Automation of a Turbine Tip Clearance Preliminary Calculation Process

by

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VI

AUTOMATISATION DU PROCESSUS DE CALCUL DU JEU EN BOUT D'AUBE DANS UNE TURBINE

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RÉSUMÉ

La conception d'un turboréacteur est un processus itératif et multidisciplinaire qui requiert une interaction efficace entre les outils et processus des différentes disciplines impliquées afin d'obtenir le meilleur compromis possible. La conception de ces moteurs d'aéronefs comprend deux étapes principales: les phases de conception préliminaire et détaillée. Pendant la première phase, le temps est la principale contrainte ce qui peut impacter la précision des résultats. Cela peut compromettre la capacité de l'ingénieur à explorer complètement tous les designs potentiels et cela pourrait donc mener à la sélection d'un assemblage non-optimal. Étant donné le temps requis pour effectuer les analyses durant la phase de conception détaillée, il y est très difficile de corriger un concept insatisfaisant. L'utilisation de techniques d'optimisation multidisciplinaire (MDO en anglais) durant la phase de conception préliminaire (PMDO) permet de résoudre ce problème en investissant plus d'efforts tôt dans le processus de conception afin de prévenir la sélection d'un concept insatisfaisant. L'implémentation d'un système de PMDO requiert de produire autant de données que possible au début de la conception puisque c'est durant cette phase que la liberté de modifier le design est maximale. Cela impose d'utiliser des outils de plus haute fidélité qui communiquent efficacement entre eux. Étant donné l'impact du jeu à l'extrémité des aubes de la turbine sur l'efficacité d'un turboréacteur, avoir un outil permettant de prédire efficacement celui-ci est une étape nécessaire dans l'implémentation d'un système de PMDO pour la turbine. De plus, la prédiction de ce jeu est un bon candidat pour l'implémentation de pratiques de PMDO à plus petite échelle étant donné que celle-ci requiert différentes sortes d'analyses menées à la fois sur le rotor et le stator. Cette thèse présente les résultats obtenus grâce au développement d'un processus automatisé exécutant des analyses de température et de contraintes sur le rotor et le carter de la turbine, et menant au calcul de la variation du jeu en bout d'aube de la turbine durant une mission de vol donnée ainsi qu'à la prédiction du jeu nécessaire à froid. Afin d'automatiser les analyses effectuées sur le rotor et le carter, le processus proposé intègre un calculateur de conditions aux frontières et interagit avec un générateur de système d'air secondaire simplifié ainsi qu'avec plusieurs outils de conception basé sur des modèles CAD paramétrés. Comparé à un processus préliminaire régulier de prédiction du jeu en bout d'aubes, le processus proposé offre une amélioration considérable de la précision des résultats puisqu'ils se sont révélés être proches de ceux générés durant la phase de conception détaillée et utilisés comme objectifs. De plus, ce processus de conception s'est révélé être plus rapide qu'un processus préliminaire régulier tout en menant à une réduction du temps nécessaire durant la phase de conception détaillée. Le système étant automatisé et plus rapide que celui d'un processus régulier d'étude préliminaire, il a été possible d'effectuer des itérations afin d'obtenir la mission de vol la plus critique en terme de fermeture du jeu en bout d'aubes. Le système proposé a également permis d'effectuer une

VIII

analyse de sensibilité ciblée qui a amené à l'identification de plusieurs paramètres d'importance lors de l'optimisation du jeu en bout d'aubes. Finalement, requérant moins de données de la part de l'utilisateur, ce système réduit le risque d'erreurs humaines tout en laissant les décisions importantes au designer.

Mots-clés: Optimisation multidisciplinaire et préliminaire; turboréacteur; jeu en bout d'aube; Analyses automatisées

AUTOMATION OF A TURBINE TIP CLEARANCE PRELIMINARY CALCULATION PROCESS

Maxime MORET

ABSTRACT

An aeronautical gas turbine engine design is a multidisciplinary iterative process requiring an efficient interaction between each discipline tool and process in order to find the best compromise satisfying all the conflicting domains involved. The gas turbine engine design traditionally has two main stages: the pre-detailed design and the detailed design phases. During the first phase of the design, time is the main concern and the fidelity of the results may be impacted. This may compromise the engineers' ability to thoroughly explore the envelope of potential designs and thus lead to the selection of a sub-optimal system concept. Considering the time-consuming analysis and resources-intensive tools used during the detailed design phase, it is extremely difficult to correct an unsatisfactory concept at that stage of an engine's design. The use of Multidisciplinary Design Optimization techniques at a preliminary design phase (Preliminary MDO or PMDO) allows correcting this by investing more effort at the pre-detailed phase in order to prevent the selection of an unsatisfactory concept early in the design process. PMDO system implementation requires bringing as much knowledge as possible in the early phases of the design where the freedom to make modification is at a maximum. This imposes the use of higher fidelity tools that communicate effectively with each other. Considering the impact of the turbine tip clearance on an engine's efficiency, an accurate tool to predict the tip gap is a mandatory step towards the implementation of a full PMDO system for the turbine design. Moreover, tip clearance calculation is a good candidate for PMDO technique implementation considering that it implicates various analyses conducted on both the rotor and stator. This thesis presents the results obtained by developing an automated process to execute thermal and stress analyses on a turbine rotor and housing, and leading to the computation of the turbine stage's closure during a given mission along with its cold build clearance. For the analyses automation of the rotor and housing, the proposed conceptual system integrates a thermal boundary conditions automated calculator and interacts with a simplified air system generator and with several design tools based on parameterized CAD models. Compared to a regular preliminary tip clearance calculation process, the proposed conceptual system offers a considerable increase in the accuracy of the results as they revealed to be close to the one generated by the detailed design tools used as target. Moreover, this design process revealed to be faster than a common preliminary design phase while leading to a reduction of time spent at the detailed design phase. The system being automated and faster than the one of a regular pre-detailed design phase, it was possible to run iteration loops in order to determine the worst mission in terms of tip clearance closure. The proposed system also allowed running a targeted sensitivity analysis of the tip clearance leading to the identification of parameters that should be focused on when optimizing a turbine's tip clearance. Finally, by requiring fewer user inputs this system decreases the risk of human errors while entirely leaving the important decisions to the designer.

X

TABLE OF CONTENTS

LIST OF TABLES

LIST OF FIGURES

XVI

LIST OF ABREVIATIONS

2D/3D	Two/Three dimensional
CAD	Computer-aided design
CAE	Computer-aided engineering
HPT	High pressure turbine
LCC	Life cycle cost
LPT	Low pressure turbine
MDO	Multidisciplinary design optimization
NATO	North Atlantic Treaty Organization
OOP	Object oriented programming
PDDS	Pre-Detailed Design System
PMDO	Preliminary and multidisciplinary design optimization
P&WC	Pratt & Whitney Canada
SAS	Secondary air system
SFC	Specific fuel consumption
TIT	Turbine inlet temperature

LIST OF SYMBOLS AND UNITS

%	Percent
mm	Millimeter
c	Gap closure
δc	Transient gap closure
D	Impingement jet diameter
D_h	Hydraulic diameter
h	Heat transfer coefficient
h_{blade}	Blade height
k	Gas thermal conductivity
L	Length of the conduct
$\dot{m}$	Mass flow rate
Nu	Nusselt number
Pr	Prandtl number
r	Radius at which the calculation is done
$r_{blade\ tip}$	Blade tip radius
r_{rotor}	Initial blade tip radius
r_{stator}	Initial shroud segment radius
δr_{stator}	Transient radial growth of the shroud segment
δr_{rotor}	Transient radial growth of the airfoil tip
Re	Reynolds number
s	Average spacing between the co-rotating discs

s_{cold}	Cold build clearance
Δs	Transient tip clearance
$t_{blade\ tip}$	Blade tip thickness
T_c	Cooling air temperature
t_{cool}	Cooling air response time
T_{eff}	Effective (or bulk) temperature
$t_{ground\ idle}$	Dwell time prior to re-slam
T_h	Hot gas temperature
U_{rel}	Fluid's relative tangential velocity
V_{ax}	Fluid's axial velocity
V_{rel}	Fluid's relative velocity
x	Length of influence of an impingement jet
$\#_{blades}$	Number of blades
α	Impingement angle
η_c	Cooling effectiveness
ρ	Density of the fluid
σ	Time fraction
ϕ	Azimuth in cylindrical coordinate system
ω	Angular velocity of the co-rotating discs
Ω	Rotational speed of the disc

INTRODUCTION

The design of a gas turbine engine is a multidisciplinary and iterative problem in which the best compromise has to be found between the conflicting disciplines involved: thermal, structural, aerodynamics, manufacturing, cost, weight, etc. The design of aero-engines traditionally follows two main stages: preliminary design and detailed design. At the pre-detailed stage, a few groups are involved to design and analyse the turbine concept's components and sub-systems. However, the Science and Technology Organisation of the North Atlantic Treaty Organization (NATO) showed that decisions taken early in the design process are often based on low fidelity models and when only little information (data, requirements, etc.) is available. This may compromise the engineers' ability to select the optimal design. At the detailed design phase, more groups are involved having their own set of specialized tools and methodologies, and the process is thus even more segmented within the groups to form sub-disciplines' specialists (NATO, 2006).

Optimizing a turbine design involves modifying design variables (parts' shapes, sizes, materials, performance data, etc.) to attain the design objectives (thrust, life, fuel consumption, weight, cost, etc.) with respect to several constraints (structural limits, temperatures, limits of physics models, etc.). The total number of variables combinations represents the design space, which can be enormous for gas turbine engines considering that they have thousands of parts.

Even though knowledge increases during the design process, the freedom to modify any part of the design decreases as shown in Figure 0.1, and/or induces major delays in the planning and a rise of design costs due to necessary redo's or new analyses that need to be executed for example (Panchenko, Patel, Moustapha & Dowhan, 2002). These decisions made early in the design process are indeed almost irreversible considering that it takes higher fidelity tools to increase knowledge, with which comes an exponential increase of the staff required to explore the design space. And finally, having that many designers involved produces so much inertia that global changes are practically impossible (NATO, 2006). It is consequently

hard to correct a non-optimal concept at a detailed design phase. To answer this problem, the use of Multidisciplinary Design Optimization (MDO) at the preliminary design phase (i.e. PMDO) is suggested, since it is at that stage that the biggest influence on the final product configuration is made (Panchencko et al, 2002). Moreover, at a pre-detailed design phase, the design space is easier to search as the number of design variables is reduced (the geometry is for example less complicated and refined) and the means to evaluate the design objectives (i.e. CAE analyses, rules of thumb, use of reference data, etc.) are simpler and therefore faster to execute.
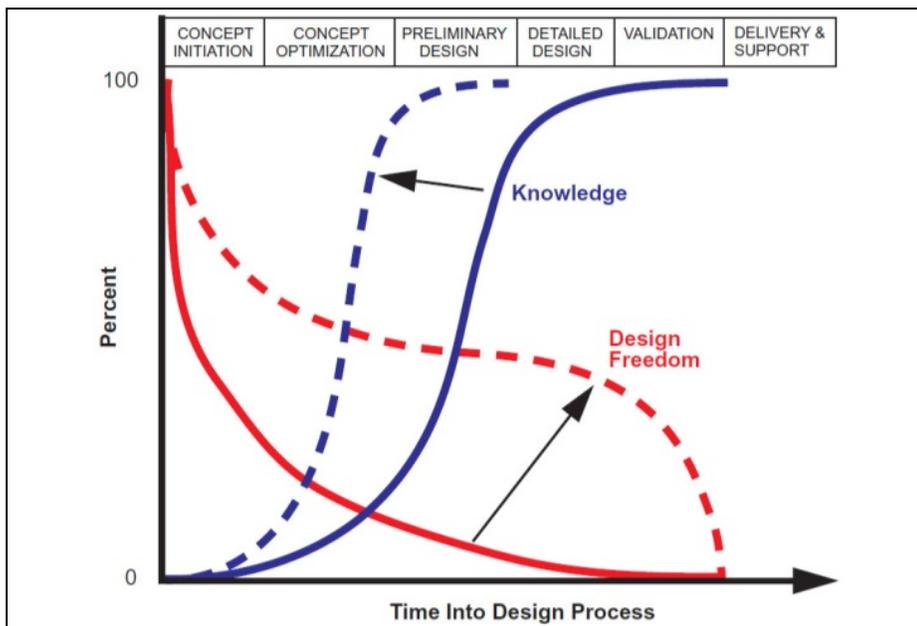
Figure 0.1: Knowledge vs. design freedom during the design process
Taken from NATO (2006)

The concept of MDO has been widely studied during the past fifty years (Martins & Lambe, 2013). However, there is a lack of information in the literature about using this methodology during the early stages of design. An increase of the efforts and knowledge during the pre-detailed design phase implies involving directly the specialist groups instead of waiting until the design phase. However, it results in significant delays of the concept evolution since many interactions between several groups are then required (Panchencko et al., 2002). This is where PMDO systems come into play in order to automate and ease this iterative process.

Referring to Panchencko et al. (2002), NATO (2006), and Korte, Weston & Zang (1998), the following steps are required to implement a PMDO system:

1. Develop a robust tool base, i.e. design tools based on parametrized CAD models and advanced physics analysis tools which includes the development or improvement of correlations;

2. Apply single discipline optimization to individual analytical tools;

3. Create an integration framework, i.e. a software architecture enabling integration, communication and execution of several tools;

4. Implement multidisciplinary optimization with a clear statement of the design objectives, constraints and variables, and an appropriate selection of the algorithms.

It is mandatory for the integration framework to efficiently manage and automate the data transfer between each tool (Panchencko et al., 2002). In order to do so, having a unique set of data in the system (and not a specific set for each separated tool) is suggested as it eliminates the non-value-added task of managing them while reducing the chance of using incorrect data. This approach offers many benefits such as increasing the data robustness, facilitating data archiving, enabling future addition of new tools or deleting the need for discipline-specific data translators (NATO, 2006).

A collaborative program, called Pre-Detailed Design System (PDDS), was initiated between Pratt & Whitney Canada (P&WC) and the École de Technologie Supérieure to implement a PMDO system for designing turbines at the pre-detailed design phase. The driver for P&WC to initiate such project is that they execute about thirty to forty preliminary design studies per year. It is therefore mandatory for the company to have a fast and efficient process to carry out all these pre-detailed studies. The architecture of PDDS, created by Benoit Blondin at P&WC, is shown in Figure 0.2. This figure shows all the different modules involved in PDDS following the requirements of a PMDO system introduced previously. Of importance for this work, one can observe in Figure 0.2, below the PDDS interface, all the discipline specific applications (turbine components design, cooling, lifing, thermal analysis, tip clearance, meanline design, etc.).
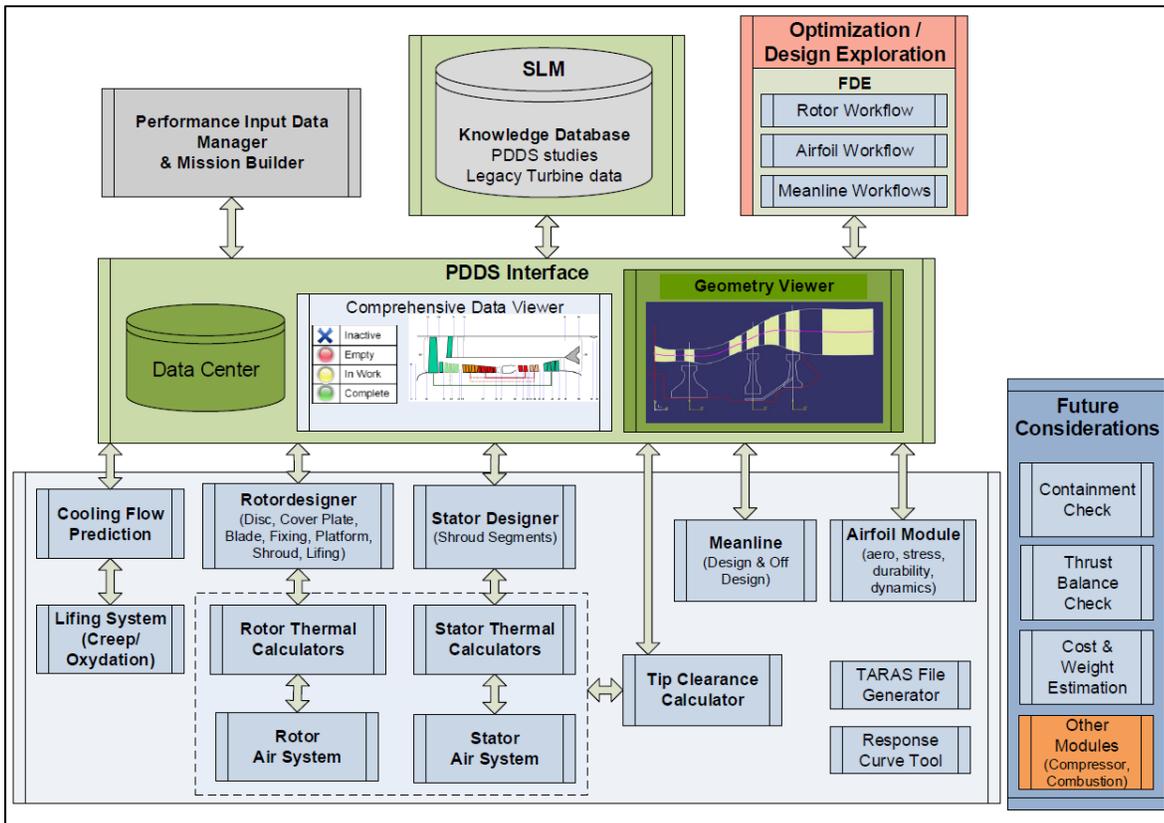
Figure 0.2 : PDDS Architecture by B. Blondin

Until now, the process to evaluate the tip clearance at a pre-detailed design phase was not optimal. This process required lots of manual steps from the analyst in charge of doing the evaluation, along with various approximate estimations (rule of thumb, scaled reference values, etc.). Considering the impact of the tip clearance on an engine's efficiency, and therefore on its specific fuel consumption (SFC), an error in the estimation of the tip clearance can cost a lot to the industry. If the tip clearance predicted cannot be met, the target SFC committed to a client will not be achieved. In order to meet the target SFC, redesign and development work is required with a heavy cost to the engine manufacturer.

In order to answer this uncertainty in tip clearance prediction, the present research intends to propose and implement a solution to automate the tip clearance calculation process integrated within PDDS. In order to do so, this Ph.D. project can be divided in three phases: (1)

automate the prediction of the rotor's thermal and centrifugal growth, (2) automate the prediction of the static components' thermal growth, and finally (3) calculate the radial gap between the blade tip and the shroud segment.

This thesis is divided in six chapters. The first chapter puts this work into context by presenting the topics directly connected to this work's problematic. The turbine rotor and stator geometries are therefore introduced along with a description of what are the thermal boundary conditions, of the tip clearance importance and of the mechanisms of its size variation. The second chapter presents this study's research problem and objectives. The third chapter describes the methodology used in order to develop the solution proposed to answer the research problem. The fourth chapter consists of the journal article written on the automation of the analyses process of the rotor and therefore addressing the first phase of this work. The fifth chapter is made of the paper written on the automation of the stator analyses and demonstrate how the second phase of this research was solved. The sixth chapter presents the article written on the third phase of this work, i.e. the tip clearance automatic prediction using the previously calculated growth of the rotor and stator.

CHAPTER 1

BASICS OF A TURBINE'S GEOMETRY, THERMAL BOUNDARY CONDITIONS AND TIP CLEARANCE

This chapter presents an introduction of the context of this Ph.D. project. It therefore introduces the reader to the turbine's rotor and stator geometries, presents what are the thermal boundary conditions made of, highlights the importance of the role of the tip clearance on the engine efficiency and introduces the physics of the clearance's size variation.

1.1   Introduction to a Turbine Rotor Geometry

The turbine blades are used to transform the hot gases high energy into mechanical energy available on the shaft. The high-pressure turbine (HPT) blades are in a very hostile environment considering that they are in the gas-path (or core flow in Figure 1.1 (a)) at the exit of the combustion chamber while rotating at high speed. One can therefore easily understand that these blades are exposed to large thermal and mechanical stresses.

The blades are fastened to the disc through the fixing, or shank in Figure 1.1 (a), as represented in Figure 1.1 (b). Generally, the gap between the blade root and the disc fixing is used to provide the blade with compressor bleed air for cooling purpose.

The turbine disc, or rotor in Figure 1.1 (a), is a large rotating component surrounded by compressor bleed air. The main role of the disc is to mechanically support the blades and to transfer the mechanical energy produced by the blades to the shaft. A cover-plate might sometime be secured to the disc and used to direct the cooling air more efficiently inside the fixing to cool the blade.
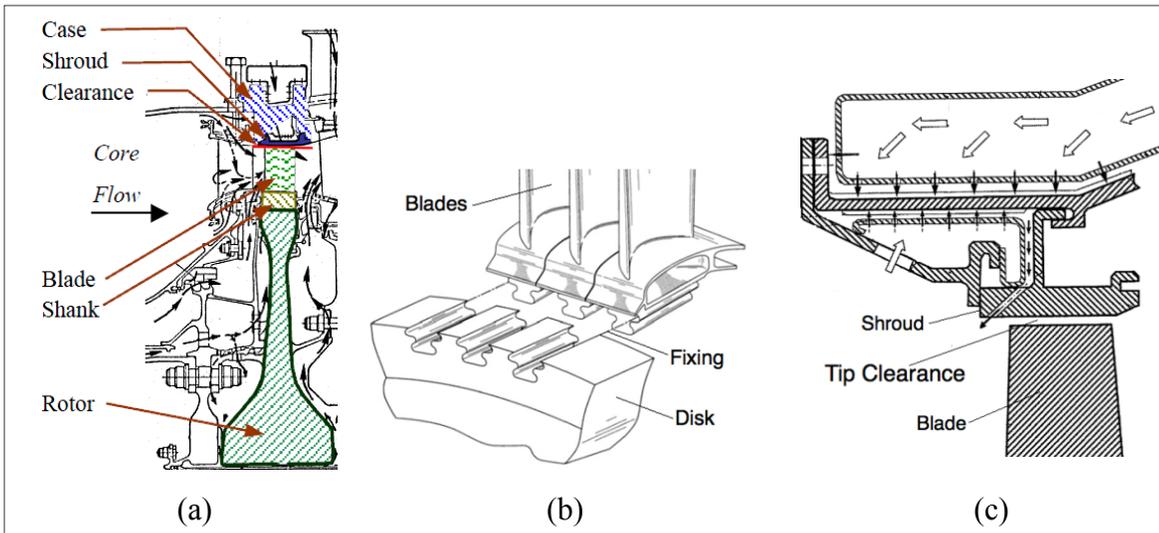
Figure 1.1: (a) Geometry of the E^3 high-pressure turbine; (b) Turbine blades fixing (c) Highlighting of the tip clearance
(a) Taken from Melcher & Kypuros (2003, p. 4); (b) Taken from Pandey et al. (2012); (c) Taken from Boswell & Tibbott (2013)

Within PDDS, the geometry of each component of the turbine rotor is provided by specific design tools based on CAD parametrized models (Ouellet, Savaria, Roy, Garnier & Moustapha, 2016; Twahir, 2013). These tools allow a user to design a wide range of turbine rotor's configurations including HPT rotors with cooled blade and cover-plates directing the cooling air, or low-pressure turbine (LPT) rotors with shrouds on top of uncooled airfoils as shown in Figure 1.2.
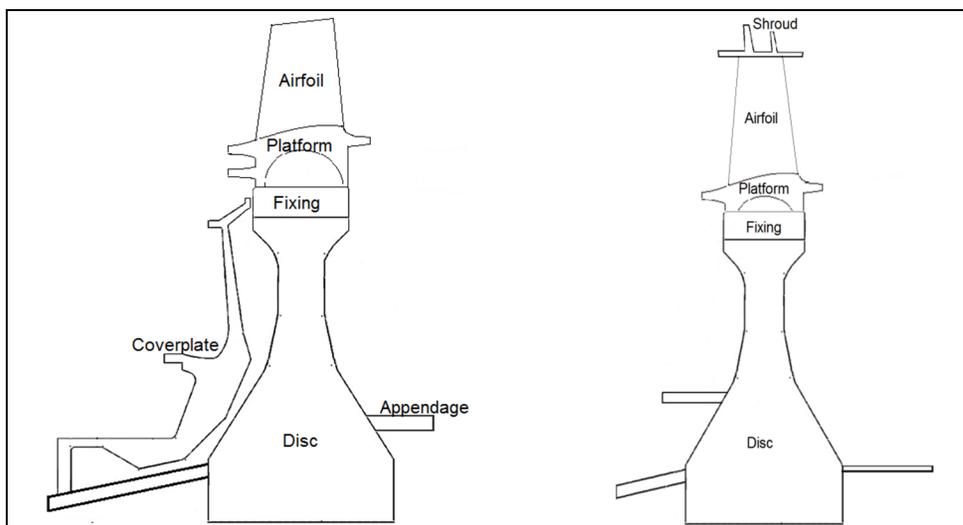
Figure 1.2 : Examples of rotor configurations (left: HPT; Right: LPT)

1.2   Introduction to a Turbine Stator Geometry

The static components above the blade tip that are of interest for this Ph.D. project are the housing (or casing) and the shroud segment (also called blade outer air seal). The assembly of these two components is referred to as "stator" in this work.

