio that became the result was the reason for the continued study being conducted around the fixed bypass turbofan (cold and hot streams mixed inside the engine, see Fig. 2.

8.2 Off-design Calculations

Since the fixed or variable bypass engine was scaled by the dash requirements quite a big engine became the result. The engine design was then run in off-design mode with a series of T4 temperatures as an input with the afterburner turned off. The resulting thrust values were then interpolated for 16kN (mission requirement), which gives the corresponding T4 value and fuel consumption. All VABIs were open.

Using the times and fuel consumptions for dash and cruise (in fact a loiter requirement was integrated together with the cruise) it is possible to calculate the total mission fuel consumption measured in kilograms.

8.3 Summary of the Calculations

A summary of important numbers for the mixed flow, fixed bypass ratio engine, including the total fuel consumption calculation, is presented in the table. The table contains a parametric variation in order to meet the very modest *cruise* condition requirement. The burner exit temperature, referred to as T_4, was therefore varied. The net thrust (FN) and the specific fuel consumption (SFC) vary accordingly. It should be kept in mind that in order to reach the dash requirement thrust figure of 300kN (see the mission requirements of Sect. 4) the intake mass flow is adapted accordingly. Although the dash requirement did not constitute one of the Gasturb formal design criteria it does set the large mass flow (=large engine) as elaborated in Sect. 5. As seen the cruise requirement of 16 kN occurs at the very low value of T4 = 1235°K.

Table 2 – Compact performance summary

T4	FN	SFC			Duration,sec	Fuel cons.	
1900	43	25.26		Cruise+loiter:	27700	9266.293	kg
1700	35.52	23.48		Dash:	300	4910.283	kg
1500	28.89	21.66		Total mission fuel =		14176.58	kg
1300	19.67	20.607					
1235	16	20.9077					
1200	14.19	21.2					

The afterburner-OFF numbers, from the table, are plotted in Fig. 4. It is satisfying that the specific fuel consumption need occurs at or near its minimum. This is conducive to low mission fuel consumption.

Without knowledge of the power need onboard the UAV 100kW of power was assumed and the default method was assumed for its extraction. It is not clear from what source (1st shaft, 2nd shaft or air vent) the 100kW were derived. The program allows the user to specify from which shaft to extract the onboard power. When this is not utilized an automatic choice is made and the power extraction can always be specified regardless of source.

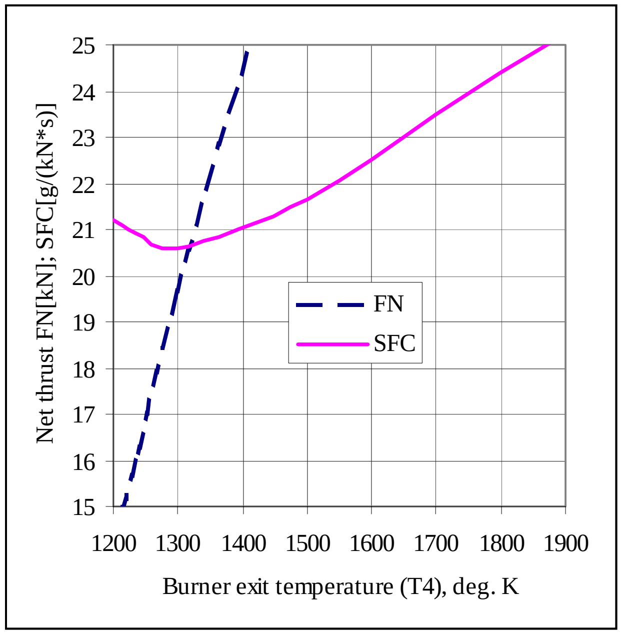

Figure 4 – Thrust and specific fuel consumption

A pictorial part of the output is presented below, while the actual numbers for input and output in the form of thrust[kN] and thrust specific fuel consumption TSFC [g/(kN*sec)] are placed in ascii file(s) not visible in this document.

The diagrams of thrust and fuel consumption, on the following pages, enclose all points that were successfully calculated for the afterburner lit cases, where only one temperature (=1900°K) for the main burner was used.

All diagrams were copied from the Gasturb10 output.

It may be disturbing to see that some calculated points fall outside the workable diagram region. Upon scrutiny, however, it was found that they correspond to combinations of altitude and Mach number, which can be avoided. One example of such a point is the take-off condition where a restriction on primary burner temperature may be imposed. Otherwise the low pressure shaft will overspeed. The afterburner may be employed to raise the take-off thrust to a satisfactory level. It is generally the low Mach numbers, combined with high burner temperature , which cause the high RPMs.

Afterburner ON

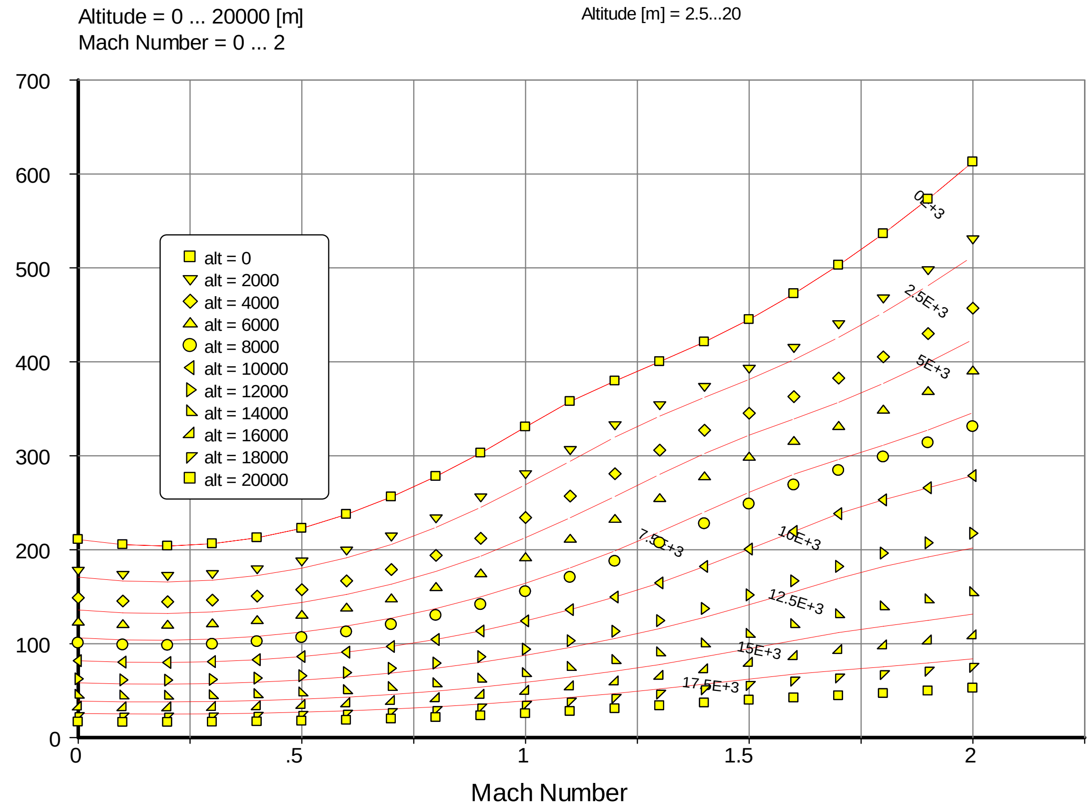

Figure 5 - Afterburner on – Thrust, kN - FN = FN(M,H)

