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Sizing Methodology for Reaction-Driven, Stopped-Rotor Vertical Takeoff and Landing Concepts

William A. Crossley*
Arizona State University, Tempe, Arizona 85287-6106
and
John W. Rutherford†
McDonnell Douglas Helicopter Company, Mesa, Arizona 85215

Reaction-driven, stopped-rotor, vertical takeoff and landing (VTOL) aircraft concepts offer great potential for application by allowing the use of a common propulsion system for high-speed cruise and low-speed powered lift. Using the rotor for lift in both flight modes increases its utility. Existing analysis and design methodology for shaft-driven VTOL concepts, like helicopters or tilt-rotor and tilt-wing aircraft, cannot be applied to reaction-driven, stopped-rotor concepts. The methodology for sizing this type of vehicle is presented, addressing both internal gasdynamics and external aerodynamics, along with consideration of their noncomplimentary interactions. Aerodynamic compromises required to obtain flight in both fixed- and rotary-wing modes are presented. Lessons about design issues and tradeoffs learned during the development of this methodology and its implementation are discussed. Finally, the results of applying this methodology to an example of this type of vehicle, the canard rotor/wing, show its utility.

Nomenclature

A = cross-sectional area, or rotor disk area
b = number of rotor blades
C_D = discharge coefficient
C_L = planform lift coefficient
C_{L_α} = planform lift–curve slope
C_M = planform moment coefficient
C_T = rotor thrust coefficient
c = chord
$\bar{c}$ = mean aerodynamic chord
$\mathcal{D}$ = duct hydraulic diameter
F_N = force (thrust) produced by nozzle
f = Fanno friction factor
HP = horsepower
k = structural thickness requirement
l = moment arm length from aircraft c.g.
M = Mach number or total aerodynamic moment
$\dot{m}$ = mass flow rate
n = maneuver load factor
P = total (stagnation) pressure
p = static pressure
Q = torque
q = dynamic pressure, $\frac{1}{2}\rho V^2$
R = gas constant
r = radial position from rotor center of rotation
S = planform area
T = total (stagnation) temperature
t = static temperature
t/c = thickness-to-chord ratio
V = velocity
V_T = rotor tip speed
W = weight
α = angle of attack
Γ = vortex strength
γ = ratio of specific heats
δD = incremental drag component
δF_{CF} = incremental centrifugal force component
δF_f = incremental friction force component
ε = downwash angle
η = efficiency, or dynamic pressure coefficient
ρ = density
σ = rotor solidity
Ω = rotational velocity

Subscripts

a = ambient conditions
avail = available
c = canard
co = Coriolis force
con = conversion flight mode
d = rotor disk
Eng = engine
e = exit plane
F_N = nozzle thrust
gas = tip nozzle gas
h = horizontal tail
MR = main rotor
root = blade root
w = wing

Introduction

REACTION-DRIVE helicopters operate with the same principle as Hero's steam turbine or a rotating lawn sprinkler.[1] Ducting gases through the rotor blades and changing the gas momentum vector direction at the tips appropriately produces a force that applies a driving torque to the rotor. This reaction-drive system requires only a gas generator, obviating the need for a mechanical transmission which, in turn, eliminates the need for an antitorque system. The inherent simplicity attracted past designers and several companies built reaction-driven helicopters. Today, however, the

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*Doctoral Candidate, Department of Mechanical and Aerospace Engineering; currently Assistent Professor, School of Aeronautics and Astronautics, Purdue University, West Lafayette, IN. Member AIAA.

