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Technical Notes

GasProp: Reducing Blade Compressibility Effects Through Waste-Heat Recovery in Aircraft Engines

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I. Introduction

GAS-TURBINE aircraft engines use a gas turbine to produce high-pressure hot gasses. The available internal energy of the hot gasses is then converted into jet kinetic energy through nozzles or extracted from a turbine to produce thrust by means of fans and/or propellers. Propulsive-efficiency requirements have resulted in the continuous increase of engine bypass/core ratios experienced by modern high-bypass-ratio turbofans, turboprops, and unducted fans or propfans [1]; whereas the improvement of the core-compression ratios allows to obtain thermodynamic efficiencies higher than 0.6.

Propeller/fan blades act as rotating wings (see, e.g., Ref. [2]), accelerating the air flow in the flight direction by means of aerodynamic interaction. In their proximity, they further accelerate the flow because of their induced pressure field. This combined effect leads to local Mach number values equal to or higher than one, even for helical Mach numbers (given by the combination of blade tangential velocity and airplane flying speed) of $M_{\mathrm{he}} < 1$. In the early transonic regime, shock waves start occurring. At an even higher flow velocity, after reaching the drag-divergence Mach number M^{DD}, the shock pressure jump causes shock-induced boundary-layer separation, which eventually leads to reduced propulsive efficiency [3], increased aeroacoustic emissions [4,5] and enhanced structural fatigue related to buffet and aeroelastic instability [6]. To lower these effects, the propeller/fan power loading is limited, reducing the shaft angular velocity or the aerodynamic load of the blades. As a consequence, the blade-element design is adjusted by limiting either the pitch, the camber, or the thickness. The propeller/fan radius is also constrained to limit the blade-section tangential velocity. All these solutions cap the total thrust exerted by the propeller/fan. Finally, the aircraft flight speed is also restrained (for turboprops, to cruise Mach numbers of approximately $M_\infty = 0.6$) with all the consequences that this has on propeller-aircraft performances (see, e.g., Ref. [4]). To mitigate these effects, unducted fans or propfans use significantly swept blades and supercritical airfoils in order to operate at flight Mach numbers comparable to those of traditional ducted turbofans [4]. At the rotor hub [6,7], compressibility effects are caused by the larger blade section's thickness and camber, as well as by the higher solidity and related flow blockage. These effects at the blade's hub can be mitigated with ad-hoc engine nacelles designed to locally decelerate the flow facing the blades (see, e.g., Refs. [4,6,7]).

II. *GasProp*

The present work proposes an engine modification, named *GasProp*, which uses the waste heat of the engine-core thermodynamic cycle to increase the temperature and the speed of sound of the flow ingested by the propeller/fan, therefore reducing the mentioned compressibility effects. This potentially enables engines with improved aerodynamic, structural, and aeroacoustic performances; or it enables them to operate at higher power loading or at higher M_{he}. Turboprop/unducted-fan engines in pushing configurations (see Fig. 1) are here considered for application of the proposed concept because this appears as the most straightforward and efficient implementation: the propelling nozzle of a common turboprop (see Fig. 2a) is substituted with an annular duct around the propeller axis, which is designed to guarantee the optimal mixing of the hot gasses with the incoming air during cruise flight (see also Ref. [8], where a similar modification, although related to different flow mechanisms, was presented). Two configurations are considered in which the treated propeller sections are the blade root and tip regions (Fig. 2b; top and bottom, respectively), where the flow-compressibility effects are more pronounced. The *GasProp* concept for the blade root sections can be obtained as a rather simple circular slit around the engine external casing, which is designed to direct the hot gasses upstream of the blade roots (see Figs. 1 and 2b, bottom); whereas for the blade-tip region, *GasProp* is deemed of more cumbersome implementation: in this case, the hot-gasses duct needs a shroud of large diameter (Fig. 2b, top) that would cause a drag augmentation due to its skin-friction contribution and to its higher weight leading to increased induced drag.