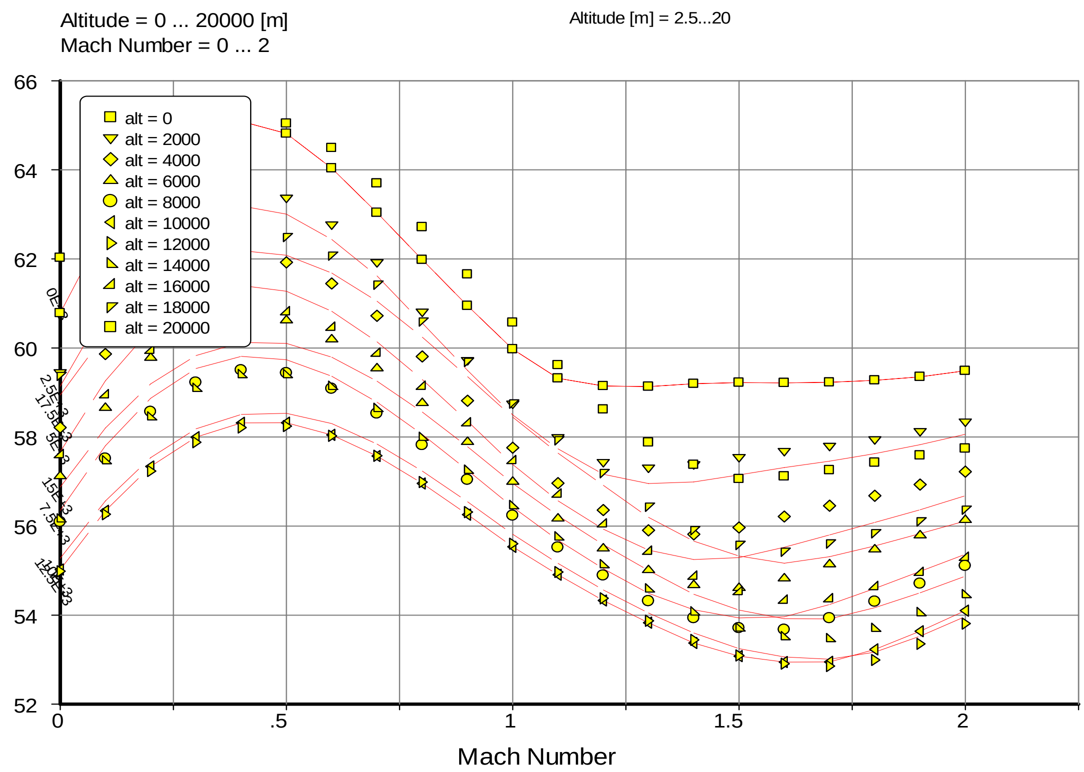

Figure 6 - Afterburner on – Thrust specific fuel consumption - TSFC = TSFC(M,H)

The following diagrams show examples of the off design data in the form of so called “maps” for compressors and turbines with the performance points represented by symbols. Not all combinations of H and M are thermodynamically feasible for engine operation. It is seen that the points generally cluster reasonably close to the high efficiency region.
Abbreviations, for the interpretation of the maps, follow first:

LPC = Low pressure compressor
HPC = High pressure compressor
IPC = Intermediate pressure compressor
HPT = High pressure turbine
LPT = Low pressure turbine