The shroud segments are the inner part of the stator. This "inner casing" is generally made up of a series of arc segments joined together to form a ring and attached to the "outer casing" by some hooks as presented in Figure 1.3. These hooks allow a certain radial displacement due to thermal and mechanical loads (Hennecke, 1985). The cavity between the outer surface of the shroud segments and the housing is used to conduct compressor bleed air and to cool down the shroud by jet impingement cooling (Melcher & Kypuros, 2003).
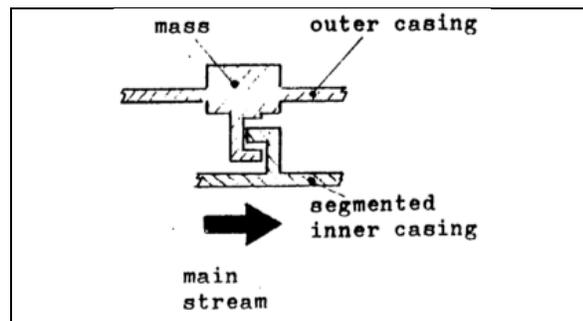
Figure 1.3: Shroud attachment to the casing
Taken from Hennecke (1985, p. 21)

Similar to the work done on the rotor, a stator design module was developed and used as a base for the static components thermal and stress analyses executed in this work (Savaria, Phutthavong, Moustapha & Garnier, 2017). Figure 1.4 shows an example of stator assembly and introduces the naming convention used in this work for each components of the stator geometry.
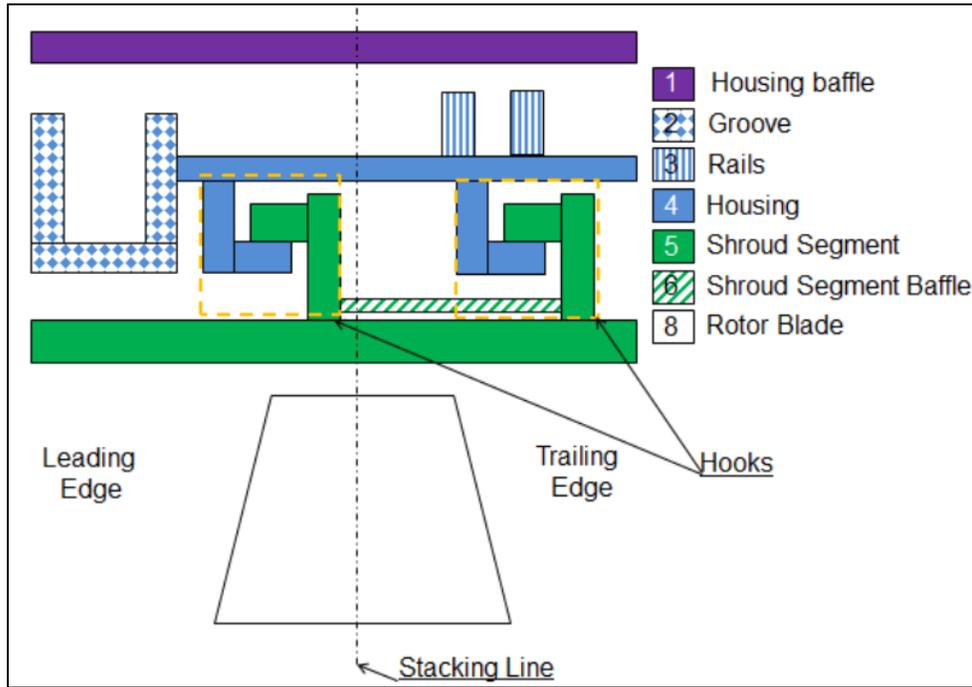
Figure 1.4: Stator Assembly Components Names
Taken from Savaria (2016, p.37)

1.3 Thermal Boundary Conditions Calculation

In order to execute the necessary thermal analyses on the rotor and stator components, thermal boundary conditions at the surface of these components need to be calculated. These thermal boundary conditions consist of a combination of a heat transfer coefficient and a fluid (i.e. air or gas depending on the turbine's zone considered) effective temperature, which allow evaluating the local heat flux. The Nusselt number is the dimensionless temperature gradient at a surface, and is therefore frequently used to measure the heat transfer coefficient on a surface using Equation (1.1) (Incropera & DeWitt, 1996).

$$h = \frac{Nu\,k}{L_c} \tag{1.1}$$

where k is the gas thermal conductivity and L_c is the characteristic length of the studied zone. The Nusselt number is generally a function of the Reynolds and Prandtl numbers, as shown by Equation (1.2) (Incropera & DeWitt, 1996; and Kreith, Manglik & Bohn, 2011).

$$Nu_L = C \, Re_L{}^m \, Pr^n \tag{1.2}$$

where C, m and n vary with the nature of the surface geometry and the type of flow, and might take any form (as an algebraic expression for example) or value (including 0). The relationship correlating these numbers is determined depending on the physic at stake and on the flow regime by analytical, numerical or experimental means.

The correlations used for each specific zone around the rotor and stator geometries are described in the CHAPTER 4 and CHAPTER 5 in the subchapters dedicated to the thermal boundary conditions (i.e. subchapters 4.5 and 5.5 respectively). One must remember that this thesis' research problem is not related to the detailed study of the heat transfer process in each component of a turbine, which is a large subject of study on its own, but focuses on evaluating the turbine components thermal and centrifugal growth at a pre-detailed design stage. Generally, preliminary analyses only require an average heat transfer coefficient on each surface of a simplified version of the turbine's geometry. And the heat transfer process of these surfaces is often modeled using basic correlations such as flat plates, isothermal rotating discs, impingement on flat surfaces, etc. Considering that, the literature review that led to the selection of the correlations presented in the subchapters 4.5 and 5.5 does not fully cover each separated subject but has for objectives to:

- Allow a better understanding of the heat transfer physics and flow patterns affecting the turbine components;
- Find how the heat transfer process on these components can be modeled;
- Gather correlations to evaluate the heat transfer coefficients on each surface.

1.4   Tip Clearance Importance

The prediction of the tip clearance size variation is important in order to avoid rubs and to prevent blades wear so as to dramatically increase the engine service life. Indeed, the deterioration of the airfoil tip and/or of the shroud segment because of rubs produces an increase of the tip clearance. It follows that the engine has to increase the turbine inlet temperature to develop the same thrust. If the disc temperature reaches its upper limit, the engine must come off the wing for maintenance. Considering that, one can easily understand that a better prediction of the tip clearance variation can reduce the engine life cycle cost (LCC). It was demonstrated that improving the tip clearance in the HPT as much more impact on the LCC than the same improvement of clearance in the low pressure turbine or the high pressure compressor (Lattime & Steinetz, 2002).

Tip clearance also has a considerable impact on the turbine efficiency and thus on the engine SFC. Indeed, the bigger the tip clearance the more gas can leak from to pressure side of the blade's airfoil toward the suction side without producing any work. Considering that work was required in order to bring that gas to its temperature and pressure, every percent of it that is not used to produce work is a loss for the whole engine.

Several tests were executed on a CF6-50C turbofan to evaluate the effect of the HPT tip clearance on the engine performance (Howard & Fasching, 1982). The authors performed these measures during both steady state and transient operations. The results showed that an increase of 0.305 mm in clearance produced a reduction of 0.7 % in turbine efficiency. In terms of the tip clearance effects on the whole engine efficiency, it was demonstrated that, once the gap between the rotor and the stator is larger than 1 % of the blade height (meaning a tip clearance of about 0.2 to 1 mm), an increase of 1 % in tip clearance produces a drop of about 1 % in efficiency (Hennecke, 1985). Finally, as explained previously the engine SFC is directly related to the HPT tip clearances. It was indeed reported that an increase of 0.254 mm in HPT tip clearance roughly produces a 1 % increase in SFC (Lattime & Steinetz, 2002).

Even if the values of the tip clearance impact on the engine efficiency or its SFC slightly vary from one reference to another, one can easily conclude that the prediction of the tip clearance size variation is mandatory in an integrated turbine design system considering the magnitude of its effect on the engine's performances.

1.5   Introduction to the Tip Clearance Size Variation Physics

As described with more details in the chapter 6.4.1, many loads are acting on the turbine's components which therefore affect the tip clearance. It was however demonstrated that the ones having the largest influence on a turbine clearance's variation are the thermal and centrifugal loads (Lattime & Steinetz, 2002; Howard & Fasching, 1982; Olsson & Martin, 1982).

The rotor's disc is a massive component surrounded by cooling air, and is therefore slow to respond to a change of condition due to its high thermal inertia. The rotor's airfoils on the other hand are exposed to the gas-path temperatures. Moreover, like the disc they spin at high rotational speed but are further from the center of rotation than the disc and therefore experience higher centrifugal forces. One therefore understands that the blades are quick to respond to a change in flight conditions and will grow (and shrink) significantly during a flight mission. The rotor overall response therefore happens in two phases: first the blade rapidly grow (or shrink) based on the change in condition and then the response of the disc slowly brings the whole rotor to a steady state.

The stator's radial growth on the other hand is dictated by the housing to which the shroud segments are attached. The housing is surrounded by cooling air and is stiff. This means that the stator will be slower to respond than the blade. However, being smaller than the disc the housing has a lower thermal inertia, and the entire stator is therefore faster to respond than the whole rotor to a change in flight condition.

The complexity of optimizing the tip clearance comes from this difference in response time between the different components of the turbine. Indeed, after a change of condition the tip clearance could suddenly close (and potentially rub) as the blade grow, and then open once the blade has stopped expanding and while the stator grows faster than the disc. Calculating the value of the cold build clearance therefore requires predicting the transient variation of the tip clearance for various flight manoeuvers in order to catch the worst possible scenario.

CHAPTER 2

RESEARCH PROBLEM, SOLUTION AND OBJECTIVES

Based on the definition of Wieringa (2009), this work is a practical problem as opposed to a knowledge problem. Wieringa explains that a practical problem calls for a change in the world to match the view of the stakeholders, as opposed to a knowledge problem which calls for a change in our knowledge about the world and what the stakeholders would like to know about it. Therefore the goal of this work is to identify the requirements of the stakeholders (i.e. the industry) regarding the research problem, to propose a new process in response to those requirements, and to implement that proposal. Evaluation of the proposed solution implies investigating if the requirements of the stakeholders were met (Wieringa, 2009).

2.1   Research Problem and Solution

As introduced previously, the problematic of this thesis is that the current process to estimate the tip clearance at a pre-detailed design phase is not accurate and is time consuming. A considerable amount of time is indeed lost manually doing unnecessary tasks (such as managing data) and guessing inputs. This leads to tip clearance prediction that could be wrong (because based on inaccurately estimated inputs for example) which implies heavy redesign along with being late on the delivery schedule of the engine (which comes with costly penalties). This potential rise of the design cost and time is not acceptable and must be solved.

This work intends to propose and implement a solution to automate the tip clearance calculation process for the pre-detailed design phase of a gas turbine engine and integrated in a PMDO system. This project is therefore a process innovation for which new practices and procedures need to be defined. The proposed solution can be divided in three phases:

Firstly, the process leading to the prediction of the rotor's thermal and centrifugal growth needs to be automated. One of the main challenges of this first step resides in the fact that

these thermal and stress analyses are usually not conducted during the pre-detailed design phase. To be able to run these analyses, all the required inputs (including some that are usually not produced before the detail design phase) must be available within the PMDO system in which this work must be integrated.

Secondly, similar to the first research problem the process allowing determining the stator's thermal growth needs to be automated. For the sake of integration, the design and analysis process of the turbine stator needs to be very similar to the one of the rotor. This might be complicated as these processes are often different in the industry.

Thirdly, once the two first research problems are answered, the tip clearance must be calculated by combining the growths of the rotor and stator predicted for specific transient manoeuvers and steady states flight conditions.

2.2   Objectives

The stakeholders' requirements, and therefore this thesis' objectives allowing verifying that the research problem has been properly answered, are:

1.   Improve the pre-detailed design process by proposing a more robust, flexible and user-friendly design and analysis system. This means that the new process has to allow more rotor and stator configurations to be modeled while requiring limited number of input from the user.

2.   The new process results should be targeting a 20 % difference when comparing the delivered tip clearance to detail design results. This 20 % accuracy comes from the fact that, based on experts' opinion, it is usually accepted for a regular pre-detailed phase to be about 30 % off when compared to the detail design phase results. This target of 20 % therefore represents an accuracy improvement of 33 % compared to the current pre-detailed design process.

3.   It is expected that the implementation of a PMDO system would lead to an increase of productivity through the automation and/or integration of the various

functionalities of the system. The whole design and analysis process developed in this work should therefore be, at least, as fast as the current pre-detailed design process.

CHAPTER 3

METHODOLOGY

3.1 Global Methodology

As introduced previously, this Ph.D. dissertation is about process innovation. This means that in order to answer the research problem and achieve the research objectives, an entirely new process for the tip clearance pre-detailed evaluation had to be defined. Indeed, due to the extreme complexity of designing a gas turbine engine, the work is generally decomposed into manageable sub-tasks assigned to specialist groups. Large companies designing entire engines are thus forced to distribute the workload between several departments in order to increase their efficiency. Even if this type of work organization has many benefits such as allowing each department to be more specialized, it limits the communication between the separate disciplinary groups which may generate some complications. Several departments are for example directly involved in this thesis and in the determination of the tip clearance. Indeed, the prediction of the turbine's performances, the static components' analysis, the rotating parts' analysis, or the determination of the secondary air system behaviour around these components to name a few, are the mandate of different departments which handle their part of the process using different tools that might not communicate efficiently with each others. This is not optimal, considering that these steps of a turbine design share many inputs and parameters, and should therefore be unified. This work must therefore make use of all the available expertise and implement it within a new process for the Turbine Design & Analysis as presented in Figure 3.1. This process is new and was developed

As introduced previously, the tip clearance calculation over an entire flight mission implies the determination of the transient radial growth, due to temperature and rotational speed, of the turbine's rotor and static components. Figure 3.1 shows the architecture of the tip clearance calculation process developed in this work, where each arrow represents data exchange. The process required in order to evaluate the turbine's components displacement includes several sub-systems, divided in three groups:

- The pre-processors which are defining the analyses inputs and creating the turbine design;
- The analysers that run the transient and steady state thermal and stress analyses required for a tip clearance prediction;
- The post-processors displaying the analyses results and allowing the analysts to pursue their study.
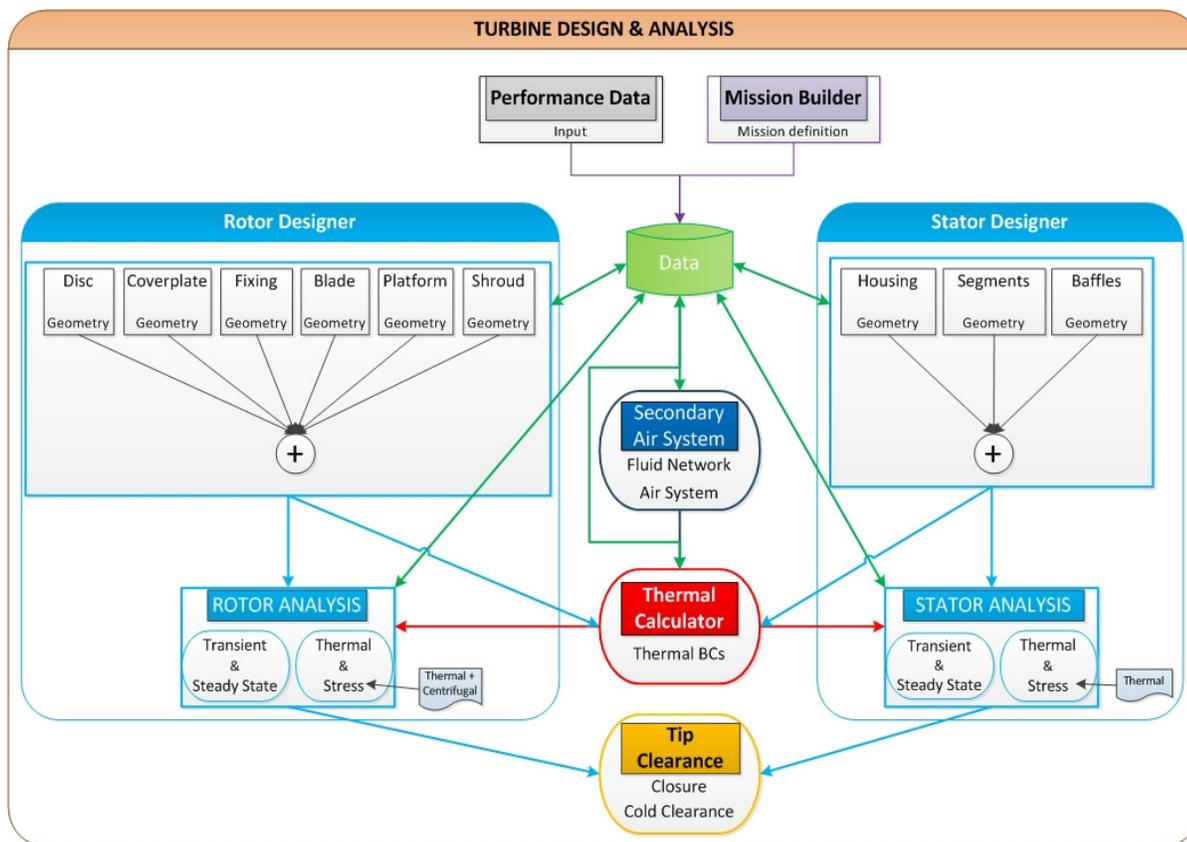
Figure 3.1 : Tip clearance calculation methodology

The first and highest-level pre-processor sub-system in this program architecture provides the initial concept's data, i.e. the Performance Data. The second one, called Mission Builder, generates the flight mission with the proper transient responses to a change in throttle position for all the air chambers around the rotor, the stator and in the gas-path. The third pre-processor are the design tools generating geometric data of the rotor and stator based on

parameterized CAD models. The sub-systems introduced so far are not the work of the authors but were developed in the collaborative program initiated between Pratt & Whitney Canada and the École de Technologie Supérieure (Ouellet et al., 2016; Twahir, 2013; Savaria et al., 2017).

If one assumes that the rotor's and stator's geometry and the aircraft operating conditions (i.e. its performances and flight mission) are known, the remaining work in the preprocessing step of a rotor and stator's thermal and stress analyses is the generation of the secondary air system (SAS) and the calculation of the thermal boundary conditions, as shown in Figure 3.1. The fourth pre-processor (and first part of this work) in this figure therefore creates the SAS based on the rotor and stator's configuration, i.e. existence of appendages along the disc's sides, presence of cover-plates, location of the housing groove, presence of a piston ring in the groove, presence of impingement baffles, etc. The last pre-processor calculates the thermal boundary conditions, i.e. the heat transfer coefficients and effective temperatures, for each zone around the rotor and stator separately. The analysis sub-systems (Rotor Analysis and Stator Analysis) automatically execute the thermal and stress analyses in a CAE software and provides the user with temperature maps and components displacements. The implementation of these sub-systems is developed in the CHAPTER 4 and CHAPTER 5. Finally, the tip clearance calculation is handled by the last module shown in Figure 3.1. This sub-system may be considered as a post-processor as it does not execute any CAE analysis but calculates the tip clearance by subtracting the results of the two analysis sub-systems and provide the analyst with all the required outputs. The tip clearance calculation process is developed in the CHAPTER 6.

The integration of all these sub-systems is handled by the use of a common and centralized data structure, and by means of object oriented programming (OOP) to create a framework repeated through all parts of the system. This enforces having a unique set of data in the system and allows their efficient management. Moreover, this removes the non-value-added task of managing data while reducing the chance of using incorrect inputs. Finally, as it was introduced before, such centralized data structure increases the data robustness, facilitates

data archiving, enables future addition of new tools (which is made even more possible by the use of OOP) and deletes the need for discipline-specific data translators (NATO, 2006).

3.2   Detailed Methodology

In order to have a deeper understanding of the process developed in this work and of how the sub-systems interact with each other, an introduction to the detailed methodology of this project is required.

In the Rotor Designer sub-system, some of the components are axisymmetric and are therefore designed purely in 2D, others are asymmetric and are designed in 3D. The 2D components are the disc (with appendage(s)) and the cover-plate(s). The fixing is not an axisymmetric component but it is of constant shape in its thickness (as shown in Figure 1.1 (b)) and is therefore modeled as a 2D component in a plane perpendicular to the plane in which the disc and cover-plate are designed. The 3D components are the platform, airfoil and shroud. Each component of the rotor is designed with its own tool as shown in Figure 3.1 (Ouellet et al., 2016; Twahir, 2013), and all these separated components' geometry are combined in the CAE software to run the thermal and stress analyses. Considering that all the analyses being run during the pre-detailed phase of a tip clearance prediction are 2D, the 3D components are cut to obtain 2D cross sections and are then cut in their thickness direction to extract thickness maps.

In the Stator Designer sub-system, both the housing and the shroud segment are designed in 2D (Savaria et al., 2017). The housing is an axisymmetric component while the shroud segment is not. The shroud segment's thickness is calculated based on its radial position and the number of segments present on the circumference.

The analyses process is the same for the rotor and stator. First the geometries are loaded in the CAE software along with their material properties. The geometry is then meshed using solid, convection and surface effect elements. The thicknesses previously extracted from the

3D models are mapped onto the 2D geometries' elements. Finally, the transient thermal boundary conditions are applied to the surface effect elements. As one can notice, the analysis process is straightforward. This is only possible because all the required input have been generated previously, as shown in Figure 3.1, and as described in the next chapters.

CHAPTER 4

AUTOMATED THERMAL AND STRESS PRELIMINARY ANALYSES APPLIED TO A TURBINE ROTOR

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4.1   Abstract

The use of Multidisciplinary Design Optimization (MDO) techniques at the preliminary design phase (PMDO) of a gas turbine engine allows investing more effort at the pre-detailed phase in order to prevent the selection of an unsatisfactory concept early in the design process. Considering the impact of the turbine tip clearance on an engine's efficiency, an accurate tool to predict the tip gap is a mandatory step towards the implementation of a full PMDO system for the turbine design. Tip clearance calculation is a good candidate for PMDO technique implementation considering that it implies various analyses conducted on both the rotor and stator. As a first step to the development of such tip clearance calculator satisfying PMDO principles, the present work explores the automation feasibility of the whole analysis phase of a turbine rotor preliminary design process and the potential increase in the accuracy of results and time gains. The proposed conceptual system integrates a thermal boundary conditions automated calculator and interacts with a simplified air system generator and with several design tools based on parameterized CAD models. Great improvements were found when comparing this work's analysis results with regular pre-detailed level tools, as they revealed to be close to the one generated by the detailed design tools used as target. Moreover, this design process revealed to be faster than a common preliminary design phase while leading to a reduction of time spent at the detailed design phase. By requiring fewer user inputs, this system decreases the risk of human errors while entirely leaving the important decisions to the designer.

