# Performance Assessment of a Pre-Cooled Turbofan for Hypersonic Vehicle Acceleration

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## Abstract

Within the framework of the EU FP7 LAPCAT II Mach 8 hypersonic transport project (Ref. 14), ULB was assigned to evaluate the performance of a turbofan accelerator as part of a so-called turbine based combined cycle (TBCC). The accelerator would be used up to Mach 5, before switching to ram/scramjet propulsion for final acceleration and cruise.

This paper describes the performance assessment of a low by-pass ratio pre-cooled turbofan, along the trajectory optimized for the corresponding hypersonic vehicle project sized by the University of Rome "La Sapienza" (UNIROMA1).

Initial engine data are based on a two-stage-to-orbit pre-cooled engine concept for in-flight liquid oxygen collection developed within the ESA FESTIP2 project (Ref. 4). Engine characteristics are then introduced into J. Kurzke's Gasturb v11 software (Ref. 8) for a first detailed performance evaluation along the trajectory. Original FESTIP2 concept fan and compressor maps are also introduced into Gasturb through its "Smooth C" tool for better predictions.

This sizing has in a next step been compared and improved by using Ecosimpro software (Ref. 9), in order to use a more precise heat exchanger model as well as to include ways to take, among others, fuel heating effects on specific impulse and ram recovery reduction due to the pre-cooler presence into account.

Finally, engine and pre-cooler weight estimations are presented for vehicle weight assessment.

## Introduction

As opposed to rocket based combined cycles, air breathing turbine based combined cycles allow for higher specific impulse during acceleration, reducing fuel consumption and hence the vehicle fuel mass fraction required at take-off. Using a turbine accelerator for as long as possible during acceleration would therefore provide a significant fuel consumption advantage, while at the same time reducing the high speed propulsion system complexity. However, limitations on compressor face temperature do not allow turbine engines using conventional compressor blade materials to go much beyond Mach 3 (Ref. 11, 12). Moreover, an increase in compressor inlet temperature leads to a higher required compression power, as well as to reduced maximum fuel flow due to turbine inlet temperature limits. Therefore, in order to keep using gas turbine engine propulsion up to Mach 5, inlet air cooling using a pre-cooler is mandatory. This pre-cooler would be fed on one side with gaseous hydrogen coming from the vehicle's liquid hydrogen fuel tank, taking advantage of its heat sink potential, before flowing to the engine combustion chamber and the afterburner.

The aim of this paper is to evaluate the performance and sizing of a pre-cooled turbofan accelerator. Vehicle designers will then be able to use these data in order to assess global vehicle performance and size.

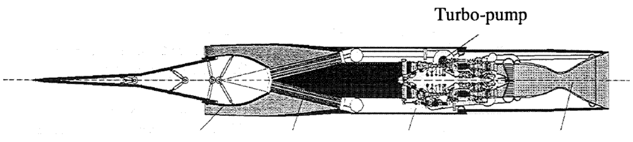


**Figure 1: Pre-cooled turbine engine concept (Ref.10)**

## Nomenclature

BPR: By-Pass Ratio
ESA: European Space Agency
FESTIP: Future European Space Transportation Investigation Project
FHV: Fuel Heating Value
HP: High Pressure
ISA: International Standard Atmosphere
ISP: Engine Specific Impulse
LHV: Lower Heating Value
LP: Low Pressure
M: Mach number
OP: Operating Point
PR: Pressure Ratio
TBCC: Turbine Based Combined Cycle
RR: Ram Recovery
SLS: Sea Level Static
TIT: Turbine Inlet Temperature
T/O: Take-Off

## Initial Performance Estimations

Sizing of this pre-cooled turbofan is based on the former FESTIP2 study by Techspace Aero, von Karman Institute, and ULB. FESTIP2 program designed a ring pre-cooled turbofan engine for cruise liquid oxygen collection operation through atmospheric air cooling, liquefaction and separation, the advantage of this operation being that no liquid oxygen tank needs to be carried by the vehicle, hence reducing its mass. This study made engine performance estimations available from take-off up to Mach 3.8 (Ref. 4). Therefore, as our turbofan engine is supposed to operate up to Mach 5, initial performance estimations beyond Mach 3.8 have been extrapolated. Moreover, since no collection is to be made by our engine, data given by the FESTIP2 study during collection have not been considered (from Mach 1.8 up to Mach 3).

