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Multi Purposes Reusable LOX/CH4 Bleed Rocket Engine

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Abstract
The design rules dealing with reusable Liquid Rocket Engines are similar as expendable engine, but in fact they are not.
The reusable engines are to be designed for numerous missions with many start up, shutdown and long life.
Liquid Rocket Engine Reusability leads to improve design rules: low stressed components meaning, low tip Speed for rotating parts and a temperature as low as possible for hot parts like Combustion Chamber.
These considerations promote cold bleed engine burning Lox/Methane propellants.
Indeed, Bleed rocket engine don't use gas generator and the Lox/Methane combination compared to LOX/LH2 according its high bulk density reduces drastically the Launch Vehicle Tanks Sizes.
More compared to Lox/Kerosene the low soot emission and good CH4 thrust cooling capacity without risk of cracking and deposit are in favour of Lox/Methane.
A preliminary design a 250 kN (56 klbf) Bleed Rocket to thrust a Nano Launch Vehicle (NLV) is the topic of this paper.
In complement a 1000 kN (224klbf) four engines (4*250 kN) cluster bay design and a comparison expendable versus reusable design are addressed. This cluster bay is to thrust a medium launch vehicle.

1. Introduction

Lox/Methane may be considered as new promising propellant combination for launch vehicles.
The virtues of methane compared to kerosene are its worldwide availability and higher ISP. Compared to LOX/LH2 its bulk density which reduces drastically the tanks sizes [1].

Dealing with the reusability the following design drivers need to be followed:
- Low stressed components meaning moderate temperature for hot parts: Thrust Chamber, Turbine and lines.
- Low tip speed for rotating parts: pump impeller and turbine blades.

A significant feature is to be emphasized: The soot formation may be significant with LOX/kerosene compared to LOX/CH4 [3].

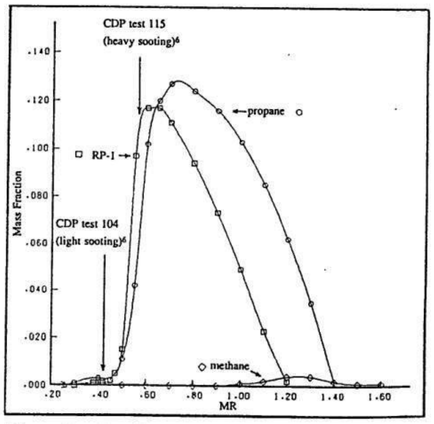

Figure 1: Soot mass fraction predictions vs MR for LOX/RP-1, Propane and Methane. Pc= 68 atm.

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We consider that the soot deposit is inacceptable owing to the performances losses and fire during engine reignition are both risks for the mission launcher. For example, Titan launcher losses 14% of performance due to carbon deposit [4].

Moreover, this is a favourable argument, LOX/CH4 authorises one single shaft turbopump: one unique turbine drives the LOX and methane pump.

The benefits of LOX/CH4 are fully documented and many developments through the world are in progress [2].

From the current cycle engines, the bleed engine offers a great merit. It doesn't use gas generator to feed the turbines driven the pumps: the enthalpy required for the turbines is pick up from the regenerative thrust chamber cooling circuit; so, by evidence no soot can be deposit.
The engine start up and shut down sequence are simple: only the combustion chamber is to be fired.
The Thrust and Mixture ratio can be easily tuned; the GG absence is a high Reliability Guaranty (see LE5 lesson learned).

For many years, several reusable launch vehicle concepts for commercial market have emerged [5] without success. Currently Space X tries to raise the challenge after the return ability success of Falcon 9.

It is to be mentioned, however the development by ASL of a reusable gas generator L0X/CH4 engine to power a suborbital space plane for tourism. [7]

According [6] the profitability of reusability is doubtful and depending of many factors: flight cadence and cost for engine health control and refurbishment between missions.
Our approach doesn't take in fact the reusability economic aspects conditions.

