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Preliminary Design of a Radial Turbine for Methane Expander Rocket-Engine

Angelo Leto[a,b,1], Aldo Bonfiglioli[b]

[a] CIRA Italian Aerospace Research Center, via Maiorise, 81043 Capua(CE), Italy
[b] University of Basilicata, V.le dell'Ateneo Lucano 10, 85100, Potenza(PZ), Italy

Abstract

The present paper summarizes recent research efforts carried out at the Italian Aerospace Research Centre, CIRA, aimed at using radial turbines in modern rocket-propulsion systems. Over the last few years, CIRA has been involved in the HYPROB program, funded by the Italian Ministry of Research (MIUR). HYPROB aims at developing competences to consolidate the national background on rocket-engine systems for future space applications. Since nowadays liquid methane represents an innovative fuel in aerospace propulsion, one of HYPROB's objectives is the development of a simulation tool for the preliminary design of the radial turbines used to drive the turbo-pumps in expander-cycle rocket-engines operating with methane.

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Keywords: Expander-cycle rocket-engine, radial turbine, methane

1. Introduction

Designing turbo-machines can be an extremely complex task, as it ultimately requires the detailed analysis of the flow behavior inside the stationary and moving vanes. The use of cryogenic liquids, such as those used in rocket-engines, further complicates the task. The turbines and pumps that operate in an expander-cycle rocket-engine are closely interlinked with other components of the engine, such as the thrust chamber and the regenerative cooling system that surrounds the chamber and nozzle. When the entire engine is being designed, computationally fast simulation methods are mandatory to be able to explore in a reasonable time-frame the design space spanned by a multitude of design parameters. In this framework, a preliminary, though still reliable, design of the turbo-machines can be achieved by using low-dimensional models, supplemented with semi-empirical correlations. Only in the final design stage, detailed CFD analyses should be used to fine-tune the geometries obtained from the simplified model.

Axial flow turbines are those more commonly used to drive the turbo-pumps that supply the fuel and the oxidizer to the thrust chamber. However, radial turbines may exhibit a number of advantages with respect to axial-flow turbines, including: compactness, high reliability at low cost and good performances even with

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Nomenclature

α	Flow angle	N_s	Specific speed
β	Blade angle	p	Pressure
ℓ	Blade height	Q	Volumetric flow rate
η	Total-to-static efficiency	T	Temperature
ω	Angular velocity	u	Blade velocity
ρ	Density	V	Absolute velocity
h	Enthalpy per unit mass	W	Relative velocity
L	Work per unit mass		

unsteady flows. Radial turbines are nowadays routinely and successfully used in automotive applications, whereas their use in aerospace propulsion has not been reported very frequently [1].

This paper describes the development of a MatLab tool, called RTGD (Radial Turbine Global Design), suitable for the preliminary design of radial turbines to be used in expander-cycle rocket-engines. RTGD relies on a one-dimensional model largely inspired by (but not identical to) Rohlik's model [2] that allows determining several turbine's features, including the turbine's geometry and its efficiency. The CoolProp [3] libraries provide the thermodynamic properties of the working fluids.

The one-dimensional model will be briefly described in Sect. 2 and it will be applied to three different expander-cycle rocket-engines in Sect. 3. The first two engines use hydrogen as fuel, whereas the third engine uses methane. Whenever available, comparisons with other computational and experimental data will be made.

2. Radial Turbine Global Design Model

The model implemented in the RTGD code relies on a one-dimensional aero-dynamic model largely based on the seminal work by Rohlik [2] and Glassman [4] to which the reader is referred for details. Hereafter we shall only recall the key design parameters used in the model.

The specific speed:

$$N_s = \frac{\omega \sqrt{Q}}{\Delta h_{id}^{3/4}} \tag{1}$$

is the parameter commonly used to describe the operating requirements of a radial turbine in terms of angular velocity, ω, volumetric flow rate at the rotor outlet, Q, and ideal work, Δh_{id}.