A. Thermodynamic Description

For turboprops, considering an ideal Brayton cycle, the waste-heat thermal power $\dot{Q}^{\mathrm{WH}}$ for an engine providing a power output $\dot{W}$ and having a thermodynamic efficiency of $\eta_t = 1 - T_\infty/T_{\mathrm{tc}}$ (where T_∞ is the static temperature at ambient conditions, and T_{tc} is the stagnation temperature at the compressor exit) can be expressed as follows:

$$\dot{Q}^{\mathrm{WH}} = \frac{1-\eta_t}{\eta_t}\dot{W} = \dot{m}_0 c_p (T_e - T_\infty) = \dot{m}_0 c_p (T_{\mathrm{tb}} - T_{\mathrm{tc}})(1-\eta_t) \quad (1)$$

where $\dot{m}_0$ is the core mass flow rate, c_p is the isobaric specific heat capacity, T_e is the gas static temperature at the nozzle exit, and T_{tb} the stagnation temperature after the combustion (here, it is assumed that the nozzle flow is fully expanded down to ambient pressure p_∞; i.e., we consider outlet exhaust gasses). As shown in the second equality in Eq. (1), this thermal power is equal to the fraction of heat input, which cannot be converted into work due to the thermodynamic cycle efficiency. If the exit velocity is taken equal to the freestream velocity, therefore not considering the nozzle contribution to thrust, the stagnation temperature variation of $T_{\mathrm{te}} - T_{t\infty}$ is equal to the static temperature (referred without the subscript t) variation of $T_e - T_\infty$. The propeller/fan rotor area exposed to the mixed gasses peculiar to the *GasProp* implementation (thus denoted with the superscript GP) can be expressed as the sum of the two former quantities:

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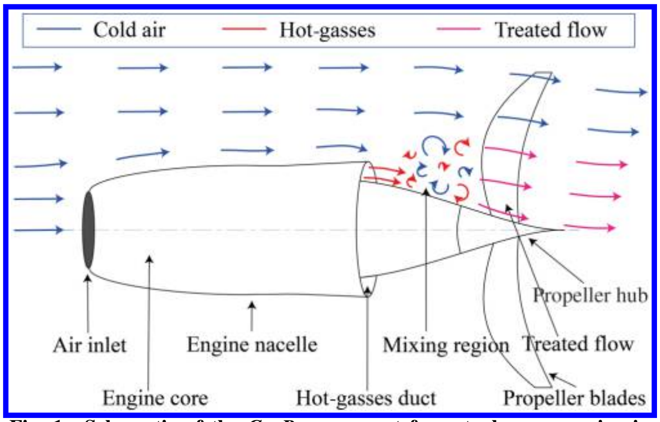

Fig. 1 Schematic of the *GasProp* concept for a turboprop engine in pushing configuration and qualitative thermofluid dynamic evolution of the flow around it.

$A^{\mathrm{GP}} = A_{\mathrm{co}} + A_e$. Following Fig. 2b, the exhaust gasses and the treated cold-flow streamtube cross sections can be parametrized as functions of the inner radius r_i and the thickness δ of the hot-gasses duct, as well as of the thickness of the cold circular streamtube δ_{co}.

The effect of the mixing on the thermodynamic parameters at the propeller/fan can be evaluated by considering mass and energy conservation through the mixing process under the simplified condition of equal freestream and hot-gasses velocities and neglecting, in a time-averaged perspective, secondary flows through the streamtubes interfaces. The actual freestream flow velocity to consider should account for the fan/propeller induction (the acceleration exerted upstream of the propeller/fan rotor) for the local acceleration/deceleration caused by the engine nacelle geometry [7] and for the mixing process effects. Yet, as a first estimate, the airplane flight velocity V_∞ can be considered, given its larger contribution, and the other effects are discarded. Under these hypotheses, the velocity in the streamtube affected by the mixing between core and ambient flows is not expected to change. From the conservation of mass, it is possible to estimate the treated mass flow rate reaching the propeller/fan $\dot{m}^{\mathrm{GP}}$ as the sum of the core mass flow rate of $\dot{m}_0 = \rho_e V_\infty A_e$ and the cold treated streamtube mass flow rate of $\dot{m}_{\mathrm{co}} = \rho_\infty V_\infty A_{\mathrm{co}}$.