4.2 Introduction

The design of a gas turbine engine is a multidisciplinary and iterative problem in which the best compromise has to be found between the conflicting disciplines involved: thermal, structural, aerodynamics, manufacturing, cost, weight, etc. The design of aero-engines traditionally follows two main stages: preliminary design and detailed design. At the pre-detailed stage, a few groups are involved to design and analyse the turbine concept's components and sub-systems. However, the Science and Technology Organisation of NATO showed that decisions taken early in the design process are often based on low fidelity models and when only little information (data, requirements, etc.) is available (NATO, 2006). This may compromise the engineers' ability to select the optimal design. At the detailed design phase, more groups are involved having their own set of specialized tools and methodologies, and the process is thus even more segmented within the groups to form sub-disciplines' specialists. Panchencko et al. explain that even though knowledge increases during the design process, the freedom to modify any part of the design decreases as shown in Figure 4.1, and/or induces major delays in the planning and a rise of design costs. It is consequently hard to correct a bad concept at a detailed design phase. To correct this, the use of MDO at the preliminary design phase is suggested, since it is at that stage that the biggest influence on the final product configuration is made (Panchencko et al., 2002). The concept of MDO has been widely studied during the past 50 years (Martins & Lambe, 2013). However, there is a lack of information in the literature about using this methodology during the early stages of design. Panchencko et al. explain that an increase of the efforts and knowledge during the pre-detailed design phase implies involving directly the specialist groups instead of waiting until the design phase. However, it results in significant delays of the concept evolution since many interactions between several groups are then required (Panchencko et al., 2002). This is where PMDO systems come into play in order to automate and facilitate this iterative process. Referring to Panchencko et al. (2002), NATO Science and Technology Organisation (2006) and Korte et al. (1998), the following steps are required to implement a PMDO system:

1. Develop a robust tool base, i.e. design tools based on parametrized CAD models and advanced physics analysis tools;
2. Apply single discipline optimization to individual analytical tools;
3. Create an integration framework, i.e. a software architecture enabling integration, communication and execution of several tools;
4. Implement multidisciplinary optimization with a clear statement of the design objectives, constraints and variables, and an appropriate selection of the algorithms.
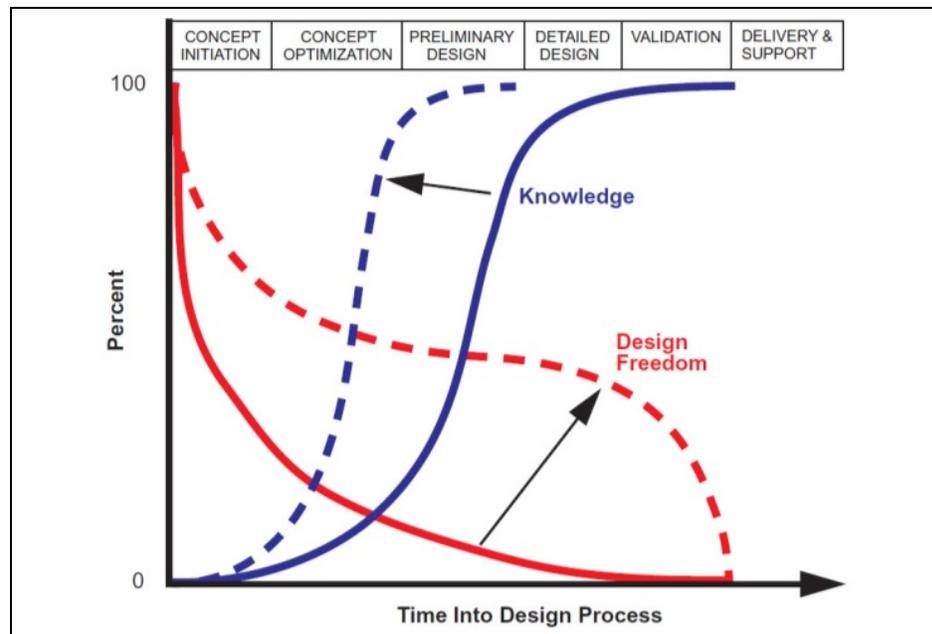
Figure 4.1: Knowledge vs. design freedom during the design process
Taken from NATO (2006)

A collaborative program was initiated between Pratt & Whitney Canada and the École de Technologie Supérieure to implement an MDO system for designing turbines at the pre-detailed design phase (PMDO). The implementation of a PMDO system generally includes four steps: parameterization of geometric and performance parameters, development or improvement of correlations, integration of disciplines and components, and finally optimization. The present work focuses on the second and third steps. As part of this collaborative program, the development of a tip clearance calculation system is a mandatory step and a perfect example for the implementation of a PMDO methodology considering that

it requires the design and analyses (thermal, structural and aerodynamic) of several turbine components as described in Figure 4.2. It was shown that the prediction of a turbine's tip clearance through a typical flight mission is essential in order to maximize an engine's efficiency and its service life (Lattime & Steinetz, 2002). Indeed, an increase of the tip clearance implies that the engine has to augment the turbine inlet temperature to develop the same thrust. If the disc temperature reaches its upper limit, the engine must be removed for maintenance. If the gap between the rotor and the shroud segments is larger than 1 % of the blade's height, an increase of 1 % in tip clearance produces a drop of about 1 % in efficiency (Hennecke, 1985). Based on this, a process for prediction of tip clearance size variation is mandatory in an integrated turbine design system.

Figure 4.2: (a) Geometry of the E^3 high-pressure turbine; (b) Highlighting of the tip clearance (a) Taken from Melcher & Kypuros (2003, p. 4); (b) Taken from Boswell & Tibbott (2013)

In order to develop a tip clearance calculator at the preliminary phase using PMDO principles, three steps can be identified: the prediction of the rotor's thermal and centrifugal growth, the prediction of the static components' thermal growth, and finally the calculation of the radial gap between the rotor and shroud. This paper focus on the first step of implementing a rotor automated analysis process and the work is presented as follows: after a description of the methodology, the secondary air system generation and the identification of the main parameters and of their effect on rim seals' efficiency are presented. The thermal

boundary conditions calculation process is then introduced, including the simplifications and assumptions made for the heat transfer coefficients calculation. As a last step of this work, the automation feasibility of the analyses is discussed. Finally, the paper concludes with a presentation of the main results.

4.3   Methodology

Tip clearance calculation over an entire flight mission implies the determination of the transient radial growth, due to temperature and rotational speed, of the turbine's rotor and static components. Figure 4.3 shows the architecture of the tip clearance calculation process being developed in this work, where each arrow represents data exchange. As one can see, the creation of a rotor analyses system is the first step toward the implementation of a tip clearance calculator, and the subject of the present article. The process required in order to evaluate the rotor's components displacement includes several sub-systems: pre-processors, an analysis system and a post-processor for visual purpose. The first and highest-level pre-processor sub-system in this program architecture provides the initial concept's data, i.e. the performance data. The second one, called mission builder, generates the flight mission with the proper transient responses to a change in throttle position for all the air chambers around the rotor and in the gas path. The third one is the design tool generating geometric data of the rotor based on parameterized CAD models. The sub-systems introduced so far are not the work of the authors. They are indeed developed in the collaborative program initiated between Pratt & Whitney Canada and the Ecole de Technologie Supérieure (Ouellet et al., 2016; Twahir, 2013). The scope of this paper is indeed limited to the following sub-systems.

The fourth sub-system is the creation of the SAS based on the rotor's configuration, i.e. existence of appendages along the disc's sides, presence of cover-plates, number and size of fixing channels, etc. This sub-system is described in the next chapter on SAS Generation. The final pre-processor sub-system calculates the thermal boundary conditions, i.e. the heat transfer coefficients and effective temperatures, of each zone around the rotor. This last pre-processor sub-system is described in the chapter on Thermal Boundary Conditions. The

Rotor Analysis sub-system automatically executes the thermal and stress transient analyses in a CAE software and provides the user with temperature maps and rotor displacements. The chapter on Automated Analyses is dedicated to this analysis sub-system. The integration of all these sub-systems is handled by the use of a common and centralized data structure, and by means of object oriented programming to create a framework repeated through all parts of the system.
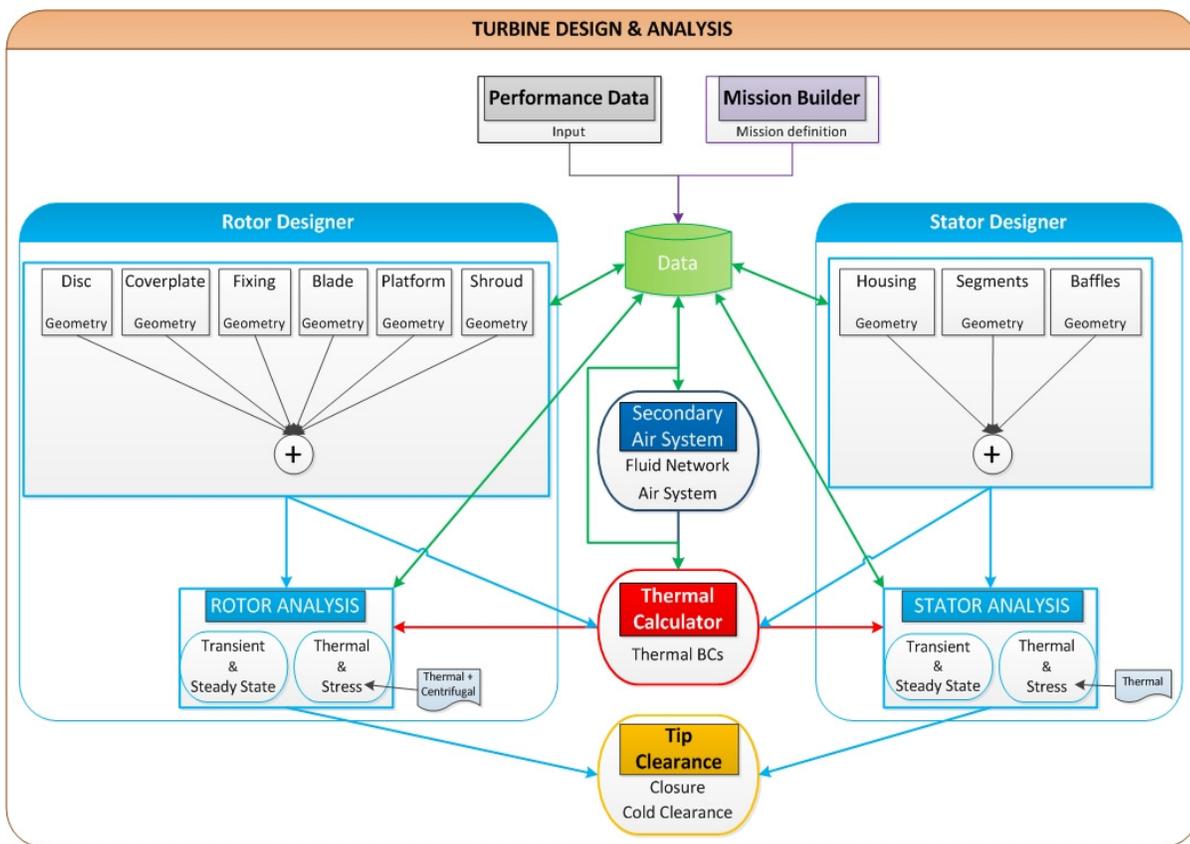
Figure 4.3: Tip clearance calculation methodology

If one assumes that the rotor's geometry (delivered by subsystems such as described by Ouellet et al. (2016) and Twahir (2013)) and the aircraft operating conditions (i.e. its flight mission) are known, the remaining work in the preprocessing step of a rotor's thermal and stress analyses is the generation of the SAS and the calculation of the thermal boundary conditions.

4.4   Secondary Air System Generation

The primary purpose of a SAS is to cool the gas turbine components which are most vulnerable to permanent damage caused by overheating. An optimal air system design is capable of maximizing rim seals' efficiencies, defined as the ratio between the mass flow rates of purge flow and hot air ingested, as shown by Equation (4.1).

$$\eta_{rim\ seal} = \frac{\dot{m}_{purge}}{\dot{m}_{purge} + \dot{m}_{ingested}} \qquad (4.1)$$

Maximizing a rim seal efficiency means limiting hot gas ingestion to achieve acceptable disc cavity temperature while minimizing the air flow consumption from the compressor stages. The latter is a crucial parameter, as it has a direct impact on the gas turbine overall performance. The two criteria mentioned above form the basis for the optimization of an air system network. The SAS parametrisation is based on the rotor configuration and geometry. The SAS generator identifies the rotor configuration, extracts the features' positions (i.e. appendages, cover-plates, etc.), and automatically assigns the correct number of chambers at the relevant locations around the rotor. Furthermore, the rotor geometry dictates the type of restrictors to be used between these chambers. The geometrical and performance parameters which have the greatest impact on the SAS behavior around rotor stages were identified through sensitivity analyses. These parameters are the air source mass flow rate, the rim seal geometry and clearance, and the flow distribution. The analyses were carried out on a high pressure turbine stage running at take-off condition. The results were obtained using an air system analysis in-house software, which is integrated within the SAS generator. The latter provides the necessary inputs and generates a simplified air system network, which is then analyzed using the in-house software.

The first two analyses were carried out on a simplified rotor configuration to investigate how the rim seal efficiency is affected by either the input mass flow rate or its gap clearance. In both cases the two most common rim seal geometries were investigated (i.e. U-shape and

axial overlap). The same results could be observed using a more complex air system network, if the proper flow distribution adjustments are made.

The first analysis aimed at identifying how the quantity of air supplied for cooling the rotor will affect its rim seal efficiency. The U-shape and axial overlap rim seal geometries were investigated with low and large radial clearance. One can distinguish two effects of the flow consumption on the rim seal efficiency in Figure 4.4. Indeed, initially the rim seal efficiency experiences a sharp increase up to ~90 % with increasing mass flow rate. After reaching ~90 % efficiency, a considerable amount of additional air flow is required in order to further increase the rim seal efficiency. While a 25 % increase to the initial mass flow rate would be sufficient to reach 90 % efficiency, a further 20 % increase of mass flow is required to gain 5 % efficiency. Figure 4.4 highlights the fact that U-shape rim seals (U) tend to require less mass flow rate to reach 90 % efficiency as compared to axial overlap rim seals (AO). Therefore U-shape rim seals are often preferred for high pressure turbine stages where the cooling air is costly in terms of engine performance. For low pressure turbine stages, it is more common to use axial overlap rim seals as they are easier to produce and as the air source comes from further upstream in the compressor, it has less penalty on engine performance.

The second study investigated how each rim seal efficiency respond to increasing radial clearance for a mass flow rate imposed by using the appropriate labyrinth seal geometry. One can observe in Figure 4.5 that the efficiency of U-shape rim seals decreases linearly with increasing radial clearance. It is also interesting to note that as the mass flow rate increases, the range of radial clearances at which the U-shape efficiency remains above 90 % is greater. On the other hand, an axial overlap rim seal offers a much smaller allowable radial clearance margin to maintain an acceptable efficiency. Moreover the rate at which efficiency drops with increasing radial clearance is greater when the efficiency drops below 90 %. This study suggests that for a given mass flow rate, U-shape rim seals offer more freedom when dealing with the rotors radial displacement constraints during operation.
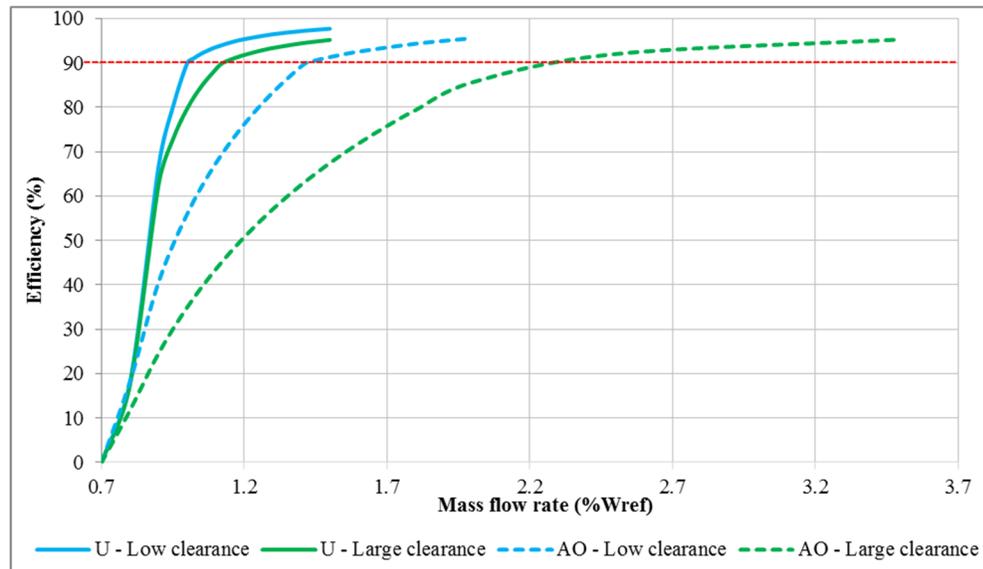
Figure 4.4: Rim seal efficiency vs. mass flow rate
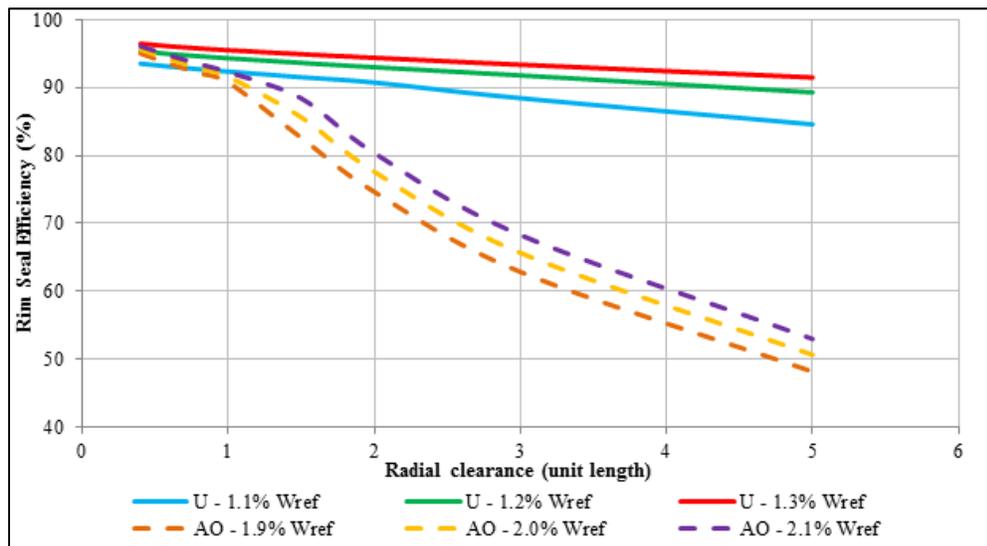
Figure 4.5: Rim seal efficiency vs. radial clearance

Given a simplified air system as shown in Figure 4.6, the total amount of cooling air injected in the system and distributed around the rotor is controlled by the labyrinth seal's clearance, the appendage's orifices, and the leakage through the blade attachments. Iterations must be executed sequentially on seal clearance and orifice dimensions until upstream and downstream cavity targets are obtained. Flow adjustment through orifices is done by finding

the adequate combination of total area, hydraulic diameter and orifice length. The orifice length is imposed by the rotor geometry and the hydraulic diameter is restricted by manufacturability requirements. The quantity of air supplied downstream was therefore adjusted by varying the total orifices' area (essentially changing the quantity of holes) for three different labyrinth seal mass flow rates. Figure 4.7 shows that most of the flow fed into the system is restricted from going downstream with a small total area, which in turn results in respectively high and low rim seal efficiencies upstream and downstream. As the number of holes increases, the rim seal efficiency downstream sharply reaches 90 % while maintaining satisfying rim seal efficiency upstream. Similarly to what was observed in Figure 4.4, further increasing the rim seal efficiency requires more flow to be fed downstream which implies a slight drop in the upstream rim seal efficiency.
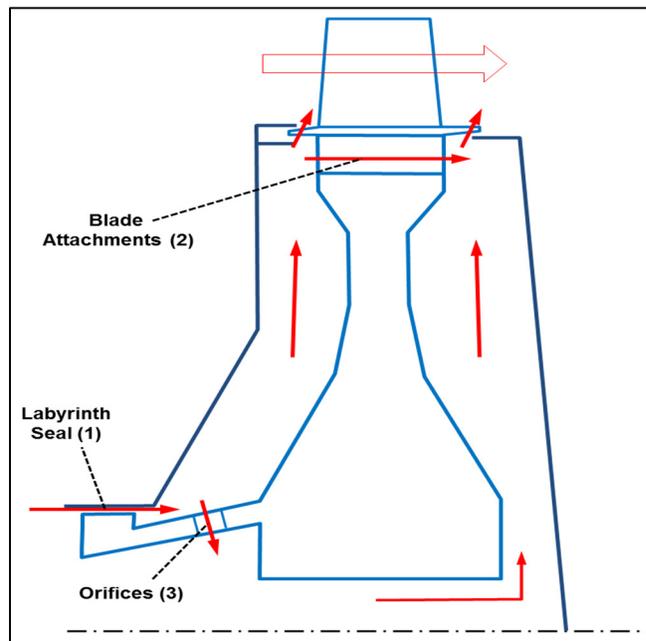
Figure 4.6: Simplified secondary air system

The sensitivity analysis revealed that an air system can be effectively optimized with few parameters. Indeed while minimizing the air consumption, one can reach the desired rim seals' efficiencies by modifying their geometry and finding the adequate orifices' sizes for optimal flow distribution.
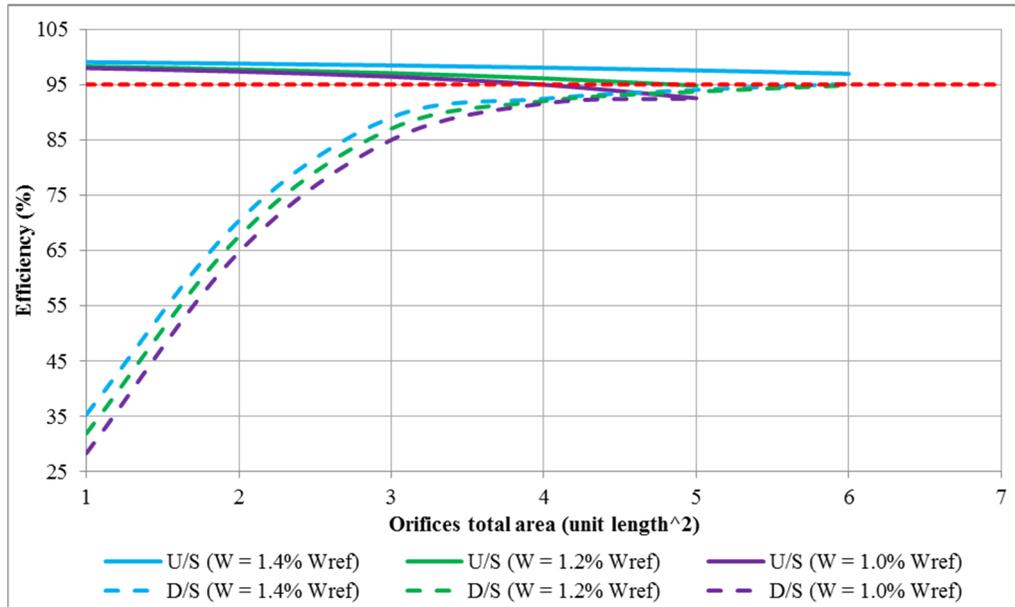
Figure 4.7: Upstream and downstream rim seals efficiency vs. flow distribution

4.5   Thermal Boundary Conditions

To execute a thermal analysis, boundary conditions are required for each specific location (i.e. zone) around the geometry. The Thermal Calculator uses for each zone a specific set of inputs obtained from all the connected sub-systems (performance data, design CAD sub-systems, the SAS generator, etc.) to calculate the thermal boundary conditions. Thanks to a Gateway software, the Rotor Analysis CAE sub-system is able to make use of the CAD parametric models (Ouellet et al., 2016; Twahir, 2013) to identify easily each zone of the rotor geometry.

In detailed design, very specific correlations are used to evaluate the boundary conditions on a large number of zones defined all around the rotor to better represent the effect of every seal, appendage, cavity, etc. At a pre-detailed level, the geometry is much simpler and therefore fewer zones are required. It was decided to define one zone per air system chamber, as presented in Figure 4.8. Thermal boundary conditions correspond to a combination of a heat transfer coefficient and a bulk temperature at the surface of each zone. Kreith et al. explained how the bulk temperature can be evaluated based on the fluid's static temperature,

its relative velocity and the Prandtl number (Kreith et al., 2011). Static temperatures around the rotor are easily computed based on the total temperatures given by the SAS generator for the cooled components, and on the gas path meanline temperature (which is part of the initial concept data) for the airfoil and platform. The real challenge here is therefore to select the proper correlations for the heat transfer coefficients considering that these are directly influenced by the zone's geometry and the physic at stake. The Nusselt number is often used to evaluate the heat transfer coefficient of a surface as these two are directly correlated. At a preliminary design stage, the zones around the rotor can be modeled using basic cases such as flat plates, rotating discs, channels, etc. The assumptions developed here are represented in Figure 4.8.
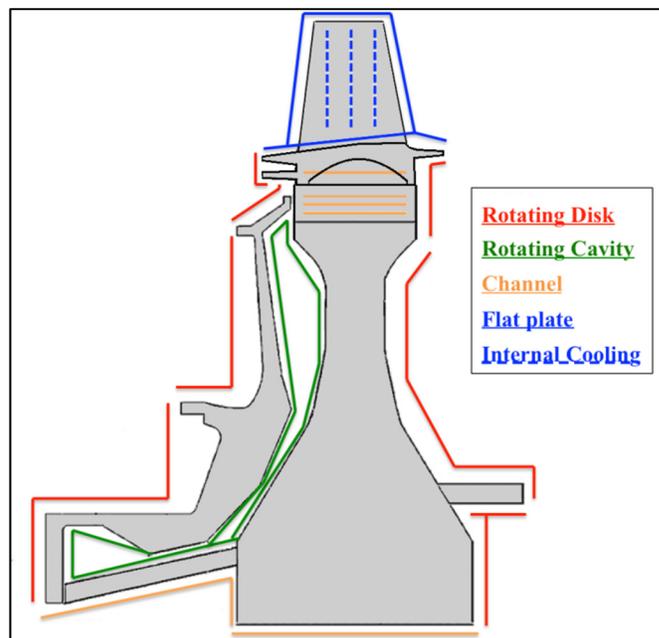
Figure 4.8: Rotor HTC zones

Despite the strong influence of the surface curvature on the thermal properties of an airfoil (Ito, Goldstein & Eckert, 1978; Goldstein, Kornblum & Eckert, 1982), several works modeled a turbine's airfoil as a flat plate in a turbulent flow (Ammari, 1989; Bhatti, Kumari, Chaitanya, Kedarinath & Kumar, 2006). For the sake of simplicity, a preliminary thermal analysis of a blade (airfoil and platform) can use the heat transfer correlations of a flat plate.