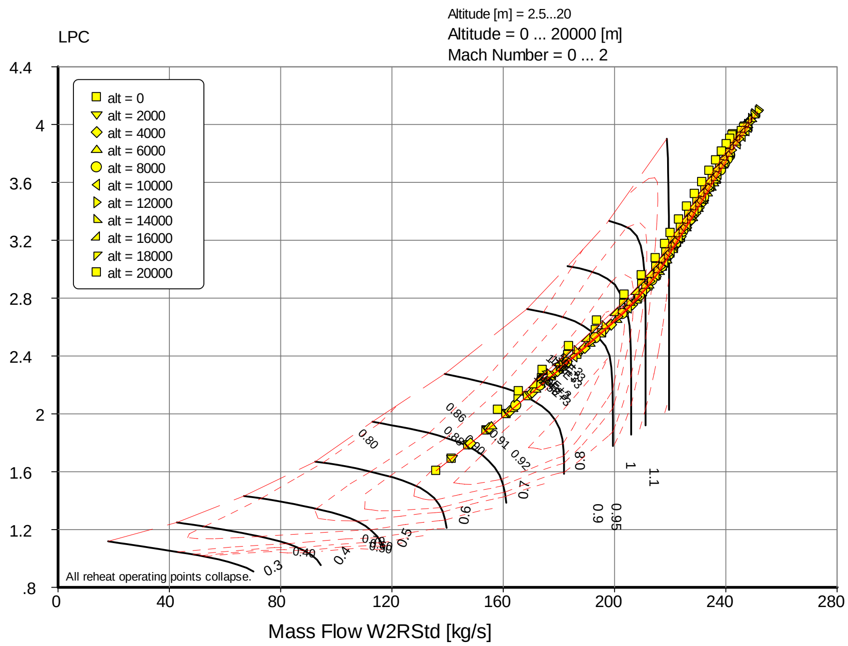

Figure 7

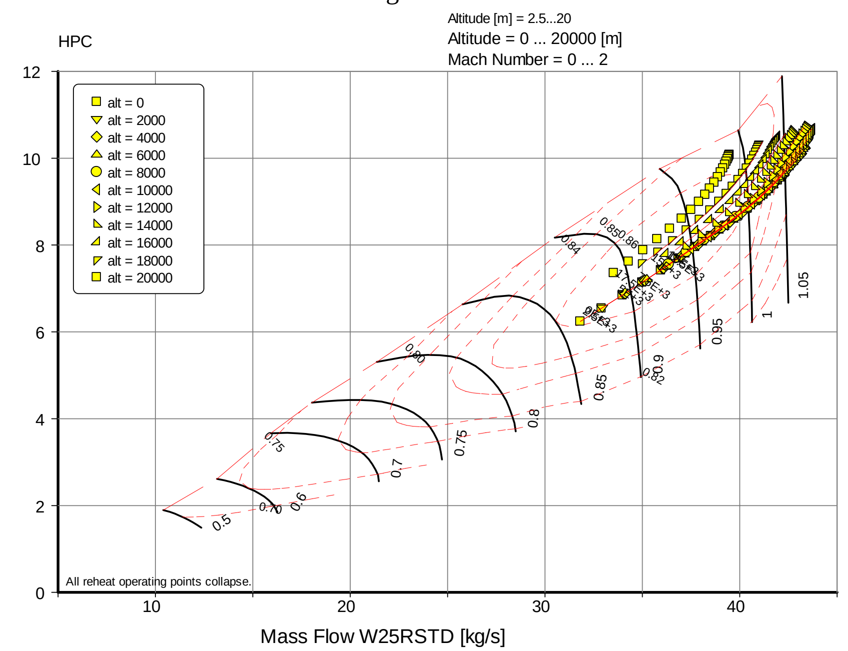

Figure 8

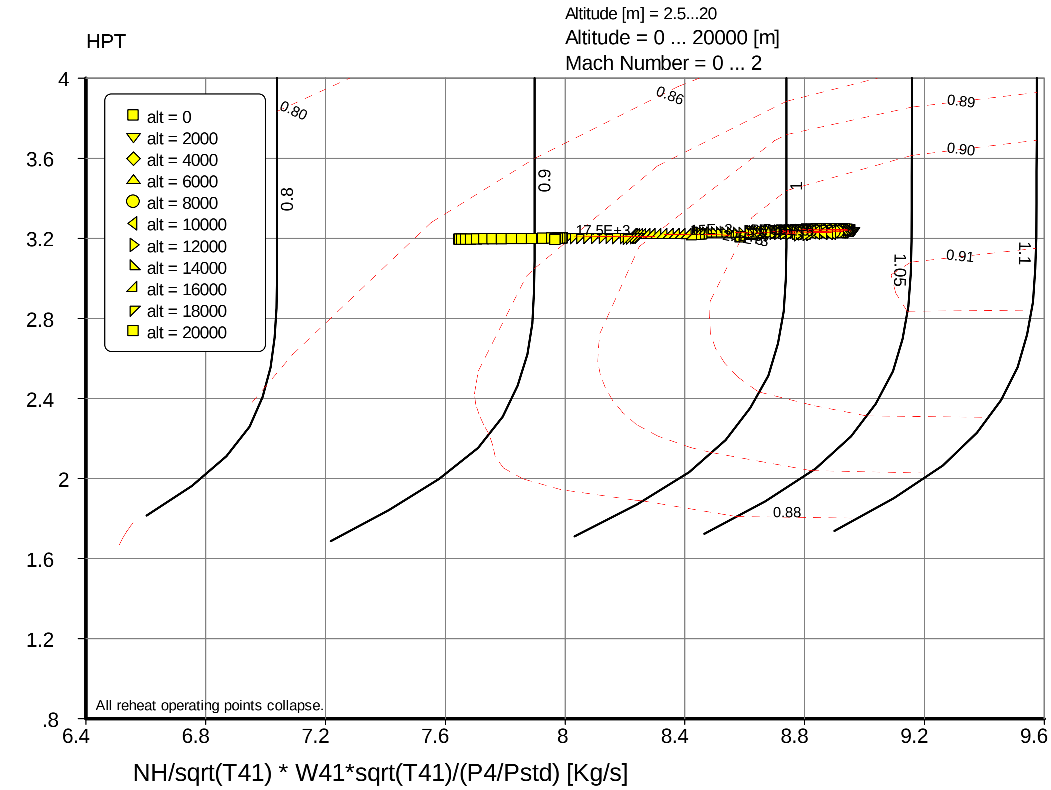

Figure 9

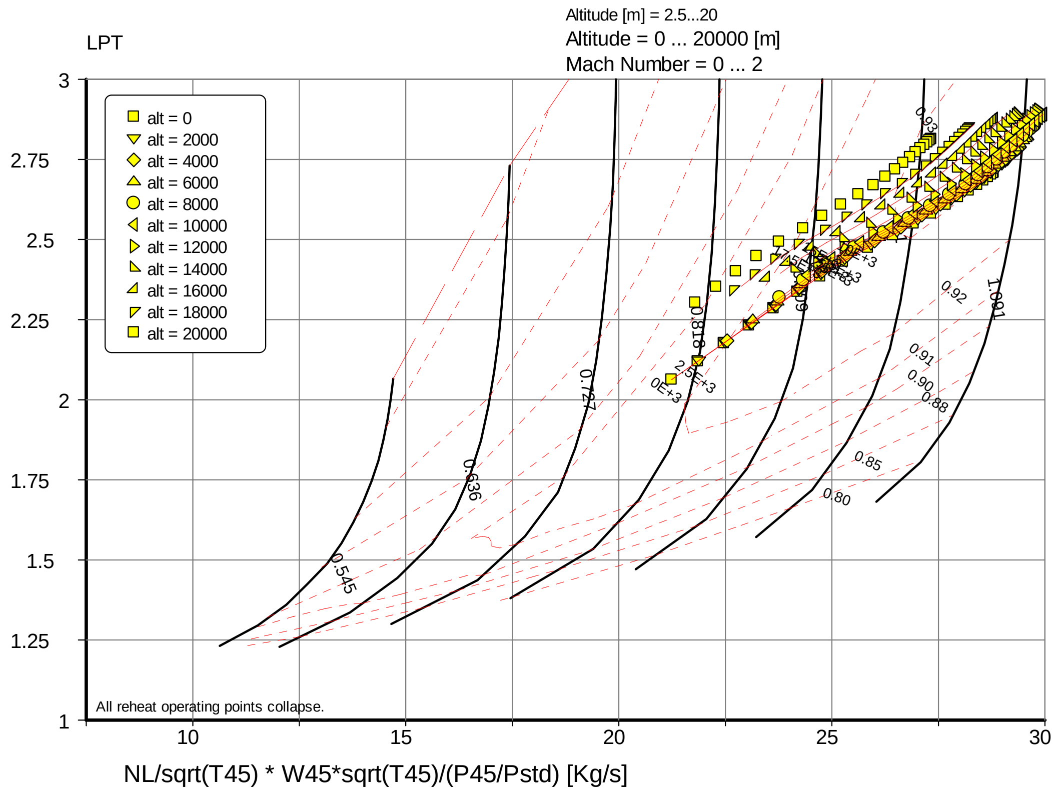

Figure 10

Afterburner OFF

The following graphs represent the dry thrust condition for a few varied conditions according to the legends in each diagram. Instead of including all calculated flight envelope points, only a few were selected for readability. The few points that did not converge (red filled circles) were checked and found to be far outside normal flying conditions. In summary it can be concluded that the engine is fairly well behaved to judge from its adequate distance to the surge lines in the compressor maps. Also the nearness to optimum efficiency (dashed contours) speaks in favor of the engine characteristics. – All four diagrams should have the pressure ratio on the y axis.