†Manager, Advanced Concept Development, 5000 E. McDowell Road. Senior Member AIAA.

shaft-driven helicopter represents the standard configuration. Despite advantages in empty weight and complexity, the power transmission efficiency from engine to main rotor of a reaction-drive rotor system falls considerably below a shaft-driven configuration. Depending on the internal losses within the duct system, Nichols[2] showed that the probable practical limit of power transmission efficiency for a reaction-drive system only reached about 50%. This implies that a reaction-drive rotor necessitates increased engine size and increased fuel consumption for the same required rotor power as the shaft-driven system; consequently, the reaction-driven helicopter is not the minimum weight solution. However, Bachmann[3] pointed out that the system became practical for very high gross weight helicopters, such as the U.S. Army's heavy lift helicopter proposed in the 1970s. Because transmission and antitorque weights of the shaft-driven vehicle continue to grow as the rotor requires more power to lift its gross weight, the payload capacity diminishes at high gross weights. The same trend in empty weight does not occur with the reaction drive since no transmission or antitorque system exists. In this case, the engines of both configurations approach the same size for a given payload, implying similar fuel consumption, but the shaft-driven vehicle's gross weight greatly exceeds that of the reaction-driven one. Therefore, practical application of reaction drive to helicopters requires a heavy lift requirement.

In an effort to improve the propulsion efficiency of reaction-drive rotors some designers attempted to eliminate duct system losses by placing small jet engines at the rotor tips. This scheme introduces its own set of problems either dynamically, acoustically, or from a fuel efficiency standpoint. Additionally, these concepts would be difficult to adapt to a stopped-rotor configuration.

Current interest in high-speed rotorcraft applications, ranging from unmanned aerial vehicles (UAV) to manned attack aircraft, proposed by several branches of the armed services, suggests potential practical applications for a reaction-drive, high-speed rotorcraft. Results of recent studies contracted by NASA[4-7] all focused on rotorcraft capable of cruise speeds of 450 kn. These studies and a separate study in Ref. 8 all indicate a severe transmission growth weight penalty for shaft-driven proprotors required to operate at these speeds. This occurs because the tip speed for cruise at high speeds requires nearly a 40% reduction from hover. Assuming the cruise power roughly equals the power required to hover (a reasonable assumption at 450 kn), the torque on the rotor increases greatly due to the reduced rpm, leading to a large weight growth in the complex drive system. The comparison in vehicle effectiveness between prop-rotor aircraft and reaction-driven, stopped-rotor aircraft at high speed then becomes analogous to the heavy lift comparison.

Canard Rotor/Wing Concept—An Example

To illustrate the utility of the methodology subsequently presented, a new high-speed rotorcraft concept called the canard rotor/wing (CRW) will serve as a model. Figure 1 shows an artist's rendition of the concept. This vehicle, described in detail in Refs. 9 and 10, provides an example of a reaction-drive, stopped-rotor concept capable of satisfying emerging high-speed rotorcraft requirements. Briefly, this concept uses gases from a turbofan engine ducted through the hub and rotor blades of a two-bladed, teetering rotor to tip jets for reaction drive. It converts from rotary- to fixed-wing flight by transferring the lift from the rotor to the aerodynamic surfaces until reaching conversion speed (about 140 kn) when the rotor is completely unloaded and stopped. Because only two blades are desired when the rotor becomes a wing in fixed-wing flight, selection of a teetering rotor minimizes vibration in rotary-wing mode while eliminating the need to pass the gas duct through a flapping hinge at the blade root. Mechanical locks control potential rotor instabilities during stopping and starting the unloaded rotor. As the rotor is

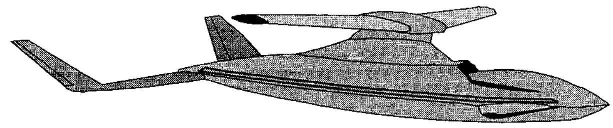

Fig. 1 CRW concept.

stopped, gases from the turbofan engines are diverted from the rotor to conventional cruise nozzles providing thrust as a conventional jet aircraft. No circulation control is used; the rotor consists of blunt leading- and trailing-edge sections and control of the rotor in rotary-wing mode occurs through conventional collective and cyclic feathering. In fixed-wing mode, the aircraft trims to share lift efficiently on all three aerodynamic surfaces.