The work output by the turbine per unit mass, L, is obtained from the ideal work via the total-to-static efficiency:

$$\eta = \frac{L}{\Delta h_{id}}, \tag{2}$$

which has been modeled either using the fit proposed by Aungier [5]:

$$\eta = 0.87 - 1.07\,(N_s - 0.55)^2 - 0.5\,(N_s - 0.55)^3\,, \tag{3}$$

which is graphically displayed in Fig. 1, or the approach described by Dixon in [6], to which the reader is referred for details. Even though all the results to be presented in Sect. 3 have been obtained using the total-to-static efficiency provided by Eq. (3), the two different efficiency estimates will be displayed alongside in order to gain some indication upon the influence of the different loss models.

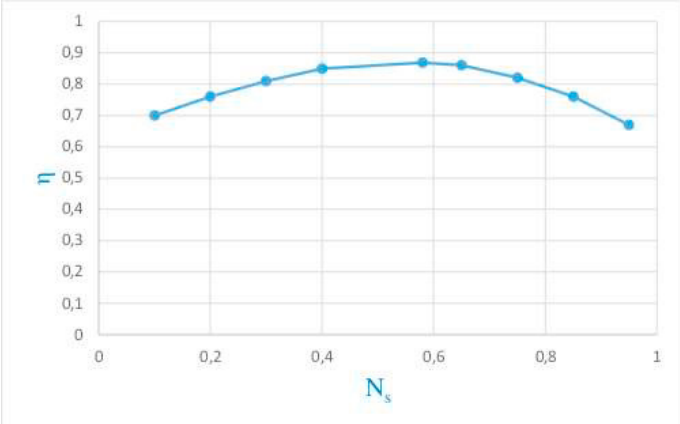

Fig. 1. Total-to-static efficiency, η, as a function of the specific speed, N_s, computed according to Eq. 3

In a typical design exercise, the power required and the rotational speed are constrained by the turbo-pumps that are driven by the turbine(s) and the volumetric flow rate depends upon the fuel flow rate needed to provide the required thrust. The examples to be presented in the following section will clarify how the design process is performed.

3. Test-cases

This section describes the application of the RTGD tool to the design of radial turbines for three different expander-cycle rocket-engines; the first two use hydrogen as fuel, the third one uses methane. The results will be compared against available simulation and experimental data, whenever available.

3.1. RL10A-3-3A Rocket-Engine

The RL10A rocket engine, whose initial design dates back to the 1960s, is still an important component of the United States space infrastructure. Two RL10 engines form the main propulsion system of the Centaur upper stage vehicle, which is currently used on the Atlas V launcher to boost commercial, scientific, and military payloads into Earth orbit and beyond. The RL10A-3-3A engine, whose main components are sketched in Fig. 2, is based on an expander-cycle. It can be seen from Fig. 2 that liquid hydrogen (the fuel) is first compressed using a two-stage centrifugal pump and then flows through the cooling system where it vaporizes, thus cooling the thrust chamber and nozzle. Finally, gaseous hydrogen expands through an axial turbine that drives both the fuel and the oxidizer (liquid oxygen, LOX) turbo-pumps. The fuel turbo-pump is directly connected to the turbine, whereas a gearbox connects the turbine and the oxidizer turbo-pump, which revolves at about half the rotational speed of the fuel turbo-pump. The interested reader is referred to Ref. [7] for an in-depth description of expander-cycle rocket-engines.

We have used the RTGD tool to design a radial turbine that replaces the axial turbine used in the original RL10 engine. Table 1 shows the input parameters: *i*) pressure and temperature at the turbine inlet, *ii*) the power required by the two turbo-pumps, *iii*) the rotational speed of the fuel turbo-pump which is directly driven by the turbine, *iv*) the fuel flow rate that guarantees 73 kN of thrust. In the RTGD simulation the turbine outflow pressure has been iteratively adjusted so as to match the required fuel flow rate. The obtained results are listed in Tab. 1 where they have been compared with those [8] of the axial turbine of the RL10 engine, obtained using a similar simulation tool (ROCETS) developed at NASA [9]. Not surprisingly, the results of Tab. 1 show that the radial turbine operates with a smaller pressure drop than that required by the axial turbine. As far as efficiency is concerned, the relative difference between Dixon's and Aungier's models is about 7%, with Dixon's model predicting higher losses than Aungier's.