2. Considerations on Bleed Rocket Cycle:

The following figure illustrates the bleed cycle flow lay out:

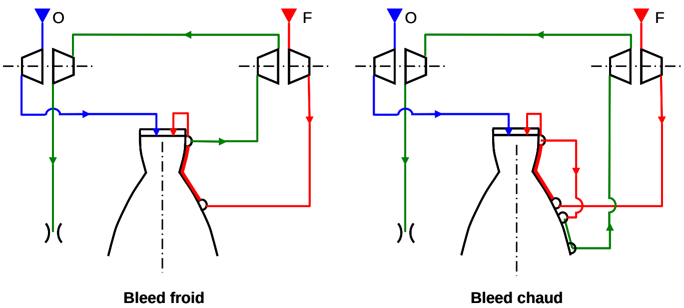

Figure 2: Cold bleed and hot bleed cycles engine.

In this cycle the gas required amount to drive the turbopump is picked up at the chamber's exit cooling system (cold bleed) or super-heated by the outer part of the nozzle extension (Hot bleed) and expanded at ambient pressure, the cycle is equivalent by its simplicity to the expander. However, due to the "by-pass flow rate" (turbine mass flow) its performance is lower.

The bleed cycle requires a fuel from which the vapor phase presents thermodynamic data (Cp and γ) predictable so this consideration eliminate kerosene.

The fact that the enthalpy available for the turbine comes from the combustion chamber only, the consequence is the turbine power limits and engine thrust. The thrust and Mixture Ratio can be easily adjusted.

All the existing Bleed engines are cryogenic (LOX/LH2). The Japan HIIa launch vehicle [8] uses such engine to thrust the upper stage. The Japanese experience is fruit full: originally the upper stage engine used a gas generator but due to GG icing and it's no ignition (LE5), the engine was modified first to a hot bleed (LE-5A) and secondly to a cold bleed (LE-5B).

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The following table refers to these Japanese engines:

	UNIT	LE-5	LE-5A	LE-5B
Engine cycle	-	GG	Hot bleed	Cold bleed
Fvac	kN	105	124	140
ISPvac	sec	450,2	452.9	450
PC	bar	37	40.6	36.4
Inlet turbine temp.	K	840	603.9	355.1
Σ Ratio	-	140	130	110

Table 1: Japanese engines

2.1 Design Bleed cycle relations

Referring to the power equilibrium between turbines and pumps the following relation [11] can be written to estimate the turbine mass flow (i_{tu}).

$$i_{tu} = \frac{\frac{Rm}{\rho o} i_F \left[\frac{K_{inj_O} \text{ x } Pc - P_{asp_O}}{\eta_{ppe_O}}\right] + \frac{i_F}{\rho_F}\left[\frac{Pc\left(K_{inj_F} + K_{reg}\right) - P_{asp\,F}}{\eta_{ppe_F}}\right]}{Tet\, C_p \left[1 - \frac{1}{\Pi_{tu}^{\frac{\gamma-1}{\gamma}}}\right]\eta_{tu}}$$

The variables are:

$$\Delta P\ pompe_o = Pc + \Delta P\ injo\ \text{-}\ Pasp$$

$$Kinj\ F = \frac{Pinj_F}{P_c}\ \text{(valeur typique 1,15)}$$

$$Kreg = \frac{\mathbf{\Delta Preg}}{\mathbf{P_C}}\ \text{(valeur typique 0,5)}$$

$$KinjO = \frac{\mathbf{Pinj\ _O}}{\mathbf{P_C}}\ \text{(valeur typique 1,4)}$$

Isp must be estimated taken in account the losses consecutive to the open cycle corresponding the turbine mass flow (i_{tu}):

$$ISP_{bleed} = ISP_{TC} * i_{tc}/i_{mot} + ISP_{echap} * i_{tu}/i_{mot}$$

The variable able to fulfil the equilibrium power is the Inlet Turbine Temperature (TET).
Clearly, higher is the TET, lower is the turbine mass flow and by consequence higher is the ISP.