Existing Design Methodology

Existing rotary-wing design methodology resides in several design codes, typically proprietary in nature, but universally used for sizing shaft-driven concepts. Representative of these codes are HESCOMP[11] and VASCOMP II[12] developed for NASA by Boeing Vertol in the 1970s. The first code contains the methodology to size a variety of conventional helicopters, whereas the second comprises the methodology to size a variety of "convertiplane"-type rotorcraft concepts. Neither of these contain the methodology necessary for the design of reaction-drive or stopped-rotor concepts. Work done by Hughes in the 1960s resulted in design methodology development oriented primarily toward reaction-drive helicopters.[13] This effort culminated in the XV-9A hot-cycle research helicopter. The renewed interest in missions for high-speed rotorcraft provides a potential application for the reaction-drive, stopped-rotor concept. Unfortunately, current rotorcraft design methodology cannot size this type of vehicle and much of the experience previously gained in this area is gone. Therefore, the reaction-drive, stopped-rotor concept does not receive adequate consideration for application to evolving missions largely because the design methodology to size it is not adequately understood or documented.

Aerodynamic Sizing Methodology

The most efficient aircraft concepts that combine attributes of rotary- and fixed-wing flight emphasize multiple use of systems throughout both flight modes. The propulsion and powered-lift systems typically require considerable design compromises due to conflicting requirements of both flight modes. A reaction-drive, stopped-rotor concept is no exception. In this case the requirements for powering the rotor influence the external aerodynamics more significantly than in shaft-driven counterparts. Including the proper relationships and effects into the sizing methodology requires a thorough understanding of the design problem. Equation (1) can be used to illustrate the difference in sizing a reaction-driven rotor compared to a shaft-driven one:

$$HP_{\mathrm{MR}} = \eta HP_{\mathrm{Eng}} \tag{1}$$

For a shaft-driven rotor, η represents installation losses and mechanical transmission losses. A typical value for η in a single main rotor helicopter is approximately 0.88. This value includes power required to drive the tail rotor, and hence, makes it unavailable to the main rotor. The value of η for the reaction-drive rotor is much less than unity, between 0.5–0.7, due to internal pressure losses and rotational flow effects, and therefore, must be maximized in the sizing process using careful modeling of the flow system.

Sizing of any high-speed rotorcraft generally makes use of several major design parameters. For high-speed rotorcraft, these major design parameters are rotor disk loading W/A_d, V_T, wing loading W/S_w, and blade loading C_T/σ. By varying the design parameters, a designer or design team can determine which combination of design parameters will result in the "best" aircraft. Typically, the best design will have the lightest gross weight while remaining capable of completing a desired mission; generally, for most concepts, the lightest weight design will also minimize cost.[14] When sizing a reaction-driven, stopped-rotor concept, the hover rotor disk loading and rotor tip speed are used directly in the sizing methodology. Wing loading, which is directly used in design studies of tilt-rotor and tilt-wing aircraft, does not directly apply to a stopped-rotor concept because the area of the rotor blades only acts as a wing while the rotor is stationary; however, the idea of a "conversion wing loading" does apply to this concept. Similarly, blade loading is commonly used in most rotorcraft design studies, but because of the interdependence of the internal gasdynamics and external aerodynamics for reaction-driven, stopped rotors, blade loading is also not directly applied. The methodology that is required for this concept and the rationale for these differences are discussed in the following sections.

Several design issues inherent to the sizing methodology for the reaction-drive, stopped-rotor concept are addressed in detail in subsequent sections and include 1) internal total pressure losses, 2) the conflicting interrelational requirements of internal gasdynamics and external aerodynamics, 3) requirements and methodology for sizing conversion lifting surfaces, and 4) design implications impacting the selection of engine characteristics.

Rotor System Sizing

The rotor system of a stopped-rotor concept is a crucial part of the design because it allows for vertical takeoff and landing capability. The rotor-sizing methodology must ensure that the engine and blade combination will allow the aircraft to hover, with an appropriate power margin for vertical climb. This is done by a matching process. First, for a given engine output condition, there is an amount of power available, and second, for a given blade geometry, there is an amount of power required. Then by varying the engine output, or the blade geometry, or both, the power available and power required are varied until they match. Unfortunately, this matching process is not as straightforward as for a shaft-driven design; design implications and tradeoffs must be judiciously examined.