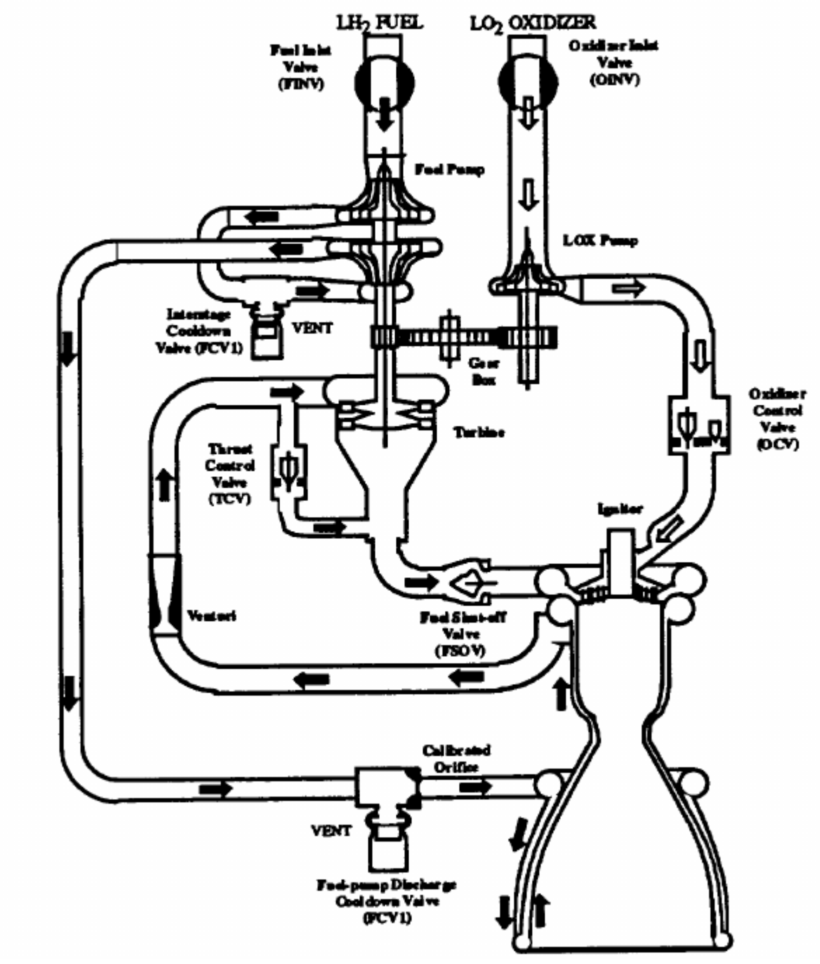

Fig. 2. RL10A-3-3A Engine System Schematic, reprinted from [8]

Table 1. Comparison between the RTGD and ROCETS simulation results for the RL10A-3-3A engine.

Model input		
Fuel (LH_2) flow rate [kg/s]	2.7945	
LH_2 turbo-pump rotational speed [rpm]	31494	
Power required [kW]	5600	
Turbine inlet pressure [bar]	56.1	
Turbine inlet temperature [K]	213.4	
Model output	**RTGD**	**ROCETS**
Turbine stator exit temperature [K]	205.3	N/A
Turbine rotor exit temperature [K]	198.5	200.5
Turbine exit pressure [bar]	43.15	40.0
Efficiency η	0.87 (Aungier) 0.81 (Dixon)	0.7353

3.2. VINCI Rocket-Engine

VINCI is a new-generation upper-stage cryogenic rocket-engine which is being developed by Snecma and other European partners in the framework of an European Space Agency (ESA) programme. Firing tests started in April 2005 and engine qualification is expected to take place in 2017.

Similarly to the RL10 engine previously described, VINCI is an expander-cycle rocket-engine fed with liquid hydrogen and liquid oxygen which is capable of delivering 180 kN of thrust when supplied with 5.8 kg/s of hydrogen. Its working cycle is similar to the one shown in Fig. 2, but not identical, because in the VINCI engine the fuel and oxidizer turbo-pumps are driven by two different turbines. We shall here concentrate on the fuel side, where a two-stage centrifugal turbo-pump compresses liquid hydrogen to 225 bar. Hydrogen vaporizes in the cooling system where it experiences a pressure drop of 45 bar so that gaseous hydrogen enters the turbine at 180 bar. Other input data needed to design the turbine that drives the fuel turbo-pump are listed in Tab. 2. Observe that only about 80% of the fuel mass flow (4.8 kg/s) flows through this turbine, whereas the remaining 20% flows through the turbine that drives the oxidizer turbo-pump.