Since this engine is supposed to be used as an accelerator only, the choice for a design point was not straightforward, as it generally is for a cruise engine, or even for the FESTIP2 study itself, which chose the design point as being the collection phase flight conditions, since those represented a significant part of the whole ascent time. In the following Gasturb and Ecosimpro assessments, the design point was chosen at Mach 2 and 43kPa dynamic pressure, very close to the FESTIP2 study design point.

Original fan diameter was kept at 1.6 meters and used for engine length and weight estimations. Therefore, vehicle designers only had to choose the required number of engines, following the thrust requirements at the most demanding point on the ascent trajectory.

Initial take-off performance (SLS ISA) as reported by the FESTIP2 study was:

- Gross thrust (reheated): 342 kN
- Airflow (SLS ISA): 325 kg/s
- BPR: 0,22
- Total PR: 11,25
- TIT: 1800 K

Initial performance for the entire trajectory are presented in the following figures for maximum gross thrust and ISP capabilities with reheat. These are pertaining to the FESTIP2 trajectory reaching the constant dynamic pressure at a much lower altitude than our selected trajectory, and therefore are not usable as such for our vehicle performance assessments:

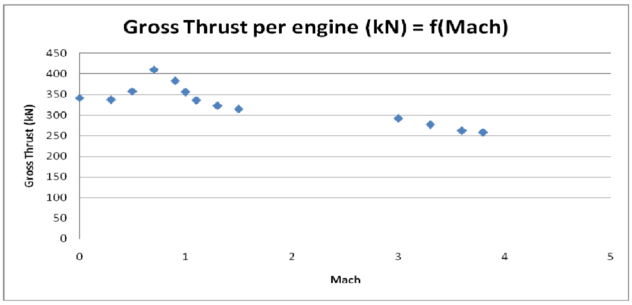


**Figure 2: FESTIP2 engine based gross thrust capability**

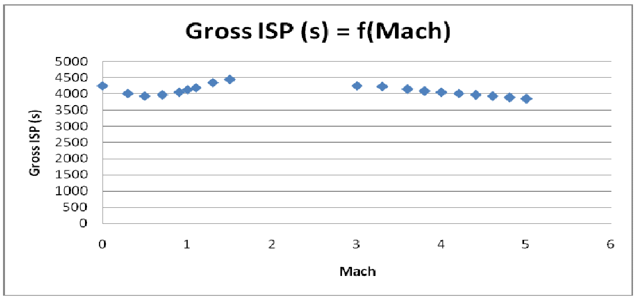


**Figure 3:FESTIP2 engine based ISP capability**

In order to take intake losses into account to calculate net performance, we used correlations for ram recovery evaluations along the turbofan operation speed range. These correlations were taken from Thomas C. Corke's "Design of Aircraft" engine selection chapter (Ref. 6) and use the following expression for relative thrust loss in function of ram recovery and a parameter named Cram:

$$\frac{\Delta T}{T} = C_{ram} \cdot \left( RR_{ref} - RR_{real} \right)$$

With following values for each parameter:

Take-off:

- Cram = 1.35
- RRref = 1 (test bench value with bellmouth)
- RRreal = 0.85

After T/O until Mach 1 included:

- no losses considered

From Mach 1 up to Mach 2.5:

- Cram = 1.35 – 0.15 * (M-1)
- RRref = 1
- RRreal = 1 – 0.075 * $(M-1)^{1.35}$

From Mach 2.5 up to Mach 5:

- Cram = 1.35 – 0.15 * (M-1)
- RRref = -0.1 * M + 1.25
- RRreal = 1 – 0.075 * $(M-1)^{1.35}$

We can see that from T/O up to Mach 1, no losses were considered, except for T/O itself, where zero flight speed ram recovery losses for a supersonic intake was estimated at 0.85.

Considering these losses, we found the following net available performance along the considered speed range as shown on Figures 4 and 5.