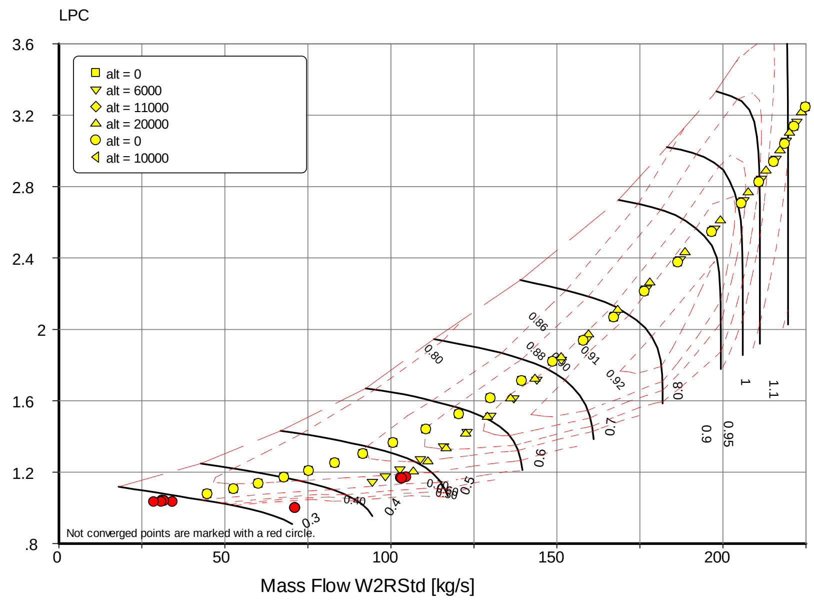

Figure 11

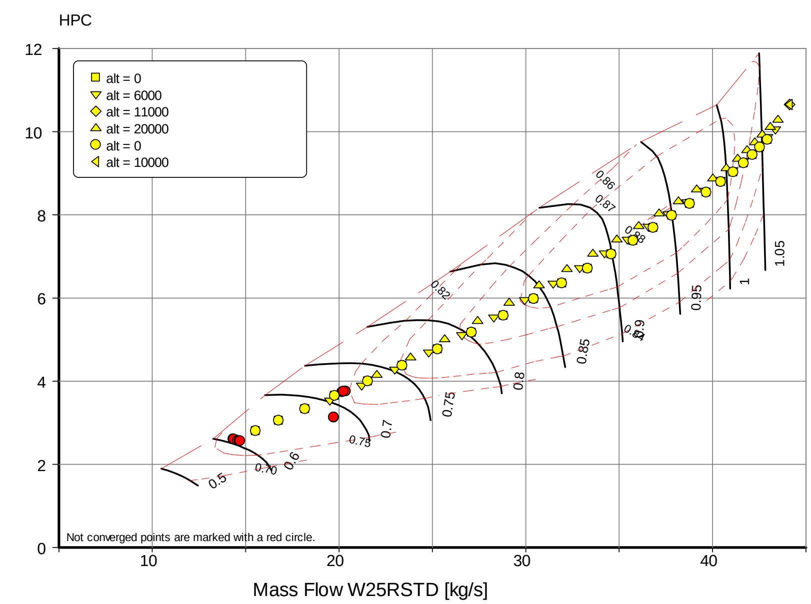

Figure 12

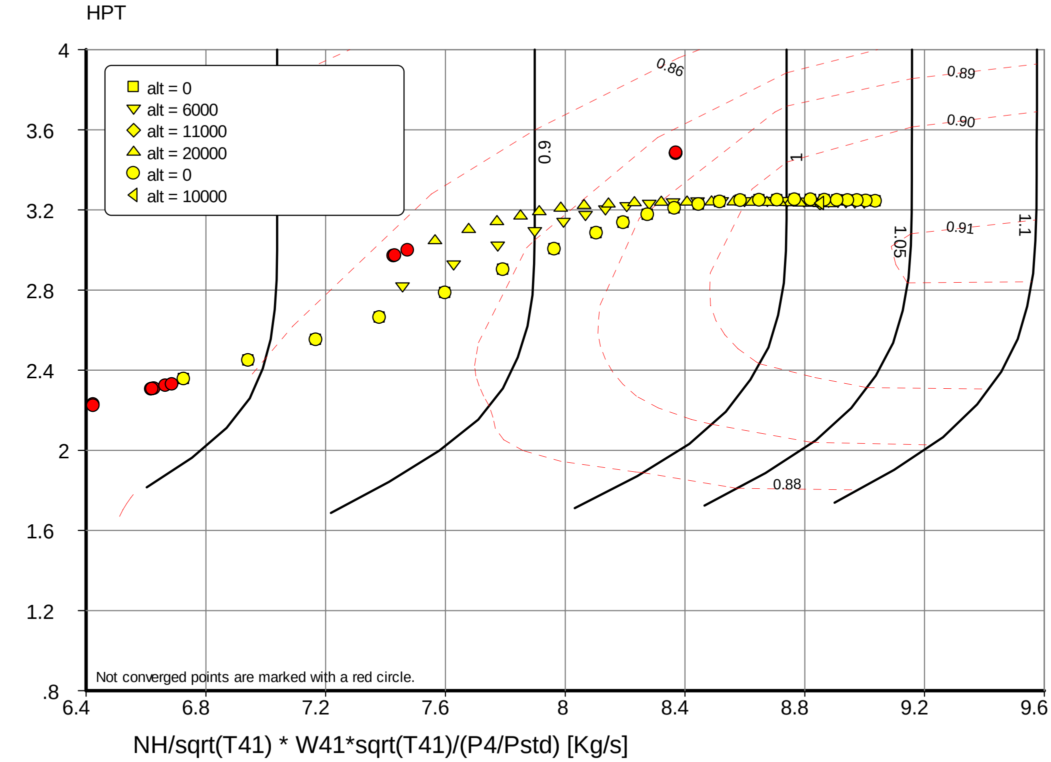

Figure 13

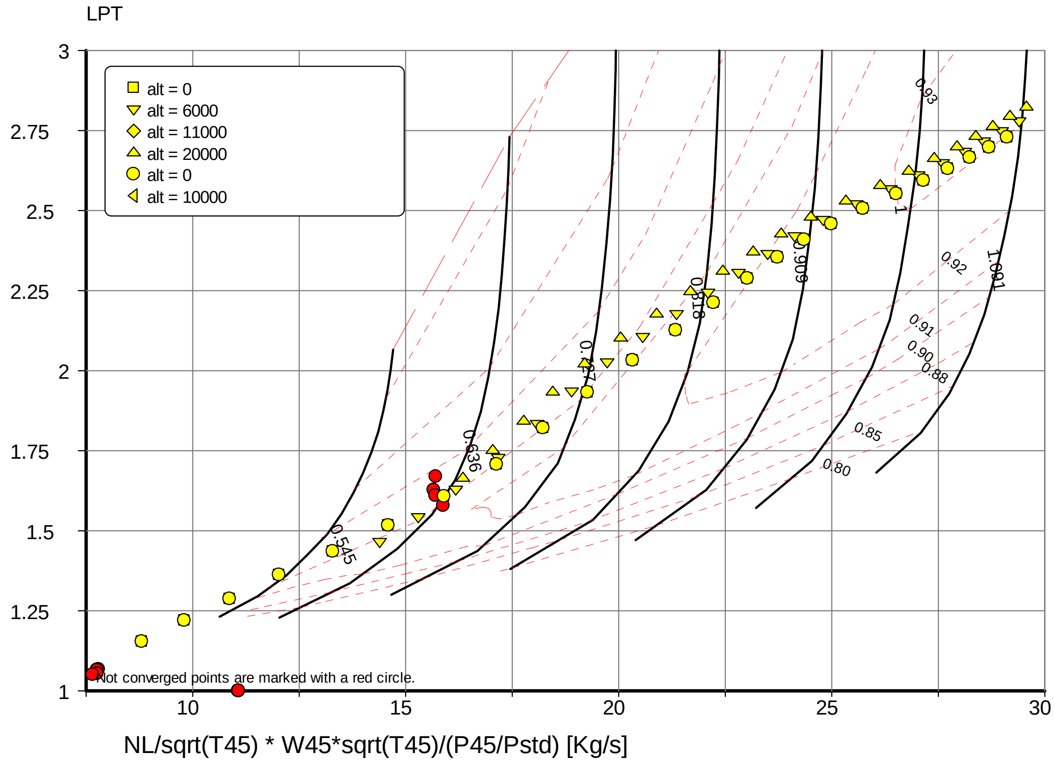

Figure 14

9.Discussion

The most complicated engine of the Gasturb10 choices was primarily chosen as a candidate for the UAV purpose. The attraction seemed to be in the possibility to close or open the VABIs for controllable bypass ratio. The observations that finally led to the change to a simpler engine were the following valid for the optimized VABI engine, see Fig. 1.