For such model, following Equations (4.2) and (4.3) can be used (Incropera & DeWitt, 1996). If the airfoil is cooled, the same heat transfer coefficients can be used but the effective temperature must be calculated based on the hot and cold gas temperatures and on the effectiveness of the cooling system, as shown by Equation (4.4). At a pre-detailed stage, this cooling effectiveness can be obtained by scaling the effectiveness of a reference engine based on the gas path temperature.

$$Laminar: Nu = 0.332 \, Re^{1/2} Pr^{1/3} \tag{4.2}$$

$$Turbulent: Nu = 0.0296 \, Re^{0.8} Pr^{1/3} \tag{4.3}$$

$$T_{eff} = T_h - \eta_c \, (T_h - T_c) \tag{4.4}$$

A turbine disc is basically a rotor surrounded by stationary elements. Rotor-stator systems have been covered many times in the literature (for example: Owen & Rogers, 1989; Kreith et al., 2011; Harmand, Pellé, Poncet & Shevchuk, 2013) and the correlations to be used for the sides of a rotating disc are therefore well known. For a laminar flow, the Nusselt number of an isothermal rotating disc will vary depending on the Prandtl number. For a Prandlt number of 0.72, the use of Equation (4.5) is suggested (Shevchuk, 2009). For a turbulent flow with a Prandtl number lower than unity one can use the Equation (4.6) (Kreith et al., 2011). If a cover-plate is present, the use of the empirical Equation (4.7) for two co-rotating discs is suggested in order to calculate the heat transfer coefficient inside the rotating cavity (Shevchuk, 2009).

$$Nu = 0.3286 \, Re^{1/2} \tag{4.5}$$

$$Nu = 0.02 \, Re^{0.8} \tag{4.6}$$

$$Nu = 0.046 \, Re^{0.5} \left(\frac{\dot{m}}{2\pi\rho s\omega r^2} \right)^{0.328} \tag{4.7}$$

For all the confined regions in the geometry (i.e. the disc bore, the fixing channels, the platform pocket, etc.), the correlation that should be used is the one of an internal channel (Incropera & DeWitt, 1996; Kreith et al., 2011). For a turbulent flow the use of the Dittus-Boelter equation (4.8) is suggested (Dittus & Boelter, 1930). If the conduct is short and the fluid turbulent, the flow cannot be considered as fully developed and the entrance effect must be accounted for by including a term in the correlation as shown in Equation (4.9). In the case of laminar flows the empirical Equation (4.10) is suggested (Sieder & Tate, 1936).

$$Nu = 0.023 \, Re^{0.8} \, Pr^{\frac{1}{3}} \tag{4.8}$$

$$Nu_{with\ entrance\ effect} = Nu \cdot \left(1 + \left(\frac{D_h}{L}\right)^{0.7}\right) \tag{4.9}$$

$$Nu = 1.86 \left(\frac{Re \, Pr \, D_h}{L}\right)^{1/3} \tag{4.10}$$

4.6   Automated Analyses

Automation of the analyses is possible thanks to the process described previously and depicted in Figure 4.3. As mentioned previously, the integration of all the sub-systems shown in Figure 4.3 is handled by the use of a common and centralized data structure. A framework was created and repeated through all parts of the system by means of object oriented programming. Automation of the analyses therefore only requires an effective way for the sub-systems to communicate with the CAD and CAE external software. Using a Gateway system, geometric data can be extracted from the CAD parametrized models and used in the sub-systems for calculation and display purpose. This Gateway software also allows direct communication between the CAD models and the CAE software used to run the 2D analyses. As the non-axisymmetric components (i.e. the platform, the airfoil, etc.) are designed using 3D models, thicknesses can be automatically extracted from CAD models by one of the sub-systems and passed to the CAE system.

A program was developed in the CAE software in order to automate the analyses process. Depending on the configuration of the studied rotor, this system loads the different components' 2D geometry, or the cross section of the 3D components, from the CAD models. The loaded geometries are meshed using the appropriate elements: thermal, fluid, convection and surface effect elements for the thermal analysis, and structural and contact elements for the stress analysis. The thermal and structural elements are defined as axisymmetric or with thickness depending on the component of the rotor they are created for. The calculated thermal boundary conditions, varying in time in accordance with the mission, are applied on the zones identified in the CAE program. For the stress analysis, boundary and load conditions are much simpler to evaluate and apply considering that they consist of fixing the disc's bore to prevent any radial translation, using the thermal results and applying the proper rotational speed. Centrifugal forces are easily computed by combining the elements' mass matrices and the rotational velocities.

4.7   Results and Discussion

A 2D transient thermal analysis process is more complex than the following stress analysis (whose boundary conditions are much easier to set than for a thermal analysis). For that reason, the presented results focus on thermals. It is indeed in the thermal analysis part of this work that improvements are made considering that executing a stress analysis is straightforward once thermals are obtained. Thermal and stress analyses were carried out using the process described in this paper for one turbine stage of three different engines. Those engines are of different configurations: a turboshaft as test 1, a high-bypass ratio turbofan as test 2 and a turboprop as test 3, and different turbine stages were chosen: two high pressure turbines and one low pressure turbine. The results obtained were compared to the final detailed design results used to create these engines. Figure 4.9 (a) and Figure 4.9 (b) show an example of results obtained for a thermal analysis done on the test case number 2.
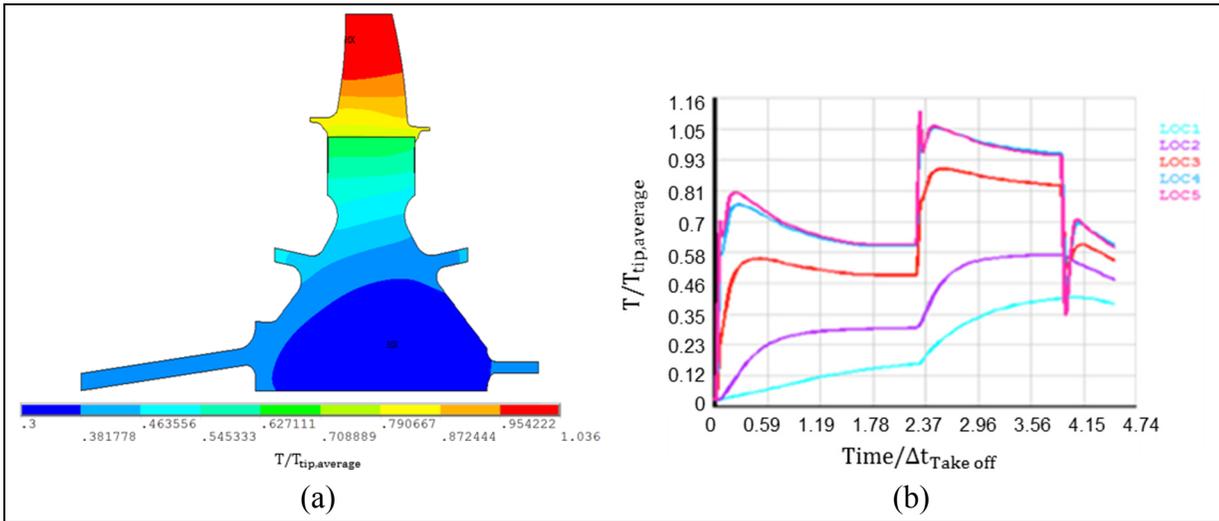
Figure 4.9: Thermal analysis results example performed on test case 2: (a) contour plot, (b) transient plot of temperatures at key locations

On the Figure 4.9 (b), one can observe the evolution of the temperature probed at five key locations around the rotor represented on the temperature contours of Figure 4.9 (a). As absolute values cannot be disclosed in this paper for intellectual property reasons, nondimensionalized scales where used in Figure 4.9 (a) and Figure 4.9 (b). In these two figures, the temperature unit was removed by means of the airfoil's tip average temperature and the time scale in Figure 4.9 (b) was nondimensionalized using the Take-off condition's duration. Figure 4.10 represents the absolute average difference between the transient temperatures obtained with the system described in this work and with detailed design level tools at various locations around each of the three turbine's rotor. Test 1 is an engine with a cover-plate explaining why seven (instead of five) locations are present for that test case. As it does not seem that any conclusion can be made on the over- or underestimation of the temperatures by the pre-detailed design system, absolute values of the differences were displayed in Figure 4.10. A maximum difference of 7 % was obtained for one test case at one location. That result is explained by the simplified geometry compare to a detailed final design (definition and size of the holes, channels and other features for example), which has an impact on both the secondary air system and the thermal boundary conditions. The second source of difference with the detail design process is the simplicity of the models used for the calculation of the heat transfer coefficients and the number of zones used. However, a

difference of less than 7 % at the pre-detailed level compared to detailed design results is excellent. Moreover, the average of the differences of the three turbines at all the locations is 2.72 % with a standard deviation of 2.13 %.
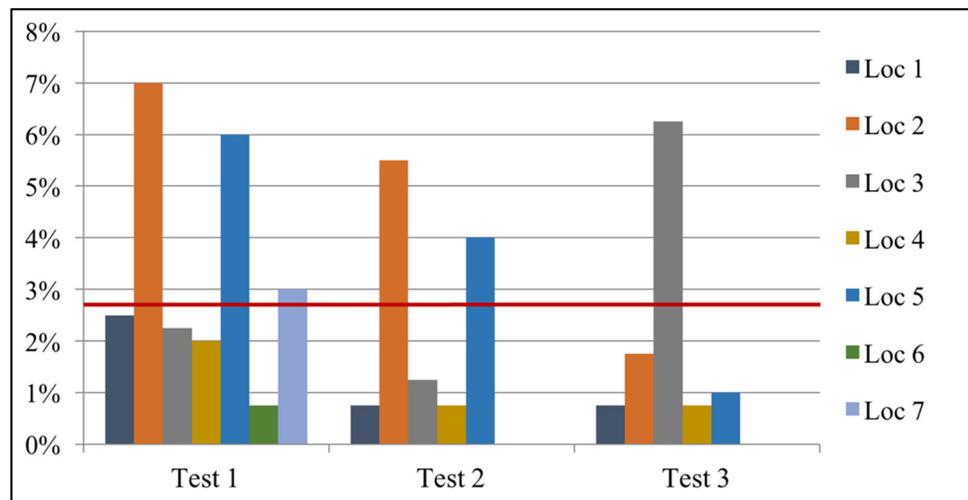
Figure 4.10: Presented work vs. Detailed Design results

For the preliminary design of a rotor such thermal analyses are not usually executed and scaled temperature maps based on legacy data are instead used. Getting the right scaling factors can be a long and tricky process, and the temperatures so obtained are often found to be off once in detailed design phase. Considering that, one understands that comparing the results obtained in this work with pre-detailed level results is more complicated. However, it is generally accepted that pre-detailed design results are about 30 % off. The average difference of 2.72 % obtained in this work is therefore a considerable improvement of the preliminary design phase's results. One has to realize two things in order to understand what is the real impact of integrating an efficient thermal analysis process into the preliminary design phase of a rotor instead of using reference maps. Firstly, thermals have a great impact on a rotor's life (materials properties vary with temperature, an unexpected delta of temperature can induce unpredicted thermal stresses, etc.). This means that a rotor's design might have to be rethought in detailed design if large discrepancies are found in the temperatures predicted during the pre-detailed design phase. Secondly, in terms of tip clearance calculation, thermals are responsible for a majority of the total growth of the rotor.

To highlight that, several steady state stress analyses were conducted on two test cases (one high pressure and one low pressure turbines) at different mission conditions. For these stress analyses, the rotational speed was set to zero in order to obtain the contribution of thermals to the rotor's growth. The obtained results were compared to complete stress analyses' results to calculate the thermal to total growth ratio. These results are summarized in Table 4.1 where the thermal and total growths were nondimensionalized using the test cases' respective blade tip radii. Depending on the rotational speed of the studied turbine stage, thermals were found to be responsible of about 78 % (high pressure turbine) to 87 % (low pressure turbine) of the total growth. One can therefore conclude that such an improvement (from 30 % to 2.72 %) on temperatures prediction will have a great impact on tip clearance calculations.

Table 4.1: Thermal contribution to rotor's total growth

	Conditions	Thermal [%]	Total [%]	Ratio [%]
Engine 1	1	0.56	0.72	76.87
	2	1.11	1.41	78.61
	3	0.97	1.24	78.17
			Average	**77.88**
Engine 2	1	0.31	0.34	89.89
	2	0.60	0.70	85.81
	3	0.53	0.62	85.74
			Average	**87.15**

Starting with an engine concept and the related data, rotors were designed and then analyzed (to obtain thermal maps, stress distribution and overall displacements) using this work's system. The time required to execute these studies was compared to what is usually taken by a P&WC expert using the existing pre-detailed design process. Although bringing more knowledge in the early stages of the design process usually increases the time spent at the pre-detailed stage, it was found that through the integration of the various disciplines involved in the design and analysis of a turbine's rotor, and due to the automation and simplification of the process (data automatic extraction/generation, reduction of the number

of inputs required from the user, etc.), the time to get a first pass of results was reduced by 80 %. It was indeed evaluated that could now be obtained in days what was usually the result of weeks of work.

In the future, it is expected that with more accurate preliminary results comes a reduction of the number of iterations required at the detailed design phase. It was estimated by several specialists that the detailed design process could be shortened by an entire design cycle. Considering that a detailed design process usually requires around three pass of the design cycle, the system introduced in this work could lead to a time reduction of about 30 % on the whole rotor detailed design phase. This estimation was made on the grounds that a more accurate pre-detailed design, obtained by taking into account more data (i.e. secondary air system, thermals, and even more disciplines incorporated in the other modules of this collaborative program between Pratt & Whitney Canada and the Ecole de Technologie Supérieure), will remove the need for a first pass in the detailed design phase to generate the missing data of a conventional preliminary process.

4.8   Conclusion

It is expected from the implementation of a PMDO system to improve the quality of pre-detailed level results considering that more knowledge is being injected early in the design process. It is also anticipated that such system would lead to an increase of productivity through the reduction of the overall design time. Through this work it was demonstrated that, when using the presented system, the time required to deliver rotor thermal and stress results in the pre-detailed design phase is reduced by 80 % due to improvements in the process efficiency. In addition, it was estimated by specialists that the higher quality and quantity of the pre-detailed data would lead to a time reduction of about 30 % during the detailed design phase.

The new system showed a substantial improvement in the quality of the pre-detailed design results, from an estimate of about 30 % to less than 3 %, when compared to detailed design

results. This means that a preliminary simplified secondary air system can be generated and will provide accurate data. It also proves that the decision of reducing the number of boundary condition zones, compared to detailed design best practices, to a minimum of one zone matching each air system chamber is valid. Finally, the selection of simplified models for the calculation of the heat transfer coefficients proved to be more than satisfying when looking at the thermal analysis results.

An evaluation of the effects of thermals over a rotor radial growth showed that stresses due to temperature only are responsible for more than 75 % of the total radial growth of a rotor. This observation highlighted the impact of integrating a thermal analysis process in the preliminary design phase.

This work also confirms that an automation of the preliminary design process is possible. The results presented in this paper suggest that one engineer could be in charge of designing and analyzing an entire rotor during the pre-detailed phase, without the need for many specialized groups to be involved.

Moreover, by requiring fewer user inputs and removing the non-value added task for an engineer to manage data, this system decreases the risk of human errors while entirely leaving the important decisions to the user.

Finally, the process described in this work closes the gap between the pre-detailed and detailed design phases meaning that results can be generated where assumptions had to be made before.

CHAPTER 5

AUTOMATED THERMAL AND STRESS PRELIMINARY ANALYSES APPLIED TO A TURBINE HOUSING ASSEMBLY

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5.1 Abstract

The use of Multidisciplinary Design Optimization (MDO) techniques at the preliminary design phase (PMDO) of a gas turbine engine allows investing more effort at the pre-detailed phase in order to prevent the selection of an unsatisfactory concept early in the design process. Considering the impact of the turbine tip clearance on an engine's efficiency, an accurate tool to predict the tip gap is a mandatory step towards the implementation of a full PMDO system for the turbine design. Tip clearance calculation is a good candidate for PMDO technique implementation considering that it implies various analyses conducted on both the rotor and stator. As a second step to the development of such tip clearance calculator satisfying PMDO principles, the present work explores the automation feasibility of the whole analysis phase of a turbine shroud segment and housing assembly preliminary design process, and the potential increase in the accuracy of results and time savings. The proposed conceptual system integrates a thermal boundary condition automated calculator and interacts with a simplified air system generator and with several design tools based on parameterized CAD models. Great improvements were found when comparing this work's analysis results with regular pre-detailed level tools, as they revealed to be closer to the one generated by the detailed design tools used as target. Moreover, this design process revealed to be faster than a common preliminary design phase while leading to a reduction of time spent at the detailed design phase. By requiring fewer user inputs, this system decreases the risk of human error while entirely leaving the important decisions to the designer.

5.2   Introduction

The design of a gas turbine engine is a multidisciplinary and iterative problem in which the best compromise has to be found between the conflicting disciplines involved: thermal, structural, aerodynamics, manufacturing, cost, weight, etc. The design of aero-engines traditionally follows two main stages: preliminary design and detailed design. At the pre-detailed stage, only a few groups are involved to design and analyse the turbine concept's components and sub-systems. However, the Science and Technology Organisation of NATO showed that decisions taken early in the design process are often based on low fidelity models and when only little information (data, requirements, etc.) is available (NATO, 2006). This may compromise the analysts' ability to select the optimal design. At the detailed design phase, more groups are involved having their own set of specialized tools and methodologies, and the process is thus even more segmented within the groups to form sub-disciplines' specialists. Panchenko et al. explain that even though knowledge increases during the design process, the freedom to modify any part of the design decreases as shown in Figure 5.1, and/or induces major delays in the planning and a rise of design costs. It is consequently difficult to correct a bad concept at a detailed design phase. To correct this, the use of MDO at the preliminary design phase is suggested, since it is at that stage that the biggest influence on the final product configuration can be made (Panchencko et al., 2002). The concept of MDO has been widely studied during the past 50 years (Martins & Lambe, 2013). However, there is a lack of information in the literature about using this methodology during the early stages of design. Panchenko et al. explain that an increase of the effort and knowledge during the pre-detailed design phase implies involving directly the specialist groups instead of waiting until the detailed design phase. However, it results in significant delays of the concept evolution since many interactions between several groups are then required (Panchencko et al., 2002). This is where PMDO systems come into play in order to automate and ease this iterative process. Referring to Panchenko et al. (2002), NATO Science and Technology Organisation (2006) and Korte et al. (1998), the following steps are required to implement a PMDO system:

1. Develop a robust tool base: design tools based on parametrized CAD models, and advanced physics analysis tools which includes the development or improvement of correlations;

2. Apply single discipline optimization to individual analytical tools;

3. Create an integration framework, i.e. a software architecture enabling integration, communication and execution of several tools;

4. Implement multidisciplinary optimization with a clear statement of the design objectives, constraints and variables, and an appropriate selection of the algorithms.

Figure 5.1 : Knowledge vs. design freedom during the design process
Taken from NATO (2006)

A collaborative program was initiated between Pratt & Whitney Canada and the École de Technologie Supérieure to implement an MDO system for designing turbines at the pre-detailed design phase (PMDO). The present work focuses on the second part of the first step described here above (i.e. the analysis portion) and on the second and third steps. As part of this collaborative program, the development of a tip clearance calculation system is a mandatory step and a perfect example for the implementation of a PMDO methodology considering that it requires the design and analyses (thermal, structural and aerodynamic) of

several turbine components as described in Figure 5.2. It was shown that the prediction of a turbine's tip clearance through a typical flight mission is essential in order to maximize an engine's efficiency and its service life (Lattime & Steinetz, 2002). Indeed, an increase of the tip clearance implies that the engine has to augment the turbine inlet temperature to develop the same thrust. If the disc temperature reaches its upper limit, the engine must be removed for maintenance. If the gap between the rotor and the shroud segments is larger than 1 % of the blade's height, an increase of 1 % in tip clearance produces a drop of about 1 % in efficiency (Hennecke, 1985). Based on this, a process for prediction of tip clearance size variation is mandatory in an integrated turbine design system.

Figure 5.2: (a) Geometry of the E^3 high-pressure turbine; (b) Highlighting of the tip clearance (a) Taken from Melcher & Kypuros (2003, p. 4); (b) Taken from Boswell & Tibbott (2013)

In order to develop a tip clearance calculator at the preliminary phase using PMDO principles, three steps can be identified: (1) the prediction of the rotor's thermal and centrifugal growth (already published in Moret, Delecourt, Moustapha, Abenhaim & Garnier, 2017); (2) the prediction of the static components' thermal growth (subject of this article); and finally (3) the calculation of the radial gap between the blade tip and the shroud segment (subject of a following article). As introduced here above, this paper focuses on

implementing an automated process to analyze the housing (or casing) and shroud segment (also called blade outer air seal) assembly, referred to as "stator" in this work.

Savaria et al. (2017) worked on the devolvement of the stator design module used as a base for the present work. Figure 5.3 shows an example of stator assembly and introduces the naming convention used in this paper for each components of the stator geometry.
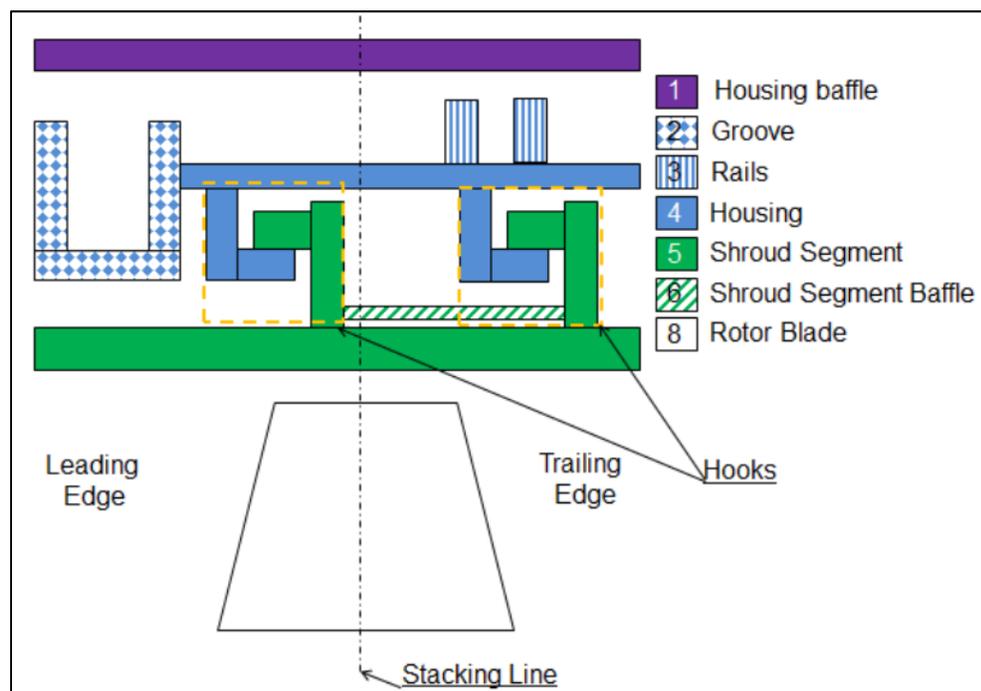
Figure 5.3 : Stator Assembly Components Names
Taken from Savaria (2016, p.37)

This work is structured as follows. The Methodology used to implement the solution is first presented. After what the SAS Generation chapter develops the automatic creation of a numerical fluid network. The Thermal Boundary Conditions calculation process is then introduced, including the simplifications and assumptions made for the heat transfer coefficients calculation. This is followed by a chapter on the Automated Analyses which develops how each part of the process communicates with each other. Finally, the paper concludes with a presentation of the main results allowing validating the proposition made in this work by meeting the stakeholders' criteria.