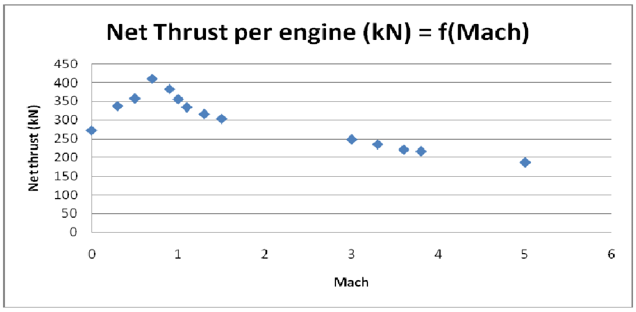


**Figure 4: Net thrust capability initial estimate**

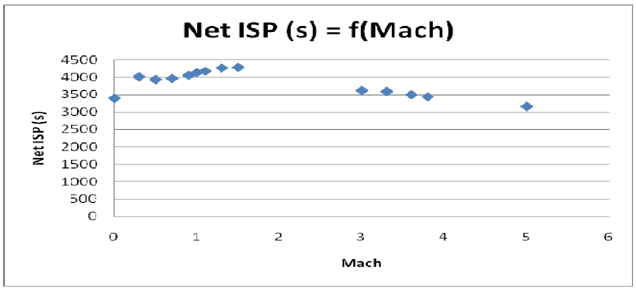


**Figure 5: Net ISP capability initial estimate**

Net ISP estimate was made based on gross ISP data given by FESTIP2 report multiplied by the ratio of net thrust to gross thrust available. Turbofan net thrust was evaluated by using thrust loss correlations cited above.

## Updated engine performance estimations using Gasturb v11

Following the first engine performance database directly taken from the ULB-TA-VKI LBPR FESTIP2 turbofan report tables, and since part of these data were unavailable due to air collection being conducted from Mach 1.8 up to Mach 3, it was decided to generate our own results based on the FESTIP2 engine characteristics using Gasturb (v11) software (Ref. 8). This study also allowed us to generate performance results specifically for the trajectory optimized by UNIROMA1 (Ref. 13), as well as for various constant dynamic pressure trajectories.

UNIROMA1's optimized trajectory was defined as:

1. Constant Mach 0.5 climb-out to 3,048 m
2. Constant altitude acceleration to Mach 0.8
3. Constant Mach 0.8 climb to 9,000 m
4. Acceleration to a 50 kPa dynamic pressure
5. Constant 50 kPa dynamic pressure climb to 30.000 m

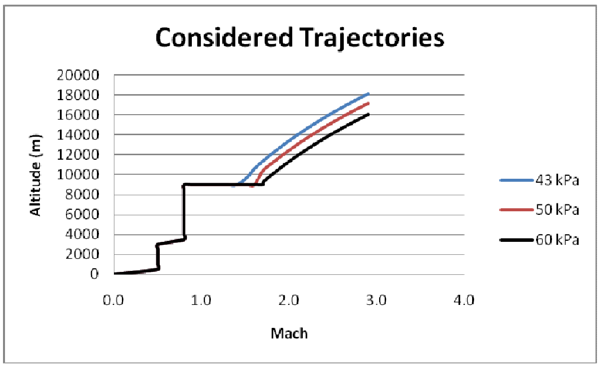


**Figure 6: Considered Mach 8 vehicle trajectories**

After the 9000m acceleration phase, three different constant dynamic pressure trajectories were considered in order to examine the effect on engine performance:

1. 43 kPa (original FESTIP2 study trajectory)
2. 50 kPa (UNIROMA1's trajectory)
3. 60 kPa

Initial goal of Gasturb calculations was to generate two sets of results, relating to two potential propulsion mode transition schemes. One without pre-cooler, transitioning to ramjet at Mach 3, and the other with pre-cooler, directly transitioning to dual-mode ramjet at Mach 5. Indeed, both concepts need to be evaluated for optimum final choice. Pre-cooling allows for extended speed range for turbofan operation while reducing dual-mode ramjet module variable geometry complexity and hence mass. On the other side, transitioning to ramjet at Mach 3 would eliminate the need for a pre-cooler, while increasing the mass of the variable geometry dual-mode ramjet system.

However, Gasturb results could not be obtained beyond Mach 3, speed at which pre-cooling would start, because of convergence problems once pre-cooling effect was introduced by imposing a negative ISA temperature deviation, no heat exchanger model being available within Gasturb. Therefore, an Ecosimpro model was later developed as described in the next section, in which specific heat exchanger models were used. As a consequence, in the set of results corresponding to the case including a pre-cooler, this section only considers its physical presence, and the related pressure recovery losses. Available FESTIP2 results were obtained with Gasturb and extended until Mach 3.8. Therefore, the slope from Mach 3 to Mach 3.8 could have been used to extrapolate them until Mach 5. However, it was not possible to estimate the first mass flow and thrust increase at pre-cooling initiation before creating the Ecosimpro model. Therefore, following results are only presented up to Mach 3, with a 0.1 step for Mach number, and a 500m step for altitude.