1. The dash requirement could be run with the VABI 2 open or closed with a very small difference in thrust and SFC.

2. The intermediate compressor was given the pressure ratio 1.0 by the optimization process.

3. The bypass ratio, as optimized, became very low, 0.1 or 0.3 depending on the VABI 2 setting

These findings led to the conclusion that the simpler fixed bypass engine option of the Gasturb10 program should be used. The search for an optimized engine therefore ended with the standard double spool bypass engine.

It may be the particular combination of requirements that happened to lead to this unexpected conclusion. Because of the multi-parametric nature of the problem there might be a solution with the VABI engine type that escaped the author. Further exercises of a similar nature should probably start with the simpler engine. After proper optimization of its key parameters the VABI engine should be tested with the same set of key parameters. Variations and optimizations should follow for a best of best engine solution.

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Appendix 1 - General Description of the Gasturb10 Computer Program

Gasturb10 deals with the thermodynamic aspects of turbojet and turbofan engines. Thus matching of compressor power to turbine power, on the same shaft, is at the core of the program equations. A number of checks are active to support the user. Error messages and warnings appear whenever an unrealistic combination of values for input variables have been requested by the user. The interface is attractive and mostly intuitive. The output includes flow, temperatures, pressures etc for the different stations in addition to the more global variables of major interest i.e. thrust and specific fuel consumption. The program does not, however, output weights and size. At least the variable cycle engine does not include size although one other engine type did present the diameter of the station 2 flow cross section. Thus weight and size will have to be dealt with as a post-processing activity using the Gasturb10 output, see sections 4 and 5 in the main text.

The program offers a set of different types of engines such as plain turbojet, ducted fan, turboprop and many more. After having selected a type of engine, the user will choose from two modes of operation. They are design mode and off-design mode. In the design mode there are several sub-mode options available. The most straightforward option is to set the “major” input variables and run. There is also an optimization mode in which the program automatically chooses the set of input variables, starting with those that the user provided initially. Examples of these input variables are pressure ratios, bypass ratios and burning temperatures. The variables to optimize and the target function to minimize or maximize are selectable by the user. Typically the specific fuel consumption will be used for minimization.

The design mode establishes the values of most input variables. When later the off-design mode is employed the design point is known such that the important matching of compressor and turbine “maps” can be carried out. In fact, what happens is that, the compressor/turbine requested design characteristics are placed in ready-made pre-stored maps delivered with the program. These maps portray pressure ratio versus mass flow and they include closed curves representing efficiency and, in the case of compressors, an extremely important line for surge. The surge line represents the limit at which compressor surge occurs. Surge is initiated by stall on the compressor blades and is followed by rapid dynamic pressure fluctuations, which can be audible. Surge can be catastrophic. At design circumstances the distance to the surge line will be reassuring. The component in question appears as a point in this map. The placement of that point defines the design in addition to the chosen or optimized input variables. The maps are quite general because the good design of compressors and turbines lead to very similar curves in the map diagrams. Although the maps are different for different pressure ratios their behavior allows a scaling which adapts (stretches or shrinks) a given map to one that better suits the preliminary design engine according to Ref. 5 in the main text.

Once the user is satisfied with the design he will go on to use the off-design mode. In off-design the compressor or turbine will experience flow conditions away from the design point in the maps. The off-design has reduced liberty for variable variation as should be expected. Now e.g. a compressor pressure ratio can no longer be defined by the user. It will be calculated. Variables that can be varied are e.g. altitude, Mach number and burner exit temperature (= “throttle”). It is easy to request the performance for an unrealistic combination of these input variables. Then the program informs the user, mostly in clear language, but sometimes the user must find the reason himself or simply avoid the variable combination. Automatic calculation of many cases, where the allowable input variables are varied according to user specification, is also available as a powerful feature. Two variables can be varied during such a run. This allows e.g. the automatic mapping of performance in the altitude – Mach number space.

One generic difficulty with engine performance calculation is to provide thrust and fuel consumption when the details of the airframe configuration are not known. The air intake and the nozzle are the main reasons for difficulties in this respect. Gasturb10 offers as an option the use of a standard formula for the intake total pressure loss as a function of supersonic Mach number. This formula represents the combined pressure losses of the different shocks that must somehow take place in order for the flow to be decelerated to subsonic speed before the engine entry. The user, however, sets the intake duct subsonic diffusion pressure loss himself (or takes the default value that is offered). See Appendix 4 for details. A better solution, in supersonic flight, is to evaluate the intake losses based on design rather than relying on the standard “blind” option.

The nozzle losses, which include effects from the detailed geometry of the hardware outlet, will have to be estimated by the user to emulate the installed engine performance.

Examples of available configurations follow.

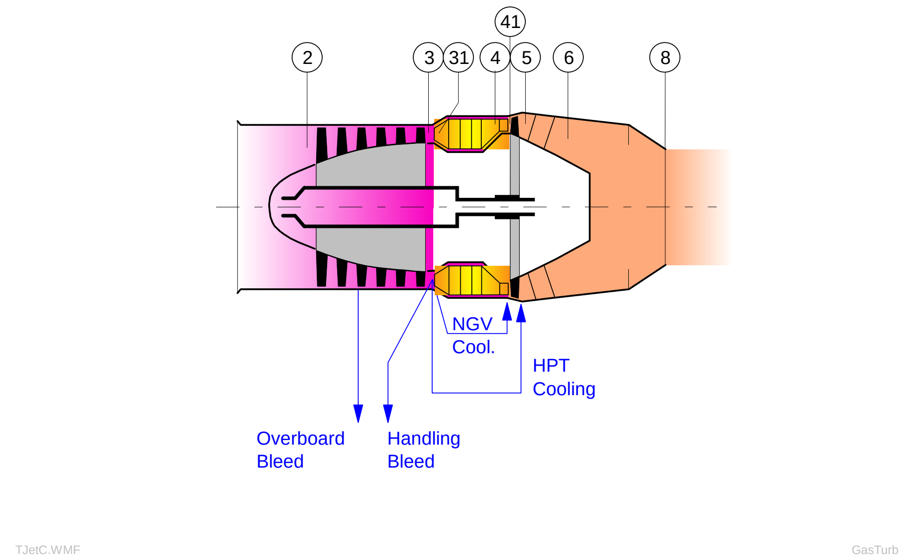

Figure A1 - 1

The figure above shows the simplest turbojet engine without an afterburner. The station notation is typical but slightly variable between types as seen in the following figures. The blue text (by the arrows), appearing under the engine, points to the availability of realistic features in the program.