5.3   Methodology

Based on Wieringa's definition, this work is a practical problem as opposed to a knowledge problem. Therefore the goal of this work is to identify the requirements of the industry regarding the research problem, to propose a new process in response to those requirements, and to implement that proposal (Wieringa, 2009). The initial requirements were to improve the pre-detailed design process by proposing a more flexible, robust and adaptable design and analysis system. This means that the new system has to allow more stator configurations to be modeled while requiring limited number of input from the user. Moreover, the new process results should be targeting within 20 % accuracy when compared to detail design results. This 20 % difference comes from the fact that, based on experts' opinion, it is usually accepted for a regular pre-detailed phase to be about 30 % off when compared to the detail design phase results. This target of 20 % therefore represents an improvement of 33 % compare to the current pre-detailed design process. Finally, the whole design and analysis process should not be slower than the current process.

As it has been developed by the authors previously (Moret et al., 2017), and is resumed here for the reader's benefit, tip clearance calculation over an entire flight mission implies the determination of the transient radial growth, due to temperature and rotational speed (for the rotor), of the turbine's rotor and static components. Figure 5.4 shows the architecture of the tip clearance calculation process being developed in this work, where each arrow represents data exchange. As one can see, the creation of a rotor analyses system is the first step toward the implementation of a tip clearance calculator (Moret et al., 2017). The present project investigates the development of the automated analyses process for the housing and shroud segment assembly. The process required in order to evaluate the stator's components displacement includes several sub-systems: pre-processors, an analysis system and a post-processor for visual purposes. The first and highest-level pre-processor sub-system in this program architecture provides the initial concept's data, i.e. the performance data. The second one, called mission builder, generates the flight mission with the proper transient responses to a change in throttle position for all the air chambers around and inside the stator,

and in the gas path. The third one is the design tool generating geometric data of the stator based on parameterized CAD models. The sub-systems introduced so far are not the work of the authors. They are indeed developed in the collaborative program initiated between Pratt & Whitney Canada and the Ecole de Technologie Supérieure. Examples of such works are described in Savaria et al. (2017) for the implementation of the stator designer sub-system or in Ouellet et al. (2016) and Twahir (2013) for the rotor design sub-systems. The scope of this paper is indeed limited to the following sub-systems.
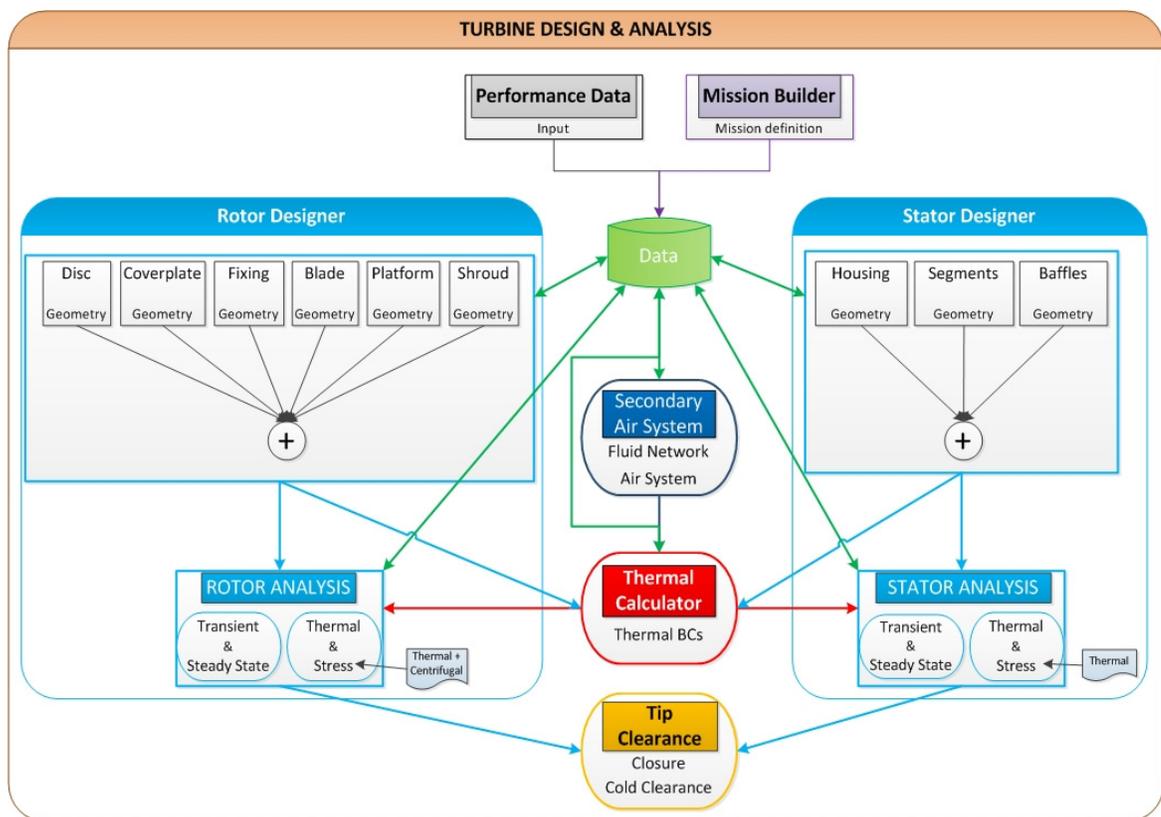
Figure 5.4 : Tip clearance calculation methodology

The fourth sub-system creates the SAS based on the stator's configuration, i.e. location of the housing groove, presence of a piston ring in the groove, presence of impingement baffles, number of housing rails, etc. This sub-system is described in the first sub-chapter of this Methodology on SAS Generation. The final pre-processor sub-system calculates the thermal boundary conditions, i.e. the heat transfer coefficients, for each zone around the stator. This last pre-processor sub-system is described in the sub-chapter on Thermal Boundary

Conditions. The Stator Analysis sub-system automatically executes the thermal and stress transient analyses in a CAE software and provides the user with temperature maps and stator displacements. The sub-chapter on Automated Analyses is dedicated to this analysis sub-system. The integration of all these sub-systems is handled by the use of a common and centralized data structure, and by means of OOP to create a framework repeated through all parts of the system.

If one assumes that the stator's and rotor's geometries (delivered by sub-systems such as described by Savaria et al. (2017), Ouellet et al. (2016) and Twahir (2013)) and the aircraft operating conditions (i.e. its flight mission) are known, the remaining work in the preprocessing step of a stator's thermal and stress analyses is the generation of the secondary air system and the calculation of the thermal boundary conditions.

5.4   Secondary Air System Generation

Studying the SAS of the stator assembly is important in order to control the thermal growth of the housing (Hennecke, 1985) and to maximize the turbine durability by avoiding hot gas ingestion that could lead to metal damage (Malak, Liu & Mollahosseini, 2015). Cooling the shroud segment assembly also allows lowering the housing temperature and therefore controlling more easily its thermal growth. Such process is described as a passive system to control the tip clearance variation in the sense that it allows to achieve a slower response of the housing similar to that of the rotor (Hennecke, 1985). The shroud segment serves as a thermal barrier taking most of the heat from the gas path and therefore protecting the housing which drives the stator's total growth.

In order to cool the stator, different solutions are used. But the most common ones are to flow the housing with compressor discharge air and to add impingement holes. The housing and shroud segment assembly's geometry can significantly vary from one configuration to another. And the cooling means chosen can also differ for a given geometry. The secondary air system will indeed vary depending on the presence of impingement holes and the number of rows of holes, if a piston ring is sealing a cavity, the orientation of the hooks, the length of

a channel, the presence of protuberances, etc. This means that the resulting secondary air systems are very different from one engine to another. A tool automatically generating a fluid network for a given geometry and cooling scheme must therefore be highly flexible.

The SAS Generator automatically assigns the correct number of air chambers at the relevant locations around the stator, and the stator geometry dictates the type of restrictors to be used between these chambers. The SAS Generator creates a simplified air system network and provides the necessary inputs to an integrated in-house air system analysis software. Figure 5.5 presents three examples of stator with their fluid network automatically generated by the proposed system. The first geometry has a piston ring in the housing groove. The presence of this piston ring seals the cavity above the housing and small leakage flows are modeled passing above and below the ring. One impingement hole also exists in the housing to cool down the shroud segment. The second stator has two impingement holes in the housing central groove with angles different from 90°. The third stator has an impingement baffle attached to the shroud segment. This feature allows reducing the impingement height and therefore increase the heat transfer, but as for drawback to limit the amount of flow that can be used to impinge the shroud segment surface. The third stator is also the only shown here that has impingement on the housing base itself. The yellow triangles at the extremities of the fluid networks are the air sources constraining the thermal model. Two are located upstream and downstream of the housing and provide cooling air, and two are located in the gas path upstream and downstream of the shroud segment. The third test case has two extra sources above the housing baffle to model the chambers upstream of the impingement holes.

As an example of the use that can be made of this SAS Generator, the impact of increasing or decreasing the amount of flow used to cool the shroud segment by impingement was studied. Secondary air systems were generated with increasing amount of air impinging the shroud segment base from 0 % (meaning no impingement jet but cooling through channel flow instead) of the reference impingement flow from the detail design analysis to 500 % of that reference flow. The stator temperatures obtained after thermal analyses run based on the
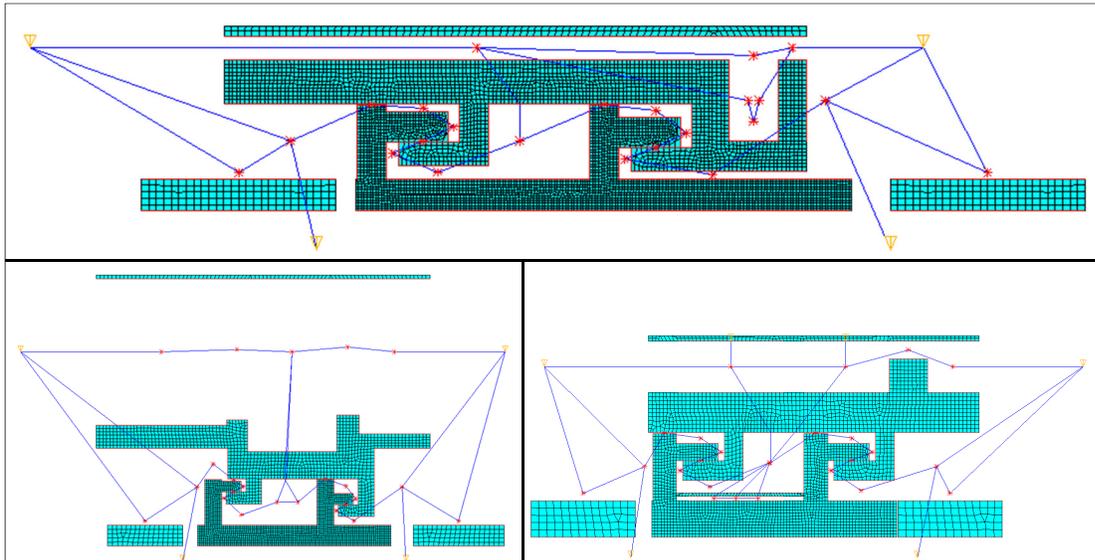
Figure 5.5: Examples of Secondary Air System

generated secondary air systems were measured and averaged on the shroud segment and housing. These temperatures are nondimensionalized using the detail design reference temperature of the housing at the reference impingement flow. As one can observe in Figure 5.6, the temperatures of both the housing and shroud segment follow the same pattern. They both decrease significantly when increasing the impingement flow from 0 % to 100 % of the detail design reference flow. Beyond that point, the temperature of the housing does not decrease much as the flow is increased further more. The shroud segment's temperature continues decreasing linearly with a steeper slope than the housing one. However, the decreasing rate of the shroud segment temperature is also much lower past 100 % compared to what it is prior to that point. This means that an optimal value can be found between the stator's temperatures (and therefore radial growth) and the amount of flow used for the impingement of the shroud segment. Minimizing the air used is important as it is extracted from the compressor and therefore represents a loss from an engine's efficiency point of view.

Following a similar procedure, the SAS Generator allows studying the need for any other geometrical or cooling features that can be created in the stator design tool by rapidly obtaining their impact on the stator temperatures and radial growths. A user can for example

add a baffle on the shroud segment and measure the difference in the obtained results. Based on these results, one can decide if there is a need for that extra feature.
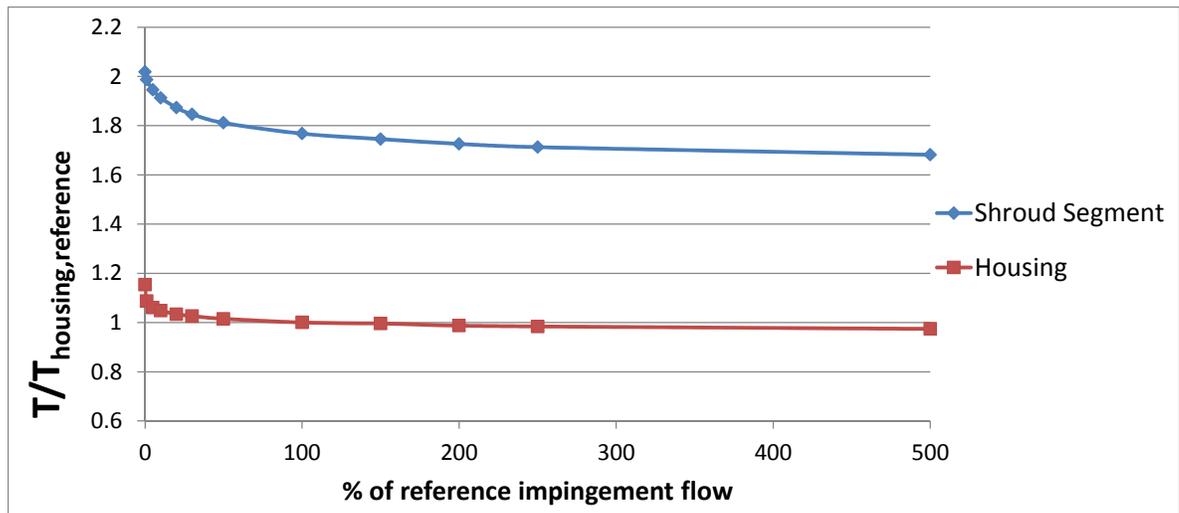
Figure 5.6 : Impingement flow effect on the Shroud Segment and Housing average temperatures

5.5   Thermal Boundary Conditions

To execute a thermal analysis, boundary conditions are required for each specific location (i.e. zone) around the geometry. The Thermal Calculator uses for each zone a specific set of inputs obtained from all the connected sub-systems (performance data, design CAD sub-systems, the SAS Generator, etc.) to calculate the thermal boundary conditions. Thanks to a Gateway software, i.e. a link through which two programs can communicate, the Stator Analysis CAE sub-system is able to make use of the CAD parametric models (cf. Savaria et al. (2017) for example) to identify and measure each zone of the stator geometry.

In detailed design, specific correlations are used to evaluate the boundary conditions on a large number of zones defined all around the 3D geometry of the stator to better represent the effect of every seal, hole, cavity, etc. At a pre-detailed level, the 2D geometry is much simpler and therefore fewer zones are defined. It was decided to define one zone per air chamber. Apart from modeling the entire housing and shroud segment assembly, it was decided to represent the tip portion of the upstream and downstream vanes in order to take

into account some heat pickup from the fluid network on the sides of the stator. An example of zone distribution around a stator's geometry is presented in Figure 5.7 along with the type of correlation used for each of these zones. The thermal boundary conditions correspond to a heat transfer coefficient defined at the surface of each zone, which are also connected to a fluid node. The fluid network connecting every fluid node is constrained by temperature source nodes at the extremities of the network as explained previously and shown in Figure 5.5. The zones in the gas path are not connected to a fluid node and have for thermal boundary conditions a combination of a heat transfer coefficient and a bulk temperature at their surface. Kreith et al. (2011) explain how the bulk temperature can be evaluated based on the fluid's static temperature, its relative velocity and the Prandtl number. Static temperatures around the stator are easily computed based on the gas path mean line temperature (which is part of the initial concept data). The real challenge here is therefore to select the proper correlations for the heat transfer coefficients all around the stator considering that these are directly influenced by the zone's geometry and the physics at stake. The Nusselt number is often used to evaluate the heat transfer coefficient of a surface as these two are directly correlated. At a preliminary design stage, the zones around the stator can be modeled using basic textbook correlations such as flat plates, channels, impingement, etc. The types of assumptions developed here are represented in Figure 5.7.
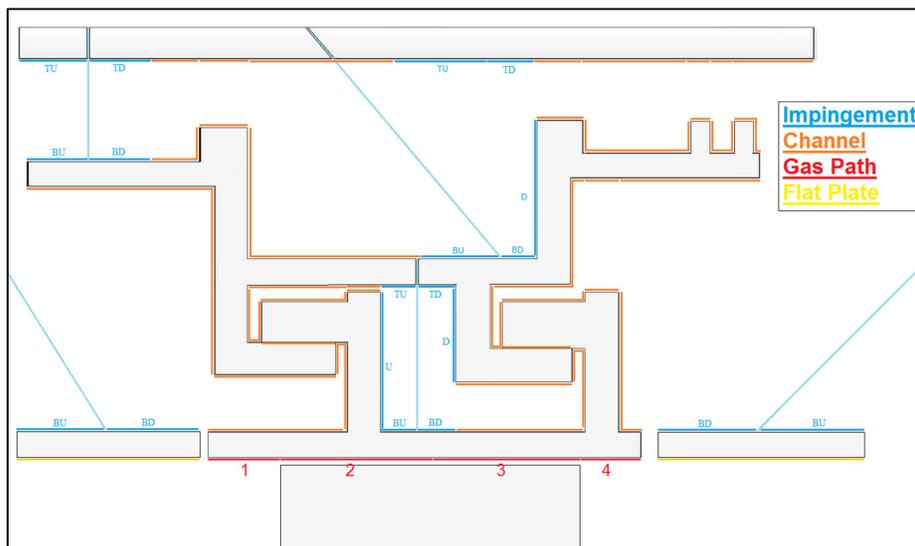
Figure 5.7: Stator thermal zones

5.5.1 Channels

The heat transfer process of channel flows is well known and has been widely covered in the literature (Incropera & DeWitt, 1996; or Kreith et al., 2011). Channels are present all around the stator geometry and represent the majority of the cooling means used to control the housing temperature. However, some very narrow channels can exist in a stator's geometry (i.e. housing and shroud segments hooks, the piston ring, etc.) for which a simple channel correlation would lead to unrealisticly high heat transfer coefficients because of their small hydraulic diameters. Indeed, some of the channels modeled during the boundary condition calculation are in the range of the meso channels (100 μm to 1 mm) (Mehendale, Jacobi & Shah, 2000). for this type of channels, Wang & Peng (1994) consistently obtained Nusselt numbers lower than the ones calculated using Dittus-Boelter Equation (5.1) (Dittus & Boelter, 1930). Wang and Peng came up with Equation (5.2) (Wang & Peng, 1994) which has a reduction factor of 0.35 compared to a regular channel flow correlation (5.1).

For a turbulent flow in regular channels the use of the Dittus-Boelter Equation (5.1) is suggested (Incropera & DeWitt, 1996; Kreith et al., 2011). If the conduit is short and the fluid turbulent, the flow cannot be considered as fully developed and an entrance effect must be accounted for by including a term in the correlation as shown in Equation (5.3). In the case of laminar flows the empirical Equation (5.4) is suggested (Sieder & Tate, 1936).

$$Nu = 0.023 \, Re^{0.8} \, Pr^{1/3} \tag{5.1}$$

$$Nu = 0.00805 \, Re^{0.8} \, Pr^{1/3} \tag{5.2}$$

$$Nu_{with\ entrance\ effect} = Nu \cdot \left(1 + \left(\frac{D_h}{L}\right)^{0.7}\right) \tag{5.3}$$

$$Nu = 1.86 \left(\frac{Re \, Pr \, D_h}{L}\right)^{1/3} \tag{5.4}$$

5.5.2   Impingement Jets

Cooling impingement jets are commonly used to provide high local heat transfer in modern gas turbine engine and effectively cool down a specific region of the geometry. Goldstein and Franchett experimentally determined the heat transfer to a jet impinging at different spacing and angles to a plane surface. The authors obtained the empirical correlation given in Equation (5.5) for the local Nusselt number on the plane surface (Goldstein & Franchett, 1988).

$$\frac{Nu}{Re^{0.7}} = A\ e^{-(B+C\ cos\phi)\left(\frac{r}{D}\right)^{0.75}} \tag{5.5}$$

In Equation (5.5), D is the jet diameter, r and ϕ are the cylindrical coordinates of contours of constant Nu and the coefficients A, B and C are given in the Table 5.1 depending on the impingement angle α and the distance L from jet orifice to surface. Compared to other research work done on the heat transfer process of impingement jets, using Equation (5.5) allows taking into account the effect of the impingement jet's angle which is necessary in housing and shroud segment assemblies where the hardware design often require the use of non-vertical impingement jet.

Table 5.1: Goldstein and Franchett coefficients
Taken from Goldstein & Franchett (1988, p. 87)

α	A			B	C
	L/D = 4	L/D = 6	L/D = 10		
90°	0.159	0.155	0.123	0.37	0
60°	0.163	0.152	0.115	0.4	0.12
45°	0.161	0.146	0.107	0.47	0.23
30°	0.136	0.124	0.091	0.54	0.34

It was proved that when surrounded by crossflow, which is often the case in a stator assembly where channel flows are omnipresent, two heat transfer areas can be defined: the

first area being driven by the impingement jet and the second being the area of crossflow influence. These two areas are illustrated in Figure 5.8. It is suggested to define the area of influence of an imping jet as the region where the Nusselt number values are above 50 % of the stagnation point value (Van Treuren, Ireland, Jones & Kohier, 1996). However, when automating an analysis process, one needs to define the zone affected by an impinging jet prior any analysis. When using Goldstein and Franchett contours of constant Nusselt number (Goldstein & Franchett, 1988), one can measure that 50 % of the Nusselt number at the stagnation point for an angle of 90° correspond to a distance of about 3.125 times the hole diameter from the stagnation point. If measuring this length of influence for the downstream side of the stagnation point depending on the angle of impingement, one comes up with Table 5.2.
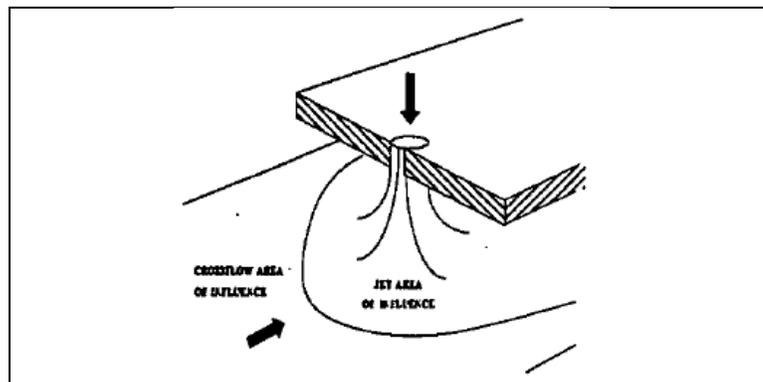
Figure 5.8 : Areas of Influence for the Impinging Jet Array
Taken from Van Treuren et al. (1996)

In Table 5.2, the values for the angles of 120°, 135° and 150° were obtained by using the length of the upstream side (i.e. right of the stagnation point) of the contours for the angles of 60°, 45° and 30° (respectively). When executing a polynomial regression on Table 5.2, one obtains the Equation (5.6) that has an R^2 of 0.9924.

Table 5.2: Length of influence
for an impingement jet

Angle [°]	x/D @ 50 % Nu
30	6.875
45	5.938
60	4.375
90	3.125
120	2.375
135	2.187
150	1.563

$$\frac{x}{D} = -3 \ 10^{-6} \ \alpha^3 + 0.0012 \ \alpha^2 - 0.1689 \ \alpha + 11.093 \qquad (5.6)$$

The impingement zones are therefore defined by calculating the length of influence of the impingement jet upstream and downstream of the impact point. The zones surrounding the impinged surface and inside the area of influence may not be directly impinged but they are still influenced by the impingement jet heat transfer coefficients due to the jet reflection. Heat transfer coefficients on these zones are generally defined as a percentage of the directly impinged zone's heat transfer coefficient depending on the quantity of cooling flow reflected toward them. This can be observed in Figure 5.7 where the second impingement jet (oblique) on the housing groove has five zones affected by the jet: the two zones directly upstream and downstream of the jet (BU and BD), the zone on the right sidewall of the groove (D) which is in the area of influence of the jet and the two zones above the impingement jet (TU and TD). On the other hand, the first impingement jet (vertical) on the housing does not have a zone on the nearest sidewall because this one is outside of jet area of influence.