Gasturb v11 software includes standard compressor maps, but in order to be as close as possible to the performance our FESTIP2 based engine could reach, and since specific low and high pressure compressor maps were available (Ref. 4), it was decided to introduce them into Gasturb by using its specific map generating tool named "Smooth C". However, standard turbine maps were kept, since those will be choked at the required maximum thrust points we wanted to investigate.

Figures 7 and 8 present FESTIP2 low and high pressure compressor maps introduced in Gasturb's off design calculation module.

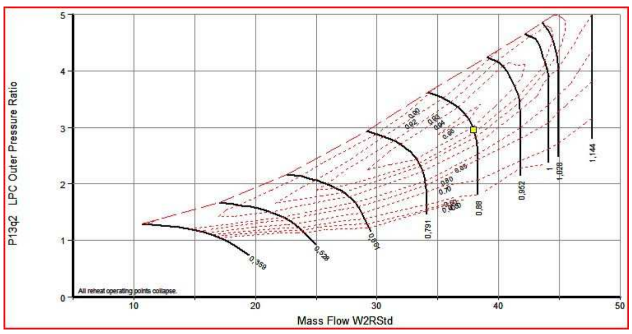


**Figure 7: FESTIP2 outer LPC map (Ref. 4)**

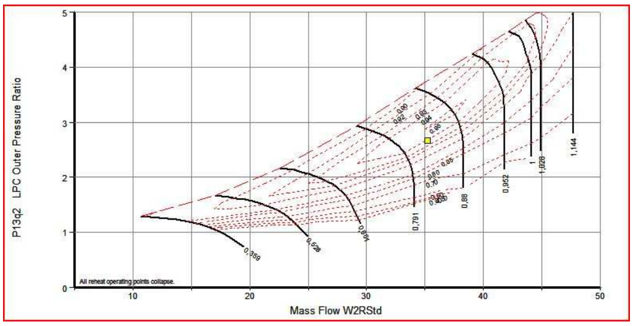


**Figure 8: FESTIP2 HPC map (Ref. 4)**

Figure 7 shows the outer fan map, meaning the part of the fan compressing the secondary flow. In this study, inner fan map has been derived from the outer one by applying a constant ratio determined at design point between outer and inner fan pressure ratios and efficiencies. As can be seen, our design points had to be moved to lower relative corrected spool speeds since we used a non-precooled design point, contrary to the FESTIP2 study. Had we kept the initial design points, the off-design operating point would have headed to excessive corrected spool speeds at pre-cooling start.

Considered design point characteristics were the following:

- Mach=2.0
- Dynamic pressure = 43 kPa
- BPR=0.22 with afterburning
- Mass flow rate: 225 kg/s
- Compressors pressure ratio and isentropic efficiency
  - Outer LPC : 2.18 ; 0.937
  - Inner LPC : 2.07 ; 0.879
  - HPC : 2.66; 0.924
- LH2 fuel
- TIT: 1500 K compared to FESTIP's 1800 K
- Reheat temperature: 2400 K
- $N_{LP}$ = 5600 rpm ; $N_{HP}$ = 9275 rpm
- Burner efficiency: 0.99
- Reheat efficiency:0.90

Off-design hypothesis:

- *Ram recoveries*:
  - Take-off: 0.85 without pre-cooler losses
  - In the pre-cooler equipped case, additional pressure losses were estimated from the following figure:

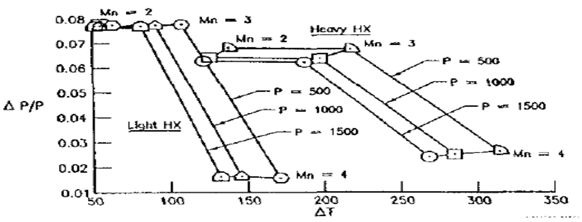


**Figure 9: Pre-cooler pressure losses estimation (Ref. 7)**

Figure 9 shows that maximum pressure losses can be estimated to be about 7%. This maximum value has been taken all along the trajectory.