Table 2 shows a comparison between the RTGD results and the few available firing-test data for the fuel pump/turbine group. The experimental data refer to an axial turbine. Conclusions are similar to those drawn

Table 2. Comparison between the RTGD simulation results and experimental data for the VINCI engine.

Model input		
Fuel (LH_2) flow rate [kg/s]	4.8	
LH_2 turbo-pump Rotational speed [rpm]	90000	
Power required [kW]	2500	
Turbine inlet pressure [bar]	180	
Turbine inlet temperature [K]	240	
Model output	**RTGD**	**Experimental data**
Turbine stator exit temperature [K]	222.2	N/A
Turbine rotor exit temperature [K]	204.3	N/A
Turbine exit pressure [bar]	98.4	90.0
Efficiency η	0.87 (Aungier) 0.82 (Dixon)	0.79

for the RL10 engine as far as the pressure drop and efficiencies are concerned.

3.3. Methane Expander-Cycle Rocket-Engine

In this section the RTGD tool will be used to design the radial turbines of a rocket-engine operating with methane. Methane is a relatively new fuel in rocket-engines; it is a low-density hydrocarbon that has advantages over both kerosene and hydrogen and, therefore, can be a convenient trade-off between these two. Methane is a cryogenic fuel, but requires less insulation and raises fewer handling concerns than hydrogen. Its density is about six times higher than that of hydrogen, and half that of kerosene. Therefore, methane propellant tanks will weight much less than hydrogen tanks. In addition, methane is not corrosive, has low toxicity and it is relatively easy to extract from natural gas. Finally, methane gives the highest specific impulse among all hydrocarbons. Therefore, a liquid oxygen/methane expander-cycle rocket-engine could provide an ideal, low-cost option for the upper-stage engine. It is for these reasons that an international joint venture between the Italian company AVIO and the Russian company KBKHA is currently developing a Methane Expander-Cycle Rocket-Engine (MECRE), namely the LM10-MIRA demonstrator [10].

Since the available information for the LM10-MIRA methane-oxygen rocket is limited to the thrust only, the Rocket Propulsion Analysis (RPA) [11] software has been used to guess the missing thrust-chamber data, which are listed in Table 3.

Table 3. Data obtained for the LM10-MIRA MECRE using the RPA software

Thrust [kN]	70.0
Specific impulse [s]	371.3
Total mass flow rate [kg/s]	26.5
Oxidizer mass flow rate [kg/s]	20.4
Fuel mass flow Rate (CH4) [kg/s]	6.1
Pressure chamber [bar]	55.0

The MECRE features a parallel feed system configuration, consisting of two turbo-pumps, one for methane and one for LOX. Both pumps are directly connected to the radial turbine by which they are driven; the fuel pump has to increase the pressure of liquid methane of 160 bar, because further losses occur in the cooling system; the oxidizer pump has to increase the pressure of LOX only by 80 bar. The oxidizer and fuel pumps have been designed using a similar tool called Pump Global Design (PGD), which has been described in [12]. The RTGD tool has been used for the preliminary design of the two radial turbines: Table 4 shows both the required input data and the computed results. As usual, subscripts 0, 1 and 2 respectively refer to the stator inlet, stator outlet or rotor inlet, rotor outlet. Observe that the methane mass flow at the discharge of the regenerative cooling system is split between the two radial turbines: 80% of it flows through the turbine that drives the fuel pump and the remaining 20% is conveyed into the turbine that drives the LOX pump. For this engine the are no available experimental or simulation data to compare with.