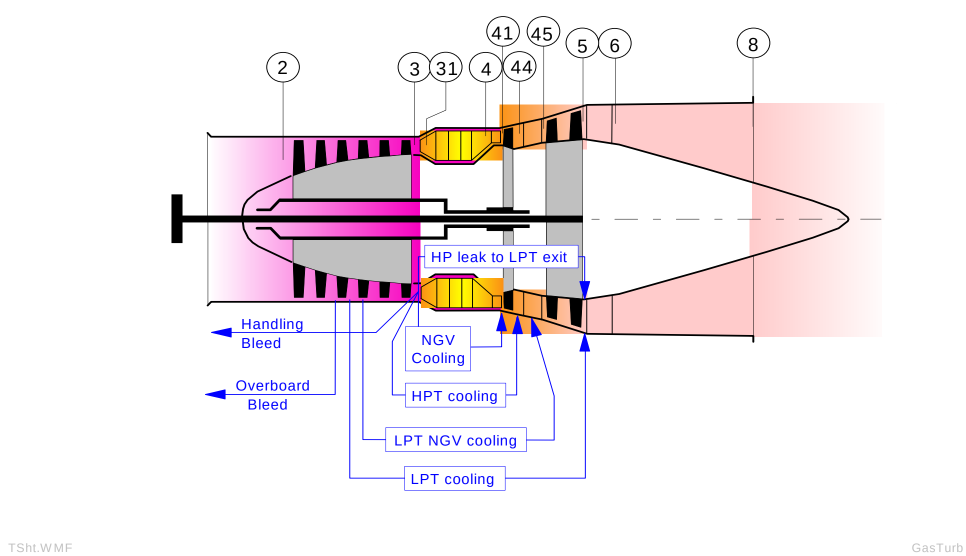

Figure A1 – 2

This engine is referred to as a turboprop depending on its typical application possibility to drive a propeller. It has two shafts (the one inside the other). It can serve as an engine on an airplane, then driving a propeller, or it could theoretically drive a stationary electrical generator on the ground, although it would be better not to use a shaft within a shaft to for the latter purpose. The shaft protruding out to the left signifies the torque attachment point. The shaft carrying the compressor should be seen as part of the gas generator whose purpose it is to provide excess power in the gases exiting from its turbine. The excess power, from the gas generator is absorbed by the subsequent low pressure turbine which, driven by the gases, provides the propeller torque. Typically such designs leave very little power in the final exit gases barely worth while to direct backward on an airplane if extra propulsion is contemplated.

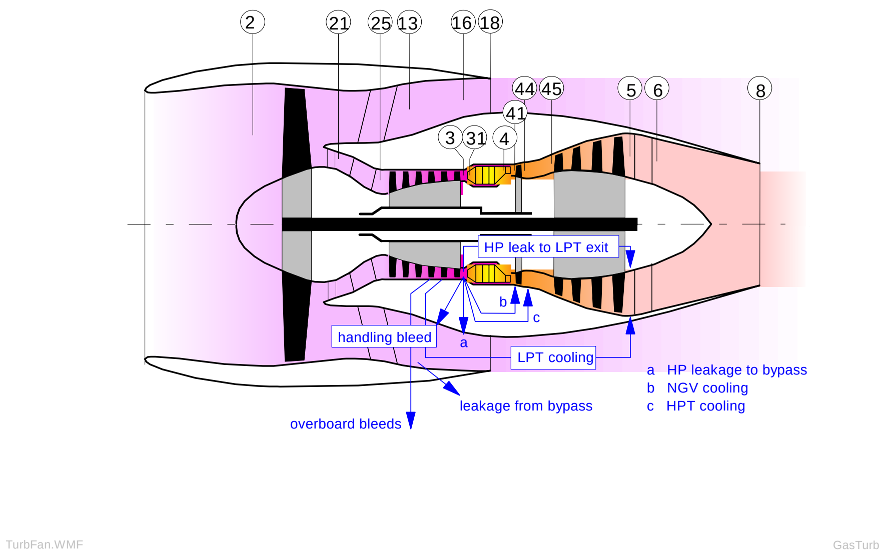

Figure A1 – 3

Figure A1 - 3 depicts a two-shaft engine, in principle similar to that of Fig.A1 - 2. But, here the output shaft drives a ducted fan as frequently seen on commercial airplanes. Another difference compared to the turboprop engine is that here the core engine exit flow generates a significant contribution to the thrust.

Gasturb10 contains 22 different airplane engines and 10 industrial engines, all based on the compressor/turbine technique.

Appendix 2 – Tabled engine weight data

File: Torenbeek method calculations.xls

Engine data from Jane's Aero Engines (2002)

Modified: Added 4 military engines at the bottom, Nov,15 MeB

Weight formula constants

C2	0.12
A	10
B	0.25
C1=A*OPR^B	

	Reli-			Year of		Jane's			Inlet			Specific						
	abil-			certifi-		engine			mass			thrust						
												at TO						
No. Engine Notation	ity	Engine Type	Manufacturer	cation	Powering	wt, kg	OPR	BPR	flow kg/s	TOthrust	Rate Type	N*s/kg	C1	Eng. wt,kg	dev%	abs	dev^2	T/W
1 JT9D-7	5	2-shaft turbofan	P & W JT9D	1971	747-200B, C	4014	22.2	5.1	698	202800	T-O Rating to 26.7 ˚C	290.54	21.706	3835.16	-4.5	4.5	19.9	52.9
2 D36	4	3-shaft turbofan	MotorSichUSSR	1977	Yak-42	1109	20	5.6	255	63740	T-O Rating	249.96	21.147	1254.84	13.2	13.2	172.9	50.8
3 RB211-22B	5	3-shaft axial trbfan	RR RB211	1979	L-1011-1 (-1	4171	25	4.89	626	186800	T-O Rating to 28.9 ˚C	298.4	22.361	3603.89	-13.6	13.6	184.9	51.8
4 JT8D-217	4	2-spool turbofan	P & W	1980	MD82	2052	18.2	1.74	219	88960	Rated at…	406.21	20.655	2022.3	-1.4	0.1	0.0	44.0
5 CF6-80C2A1	3	2-shaft turbofan	GE CF6	1985	A300-600	4144	30.4	5.06	802	257400	T-O Rating to 30 ˚C	320.95	23.481	4818.3	16.3	16.3	264.8	53.4
6 Tay 611	5	2-shaft turbofan	RollsRoyce Tay	1986	Gulfstream 4	1422	15.8	3.04	186	61610	T-O rating to 30 C	331.24	19.937	1255.41	-11.7	11.7	137.2	49.1
7 PW4056	4	2-shaft turbofan	P & W	1986	B747-400, -4	4273	30	4.9	773	252300	T-O rating to 33.3 ˚C	326.39	23.403	4725.11	10.6	10.6	112.0	53.4
8 CFM56-5A4	4	2-spool boosted sub	CFM Intern.	1986	A319	2266	29	6.2	370.1	97900	T-O rating to 30˚C	264.52	23.206	1886.59	-16.7	16.7	280.3	51.9
9 V2527-A5	4	2-spool boosted sub	IEA	1992	A320	2370	28.5	4.8	355.6	117900	T-OratingISA+10K(cryopl	331.55	23.105	2186.36	-7.7	7.7	60.0	53.9
10 TRENT 772	4	3-shaft turbofan	RR Trent	1994	A330	4785	35.5	5	919	316300	T-O Flat rated to 33 ˚C	344.18	24.409	5832.55	21.9	21.9	479.3	54.2
11 CFM56-7B26	4	2-spool boosted sub	CFM Intern	1996	B737-600/-7(	2384	27.61	5.1	353.35	116990	T-O Rating	331.09	22.923	2107.4	-11.6	11.6	134.6	55.5
12 PW4090	5	2-shaft turbofan	P & W	1996	B777-200IGV	7140	38.3	6.3	1233.8	408300	T-O rating to 30˚C	330.93	24.877	7111.67	-0.4	0.4	0.2	57.4
13 BR700-715C1-30	2	2-shaft turbofan	Rolls-Royce	1998	B717-200	2114	37.6	4.5	288.5	102300	T-O Rating	354.59	24.763	1952.02	-7.7	7.7	58.7	52.4
14 GE F110-GE129	5	2spoolFan	GE	1990	F16 F15	1791	30.4	0.76	122.4	75700	St SL dry	618.46	23.481	1819.97	1.6	1.6	2.6	41.6
15 F404GE402	3	2spoolFan	GE	1991	F18	1035	26	0.27	66.2	48000	St SL dry calcGastrb	725.08	22.581	1228.77	18.7	18.7	350.5	39.1
16 Snecma M88	3	2spool trbofan	Snecma	2003	Rafaele	897	26	0.27	65	60000	TO rating	923.08	22.581	1220.37	36.0	36.0	1299.6	49.2
17 F100-PW-229	5	"	P & W	1991	F16 F15	1700	32.2	0.36	146	79180	TO rating	542.33	23.821	2666.38	56.8	56.8	3231.5	29.7
														Sum/Ave -	99.8	16.6		
														StdDev ->			20.0	