- *Limiters* were also set on spool speeds at 105% design speed, as well as on turbine inlet temperature which was limited to 1500K, compared to FESTIP's 1800K for improved MTBO (Mean Time Between Overhaul)

- *Convergent-Divergent Nozzle area ratio* A9/A8 was automatically set to the adapted nozzle value at each trajectory point. However, it has to be kept in mind that this hypothesis implies the use of complex and potentially heavy variable geometry systems

From take-off up to maximum dynamic pressure acceleration at 9000m, thrust and ISP data now depend on altitude and Mach number, resulting in a 3D graph, but since the considered trajectories were already presented, Figures 10 to 12 give performance results in function of Mach number only.

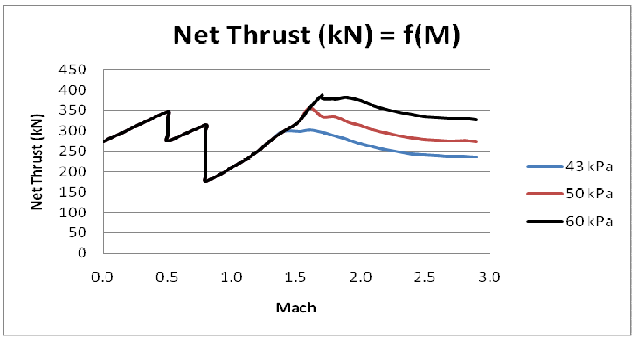


**Figure 10: Net thrust capabilities along considered trajectories**

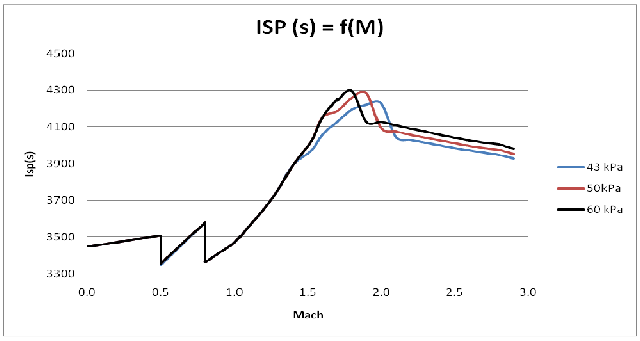


**Figure 11: Net ISP capabilities**

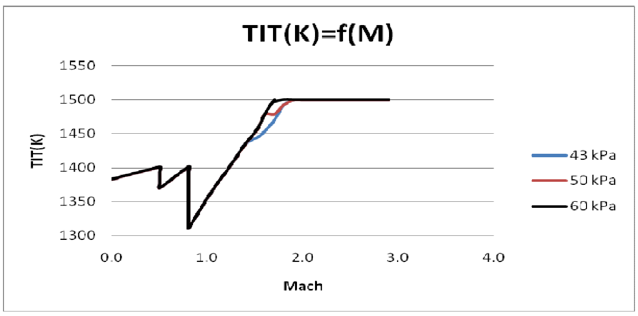


**Figure 12: TIT limitations**

From these figures, it can be seen that TIT limit had to be set lower than the chosen design 1500K value below a certain Mach number. This effect is a consequence of having to set a design point at such a high Mach number. The reason for this limitation lies in the low pressure compressor spool speed limit and will be explained in more details in the next section thanks to the low pressure compressor maps generated with Ecosimpro.

## Updated engine performance estimations using Ecosimpro

With the help of an Ecosimpro library named "HYDRA" and developed by D. Verstraete as part of his PhD thesis (Ref.1), a model of our pre-cooled engine could be created. More precisely, two separate models were used, one non-pre-cooled (but including additional ram recovery losses due to the presence of the pre-cooler in the inlet) for performance estimations below Mach 3, and another including an actual heat exchanger model for estimations after pre-cooling initiation. Design point choice was made as in Gasturb calculations at Mach 2 on a 43 kPa trajectory, and the resulting engine design was then used in both models. All characteristics as ram recovery evolution with Mach number, pre-cooler ram recovery losses, efficiencies, compressors maps, limiters on TIT and spool speeds, HP turbine cooling, as well as nozzle auto-adaption with ambient conditions were kept. However, although both compressors map design point positions were also kept, LP turbine design point was moved to an increased β angle of 0.7 compared to 0.5 in the Gasturb model in order to allow for an extended convergence range since the initial value lead to negative angles above Mach 3.8 (β angle represents the LP turbine map operating point's angular position from the origin, 0 and 1 values representing the map limits). This modification as well as an increased TIT value beyond Mach 3.8 allowed us to reach convergence up to Mach 4.7 as well as positive values for LP turbine β angles up to Mach 4.5. However, the validity of results beyond about Mach 4 still needs to be fixed.