The results obtained for the velocity triangles are summarized in Table 5 which uses the nomenclature adopted by Rohlik [2]: angles are measured with respect to the radial direction, rather than the tangential one, as sketched in Fig. 3. Observe in Table 5 that the angle β_1 is small, but the blade is not necessarily radial at the rotor inlet. This is because when using cryogenic fuels the turbine inlet temperature is relatively low, see Table 4, so that the impeller blades need not to be radial for structural reasons.

4. Conclusions

This paper deals with the design of radial turbines, in place of the more commonly used axial ones, to drive the turbo-pumps that supply both the fuel and the oxidizer to the thrust chamber of an expander-cycle rocket-engine operating with methane as liquid fuel.

The design method relies on a low-dimensional model, originally developed at NASA [2, 4]. The choice of a simple and, therefore, computationally inexpensive simulation tool was motivated by the fact that the

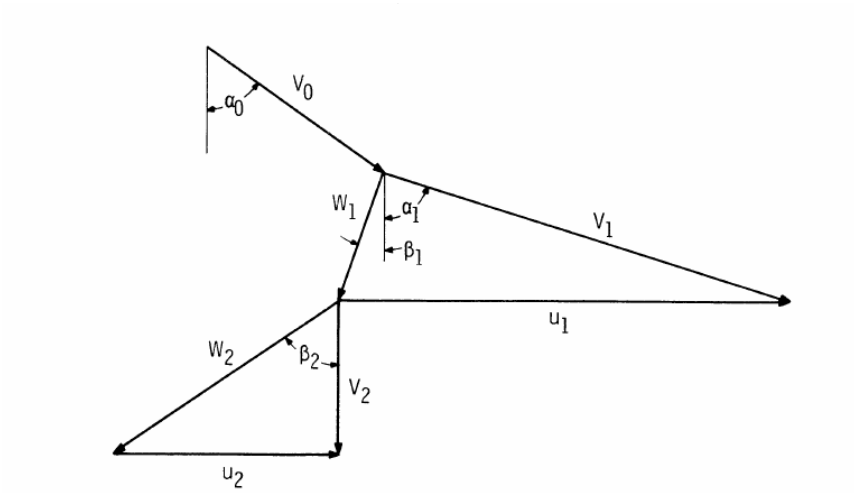

Fig. 3. Velocity triangle, re-printed from [2].

Table 4. Turbine input and output data for both turbines driving the fuel and oxidizer turbo-pumps

Model input	**Fuel**	**Oxidizer**
Turbine inlet total temperature T_0^0 [K]	450	450
Turbine inlet total pressure p_0^0 [bar]	105	105
Fuel mass flow rate (CH4) [kg/s]	4.8	1.3
Power [KW]	305	134
Number of revolutions [rpm]	52970	15500
Model output	**Fuel**	**Oxidizer**
Rotor inlet pressure p_1 [bar]	94	88.4
Rotor exit pressure p_2 [bar]	83	72
Efficiency η (Aungier)	0.87	0.86
Efficiency η (Dixon)	0.84	0.85
Number of blades z	14	14
Inlet rotor temperature T_1 [K]	438	432
Exit rotor temperature T_2 [K]	428	416
Inlet rotor gas density ρ_1 [kg/m^3]	42	40
Exit rotor gas density ρ_2 [kg/m^3]	38	34
Rotor diameter D_1 [m]	0.10	0.413
Average exit diameter $D_{2,m}$ [m]	0.035	0.103
Exit blade height ℓ_2 [m]	0.011	0.062
Tip exit rotor diameter $D_{2,e}$ [m]	0.045	0.165
Hub exit rotor diameter $D_{2,i}$ [m]	0.024	0.041

RTGD MatLab tool that has been developed is going to be one of the building blocks of a Concurrent Design Facility aimed at designing all the different components of the entire rocket-engine.

The RTGD model has been developed with the intent to predict the geometry and performance of the turbine, thus providing useful information for the design of a radial turbine for future application in an expander-cycle rocket-engine demonstrator that will be designed and tested at CIRA.

Table 5. Velocity triangles for the fuel turbine

Symbol	Inlet	Exit
Flow angle α [°]	75	0
Absolute rotor velocity V [m/s]	261	110
Blade rotor velocity u [m/s]	269	94
Relative rotor velocity W [m/s]	73	145
Blade rotor angle β [°]	14	42

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