Appendix 3 - Supersonic Intake Compression

At supersonic Mach numbers the pre-intake and at-intake detailed circumstances represent a very complicated part of the airplane development. The Gasturb program uses a simplified formula, however. The background for this formula is discussed in this section.

A crude estimate of the geometry affecting the compression, external to the intake, will be used for the following discussion. The compression occurs in three steps. The fore-body provides a "very three-dimensional" compression, with details of oblique shocks and volumes of different pressures. The detailed analysis of this flow is certainly outside the scope of the present study. It will nevertheless be dealt with in an approximation based on 2D shock flow theory. The next shock, which is also oblique, should be placed in the plane of the intake lips. Even this shock has the complications of three-dimensionality but a two-dimensional approximation gives results which are closer to reality than those of the first compression. Finally a normal shock rounds out the supersonic compression. Then the subsonic intra-duct compression takes place down to a Mach number, which is assumed to be in the interval $0.4 < M2 < 0.6$ at the compressor entry.

Using somewhat arbitrary corrections to this sequence of calculations will compensate for the lack of 3D analysis, although with a small error of unknown absolute size.

The configuration of the KoKoS UAV is shown in the figure below. Its geometry will be discussed in conjunction with the estimate of the compression.

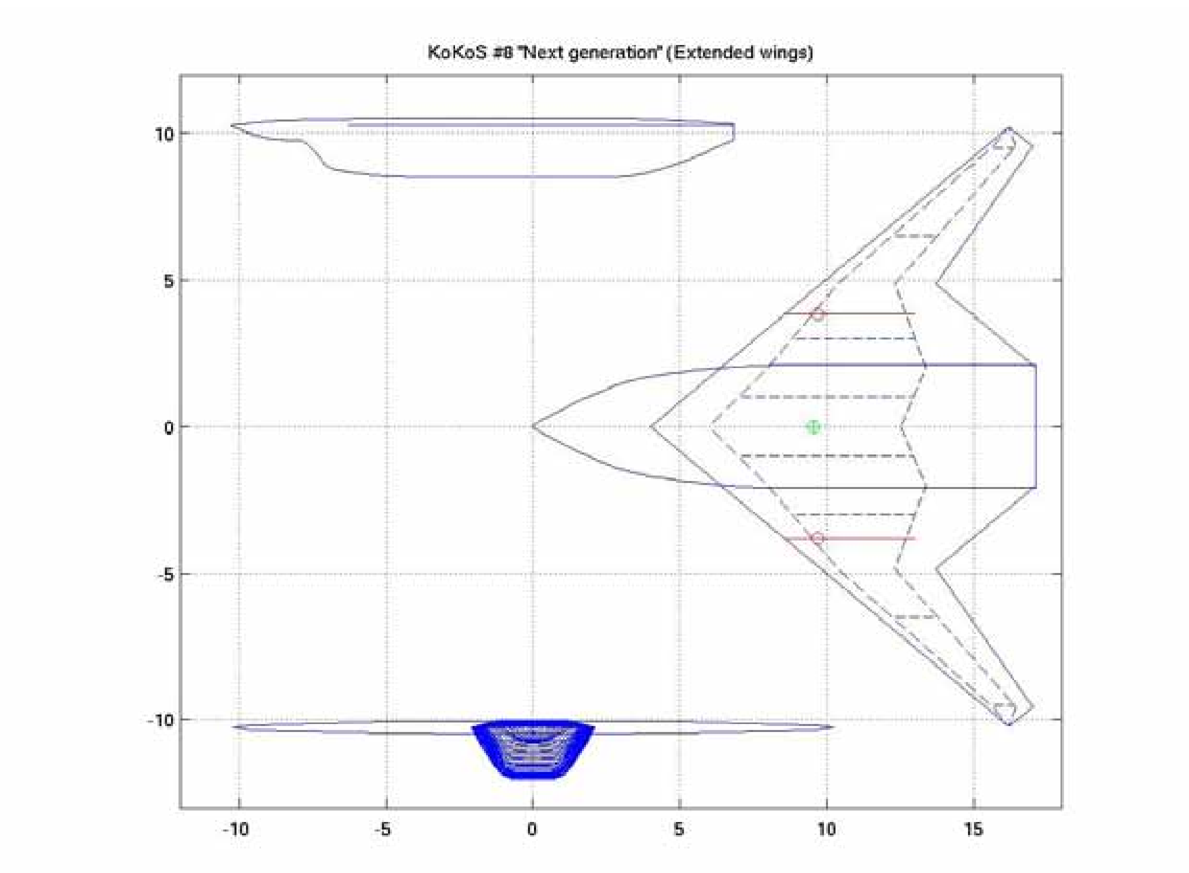

Figure A3 – 1 - The KoKoS configuration hinting at the air intake geometry on the upper left

The idea behind the compression can be explained with support of the following figure, which only emphasizes the positioning of the shocks and the shock inducing ramps.

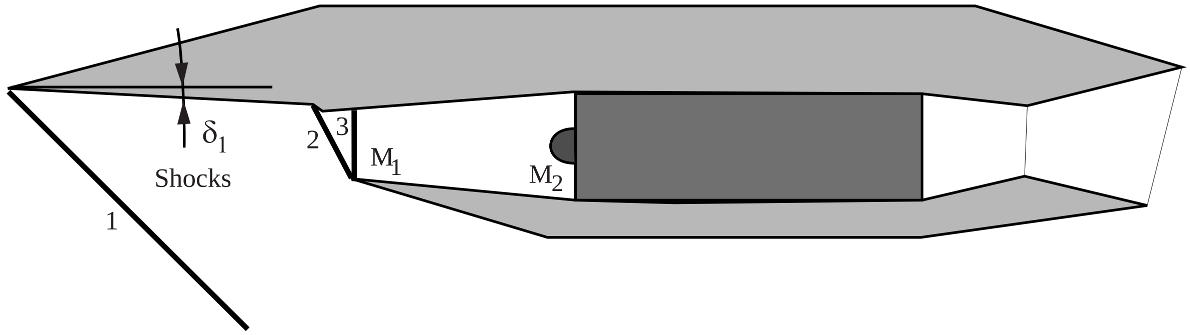

Figure A3 - 2 - Schematic showing elements of the supersonic compression

The first shock 1 can be trimmed to provide a suitable Mach number between shock 1 and shock 2 by proper selection of the material angle δ_1. The intake lips can be given an angle in a plane which equals the shock plane 2 angle. The compression to subsonic speed through a normal shock 3 should follow immediately behind. By proper trimming of the shock angles, using suitable material angles the total pressure loss can be designed to be minimal for given design circumstances. Off-design detrimental effects can be mitigated by the use of a variable geometry intake.