Pre-cooler model is based on the ε-NTU method, with an exchanger effectiveness estimated at a value of 0.7. Indeed, since its efficiency is inversely proportional to its weight, it is important in our case to choose an efficiency value that is not too high.

Starting with the design point geometry, off-design calculations were then conducted down to take-off conditions, then from take-off up to the highest possible altitude on a 50 kPa trajectory above 9000m. The reason a downward trajectory calculation until take-off conditions was calculated before the final considered trajectory was that Ecosimpro requires initial conditions to be inserted for iterative loops solving. Since values as close as possible to the final value allow for easier convergence, it was necessary to have Ecosimpro calculate all required initial conditions parameters values at take-off beforehand.

Ecosimpro model results are presented in Figures 13 to 15, together with Gasturb results for net thrust capability, ISP, and TIT for comparison.

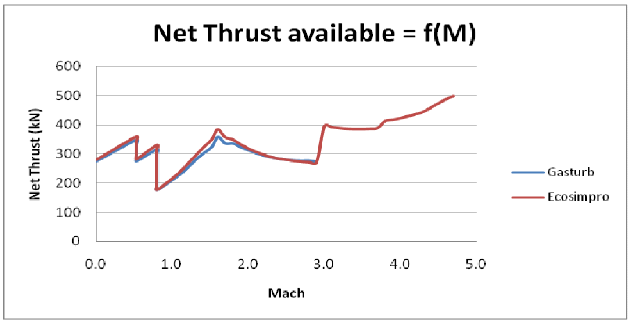


**Figure 13: Net trust available on the 50 kPa trajectory**

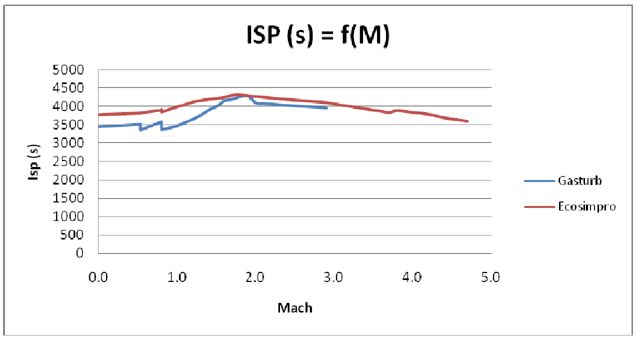


**Figure 14: Net ISP estimates along the 50 kPa trajectory**

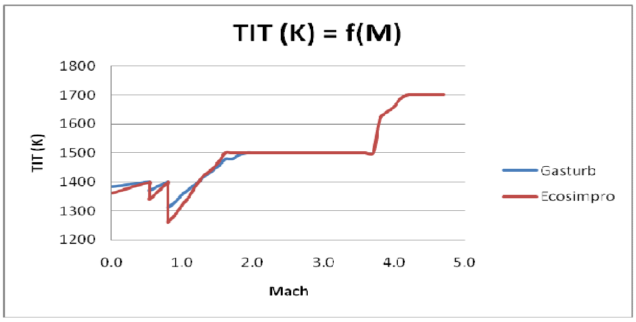


**Figure 15: TIT limitations**

As can be seen in these figures, thrust estimations are very much alike, except that we now have results above Mach 3.0. Differences in ISP estimations can be mainly attributed to Gasturb calculations including more loss sources than were modeled in the Ecosimpro model as well as to fuel injection temperature effect being taken into account in the Ecosimpro model only. Indeed, since fuel LHV are based on a temperature of 298K, the large fuel temperature differences in pre-cooled operations can lead to significant differences in FHV at injection point. This effect is even more significant with hydrogen fuel since its specific heat Cp is about 1000 times higher than kerosene, leading to much higher enthalpy variation with its temperature (Ref. 1, 3).

Figure 13 clearly shows the effect of pre-cooling initiation at Mach 3. Net thrust available suddenly increases from 270kN to about 390kN, while fan inlet temperature decreases from 600K to 480K with a hydrogen temperature increasing from 150K at pre-cooler entrance to about 460K, increasing fuel injection temperature and hence fuel heating value at the same time.