However, TET is also the outlet temperature of the thrust chamber cooling circuit and depends of the inner hot gas heat transfer which is mainly driven by pressure combustion ($\Phi \approx 1.9 Pc0.8$). TET is limited by the inner hot gas side, to about 800K for expendable Thrust Chamber, to save its life time [11].

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2.2 Design of a 250 kN first stage Reusable Bleed Rocket Engine

The reference [14] informs that from 2000 to 2750 nano/micro sats would be launched in 2016- 2020 for a 702M$ revenue their mass will of 50kg class. The references [12] and [13] present an overview of this market.

The 250 kN first stage engine is foreseen to thrust a Nano Launch Vehicle (NLV) the reference mission considered is 100 kg payload class mass on a 600 km altitude SSO orbit.

The following parameters are to be selected:

- Combustion Pressure Chamber (Pc)
  To avoid any issue related to two phases flow in cooling circuit, to avoid droplets in combustion, source of high frequency instability it is selected a Pc >Pcritical (50bar for CH_4). According that, the turbine power is limited to the heat flux and enthalpy picked up to the Chamber wall, the pressure chamber for this engine is selected to 60 bar.
- Rmep (Inlet mixture ratio =io/if) is selected to optimize ISP considering the constrain: Thrust Chamber mixture ratio RmTC =io/(if-itu) must be lower than 4 (stochiometric).
- With a pressure chamber at 60 bar, the nozzle extension area ratio (Σ) is selected to avoid jet separation at sea level, so Σ =30.
- To mitigate the thermal stresses, the TET considered is 400K (cold bleed).
- To avoid low cycle fatigue (number of start and shut down) and creeping, according reusability criteria: the maximum combustion chamber hot gas side (Twall) have to be lower than 800K.

According theses parameters, the following chart presents the main engine characteristics :

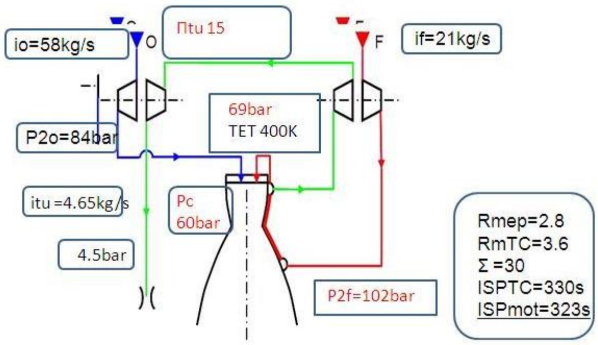

Figure 3: Preliminary design Bleed - LOX/CH4 - 250 kN.

The following figure presents the methane evolution on Mollier Chart :

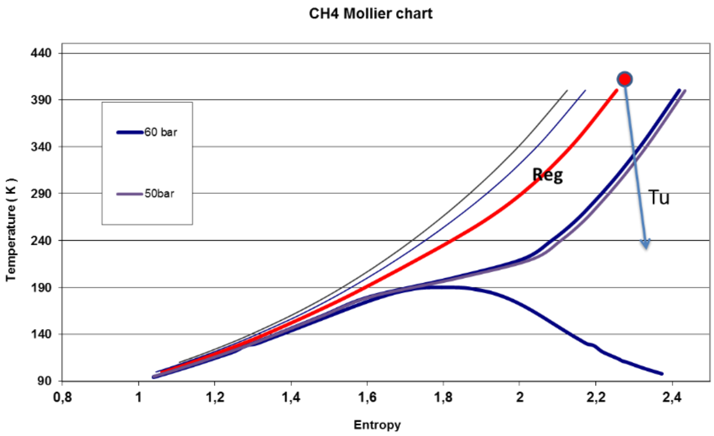

Figure 4:Methane Mollier Chart.

We may observe, the methane is always gaseous (no two phases flows in thrust chamber cooling circuit).