Accordingly, the air density of the treated flow seen by the propeller/fan is equal to the area-weighted density of the mixed flows; whereas due to the conservation of total enthalpy, the stagnation temperature of the treated flow at the propeller/fan T_t^{GP} is the mass-weighted average between the core and the ambient mass flow rate stagnation temperatures: T_{te} and $T_{t\infty}$, respectively. The stagnation temperatures of the flow seen by the blades are equal to $T_{t\infty}$ and T_t^{GP}, respectively, without and with the *GasProp*; whereas, we neglect the effects of the *GasProp* operation on the local velocity. The local static temperature at any point can be obtained from isentropic relations, given the local value of the Mach number. Considering all the contributions to the flow acceleration, the thermodynamic quantities evaluated under these conditions (maximal flow acceleration) are herein labeled with a tilde. Employing the conservation of total enthalpy and the expressions of the isobaric and isochore heat capacities as well as that of the speed of sound, we can express the thermal power needed to achieve a certain Mach number variation (and therefore useful for the *GasProp*) as

$$\begin{aligned}\dot{Q}^{\mathrm{GP}} &= \dot{m}^{\mathrm{GP}} c_p (T_t^{\mathrm{GP}} - T_{t\infty}) \\ &= \dot{m}^{\mathrm{GP}} c_p \left[\frac{V_\infty^2}{\gamma R \tilde{M}^{\mathrm{GP}^2}} + \frac{V_\infty^2 0.5(\gamma-1)}{\gamma R} - \frac{V_\infty^2}{\gamma R \tilde{M}^2} - \frac{V_\infty^2 0.5(\gamma-1)}{\gamma R} \right] \\ &= \dot{m}^{\mathrm{GP}} \frac{V_\infty^2}{\gamma - 1} \left(\frac{\tilde{M}^2 - \tilde{M}^{\mathrm{GP}^2}}{\tilde{M}^2 \tilde{M}^{\mathrm{GP}^2}} \right)\end{aligned} \tag{2}$$

B. Feasibility Analysis

Consider as an example the case of the Piaggio Aerospace Avanti EVO. It is propelled by two Pratt and Whitney Canada PT6A-66B turboprop engines in pushing configuration, with each spinning a five-bladed propeller with a diameter of $D = 2r_t = 85'' \approx 2.16$ m. Each engine produces 950 shaft horsepower (SHP) and 1010 equivalent shaft horsepower (ESHP), which are given by the sum of the SHP and of the (estimated) power-equivalent thrust from the propelling nozzle. Therefore, the contribution to power of the nozzle accounts for 60 ESHP. Following the *GasProp* concept, all the available energy is transferred to the power turbine driving the propeller. This brings the available SHP up to $\dot{W}^{\mathrm{SHP}} = 1010$ SHP $= 743$ kW. We consider cruise conditions at an altitude of $H = 11{,}000$ m and, from the standard atmosphere model, an ambient temperature of $T_\infty = 216.65$ K (air temperature), a speed of sound of $a_\infty = 295$ m/s, a density of $\rho_\infty = 0.36$ kg/m^3, and a isobaric specific heat capacity of $c_p = 1.004$ kJ/(kg/K) (for simplicity, this is considered constant). Following the Piaggio Aerospace Avanti EVO datasheet, we consider a cruise velocity of $V_\infty = 740$ km/h $= 206$ m/s, and thus a cruise Mach number of $M_\infty = 0.6$. Assuming that the gas generator is designed for maximum thrust (i.e., the compressor ratio is optimal) and that the turbine-entry stagnation temperature is $T_{\mathrm{tb}} = 1600$ K, the compressor exit stagnation temperature is $T_{\mathrm{tc}} = \sqrt{T_\infty T_{\mathrm{tb}}} = 588$ K [9], resulting in a thermal efficiency of $\eta_t = 0.632$. The waste-heat thermal power available for the *GasProp* can be evaluated by means of Eq. (1) and is equal to $\dot{Q}^{\mathrm{WH}} = 432.6$ kW.