The initial choice of 150K for fuel injection temperature is a minimum value imposed by fuel system constraints (Ref. 1, 3). Ideally, this temperature should be between 150K and 250K. Figures 16 and 17 show the fan inlet and fuel injection temperature evolution with Mach number.

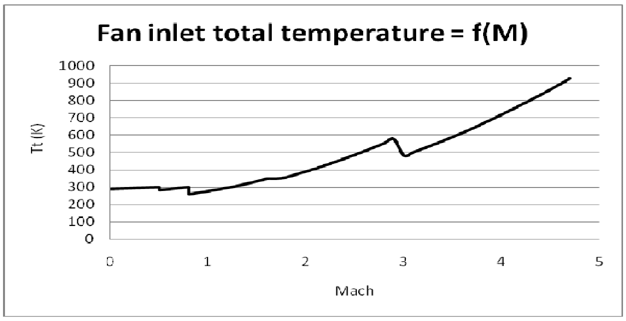


**Figure 16: Fan inlet total temperature**

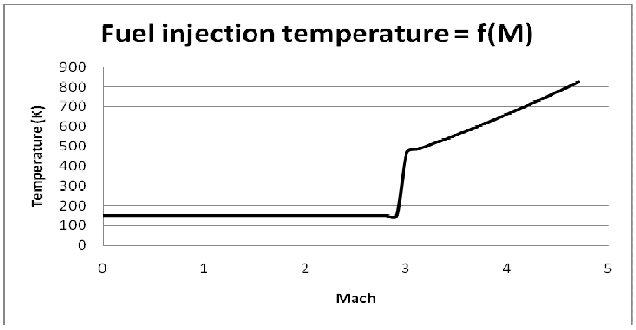


**Figure 17: Fuel injection temperature**

Figure 16 shows that the current fan blade materials temperature limit of about 600K is reached again around Mach 3.6.

Beyond about Mach 3.8, we also see on Figure 13 that net thrust available starts to increase. However, results beyond this Mach number still need to be evaluated for validity since this Mach number corresponds to the fan pressure ratio approaching 1, and therefore to a windmilling fan. Figure 18 shows the fan operating point evolution on its map, which will also allow us to explain the need for a TIT reduction at "low" Mach numbers.

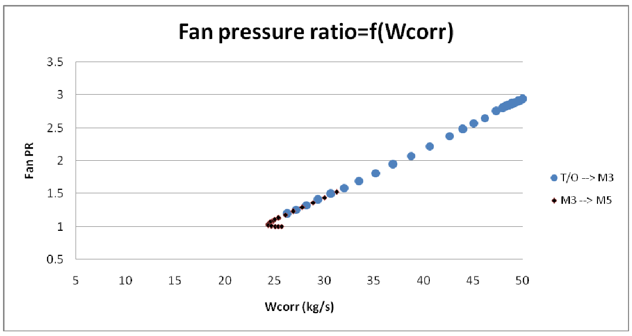


**Figure 18: Operating point evolution on the fan map**

Top right cluster of operating points corresponds to take-off conditions up to a flight Mach number of about 1.2 along our selected trajectory. From this Mach number on, the OP starts to move towards lower PR and corrected mass flow, reaching a PR of about 2 at Mach 2 design conditions, and 1.2 at Mach 2.9, just before pre-cooling initiation. Once pre-cooling starts, PR jumps to about 1.5 as can be seen in the second series of points representing the OP evolution on the pre-cooled part of the trajectory from Mach 3 to Mach 5. However, as stated above, we can see that Mach 3.8 corresponds to the fan PR reaching 1.

As can be seen in Figure 19, absolute air mass flow estimation at Mach 2 is very close to the take-off value (260kg/s and 250kg/s respectively).

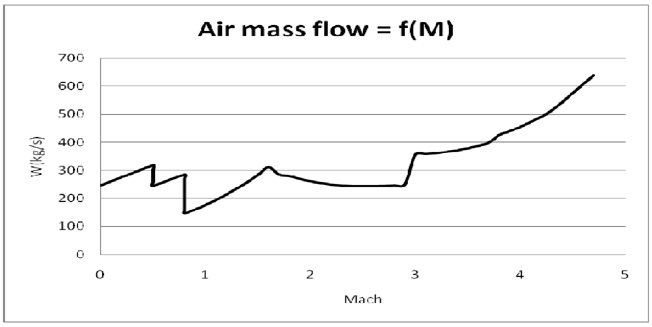


**Figure 19: Engine air mass flow rate**

Therefore, the lower corrected mass flow value at Mach 2 comes from the more significant effect of total pressure increase compared to the total temperature increase. This very important fan inlet total pressure difference between design conditions at Mach 2, and take-off conditions explains the large variations in corrected mass flow. Had we let the take-off TIT at the design point value of 1500K, the corrected mass flow would have reached extreme values, leading to engine LP spool overspeed.