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2.3 Combustion chamber design

The combustion chamber design is similar at the expander cycle. A very tricky task owing to minimize the hot wall temperature and to be able to provide the enthalpy required for the turbine.

We may observe, the hot wall gas temperature is lower than 800 K.

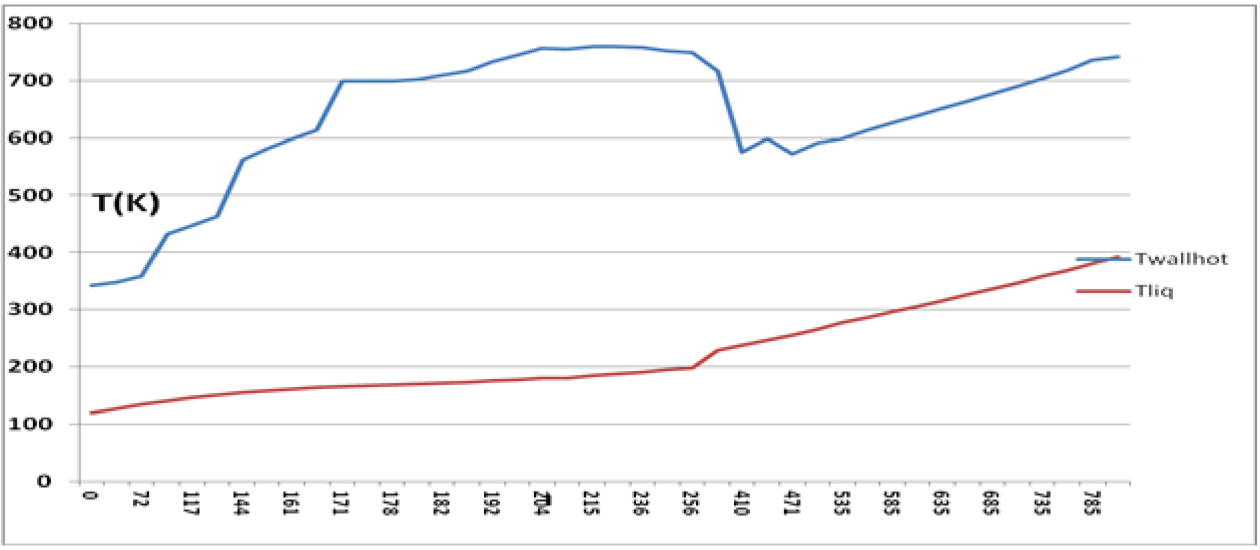

Figure 5: Hot wall gas temperature and methane liquid temperature.

2.4 Turbopump design

The following picture presents a LOX/CH4 single stage single shaft Turbopump [15]. An Impulse or Rateau or Curtiss turbine type drives the LOX pump and methane pump. Each of these pumps is single stage centrifugal type, like this:

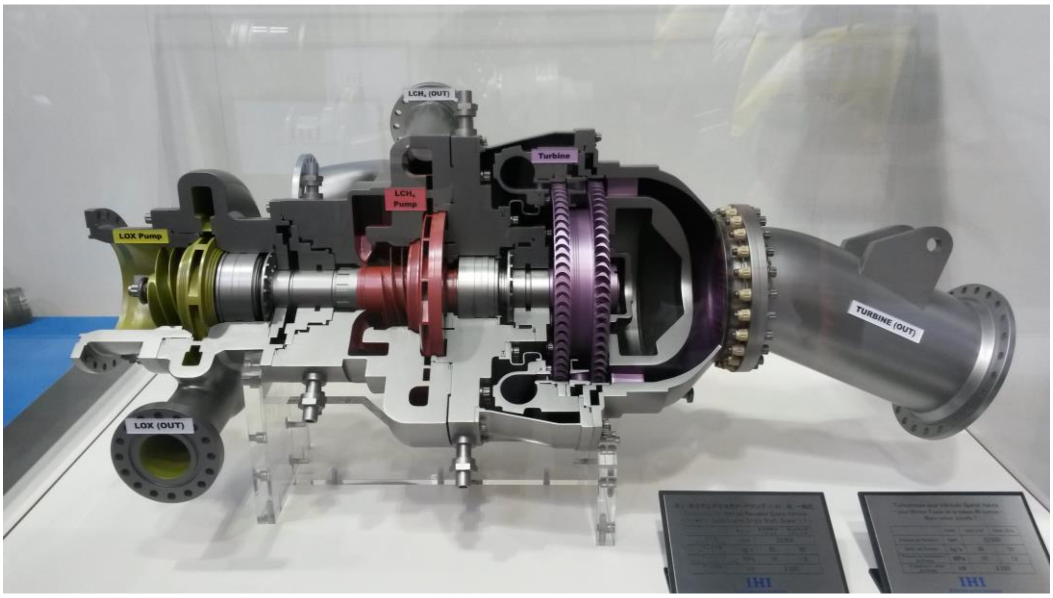

Figure 6: LOX/CH4 single stage single shaft turbopump - Le Bourget 2015

The design criteria to optimize a single shaft Turbopump is to minimize the power required for the turbine. To do this, by sweeping the rotational speed Ns (Specific Speed) and (Ds Specific Diameter) the global efficiency of each pump by empirical relation are estimated [16].

Furthermore, care to be done to the impeller tip velocity compliant with its strength material.

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It has care, the turbopump data are :

	LOX		CH4
Delta P (bar)	81		99
Rpm		25 000	
Tip speed (m/s)	130		205
Ns (in US unit)	2090		834
Ds	0,08		0,172
η_{global}	0,83		0,72
W tu (kW)		**1072**	

Table 2 : Turbopump calculation

We may note, that only one single stage is required and the tip speeds are low for each propellant.
The following figure chart presents in the NASA chart the status of each Pump [16]:

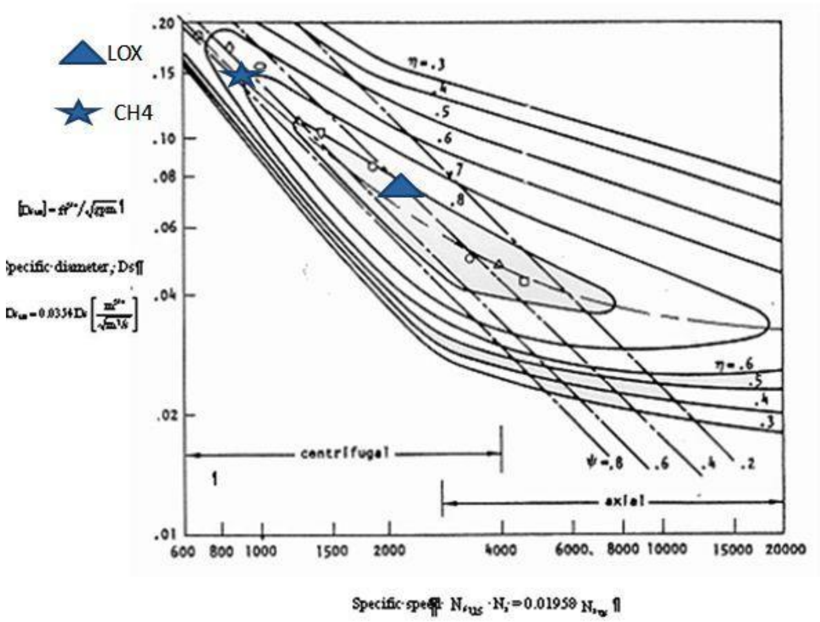

Figure 7: Representative Ns-Ds diagram for centrifugal and axial flow turbopumps.