It is appropriate to assume that the flow acceleration caused by the propeller induction, by the nacelle pressure field and by the blades' sections (including eventual blockage effects between neighboring blades), pushes the flowfield highest Mach number $\tilde{M}$ around the blades up to a value such that the blade elements are operated in the transonic regime, e.g., $\tilde{M} = 1.1$. We now consider the

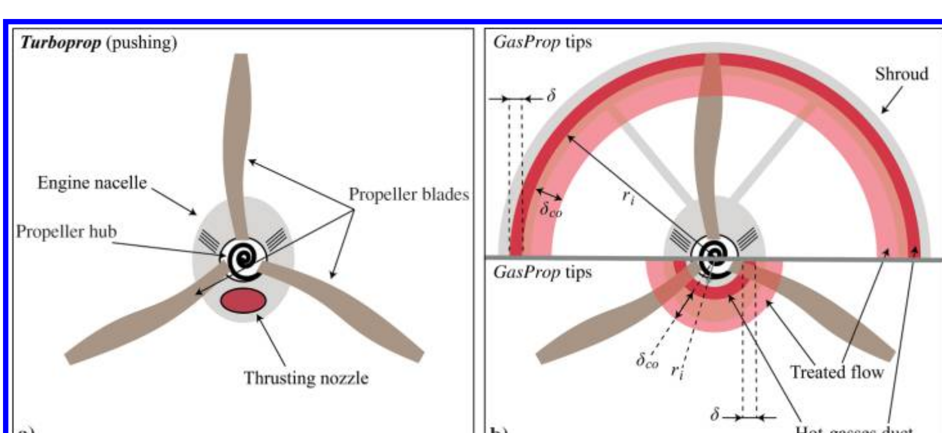

Fig. 2 Qualitative schematic of a common turboprop in a pushing configuration (Fig. 2a) and of the *GasProp* concept applied to the blades tip (Fig. 2b, top) and root (Fig. 2b, bottom) region. All views are from downstream.

implementation of the *GasProp* for the blades' root region, and we aim to treat a limited portion of the blades: ≈0.15 propeller radii (0.162 m). From the engine-core parameters, the core mass flow rate $\dot{m}_0$ can be roughly estimated as the available power ESHP divided by η_t, by c_p, and by the temperature difference through the burner ($T_{\mathrm{tb}} - T_{\mathrm{tc}}$), which is equal to about 1.16 kg/s. If all the available energy is provided to the turbine, the exit temperature will be equal to $T_e = T + (1 - \eta_t)(T_{\mathrm{tb}} - T_{\mathrm{tc}}) = 589$ K. For pressure-matched exit conditions, the exit flow density becomes $\rho_e = 0.134\ \mathrm{kg/m^3}$ and the area A_e should be equal to approximately $0.042\ \mathrm{m^2}$. Considering a hub radius of $r_i = 0.15$ m, this would correspond to $\delta = 3.9$ cm. If we consider the objective of heating a total annular front area A^{GP} with $\delta_{\mathrm{co}} = 12.3$ cm such that $\delta + \delta_{\mathrm{co}} = 16.2$ cm (see Fig. 2b), we retrieve $\dot{m}_{\mathrm{co}} = 14.35$ kg/s, and therefore $\dot{m}^{\mathrm{GP}} = 15.50$ kg/s. For the considered flow case, setting $\tilde{M}^{\mathrm{GP}} = 0.95$ (a desired subsonic value of the highest local Mach number after the mixing of the hot gasses with the cold air to guarantee the $M_{\mathrm{he}} < M^{\mathrm{DD}}$ condition), the thermal power results from Eq. (2) in $\dot{Q}^{\mathrm{GP}} \approx 463$ kW. Given the available thermal power of $\dot{Q}^{\mathrm{WH}} \approx 432.6$ kW, the *GasProp* concept could be efficiently applied to reduce the local flow Mach number of $\Delta\tilde{M}^{\mathrm{GP}} = \tilde{M} - \tilde{M}^{\mathrm{GP}} = 0.15$ to a rotor-hub region with a thickness slightly smaller than 0.15 radii (considering 0.1434 radii will lead to $\dot{Q}^{\mathrm{GP}} = \dot{Q}^{\mathrm{WH}} \approx 432.6$ kW). It has to be remarked here, however, that the effectiveness of the mixing process was not considered in this preliminary analysis and that several assumptions were made.