## A first engine sizing

Engine volume was estimated by determining engine length and diameter, and considering it as a cylinder. Length was estimated by relying on similar existing LBPR turbofan engines fan diameter to length ratio (from fan entrance to nozzle end). Considered existing engines were the following ones with their respective diameter/length ratio:

- Snecma M88 → 0.2
- GE F-404 & F-414 → 0.21
- EJ200 → 0.19
- PW F100 → 0.24

FESTIP engine fan diameter was 1.6m. Therefore, with an average diameter to length ratio of 0.21 we found an engine length of about 7.6 meters. Fan inlet diameter was increased by 10% in order to include the volume occupied by the engine casing and the various accessory boxes. Hence, total diameter was estimated at about 1.76 meters, leading to an individual engine volume of about 18.5 $m^3$.

Engine weight was estimated by considering similar existing LPBR turbofan engines gross SLS ISA thrust-to-weight ratio. Considered existing engines were the following ones with their respective thrust-to-weight ratio and mass:

- Snecma M88→ 8.5 / 900kg
- GE F-404→ 7.8 / 1040kg
- GE F-414 → 9.0 / 1100kg
- EJ 200→ 9.2 / 990kg
- PW F100 → 7.8 / 1700kg

Therefore, with an average T/W ratio of about 8.5, and our engine's SL ISA take-off gross thrust of 390kN, we found an estimated individual engine weight of about 4.7 tons.

In order to estimate pre-cooler weight, we used data available in a FESTIP2 report (Ref. 5), giving pre-cooler weight as well as various pre-cooler accessories weight in function of engine air mass flow as follows:

- Pre-cooler: 6.8 $kg/(kg/s)_{air}$
- Manifolds: 4.4 $kg/(kg/s)_{air}$
- Control devices: 0.2 $kg/(kg/s)_{air}$
- Support structures: 1 $kg/(kg/s)_{air}$

Considering the calculated Mach 3 air mass flow with pre-cooling of about 355 kg/s, we found an estimated pre-cooler weight of about 2.1 tons, a fully equipped pre-cooler weight of about 4.4 tons, and hence an equipment weight of 2.3 tons.

Another source was found in order to determine a second pre-cooler weight estimation. Ref. 7 referred to two kinds of pre-coolers, the heaviest but most effective one weighing 725kg for a Mach 3 corrected air mass flow of 113kg/s. Our engine's calculated corrected Mach 3 air mass flow of 200kg/s therefore leads to an estimated pre-cooler weight of about 1.3 tons compared to the 2.1 tons estimation made via the FESTIP2 report.

Therefore, total pre-cooled engine weight is estimated between 6 and 6.8 tons without pre-cooler equipment included, and between 8.3 and 9.1 tons everything included.

## Conclusions

More than estimating high speed pre-cooled turbofan performance, this work allowed to point out this engine type's specific features, the most important ones being:

- The very large corrected mass flow, and hence compressor map operating point variations all along the trajectory.
- The resulting requirement for a TIT limitation during the "lower" part of the trajectory.
- Without a fan map and hence geometry change compared to the standard maps, the need for a TIT increase beyond a certain Mach number in order to keep the fan from starting to windmill.

However, engine simulation results validity beyond Mach 4 will have to be investigated. Moreover, following questions also remain to be evaluated:

- Is it realistic to consider nozzle adaptation on the whole trajectory, or would this lead to unacceptable variable geometry nozzle system weight?
- Are the very high fan inlet and fuel injection temperatures beyond Mach 4 acceptable?

Finally, after engine performance integration into the selected vehicle, it remains to be assessed by vehicle designers if a pre-cooled turbofan's increased weight drawback will be compensated by its increased performance compared to a non pre-cooled turbofan switching to ramjet propulsion around Mach 3, but requiring a more complex and heavier dual-mode ramjet module variable geometry.

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