5.5.3   Flat Plates

For the sake of simplicity, the gas path side of the vanes can be modeled as flat plates. For such model, one can use the following Equations (5.7) and (5.8) (Incropera & DeWitt, 1996).

$$Laminar: Nu = 0.332 \, Re^{1/2} Pr^{1/3} \tag{5.7}$$

$$Turbulent: Nu = 0.0296 \, Re^{0.8} Pr^{1/3} \tag{5.8}$$

5.5.4   Shroud Segment Gas Path Side

Many references develop the behavior of the secondary flow within the blade tip and the shroud segment gap (Azad, Han, Teng & Boyle, 2000; Prasad & Wagner, 2000). It was proved that the unsteadiness of the flow seems to be mostly confined to the gap between the blade tip and the shroud segment (Prasad & Wagner, 2000). Based on this observation, one can conclude that for a simplification purpose of the problem at stake, the shroud segment gas path side can be separated in two regions:

The first one is composed of the surfaces preceding and following the blade. These zones are the numbers 1 and 4 in Figure 5.7. They are exposed to the main flow and can be modeled as a flat plate (using Equations (5.7) and (5.8)) considering that the blade tip does to affect them.

The second region is the portion of shroud segment located above the blade tip and numbered 2 and 3 in Figure 5.7. The zones of this second region cannot be modeled as easily as for the first region. The reason is that when the blade is underneath the shroud, the shroud – blade's tip system can be considered as two concentric cylinders, with the shroud a stationary one. While when no blade is under the shroud, this one can be seen as a flat plate as in the first region. To consider these two cases, one can calculate a ratio of the time the shroud segment is facing the tip to the time it is not.

Rademaker, Huls, Soemarwoto & van Gestel (2013) studied the heat loading of a shroud segment and more specially the effect of the blade tip's windage and the radial outflow of cooling flow at the blade squealer tip. This study of the windage effect is interesting for the present work because it highlights the periodic variation of the shroud temperatures with the passing of the blades. The authors introduce an equation allowing them to calculate the reduction factor in thermal loading to the shroud surface. If one considers this concept and applies it to the present problem of calculating the heat transfer coefficient at the surface of the shroud segment, one can come up with Equation (5.9).

$$h = \sigma \, h_{concentric\ cylinders} + (1 - \sigma) \, h_{flat\ plate} \qquad (5.9)$$

where σ is the time fraction for which the shroud segment is covered by the blades. Using basic geometry as shown in Figure 5.9, this time fraction can be calculated using Equation (5.10).
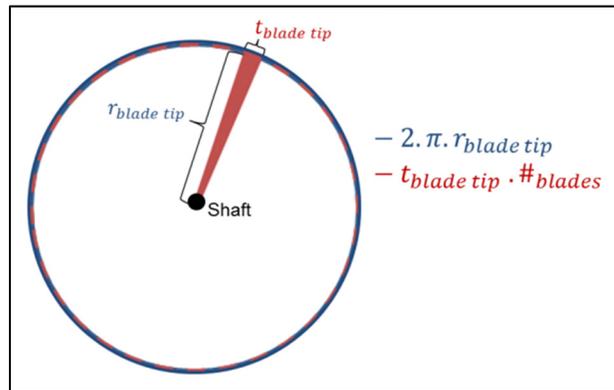
Figure 5.9: Time Fraction Theory

$$\sigma = \frac{t_{blade\ tip} \ \#_{blades}}{2 \, \pi \, r_{blade\ tip}} \qquad (5.10)$$

where t_{blade} is the blade's thickness and $\#_{blades}$ is the number of blades.

The heat transfer coefficient for concentric cylinders can be calculated using the Dittus-Boelter Equation (5.1) but with the Reynold's number calculated based on the fluid's relative velocity given by Equation (5.11).

$$V_{rel} = \sqrt{U_{rel}^2 + V_{ax}^2} \qquad (5.11)$$

with U_{rel} the fluid's relative tangential velocity between the two concentric cylinders and V_{ax} the fluid's axial velocity.

5.6 Automated Analyses

Automation of the analyses is possible thanks to the process described previously and depicted in Figure 5.4. As mentioned previously, the integration of all the sub-systems shown in Figure 5.4 is handled by the use of a common and centralized data structure. A framework was created and repeated through all parts of the system by means of OOP. Automation of the analyses therefore only requires an effective way for the sub-systems to communicate with the CAD and CAE external software. Using a Gateway system, geometric data can be extracted from the CAD parametrized models and used in the sub-systems for calculation and display purposes. This Gateway software also allows direct communication between the CAD models and the CAE software used to run the 2D analyses. The thickness of the non-axisymmetric components (i.e. the shroud segment, and its baffle if any) are calculated based on the number of segments on the circumference, and applied on the meshed geometry in the CAE system.

A program was developed in the CAE software in order to automate the analyses process. Depending on the configuration of the studied stator, this system loads the different components' 2D geometry from the CAD models. The loaded geometries are meshed using the appropriate elements: thermal, fluid, convection and surface effect elements for the thermal analysis, and structural and contact elements for the stress analysis. The thermal and structural elements are defined as axisymmetric or with thickness depending on the component of the stator they are created for. The calculated thermal boundary conditions,

varying in time in accordance with the mission, are applied on the zones identified in the CAE program. Temperatures are set at the boundary nodes of the fluid systems and air flows are applied on the fluid elements around the geometry. For the stress analysis, boundary and load conditions are much simpler to evaluate and apply considering that they consist of axially fixing the housing to prevent any axial translation, and applying the thermal results.

5.7 Results and Discussion

A 2D transient thermal analysis process is more complex than the following stress analysis. For that reason, the presented results focus on thermals. It is indeed in the thermal analysis part of this work that improvements are made considering that executing a stress analysis is fairly straightforward once thermal analysis results are obtained. Thermal and stress analyses were carried out using the process described in this paper for one turbine stage of three different engines. Those engines are of different configurations: a turboprop as test 1, a turboshaft as test 2 and a high-bypass ratio turbofan as test 3. For all three test cases, this study focused on the first stage of the high pressure turbine. The results obtained were compared to the final detailed design results used to create these engines. Figure 5.10 (a) and Figure 5.10 (b) show an example of results obtained for a thermal analysis done on the test case number 1. In Figure 5.10 (b), one can observe the evolution of the temperature probed at five key locations around the stator represented on the temperature contours in Figure 5.10 (a). As absolute values cannot be disclosed in this paper for intellectual property reasons, nondimensionalized scales were used in Figure 5.10 (a) and Figure 5.10 (b). In these two figures, the temperature unit was removed by means of the shroud segment maximum metal temperature and the time scale in Figure 5.10 (b) was nondimensionalized using the Take-off condition's duration.

Figure 5.11 represents the absolute average difference between the temperatures obtained with the system described in this work and with detailed design level tools. As it does not seem that any conclusion can be made on the over- or underestimation of the temperatures by the pre-detailed design system, absolute values of the differences were used. The

temperatures were measured at 10 different key locations around each of the three turbine's stator geometry. The three first locations (in blue) are in the shroud segment base. The four next (in red) are in the hooks (one per hook). And the last three locations are in the Housing base/Groove. One can see in Figure 5.11 that results are below (or equal to) the 20 % difference set as target by the stakeholders during the project requirements definition introduced in the Methodology.
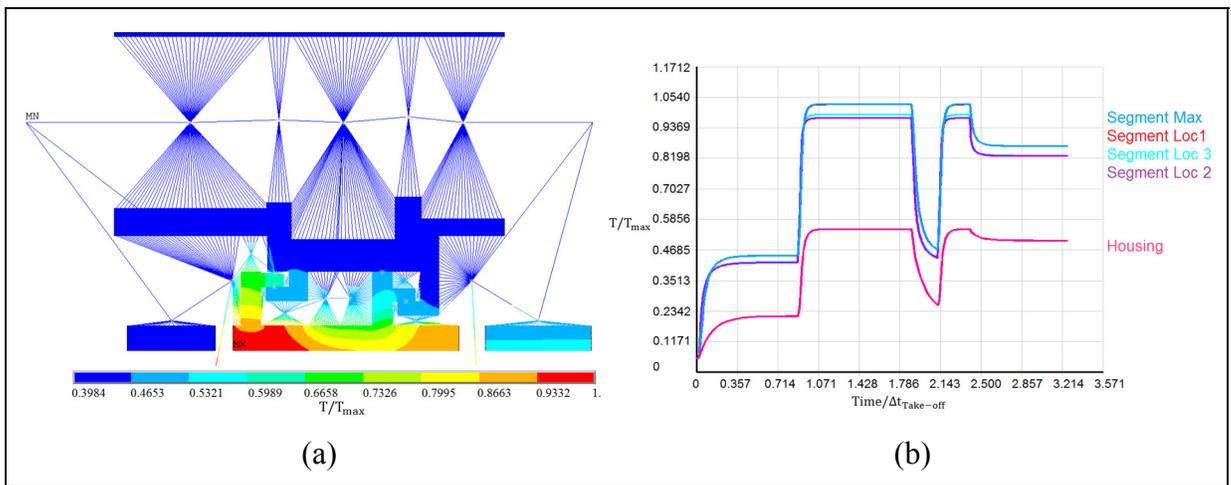
Figure 5.10: Thermal analysis results example performed on test case 1: (a) contour plot, (b) transient plot of temperatures at key locations
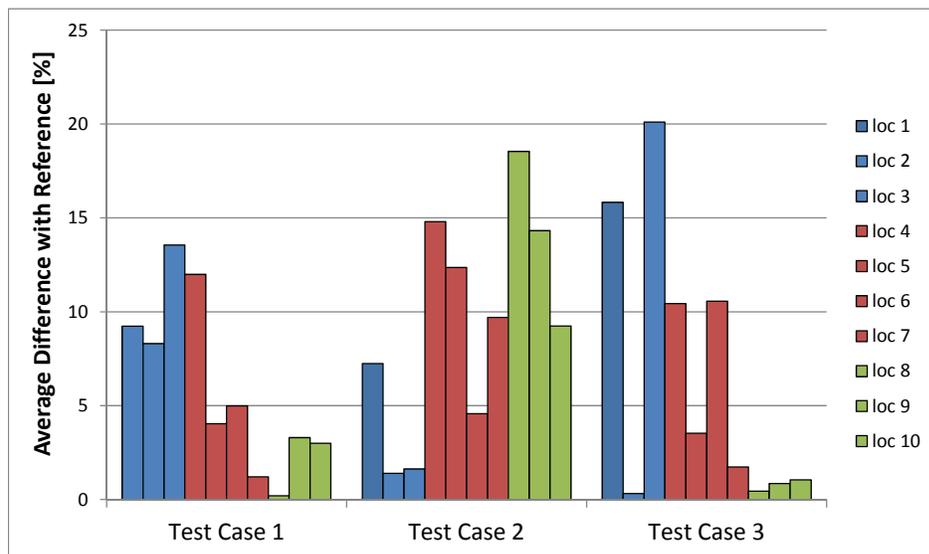
Figure 5.11: Presented work vs. Detail Design results

66

Considering that local effects are not of much interest for this study since they do not have a significant influence on the stator radial growth, the results were averaged per component (i.e. the shroud segment, hooks and housing) to highlight the main sources of discrepancies in each test case. Figure 5.12 shows these averaged results along with the total differences per component averaged on all three test cases.

One can observe in Figure 5.12 that it is in the shroud segment that differences are at a maximum in the first and third test cases. Indeed, an average difference of 10.36 % was obtained in the shroud segment of the first test case and an average difference of 12.08 % was reached in the same component of the third test case. For the second test case, the Housing is where the differences mainly originate from with an average difference of 14.03 %. The total average difference of all test cases for all the components is 7.26 %, with a standard deviation of 4.4 %, which exceeds the expectations that led to an initial target of 20 % difference. However these results can be even more improved if the main source of discrepancies of each test case are studied and answered to.
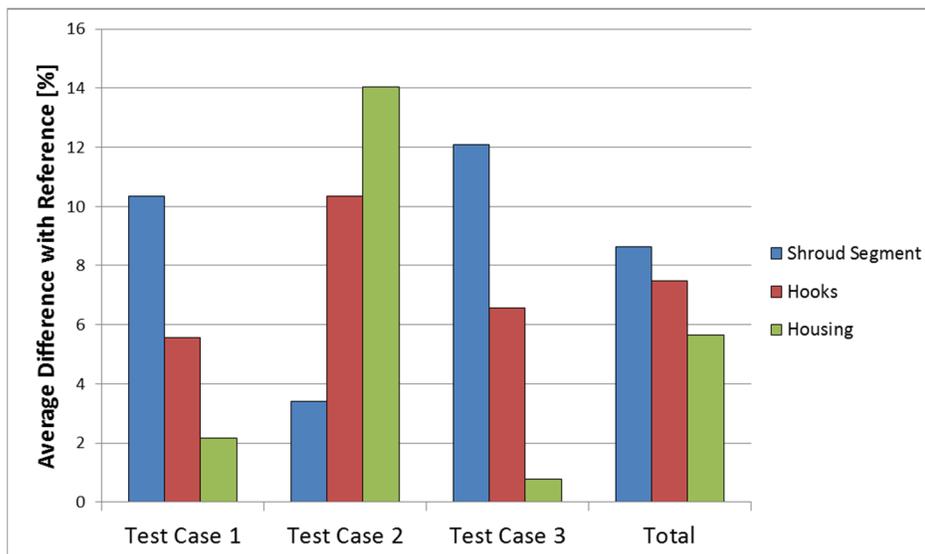
Figure 5.12: Comparison results averaged per component

The shroud segment of the test cases 1 and 3 are internally cooled using channels inside the shroud segment base which conduct air from the shroud/housing cavity downstream of the

shroud segment. These channels are modeled in the detail design analysis, but were not part of the initial requirements of the system introduced in this work. This explains why the results obtained here for the shroud segment base are hotter than the reference detailed design ones. In the second test case, the shroud segment is not internally cooled and the results for the shroud base are much closer to the reference values. However the housing ones are not. This is due to the fact that the detail design analysis models a bigger portion of the geometry of the housing. Indeed, instead of limiting the analysis to the portion of the housing above the shroud segment as it is done in this work, the detail design analysis models the entire housing up to its flange attachment. This leads to a reduced average temperature of the housing by conduction considering that the upstream section of the housing is not affected by any heat from the gas-path.

As a second iteration of the preliminary design process, small modifications were made to the model to better match the reference results. Cooling channels were modeled inside the shroud segment base for the test cases 1 and 3, and a zone was added on the upstream side of the housing of the second test case in order to mimic the effect of conduction through the upstream portion of the component. Figure 5.13 shows the results obtained after this second iteration of the preliminary design process. A maximum average difference of 9.2 % is obtained in the hooks of the second test case. That result is explained by the use in this work of a simplified geometry and air system as compared to the detailed final design (modeling of leakage flows, definition of extra holes, extra channels and other features for example), which have an impact on the thermal boundary conditions. The second source of difference with the detail design process is the simplicity of the models used for the calculation of the heat transfer coefficients and the number of zones used. However, a difference of less than 10 % at the pre-detailed level compared to detailed design results is excellent. Moreover, the average of the differences of the three turbines at all the locations is 3.78 % with a standard deviation of 2.5 %.

Comparing the system presented in this work with the existing pre-detailed design process is not easy because the proposed solution goes beyond what is usually expected from a

preliminary design phase. Only few data are therefore available that allow comparing the presented solution with both a usual pre-detailed solution and at the same time with what was finally obtained during the detailed design phase for the same test case. One engine was however identified that went through both these steps using recent methods and analytical tools from beginning to end. The third test case introduced previously allows the authors to measure the improvement brought by the system introduced in this paper by comparing it to the process that it is supposed to replace.
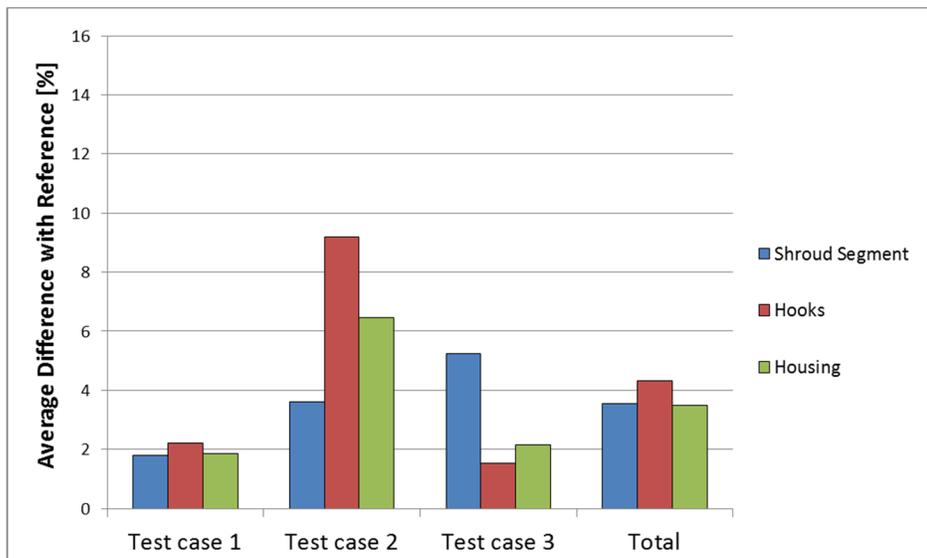
Figure 5.13: Presented work vs. Detail Design improved results

As one can see in Figure 5.14, a first iteration (i.e. not the improved results) of the system described in this work delivers more accurate results than the current process when compared to a detail design analysis. For the shroud segment, the authors' proposal is able to deliver results 20 % more accurate than the current process. The proposed system also delivers results 5 % more precise in the stator hooks, and 2 % in the housing. The housing temperatures are predicted with a similar precision by the two systems because the third test case has a simple housing configuration (no groove or piston ring for example) meaning that the current process is able to properly model the physics at stake. But the shroud segment physics is more complex (impingement baffle with several rows of holes, gas path, etc.)

which cannot be captured by the current process with as much fidelity as the proposed solution.
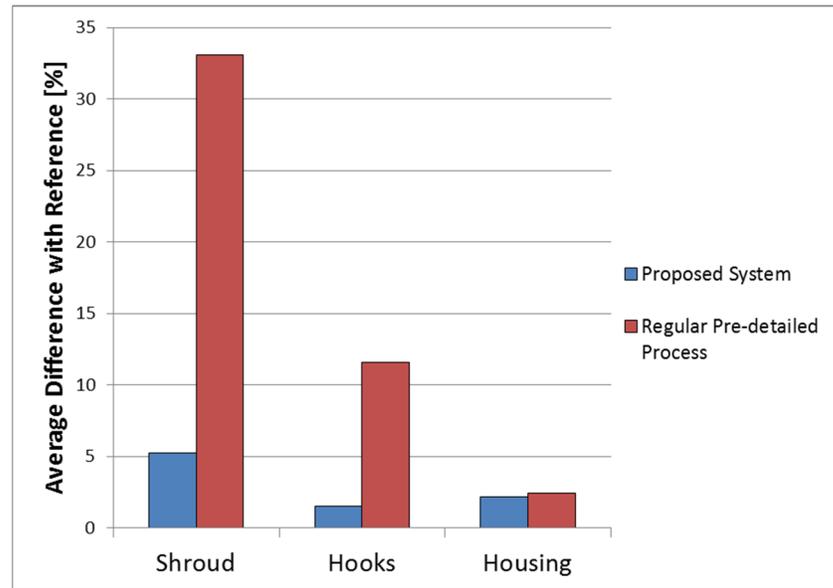
Figure 5.14: Comparison of the results of the current and
proposed processes with detail design results for the test case 3

Starting with an engine concept and the related data, stators were designed and then analyzed using system described in this paper. The time required to execute these studies was compared to what is usually taken by a P&WC expert using the existing pre-detailed design process. Although bringing more knowledge in the early stages of the design process usually increases the time spent at the pre-detailed stage, it was found that the time to get a first pass of results was reduced by 80 %. It was indeed evaluated that could now be obtained in days what was usually the result of weeks of work. This improvement comes from the integration of the various disciplines involved in the design and analysis of the turbine's stator, the increased flexibility of the system which removes the need for manual intervention from the user in some parts of the current process, and due to the automation and simplification of the process (data automatic extraction/generation, reduction of the number of inputs required from the user, etc.).

As introduced previously, it is expected that with more accurate preliminary results comes a reduction of the number of iterations required at the detailed design phase (Moret et al., 2017). It was estimated by several specialists that the detailed design process could be shortened by an entire design cycle. Considering that a detailed design process usually requires on average three iterations of the design cycle, the system introduced in this work could lead to a time reduction of about 30 % on the whole housing and shroud segment assembly detailed design phase. This estimation was made on the grounds that a more accurate pre-detailed design, obtained by taking into account more data (i.e. secondary air system, thermals, and even more disciplines incorporated in the other modules of this collaborative program between Pratt & Whitney Canada and the École de Technologie Supérieure), will remove the need for a first pass in the detailed design phase to generate the missing data of a conventional preliminary process.

5.8   Conclusion

It is expected from the implementation of a PMDO system to improve the quality of pre-detailed level results considering that more knowledge is being injected early in the design process. It is also anticipated that such system would lead to an increase of productivity through the reduction of the overall design time. Through this work it was proven that, when using a PMDO system, the time required to deliver a housing and shroud segment assembly's thermal and displacement results in the pre-detailed design phase is reduced by 80 % due to improvements in the process efficiency. It was also estimated that the higher quality and quantity of the pre-detailed data would lead to a time reduction of about 30 % during the detailed design phase.

The new system showed a substantial improvement in the quality of the pre-detailed design results, from an estimate of about 30 % to less than 8 %, when compared to detailed design results. The accuracy of the results was further improved during a second iteration on the pre-detailed model by adding some extra features to the model, which led to an average difference of less than 4 %. These results prove that the decision of reducing the number of

boundary condition zones, compared to detailed design best practices, to a minimum of one zone matching each air system chamber is valid. Finally, the selection of simplified models for the calculation of the heat transfer coefficients proved to be valuable when looking at the thermal analysis output.

The results showed in this paper answered the knowledge sub-problem of identifying the predominant phenomena in the thermal analysis of a housing and shroud segment assembly. As expected, the main ones were the impingement jet cooling, the heat transfer from the gas-path, and all the channel flows around the stator. But the system introduced here allowed highlighting the importance of modelling the cooling channels inside the shroud segment base and the conduction in the housing if it has been cut, which was not initially anticipated.

This work also confirms that an automation of the preliminary design process is possible. The results presented in this paper suggest that one analyst could be in charge of designing and analyzing an entire stator during the pre-detailed phase, without the need for many specialized groups to be involved.

Moreover, by requiring fewer user inputs and removing the non-value added task for an analyst to manage data, this system decreases the risk of human errors while entirely leaving the important decisions to the user.

Finally, the process described in this work closes the gap between the pre-detailed and detailed design phases meaning that results can be generated where assumptions had to be made before.

CHAPTER 6

AUTOMATION OF A TURBINE TIP CLEARANCE PRELIMINARY CALCULATION PROCESS

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6.1   Abstract

An aeronautical gas turbine engine design is a multidisciplinary iterative process requiring an efficient interaction between each discipline tool and process in order to find the best compromise satisfying all the conflicting domains involved. The gas turbine engine design traditionally has two main stages: the pre-detailed design and the detailed design phases. During the first phase of the design, time is the main concern and the fidelity of the results may be impacted. This may compromise the engineers' ability to thoroughly explore the envelope of potential designs and thus lead to the selection of a sub-optimal system concept. Considering the time-consuming analysis and resources-intensive tools used during the detailed design phase, it is extremely difficult to correct an unsatisfactory concept at that stage of an engine's design. The use of Multidisciplinary Design Optimization techniques at a preliminary design phase (Preliminary MDO or PMDO) allows correcting this. PMDO system implementation requires bringing as much knowledge as possible in the early phases of the design where the freedom to make modification is at a maximum. This imposes the use of higher fidelity tools that communicate effectively with each other. Considering the impact of the turbine tip clearance on an engine's efficiency and on preventing blades wear, an accurate tool to predict the tip gap is a mandatory step towards the implementation of a full PMDO system for the turbine design. Tip clearance calculation is a good candidate for PMDO technique implementation considering that it implicates various analyses conducted on both the rotor and stator. As a third and final step to the development of such tip clearance calculator satisfying PMDO principles, the present work integrates the results of the turbine

74

rotor and housing thermal and stress analyses presented in two previous papers. These processes' integration leads to the automatic calculation of a turbine stage's closure during a given mission and the computation of the cold build clearance. Compared to a regular preliminary tip clearance calculation process, the proposed conceptual system offers a considerable increase in the accuracy of results and a time reduction. The system being automated and faster than the one of a regular pre-detailed design phase, it was possible to run optimisation loops in order determine the worst mission in terms of throat closure. The proposed system also allowed running a sensitivity analysis of the tip clearance leading to the identification of parameters that should be focused on when optimizing a turbine's tip clearance. Finally, by requiring fewer user inputs this system decreases the risk of human errors while entirely leaving the important decisions to the designer.