Returning to the design case there is a theoretical possibility to use an infinite number of shocks, which without a boundary layer, would provide isentropic compression with no total pressure loss. Because of real world effects including boundary layers, however, the literature on the topic is taken as convincing evidence that a three shock solution is most practical.

The NACA Report 1135 (available from the web) provides the equations for subsonic flow and supersonic flows with shocks. Equation (139a) relates the material angle to the shock angle as follows.

$$\cot\delta = \tan\vartheta\left[\frac{(\gamma+1)M^{\;2}}{2\left(M^2\sin^2\vartheta-1\right)}\right]-1 \qquad \text{(A3}-1\text{)}$$

(A4 – 1) is an implicit expression for $\vartheta = \vartheta(\delta, M)$. The subscript 1 and 2 are used in the NACA report definition in which the authors refer to properties before (1) and after (2) the oblique shock. They should not be confused with the definition found in the intake sketch above.

After ϑ has been calculated the Mach number and all other quantities behind shock 1 can be calculated using expressions from the NACA reference. Most useful for the oblique shocks are Eqs. (131) and (142) from the reference as follows.

$$M_2^2 = \frac{(\gamma-1)M_1^2\sin^2\vartheta+2}{\sin^2(\vartheta-\delta)\left[2\gamma M_1^2\sin^2(\vartheta)-(\gamma-1)\right]} \qquad \text{(A3-2)}$$

$$\frac{p_{0_2}}{p_{0_1}} = \left[\frac{(\gamma+1)M_1^2\sin^2\vartheta}{(\gamma-1)M_1^2\sin^2\vartheta+2}\right]^{\frac{\gamma}{\gamma-1}}\left[\frac{(\gamma+1)}{2\gamma M_1^2\sin^2\vartheta-(\gamma-1)}\right]^{\frac{1}{\gamma-1}} \qquad \text{(A3-3)}$$

Running through a few calculations for two-dimensional ramps the numbers necessary to create the following diagram were generated.

The formulas can be used to successively calculate the passage of several shocks. The oblique shocks can be designed to be of variable strength depending on the flow direction change. The Mach number of the flow coming out of the last oblique shock determines uniquely the conditions for the normal shock.

The designer will also use this knowledge of the shock behaviors to calculate the air flow that can be captured.

The Gasturb10 program, however, uses an approximation for the supersonic compression. It is derived from the extensive source of military standards. Its military specification notation is MIL-E-5008B. Its equation reads:

$$\frac{P_{postShock}}{P_{preShock}} = 1 - 0.075(M - 1)^{1.35} \qquad (A3-4)$$

The corresponding curve is seen together with selected samples of compression possibilities using the flat ramp technique.

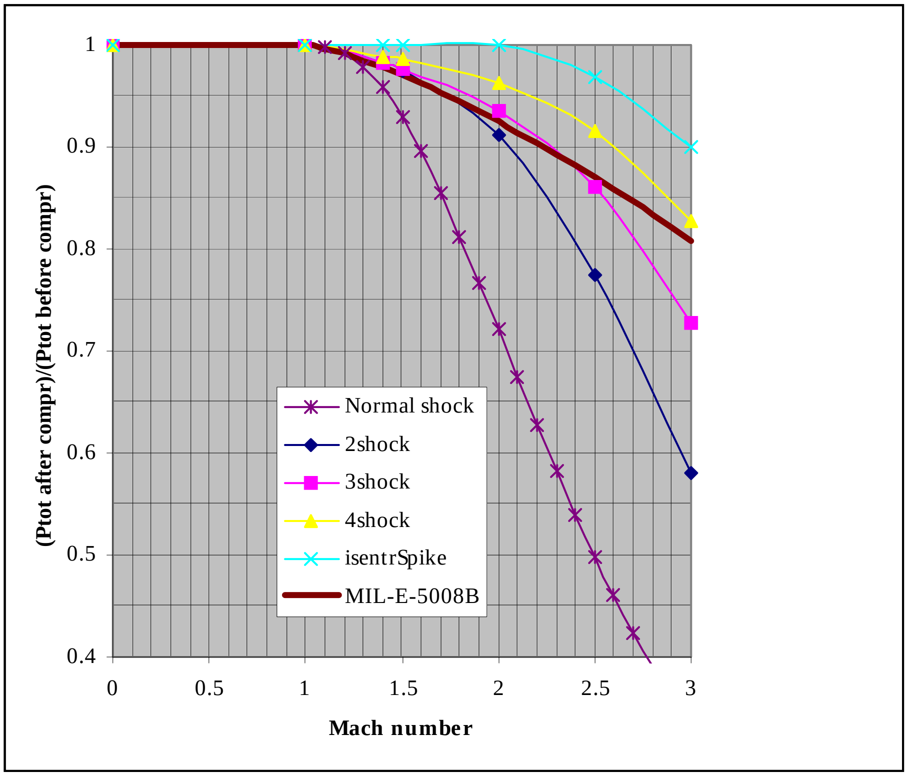

Figure A3 - 3 – Supersonic compression pressure losses

The philosophy behind a definition of a curve for all possibilities is that a designer of intakes will follow the same logic, which includes the following thoughts.

The basic concern will be to maintain as high a compression ratio as possible, where "compression ratio" is to be referred to total pressures. The reason behind this aspiration is that, for any given mass flow into the engine, it is advantageous to be able to fire up the engine as much as possible for maximum thrust. A low isentropic efficiency for the intake compression also means that the total temperature increases considerably above its corresponding isentropic value at station 2. This has the consequence that the temperature is excessively hot when the flow, after having passed the compressor enters the combustion chamber. Hence the margin from the natural compression temperature to the highest temperature that can be endured by the combustion chamber wall material, determines the maximum fuel injection rate. The higher the fuel injection rate the higher is the thrust level deliverable by the engine. In fact this describes the reason why a ram engine is more efficient than a turbojet over a certain Mach number range.

If a flying vehicle is to reach a maximum of say M=1.2 the diagram shows that the loss is small for all techniques displayed. It can be concluded immediately that one single straight shock will do. It is always the simplest and least expensive solution. A requirement to fly at e.g. M=1.6 would give a 10% loss for the straight shock solution. The designer might opt for a two-shock solution in this case.

Continuing this way of reasoning toward even higher Mach numbers will lead to levels of pressure loss that resemble the military specification curve, which is typically employed by programs dealing with engine development or adaptation, including the Gasturb program.

Using the mil spec formula allows the use of the program directly without particular attention to the entering flow shock details. In preliminary engine design work it is frequently assumed that the responsibility for this part of the total vehicle development lies with the airframe developer. This also says that at a later stage in the development recalculations of the engine performance must be carried out.