3 Comparison Expendable/reusable engine

The reusability constrains lead to modest performances. To enhance it, a solution is to increase Turbine Inlet temperature (TET) to 600K by adding a heat exchanger on nozzle extension (Hot Bleed), the gain in ISP would be more than +2s.
If we need more ISP:

- to increase up the TET at 900K, by using a gas generator. In consequence, soot deposit is to be taken in account.
- to increase combustion chamber pressure (Pc), increasing by consequence the combustion hot gas side temperature up to 800K. The engine performances would be the following:

	GG cycle [10]	Hot Bleed	Cold bleed
Pc (bar)	100	60	60
Σ	45	30	30
TET (K)	900	600	400
ISP (s)	**345**	**325**	**323**

Table 3: Cycles engines performances

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4 Nano Launch Vehicle design.

Using two stages launch vehicle, like Falcon 1[17].
No extra mass dealing with purpose to toss Back is added.
The Nano Launch Vehicle (NLV) using two stages LOX/CH4, is designed for a 100 kg payload class mass on 600 km SSO orbit. According the fact that the first stage thrust is specified and according with the state of art for Lift off Acceleration, the Glow are the same for each version. The second stage of the NLV is powered by a Pressure Fed Engine, with an Isp of 310s.

The following chart presents the impact of the engine cycle on payload mass:

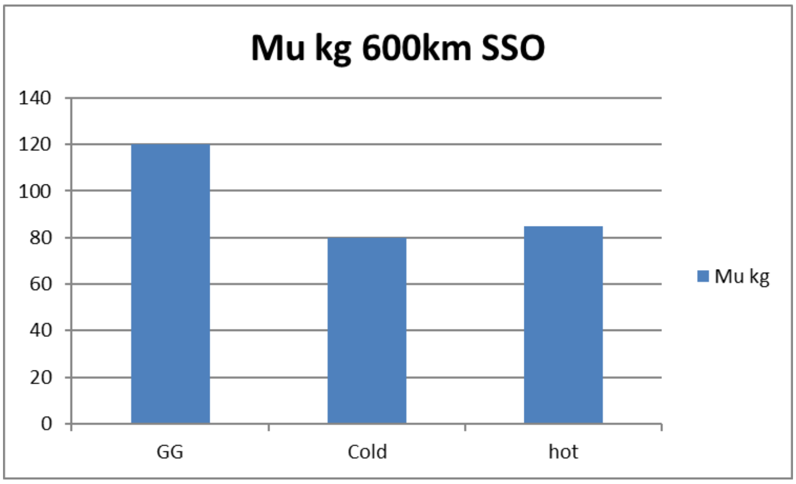

Figure 8: Engine cycle impact on payload mass

So, the cold bleed according is simpler and the best.

The design is in the small launch vehicle family:

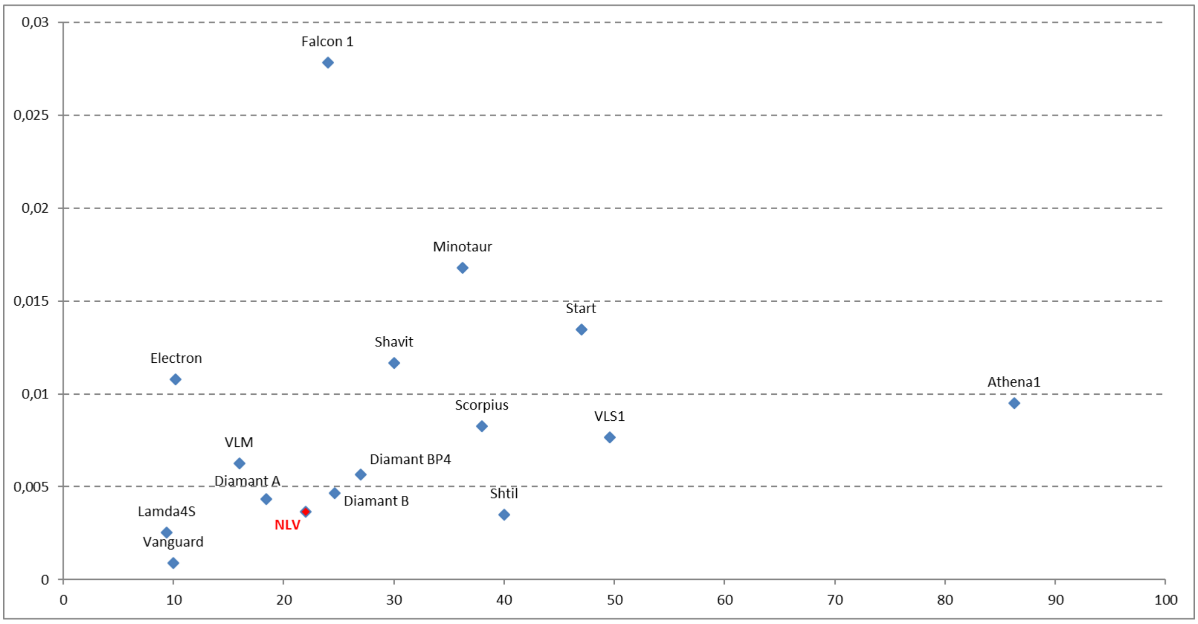

Figure 9: MU/GLOW versus GLOW

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5 1000 kN Bleed Rocket Engine Design

The 1000 kN LOX/CH4 engine is designed to power a medium launch vehicle.
The bleed cycle main issue is that the enthalpy turbine required is to be provided by the combustion chamber cooling circuit and therefore limited to the max hot wall temperature acceptable and geometry. By consequence the thrust is limited.
Considering a 1000kN engine; 20 MW thermal Power has to be provided.
So, it is better to design a high thrust engine (1000kN) by clustering the previous 250kN combustion chamber in a propulsive bay.
It would be attractive to feed the four thrust chambers by a unique turbopump.

The following engine parameters are:

- Combustion Pressure Chamber (Pc): 60 bar
- Nozzle extension area ratio (Σ) =30.
- TET considered is 400K (cold bleed).

The flow schematic is the following :

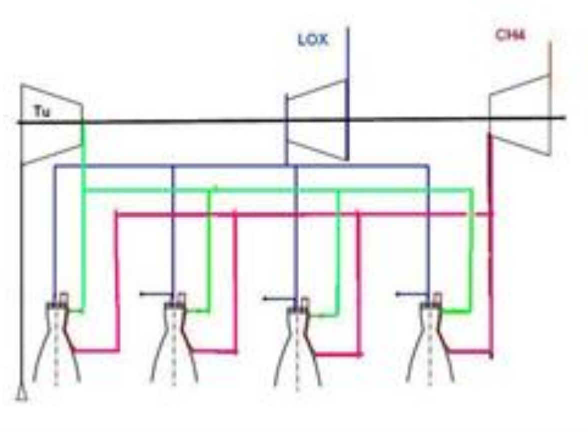

Figure 10: Four combustions chambers in propulsion bay

The Turbopump characteristics are :

	LOX		CH4
Delta P (bar)	81		99
Rpm		12 000	
Tip speed (m/s)	130		205
Ns (in US unit)	2005		801
Ds	0,082		0,178
η_{global}	0,85		0,68
W tu (kW)		**4350**	

Table 4: Turbopump calculation

6 Three Stages medium lift launch vehicle

Using on first stage with the 1000 kN Bleed (4*250kN), on second stage with the 250kN cold bleed and on third stage with an expander Lox /CH4 engine [18], it is feasible to design a very attractive launch vehicle.
This solution presents one technology (LOX/CH4) for all the stages, is a source of substantial cost reduction during development, manufacturing and operations.
The vehicle with a GLOW of 85 t will be able to orbit on 800 km SSO a 2000 kg class mass payload

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7 Summary and conclusion

A preliminary Bleed Rocket engine design considering constrains dealing with reusability; meaning long life many missions with the minimum of ground operations is presented.
A Nano Launch Vehicle and a medium launch vehicle powered by bleed LOX/CH4 engines are summarily mentioned. This paper shows the possibility to design common engine for two very attractive launchers.

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