The application of the *GasProp* concept to the blade-tip region is instead deemed not feasible because of energetic reasons: considering the same 0.162-m-wide $\delta + \delta_{\mathrm{co}}$ slit but with an inner radius of 0.918 m (and considering only the mixing of the outer hot streamtube with the inner cold one as shown in the top of Fig. 2b), the required thermal power would be $\dot{Q}^{\mathrm{GP}} \approx 1.87$ MW (given $\dot{m}^{\mathrm{GP}} = 62.48$ kg/s).

C. Discussion

The beneficial effect of the *GasProp* on the propeller/fan aerodynamic performance can be evaluated through the classical blade-element momentum (BEM) theory [2]. The propeller/fan propulsive efficiency $\eta_p = (FV_\infty/P)$, where F is the exerted thrust and P is the shaft power. These two quantities are obtained, according to BEM theory, as the radial integral between the hub radius r_h and the tip radius r_t of the thrust $f(r)$ and power (i.e., torque $q(r)$ times rotational speed) contributions generated by the individual blade elements times the number of blades N. Thrust and power are computed by projecting the blade-element lift $l(r)$ and drag $d(r)$ on the rotor disk axial and tangential directions; they are, in general, varying along the radial coordinate to optimize the propeller performance. Accordingly, it is possible to symbolically estimate the effects of variations of the lift $\Delta l^{\mathrm{GP}}(r)$ and drag $\Delta d^{\mathrm{GP}}(r)$ distributions along the rotor disk caused by the operation of the *GasProp* on the resulting thrust F^{GP} and shaft power P^{GP} as

$$\begin{aligned} F^{\mathrm{GP}} &= F + \Delta F^{\mathrm{GP}} \\ &= N\bigg[\int_{r_h}^{r_t}[l(r)C(\phi(r)) - d(r)S(\phi(r))]\,\mathrm{d}r \\ &\quad + \int_{r_h}^{r_t}[\Delta l^{\mathrm{GP}}(r)C(\phi(r)) - \Delta d^{\mathrm{GP}}(r)S(\phi(r))]\,\mathrm{d}r\bigg] \end{aligned} \tag{3}$$

$$\begin{aligned} P^{\mathrm{GP}} &= P + \Delta P^{\mathrm{GP}} \\ &= N\omega\bigg[\int_{r_h}^{r_t}[l(r)S(\phi(r)) + d(r)C(\phi(r))]r\,\mathrm{d}r \\ &\quad + \int_{r_h}^{r_t}[\Delta l^{\mathrm{GP}}(r)S(\phi(r)) + \Delta d^{\mathrm{GP}}(r)C(\phi(r))]r\,\mathrm{d}r\bigg] \end{aligned} \tag{4}$$