6.2   Introduction

The design of a gas turbine engine is a multidisciplinary and iterative problem in which the best compromise has to be found between the conflicting disciplines involved: thermal, structural, aerodynamics, manufacturing, cost, weight, etc. The design of aero-engines traditionally follows two main stages: pre-detailed design and detailed design. At the pre-detailed stage, only a few groups are involved to design and analyse the turbine concept's components and sub-systems. However, the Science and Technology Organisation of NATO showed that decisions taken early in the design process are often based on low fidelity models and when only little information (data, requirements, etc.) is available. This may compromise the analysts' ability to select the optimal design. At the detailed design phase, more groups are involved having their own set of specialized tools and methodologies, and the process is thus even more segmented within the groups to form sub-disciplines' specialists (NATO, 2006). Panchenko et al. explain that even though knowledge increases during the design process, the freedom to modify any part of the design decreases as shown in Figure 6.1, and/or induces major delays in the planning and a rise of design costs. It is consequently difficult to correct a bad concept at a detailed design phase. To correct this, the use of MDO at the preliminary design phase is suggested, since it is at that stage that the

biggest influence on the final product configuration can be made (Panchencko et al., 2002). The concept of MDO has been widely studied during the past 50 years (Martins & Lambe, 2013). However, there is a lack of information in the literature about using this methodology during the early stages of design. Referring to Panchenko et al. (2002), NATO Science and Technology Organisation (2006) and Korte et al. (1998), the following steps are required to implement a PMDO system:

1. Develop a robust tool base: design tools based on parametrized CAD models, and advanced physics analysis tools which includes the development or improvement of correlations;

2. Apply single discipline optimization to individual analytical tools;

3. Create an integration framework, i.e. a software architecture enabling integration, communication and execution of several tools;

4. Implement multidisciplinary optimization with a clear statement of the design objectives, constraints and variables, and an appropriate selection of the algorithms.

Figure 6.1: Knowledge vs. design freedom during the design process
Taken from NATO (2006)

A collaborative program was initiated between Pratt & Whitney Canada and the École de Technologie Supérieure to implement an MDO system for designing turbines at the pre-detailed design phase. As part of this collaborative program, the development of a tip clearance calculation system is a mandatory step and a perfect example for the implementation of a PMDO methodology considering that it requires the design and analyses (thermal, structural and aerodynamic) of several turbine components as described in Figure 6.2. It was shown that the prediction of a turbine's tip clearance through a typical flight mission is essential in order to maximize an engine's efficiency and its service life (Lattime & Steinetz, 2002). Indeed, an increase of the tip clearance implies that the engine has to augment the turbine inlet temperature to develop the same thrust. If the disc temperature reaches its upper limit, the engine must be removed for maintenance. If the gap between the rotor and the shroud segments is larger than 1 % of the blade's height, an increase of 1 % in tip clearance produces a drop of about 1 % in efficiency (Hennecke, 1985). Based on this, a process predicting the tip clearance variation during a flight mission and calculating the cold build clearance necessary to avoid any rub is mandatory in an integrated turbine design system.

(a)   (b)
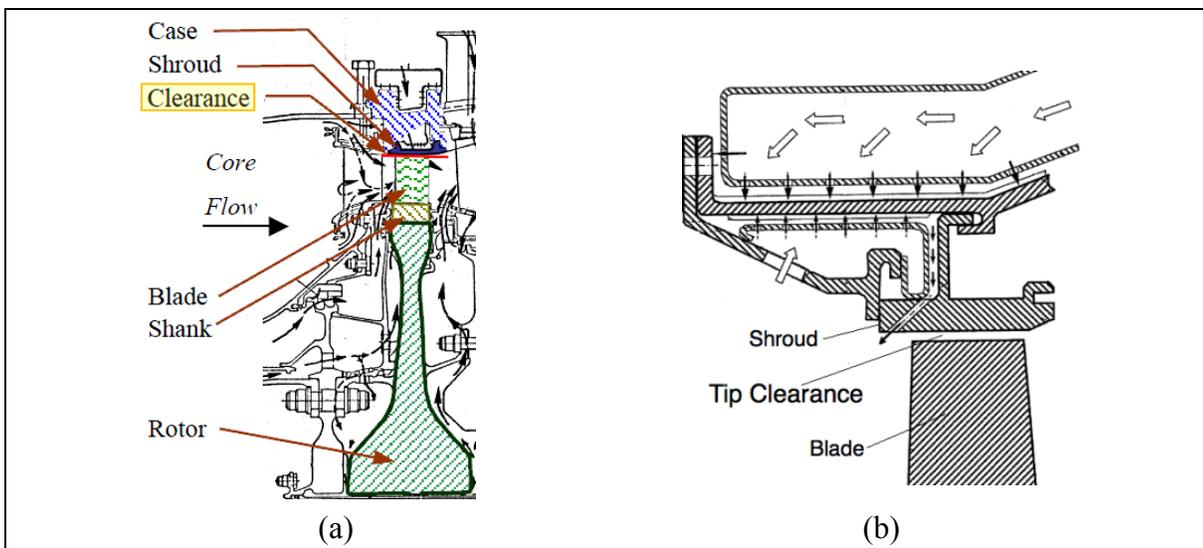
Figure 6.2: (a) Geometry of the E^3 high-pressure turbine; (b) Highlighting of the tip clearance
(a) Taken from Melcher & Kypuros (2003, p. 4); (b) Taken from Boswell & Tibbott (2013)

In order to develop a tip clearance calculator at the preliminary phase using PMDO principles, three steps can be identified: (1) the prediction of the rotor's thermal and centrifugal growth (Moret et al., 2017); (2) the prediction of the static components' thermal growth (Moret, Moustapha, Phutthavong & Garnier, 2018); and finally (3) the calculation of the radial gap between the blade tip and the shroud segment (Moret, Garnier, Phutthavong & Moustapha, 2018).

This work is structured as follows: the methodology used to implement the proposed solution is first presented. This is followed by a chapter developing how the tip clearance is calculated and what is the mechanism of the gap size's variation during a flight mission. This same chapter also develops how the cold build clearance can be determined. Finally, the paper concludes with a presentation of the main results obtained using the process introduced in this work. These results show the possibility to use this process to run iteration loops in order to calculate the cold build clearance. This final chapter also presents the first results obtained when running a targeted sensitivity analysis, thus demonstrating the optimization capability of the tool. These results finally allowed validating the proposition made in this work by exceeding the stakeholders' criteria in terms of time gain and accuracy of the results.

6.3   Methodology

Based on the definition of Wieringa, this work is a practical problem as opposed to a knowledge problem. Therefore the goal of this work is to identify the requirements of the industry regarding the research problem, to propose a new process in response to those requirements, and to implement that proposal (Wieringa, 2009). The initial requirements were to improve the pre-detailed design process by proposing a more flexible, robust and adaptable design and analysis system. This means that the new system has to allow more rotor and stator configurations to be modelled while requiring limited number of input from the user. Based on standard design practice it is usually accepted for a regular pre-detailed phase to be about 30 % off when compared to the detail design phase results. The new process aims at reducing the gap between pre-detailed and detailed design processes, and it

should therefore improve as much as possible this 30 % accuracy. Finally, the whole design and analysis process should not be slower than the current process.

As it has been developed previously by Moret et al. (2017) and Moret et al. (2018), and is summarized here for the reader's benefit, tip clearance calculation over an entire flight mission implies the determination of the transient radial growth, due to temperature and rotational speed (for the rotor), of the turbine's rotor and static components. Figure 6.3, first introduced by Moret et al. (2017), shows the architecture of the tip clearance calculation process being developed in this work, where each arrow represents data exchange. As one can see and as mentioned in the Introduction, the creation of a rotor and a stator analyses system are the two first steps toward the implementation of a tip clearance calculator. For reminder, these steps were published by Moret et al. (2017) and submitted for publication by Moret et al. (2018).
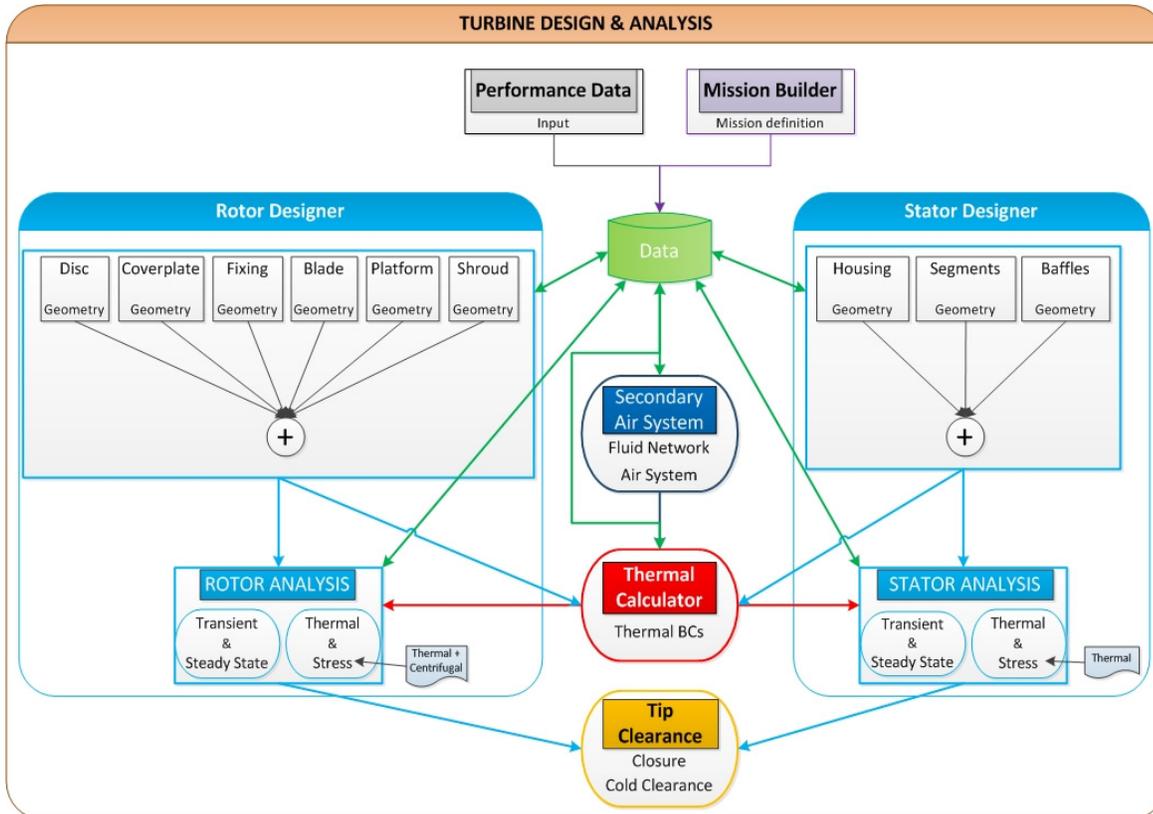

Figure 6.3 : Tip clearance calculation methodology

The integration of all the sub-systems shown in Figure 6.3 is handled by the use of a common and centralized data structure, and by means of OOP to create a framework repeated through all parts of the system.

6.4 Tip Clearance Calculation

As introduced here above, the tip clearance variation during a specific mission can be calculated based on the growth of the rotor and stator components. These allow calculating two parameters: the transient closure of the gap between the airfoil's tip and the shroud segment, and the initial value of the clearance, also called cold build clearance.

$$
\begin{aligned}
\delta s &= (r_{stator} - r_{rotor}) + (\delta r_{stator} + \delta r_{rotor}) \\
&= s_{cold} + \delta c
\end{aligned}
$$

(6.1)

where r_{stator} and r_{rotor} are respectively the initial radius of the shroud segment and airfoil tip, δr_{stator} and δr_{rotor} are respectively the radial growth of the shroud segment and airfoil tip, s_{cold} is the cold build clearance and δc is the gap closure. The closure is here defined as a negative value considering that the rotor's growth is larger than that of the stator. The reason why the closure is defined as a negative value is that it is clearer when graphically representing the tip clearance variation. The clearance starts at its cold build value and gets closer to 0 as the gap between the rotor and stator closes (i.e. the closure becomes more negative). The tip clearance graph is therefore a vertical translation of the graph of the closure, up by the value of the cold build clearance.

To calculate the tip clearance, locations need to be chosen where the radial position of the rotor tip and of the stator surface are to be evaluated. For the rotor, the node (from the finite element model) of the airfoil tip that experiences the maximum radial displacement through the entire flight mission was selected. Similarly for the stator, the radial position is measured at the node of the shroud segment gas-path surface that experiences the minimum growth through the same mission. This solution is conservative considering that the maximum growth of the rotor is compared to the minimum growth of the stator, which leads to the most severe closure value.

80

6.4.1 The Physics of Tip Clearance Size Variation

As seen in several references such as Lattime & Steinetz (2002), Howard & Fasching (1982) or Olsson & Martin (1982), many loads are acting on the turbine components and thus affect the tip clearance. These loads can be separated into two families: the engine loads and the flights loads. The engine loads include thermal, centrifugal, internal pressure difference and thrust loads; while flight loads include inertial, aerodynamic (external pressure) and gyroscopic loads. Some examples of these loads' effect on the engine are shown in Figure 6.4. All these loads can be either axisymmetric or asymmetric, and their effect on the engine is presented in Figure 6.5. It was demonstrated that the axisymmetric loads with the most influence on the turbine clearance's variation are the thermal and centrifugal loads (Lattime & Steinetz, 2002; Howard & Fasching, 1982; Olsson & Martin, 1982). The two first part of this work published by Moret et al. (2017) and submitted by Moret et al. (2018) develop the transient growth analysis process of the rotor and static components due to these main loads. As developed later in this paper, the main asymmetric loads are accounted by considering their 3D effects on the engine when calculating the cold build clearance.
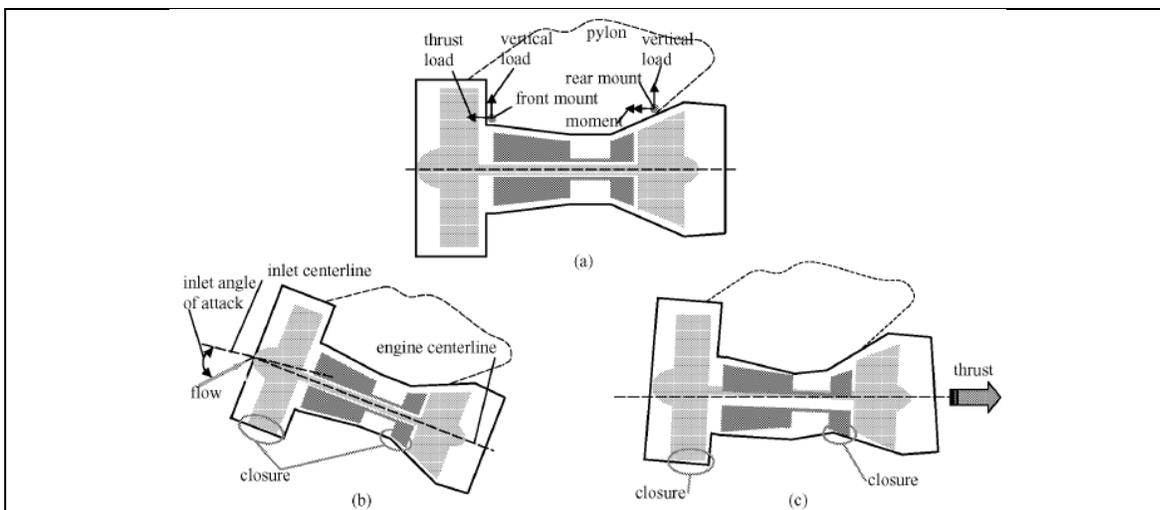

Figure 6.4: (a) Engine mounts and load paths, (b) Closures due to aero loads, (c) Closures due to thrust loads
Taken from Lattime & Steinetz (2002, p. 7)
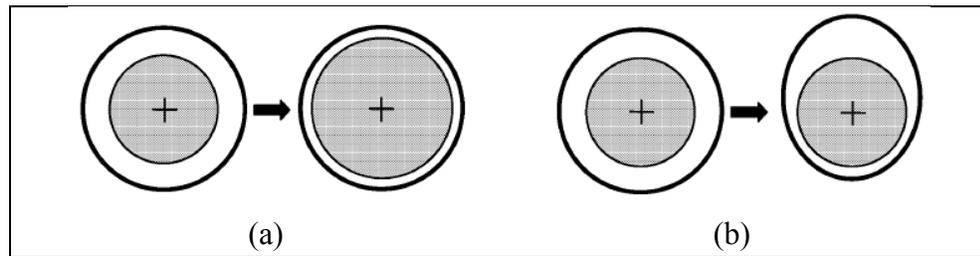
Figure 6.5: (a) Effect of axisymmetric loads, (b) Effect of asymmetric loads
Taken from Lattime & Steinetz (2002, p. 3)

6.4.2   Determination of the Cold Build Clearance

The cold build clearance is the size of the tip clearance when the engine is not running. It is set by the most severe transient engine operation in terms of loads in order to avoid any rubs between the airfoil and the shroud segment during these critical manoeuvres. For that reason, aircraft acceptance tests include flight missions susceptible of producing such loads (Lattime & Steinetz, 2002).

Figure 6.6 presents the engine rotational speed and the high pressure turbine tip clearance variation during a certain flight mission profile selected by Lattime and Steinetz (2002) for its high potential in terms of engine loads. As one can observe, two pinch points, i.e. points of maximum closure, are identified. The first one occurs during start-up, when the radial growth of the airfoil due to the thermal and centrifugal loads is maximal while the thermal expansion of the stator has just started. The second pinch point corresponds to a re-acceleration to take-off, also called re-slam, after a certain time at ground idle, called dwell time. This second pinch point can be more severe than the first one because the thermal and centrifugal expansion of the airfoil is maximal while the housing is still contracting around the rotor due to its cooling during the dwell time at ground idle. In particular, it was developed that this dwell time characterizes the severity of a re-slam manoeuvre (Howard & Fasching, 1982). Based on this, one can conclude that an "optimum" must be found for the dwell time that leads to the "worst" re-slam possible. And this worst re-slam, if leading to the most severe pinch point, allows calculating the cold build clearance. To determine this

"optimum" dwell time, iterative runs of the growth transient analyses of the rotor and housing assembly need to be executed.
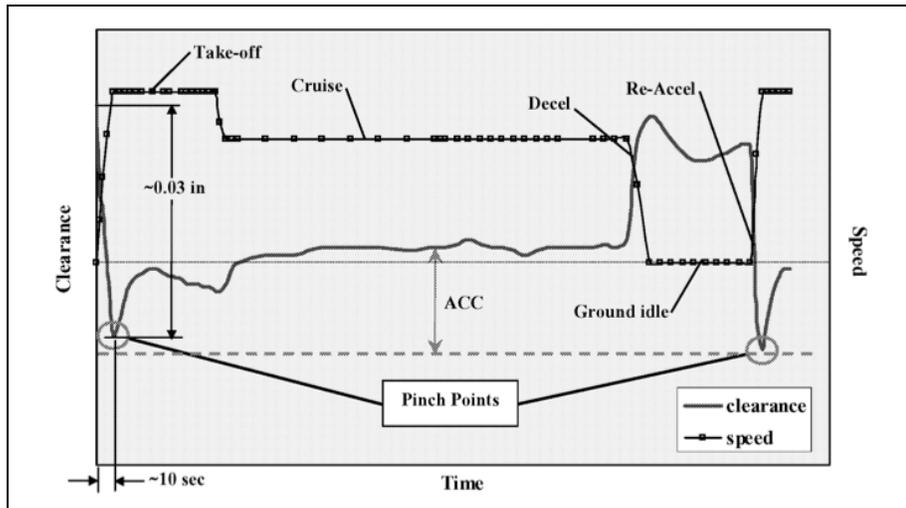
Figure 6.6: HPT tip clearance variation as function of time
Taken from Lattime & Steinetz (2002, p. 5)

Figure 6.7 shows the transient growth of the airfoil tip and of the shroud segment gas-path side, and the transient closure of the gap between them. This graph was obtained using the process described in this work for one of the test case that will be introduced later in this paper. As absolute values cannot be disclosed for proprietary reasons, the units of the Y-axis were normalized using the airfoil tip leading edge initial radius. In Figure 6.7, one can clearly observe the differences in response time of the rotor and stator for a change in throttle condition. It is these differences in response time that leads to the pinch points of critical closure.

Depending on the case, the re-slam manoeuvre may not always be the worst scenario during a flight, and this is why the entire flight mission needs to be simulated. This mission has to include specific steady-state and transient manoeuvres such as take-off, climb, cruise, shut-down in flight or re-slam, in order to capture all possible pinch points.
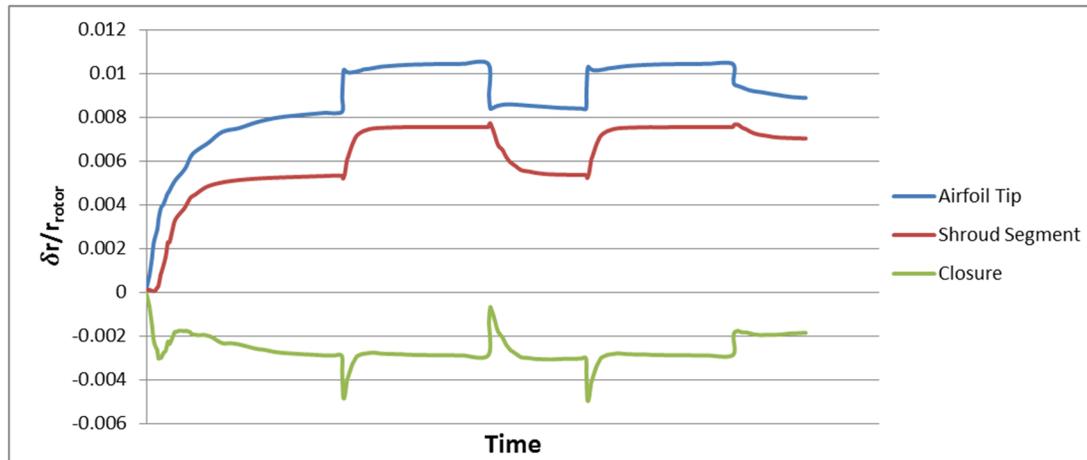
Figure 6.7: Rotor and stator transient growths and closure

Once the closure is calculated for all these manoeuvres based on the centrifugal and thermal loads, 3D effects must be added to account for the asymmetric loads (Lattime & Steinetz, 2002). Some of the main 3D effects to be considered are due to the inertia loads (gravity), aerodynamic loads, engine bending due to the thrust loads, thermal growth of the seals, or to the gyroscopic loads. These 3D effects are measured during tests on the engine for specific conditions and used as reference by new concept engines of similar build. Table 6.1 shows an example from Lattime & Steinetz (2002) where some 3D effects are added to the axisymmetric loads to get the total closure for several flight manoeuvres. Considering that it might sometimes be complicated to measure all the 3D effects' contribution separately, they can be grouped together as it is the case here with the flight load closure.

Table 6.1: Axisymmetric and asymmetric loads contribution to total closure
Taken from Lattime & Steinetz (2002, p.8)

Event	Engine load closure (in)			Flight load closure (in)	Total closure (in)
	Axisymmetric (speed & thermal)	Asymmetric			
		Thrust	Thermal		
Climb (pinch)	-0.031	-0.006	-0.005	-0.004	-0.046
In-flight restart	-0.039	-0.002	+0.001	-0.002	-0.042
Thrust reverse	-0.034	-0.008	-0.001	0.000	-0.043
Hard turn	-0.033	-0.007	-0.001	-0.005	-0.046
Airplane stall	-0.031	-0.008	+0.002	-0.007	-0.044

6.5   Results and Discussion

As introduced previously, the processes used to calculate the rotor and stator growths, which were developed by Moret et al. (2017) and Moret et al. (2018), have been integrated to calculate the gap closures during typical flight missions. The whole tip clearance calculation process being automated, it is possible to use it to run iteration loops. In the future, the objective is to use it, along with other similar systems dedicated to other disciplines, to run multidisciplinary optimization studies on the whole turbine design.

The tip clearance calculator was used on the first stage of the high pressure turbine of three different engines to validate its output and to produce the results presented in the following subchapters of this paper. Those engines are of different configurations: a turboshaft is used as test case 1, a turboprop as test case 2, and a high-bypass ratio turbofan as test case 3.

6.5.1   Dwell Time Iterations

As explained earlier, one of the most critical manoeuvre in terms of tip clearance is the re-slam after a specific duration at idle. Iterative runs need to be executed in order to find the most severe dwell time for the stage's tip clearance. These iterations aim at finding the "worst" dwell time possibly leading to the global pinch point. If the effect of the dwell time on the re-slam closure is ignored, the global pinch point might be missed as shown in Figure 6.8 (obtained for test case 2). Indeed, in Figure 6.8 the maximum closure for the first dwell time happens at the slam acceleration; but if the dwell time is increased, as for the three other curves, the maximum closure occurs at the re-slam. The closures were normalized using the airfoil tip leading edge initial radius.

Figure 6.9 was obtained by measuring the closure at re-slam for various dwell times for the three test cases. Considering that actual values of closures or dwell time cannot be disclosed due to proprietary reasons, the closures were normalized using each test case's respective maximum closure and the dwell times were normalized using the engines' respective

response time for the cooling air to pass from its value at take-off to its value at idle. In Figure 6.9 the values of closures are positives because of this normalizing of the Y-axis. The curves of the three test cases do not end at the same time due to the differences in their respective response time used to normalize the X-axis.
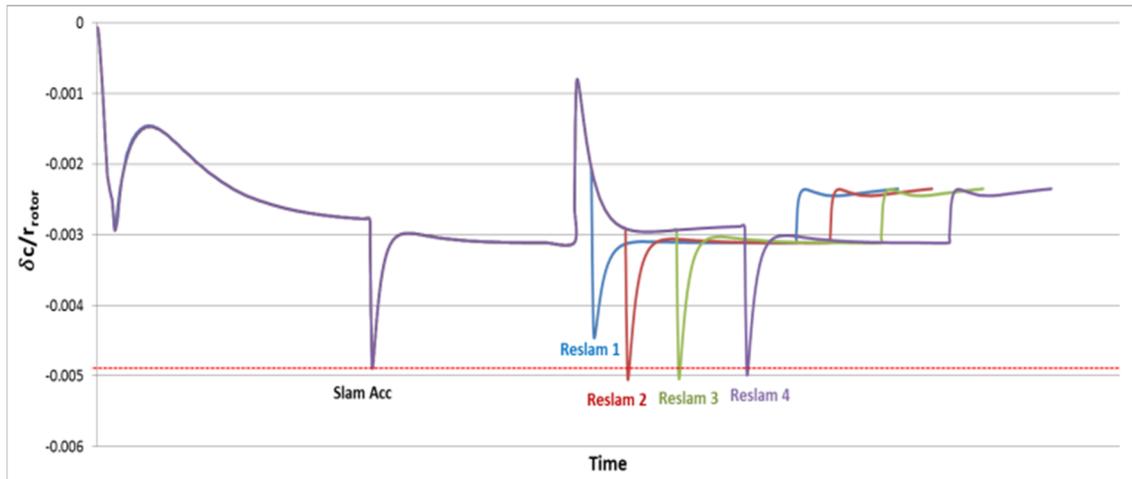
Figure 6.8: Effect of the dwell time at idle on the mission's global pinch point for test case 2
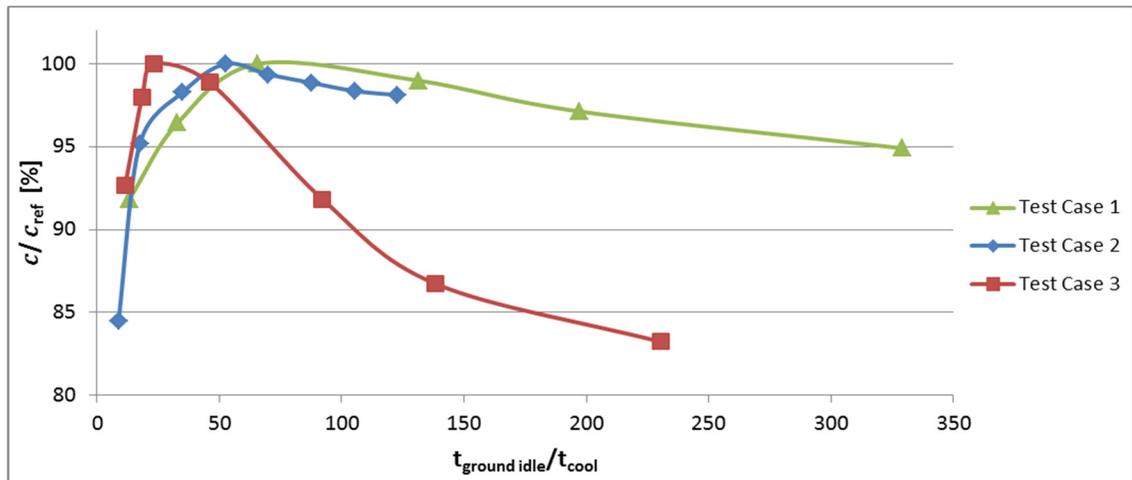

Figure 6.9: Re-slam closures depending on the dwell time

As one can see in Figure 6.9, the maximum closure does not occur at the same instant for the three test cases. However, the closure at re-slam follows the same trend for the three test cases: a non-symmetrical bell curve with a steep slope before the maximum and a flatter

slope after this point. One can conclude that the closure will always have one unique optimal value for the dwell time and that this "worst" dwell time can therefore always be identified.

6.5.2   Sensitivity Analysis

Many parameters have an impact on a turbine stage's tip clearance, but every one of them influences the tip clearance differently. However, when designing a turbine all these parameters cannot be modified to satisfy the analyst trying to optimize the tip clearance. Indeed, some parameters are set by performance requirements or by physical limitations for example. Figure 6.10, Figure 6.11 and Figure 6.12 show the difference, in percent of the reference value at re-slam of the detail design study used as target, between the closures at re-slam for different input and the reference value. The zero in the Y-axis therefore shows the detail design reference value.

Figure 6.10 shows the results of a sensitivity analysis done on the first test case for various parameters. As one can see, the airfoil size or the turbine inlet temperature have major impacts on the tip clearance. But as explained previously these parameters cannot be altered by an analyst for tip clearance optimization purpose. In the case of the airfoil size, the reason is that varying it will change the stresses the airfoil experiences and therefore its life, and will also have a major impact on its aerodynamic properties leading to a variation of the whole turbine's efficiency. For the Turbine Inlet Temperature (TIT), the reason is that it is set by the combustion chamber to meet stages performance requirements. The airfoil cooling temperature has a significant effect on the tip clearance, but similarly to the TIT it cannot be modified to optimize the tip clearance. Indeed, the cooling temperature is controlled by the compressor stage at which the air is extracted. The disc bore width on the other hand has a limited impact on the tip clearance. This is expected considering that it is the bottom of the disc, which therefore does not grow much during a mission. However, this parameter (as well as the other parameters of the disc design) is controlled by the rotor lifing requirements.

This discussion leads to the question of identifying the parameters that can be modified by a tip clearance optimization program and that would lead to significant impact. Based on several previous design studies, these parameters are:

- The cooling scheme of the housing and shroud segment assembly,
- The total amount of cooling flow injected in the housing and shroud segment assembly,
- The thickness of the housing,
- The housing material.

One can notice that these parameters are all affecting the static components, and that none impact the rotor. The reason is that the tip clearance prediction falls under the responsibility of the analysts designing the housing and shroud segment assembly. One of the main reasons justifying this is that, as it was introduced here above, the rotor's design is affected by more constrains (centrifugal stresses, vibrations, aerodynamic of the airfoil, internal cooling of the blade, etc.) than the static components. The rotor being one of the most critical parts of the turbine, it is logical not to add an extra constraint on the rotor side, but to control the tip clearance through the static components' design.

The cooling scheme can hardly be optimized by a program as it requires an analyst's input to select where to put holes to allow cooling flows to affect certain areas. A program can however study the impact of varying the amount of certain flows or varying the total amount flow used to cool down a stage to see their impact on the tip clearance. As one can see in Figure 6.10, Figure 6.11 and Figure 6.12, the amount of flow used to impinge the shroud segment has a significant impact on the tip clearance. As it was demonstrated by Moret et al. (2018), increasing the impingement flow leads to a colder average temperature of the shroud segment and the housing. Having the temperature of the housing closer to the cooling temperature means a slower response of the whole stator, and therefore an improvement of the closure as the motions of the rotor and stator tend to be more in phase. This is explained by the fact that the disc has a large thermal inertia, and is usually slower to respond than the

housing. Slowing down the housing therefore reduces the difference in response time between the rotor and stator.

As can be seen in Figure 6.10, Figure 6.11 and Figure 6.12, the total amount of cooling flow has an effect on the tip clearance which is very similar to the one of the impingement flow. This is explained by the fact that impingement cooling is the most efficient way to control the stator's temperature. And it comes from this that the effect observed in Figure 6.10, Figure 6.11 and Figure 6.12 of varying the total cooling flow mainly comes from its direct effect on the impingement flow. The impingement flow is of course directly dependent on the total amount of flow available for a stator stage. Both the impingement flow and total amount of flow used have their lower limit set to nearly 0 and are limited to a maximum value by engine efficiency requirements. Indeed, the cooling air is extracted at the compressor and therefore corresponds to a loss from the entire engine point of view. As it is shown in Moret et al. (2018), the effect of the impingement flow on the housing temperature (and therefore on its radial growth) does not change significantly passed a certain value. This means that limiting the study of the effect of the impingement or total cooling flows to a maximum value does not impact much the results shown in Figure 6.10, Figure 6.11 and Figure 6.12.

The housing thickness is limited to a minimum value set by containment (i.e. structural integrity) requirements, and to a maximum value set by the weight of the engine. One can observe that, in the tolerable range within which the thickness can vary, this parameter has a limited influence on the tip clearance.

The materials considered for this study vary mainly in their thermal expansion coefficient. Three commonly used materials were compared and their impact on the tip clearance is shown in Figure 6.10, Figure 6.11 and Figure 6.12. As can be observed, these have the largest influence on the tip clearance when compared to the other parameters that a tip clearance optimizer should be allowed to modify. For the first test case, the reference material leads to the lowest re-slam closure value as its thermal expansion coefficient is the highest of the three materials considered. This explains why there is no value under 0 for test

case 1. For the second and third test cases it is the opposite: the reference is the material having the smallest thermal expansion coefficient leading to the highest re-slam closure; which explains why there are no values above 0.
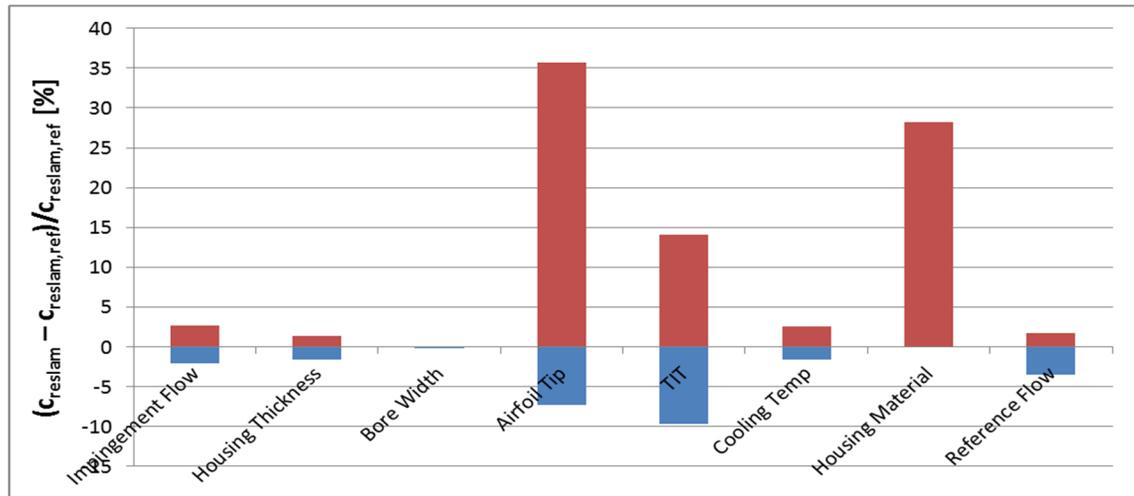

Figure 6.10: Sensitivity analysis of various parameters for test case 1
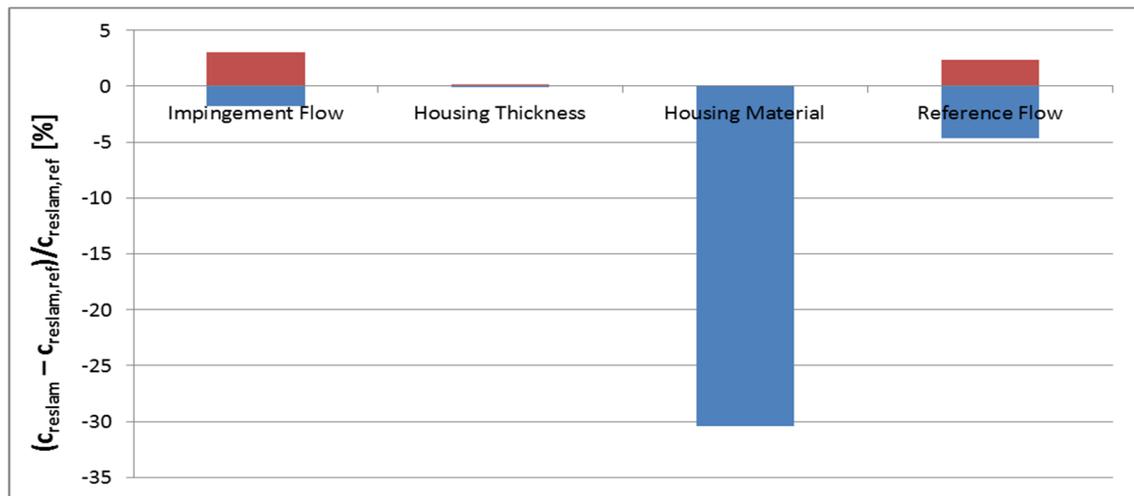
Figure 6.11: Sensitivity analysis of various parameters for test case 2
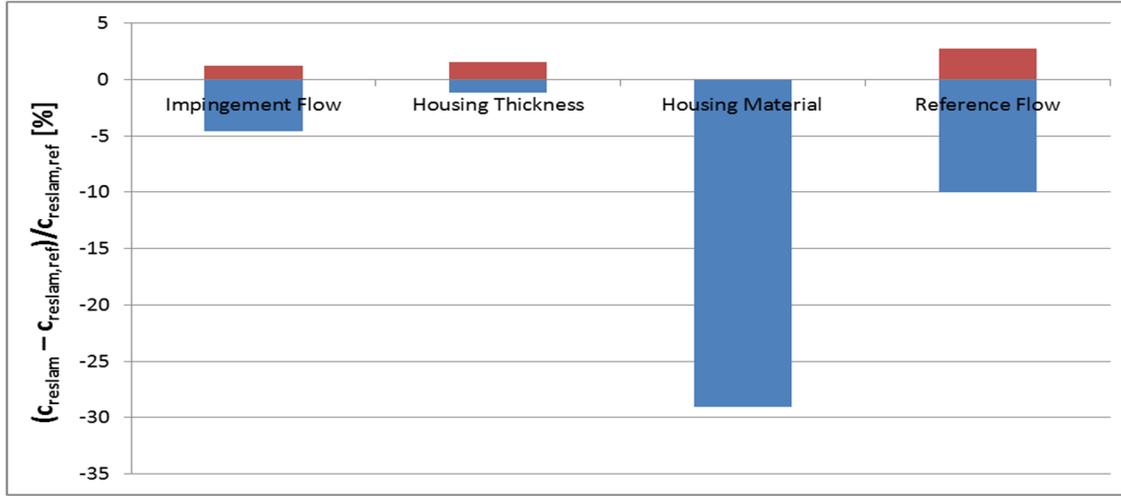
Figure 6.12: Sensitivity analysis of various parameters for test case 3

6.5.3 Speed and Accuracy Improvements

In a regular process, analysts have to manually run thermal and stress analyses on the rotor and stator in order to get the tip clearance closure. Moreover, these analyses must be executed iteratively in order to find the "worst" dwell time. This can be a long procedure considering the time required to set up and run these analyses. It was demonstrated that the proposed system considerably reduces the time required to run these analyses (Moret et al., 2017; Moret et al., 2018). By executing the same cold build clearance study using the industry current process and using the process proposed in this work, it was found that the proposed system takes 25 % of the time that is needed with the current process. Moreover, this system is executed on the click of a button where the regular process requires manual modification of input files which in turn could lead to human error.

As mentioned in the Methodology, the objective of this work was to improve as much as possible the estimated 30 % accuracy of the current pre-detailed design process. The accuracies described in this section where calculated using Equation (6.2).

$$Accuracy = \frac{(s_{cold} - s_{cold,ref})}{s_{cold,ref}} \tag{6.2}$$

When comparing the cold build clearance calculated by this work with the final design value for the three test cases using Equation (6.2), it is found that the tip clearance calculator delivers output with an average accuracy of about 6 % and with a standard deviation of 1.93%, as shown in Table 6.2. This means that by using this work's proposed process, results can be produced with a considerably improved accuracy compared to the industry current process. This accuracy of about 6 % is indeed a great improvement that outshines the currently accepted pre-detailed accuracy of 30 %.

Table 6.2 : Cold build clearance's accuracies per test case

	Test case 1	Test case 2	Test case 3
Accuracy	6.03	3.74	8.47
Average	**6.08**		

Moreover, it is accepted that at a detailed design level the cold build clearance required to avoid rubs during the engine operation is predicted with an accuracy of 0.2 % of the blade height. For the three test cases considered in this paper, this 0.2 % of the blade height is equal to an average difference between the accepted prediction and the final designs of about 11 % as shown by Equation (6.3). One can therefore conclude that this work's system produces results with an accuracy level that is not only acceptable at a pre-detailed design stage, but also has potential for detailed design standards.

$$Eq.\,(6.2) \Rightarrow \frac{\left(\left(s_{cold,ref} \pm 0.2\,\%\,h_{blade}\right) - s_{cold,ref}\right)}{s_{cold,ref}}$$

$$\Leftrightarrow \frac{0.2\,\% * h_{blade}}{s_{cold,ref}} \approx 11\,\%$$

(6.3)

6.6   Conclusion

It was demonstrated that the proposed system can automatically determine the "worst" dwell time at idle prior to a reacceleration to a take-off condition. Obtaining this time is important

in the identification of a flight mission's maximum closure which is a mandatory step in the calculation of the turbine stage's cold build clearance. It was also showed that the closure at the re-slam manoeuvre for increasing dwell times follows a bell-shaped curve with one unique maximum meaning that the optimum dwell time can always be found.

The influence of various parameters on the tip clearance was studied through a sensitivity analysis. It was however found that only a limited number of parameters that have an impact on the tip clearance can actually be modified by the analyst in charge of optimizing the tip clearance. Prior to running the sensitivity analyses on three test cases, an informed choice was made in order to limit the number of the studied parameters to the ones that could be useful to the study. These identified parameters are, by order of importance, the housing material, the total amount of flow used to cool the stator, closely followed by the impingement flow, and finally the housing thickness. This last parameter has a minor influence on the tip clearance because it is limited by containment and mass requirements. This targeted sensitivity analysis allowed giving guidelines for analysts by identifying the parameters worth being studied during a tip clearance analysis.

The study of the dwell time and the sensitivity analyses also showed the optimization capability of this system. These studies indeed highlighted the possibility of using the tools developed in this work to run iterative loops for a set of design variables. This process is therefore ready to be integrated in an optimizer along with other systems calculating more turbine output (efficiency, mass, life, etc.), and to run PMDO studies.

It is expected from the implementation of a PMDO system to improve the quality of pre-detailed level results considering that more knowledge is being injected early in the design process. It is also anticipated that such system would lead to an increase of productivity through the reduction of the overall design time. In this work it was proved that, when using a PMDO system, the time required to deliver a pre-detailed prediction of the tip clearance is reduced by 75 % due to improvements in the process efficiency and to the integration of all the sub-systems required to run this study.

When compared to a current preliminary design phase process, it was found that the proposed system delivers results with an accuracy improved by 80 %. The pre-detailed results are indeed passing from an accepted 30 % accuracy to an average of about 6 %, therefore exceeding the expectations set for this work. Finally, it was demonstrated that the accuracy of the proposed process is not only better than what was initially anticipated for such pre-detailed design system, but that it also passed the accuracy requirements for detail design results of about 11 %.

CONCLUSION

Objective 1

The present research showed that an automation of the preliminary tip clearance prediction process is possible. Moreover, by requiring fewer user inputs and removing the non-value added task for an engineer to manage data, this system decreases the risk of human errors while entirely leaving the important decisions to the user. Furthermore, the process described in this work closes the gap between the pre-detailed and detailed design phases meaning that results can be generated where assumptions had to be made before. Finally, many configurations of rotors and stators can be studied with the process proposed in this work, where only a limited number of predefined configurations could be analysed using the industry's current process. All these points are valid indications that the process proposed in this work is indeed more flexible, robust and user-friendly than the industry current process.

As mentioned before, it was proven that the set of tools implemented in this thesis reduces some non-value added tasks normally required from an analyst. As an example, an iterative process was developed to identify the "worst" dwell time by running several times the same analyses for different dwell times in order to find the worst tip clearance pinch point. This mandatory process is now fully automated and executed on the click of a button.

The targeted sensitivity analysis executed using the tip clearance calculator allowed giving guidelines to the analysts by identifying the parameters worth being studied during a tip clearance analysis.

Objective 2

It was proven that the proposed process achieved a considerable improvement in the quality of the pre-detailed design results. Indeed, the rotor's analysis module obtained an average difference of about 3 % when compared to detailed design results, as opposed to an estimated difference of 30 % for the current pre-detailed process. Similarly, the stator analysis module reached an average difference of less than 8 % when compared to detailed design results, as

opposed to the same estimated difference of about 30 % for the current pre-detailed process. In the case of the stator, the accuracy of the results was further improved during a second iteration on the pre-detailed model by adding some extra features to the model, which led to an average difference of less than 4 %. Moreover, it was proven that the tip clearance calculation overall process delivers results with an accuracy improved by 80 % when compared to a current preliminary design phase process. The pre-detailed results are indeed passing from an accepted 30 % difference to an average of about 6 %, therefore exceeding the expectations set for this work. Finally, it was demonstrated that the accuracy of the proposed process is not only better than what was initially anticipated for such pre-detailed design system, but that it also passed the accuracy requirements for detail design results of about 11 %.

Objective 3

It was demonstrated in this work that, using the proposed turbine design and analysis process at a pre-detailed design phase, the time required to obtain the rotor's and stator's temperatures and displacements was reduced by 80 % compared to the regular process. It was also established that, through the integration of this faster analyses processes and through the improvement of the tip clearance calculation process, the time required to deliver predictions of the cold build clearance was reduced by 75 %.

In addition, it was estimated by specialists that the higher quality and quantity of the pre-detailed data produced by the process described in this work would lead to a time reduction of about 30 % during the detailed design phase.

Conclusion

In conclusion, one can state that the objectives of this thesis were not only achieved but exceeded in most cases. The initial requirements gathered from the industry to allow validating the accomplishment of the present work were all met using the proposed solution, and some extra results were even produced. As an example one can name the highlighting of the contribution of thermals on the overall growth of the rotor, the identification of the major

cooling means of a stator, or the obtaining of the curve of the re-slam closure for increasing dwell time proving that one unique "worst" dwell time can always be determined.

RECOMMENDATIONS

Even though this research project was successfully completed, some more work might be required for it to reach its full potential. Here are a few recommendations:

1. To improve the stator analyses accuracy, one might consider including the modifications made on the thermal boundary conditions during the second iteration of the analyses as discussed in CHAPTER 5.

2. It could be interesting to add some new configurations to the designer modules presented in Figure 3.1. An example of new configuration for the rotor is the addition of a mini cover-plate component, and for the stator allowing the creation of a double layer housing should be considered. The rotor and stator analysis process would then need to be updated.

3. Integrate a tip clearance scaler in order to get a quick idea of the tip clearance prior to any analysis or for flight conditions that were not simulated during the tip clearance analysis. Even though the process is now much faster than a regular pre-detailed study, it still takes a certain amount of time to get a first pass of results for the tip clearance prediction. Getting a first idea of the tip clearance value without having to run any CAE analysis could be useful.

4. The process developed in this work was validated by trying to match detailed design studies and comparing it with the pre-detailed design results that were originally produced for the same study. It is necessary to use this new process to run actual pre-detailed studies (i.e. not based on legacy studies) in order to validate in real life the results raised in this work by measuring the impact of this work on the pre-detailed and detailed design phases.

5. Finally, in order to reach the full potential of the PMDO system developed by the combined initiative of Pratt & Whitney Canada and the École de Technologie Supérieure, the set of tools produced during this Ph.D. study should be used along with all the other required tools to run optimisation loops on the whole turbine design and analysis process.

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