where $C(\phi)$ and $S(\phi)$ are the cosine and sine of the angle ϕ. The latter angle is given by locally subtracting the blade section's pitch angle to the incidence angle, taking into account the rotor-induction effects [2]. Note that ω is the shaft angular velocity. The main effect of reducing the Mach number for transonic applications is to achieve a reduction of the drag ($\Delta d^{\mathrm{GP}} < 0$) exerted by the treated blade elements. This is achieved because of the lower values of both the airfoils' drag coefficient and flow density. Although the lower flow density causes a reduction of the generated lift as well, the *GasProp* operation would lead to smaller effects on the exerted lift ($|\Delta l^{\mathrm{GP}}| < |\Delta d^{\mathrm{GP}}|$). Finally, increasing the flow static temperature has smaller effects on the aerodynamic performances due to variations of the fluid kinematic viscosity. Therefore, according to Eq. (4), implementing the *GasProp* on a propeller/fan operating in the transonic regime leads to $\Delta P^{\mathrm{GP}} < 0$. For some specific geometries, it might also imply a positive thrust contribution: $\Delta F^{\mathrm{GP}} > 0$ (consider the minus sign before the delta-drag term in the expression of ΔF^{GP} in Eq. (3)). From these considerations, the operation of the *GasProp* leads to $\Delta\eta_p^{\mathrm{GP}} = \eta_p^{\mathrm{GP}} - \eta_p > 0$. Finally, given the nozzle-fan/propeller propulsive-efficiency optimization [9], swapping the energy-transformation contribution from the nozzle to the free turbine driving the propeller/fan has practically no penalties on the propulsive efficiency of the engine. The largest propulsive benefit can thus be achieved from the application of the *GasProp* concept to engines in which all the available power is provided to the propeller/fan.

For transonic propeller(s) and unducted fan(s), the main contribution to the acoustic emission is of a tonal nature and comes from the shock-induced pressure jump [5]. The operation of the *GasProp* on transonic propellers, by lowering the effective Mach number at which a portion of the propeller blades is operated, can partially suppress the occurrence of shock waves. This leads to a direct reduction of the acoustic emission. Furthermore, the interaction between the shock waves and the boundary layer causes a high-frequency variation of the flow topology around the considered aerodynamic body, which leads to an oscillation of the exerted loads, which is named transonic buffet [3]. This, coupled with the complex shapes and dynamics featured by transonic blades (e.g., sweep angles with related coupling of the structural strains, shifts of the center of mass related to the flow regime, etc.), can lead to disruptive aeroelastic phenomena (e.g., flutter instability). Airworthiness regulations limit the operation in terms of the maximum lift coefficient C_l and M of transonic aerodynamic bodies, introducing safety margins (ΔC_l and ΔM) [3]. Consequently, the *GasProp* could extend the operational range of transonic propellers/fans. The envisioned *GasProp* configuration, eliminating the periodic impacting of the blades with the nozzle jet, as it is currently occurring in pushing turboprops (see Fig. 2a), can further simplify the fatigue-life design of the propeller and reduce the propeller/fan acoustic signature in the $N\omega$ frequency band.

III. Conclusions

A novel concept for aircraft engine waste-heat recovery, named *GasProp*, is introduced in this technical note. It is proposed that the high-temperature exhaust gasses are mixed with the ambient air to locally increase the speed of sound in proximity of the propeller/fan of turboprop/unducted-fan engines. The envisioned benefits are related to the attenuation of the flow-compressibility effects, which limit the engine propulsive efficiency, increase its aeroacoustic emission and complicate the blade structural design.

The application of the *GasProp* to the propeller-blade hub of turboprops in the pushing configuration appears to be the most straightforward option. Encouraging preliminary estimations of the *GasProp* feasibility are discussed herein. Likely, maximal beneficial effects would derive from ad hoc blade designs optimally performing under the *GasProp* operation. Besides the preliminary discussion on the implementation of the proposed concept, elaborated investigations are necessary and will hopefully be the object of future research.

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