First, the maximum horsepower available to the rotor system must be calculated for comparison to the required power. This necessitates evaluation of the gas flow properties from the engine exit plane, through the hub and blades, to the blade tip nozzles. In a reaction-driven rotor, jet thrust produced at the blade tips by the ducted gas provides power available to the rotor system (Fig. 2). The gas analysis determines the magnitude of the internal losses, leading to the thrust produced at the blade tip nozzles, which ultimately results in horsepower available. Thus, increasing the thrust at the nozzles increases the power available to turn the rotor.

Losses in the rotor shaft and hub from the engine exit plane to the blade root remain relatively insensitive to changes in major aircraft design parameters, such as disk loading and tip speed. These hub losses are more likely to be affected by the number of turns in the ducts and the engine exit flow conditions. As an estimate, test data from the XV-9A hot-cycle, reaction-drive helicopter[15] suggest mass flow leakage of less than 0.5% and a total pressure loss of approximately 3% occur in the hub section. This approximation provides a simple method to account for hub losses. Alternatively, the shaft and hub can be modeled as a combination of several ducts, turns, and flow branches as appropriate. This second approach allows for calculation of minor and friction losses using graphs or tables of loss coefficients; representative graphs can be found in Ref. 16.

Disk loading and tip speed significantly affect the gas flow in the blades; therefore, the blade ducts require a more complete model. Internal flow analysis for the rotor blade ducts must evaluate potentially high-temperature exhaust flow in a rotating duct; this calls for equations dealing with generalized, one-dimensional flow of an imperfect gas, including rotational effects. The governing equation for flow through the blade duct with centrifugal pumping was derived in a manner similar to derivations of both Henry[17] and Bachmann.[3] Beginning with a sum of forces on an arbitrarily small segment of flow (Fig. 3), the governing equation for flow through the straight duct in the rotor blade is

$$\frac{dM}{dr} = \frac{M\left\{1 + \left[\frac{(\gamma - 1)}{2}\right]M^2\right\}}{(1 - M^2)}\left(-\frac{1}{A}\frac{dA}{dr} + \frac{\gamma M^2}{2}\left(\frac{4f}{\mathcal{D}}\right) + \frac{(1 + \gamma M^2)}{2T}\frac{dT}{dr} - \frac{\left\{1 + \left[\frac{(\gamma - 1)}{2}\right]M^2\right\}}{RT}\Omega^2 r\right) \tag{2}$$

The four terms in the oversized parentheses account for variations in flow Mach number caused by area change in the duct, losses due to friction, changes in total temperature, and centrifugal pumping, respectively. These driving potentials are considered to be the only factors influencing the flow, body forces due to gravity are generally neglected.

Area change of the rotor blade duct affects the flow as if a contraction or expansion were encountered. Since the flow in the rotor blade will generally be subsonic, an expansion of the duct area acts to decrease the flow Mach number; conversely, a contraction acts to increase the flow Mach number.

Friction forces encountered by the flow increase the Mach number of the flow. These frictional effects are typically seen as an increasing boundary-layer thickness. As the boundary-layer thickness increases, the effective duct area decreases. Friction is an important driving potential, and the rotor blade ducts should be as smooth as practical to reduce the deleterious effects. The smoothness (or roughness) of the duct's interior results from the manufacturing process. If a hot- or

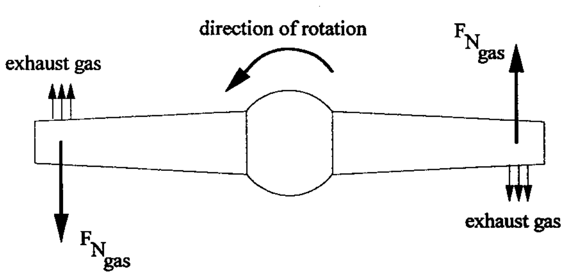

Fig. 2 Jet thrust produced at blade tips.

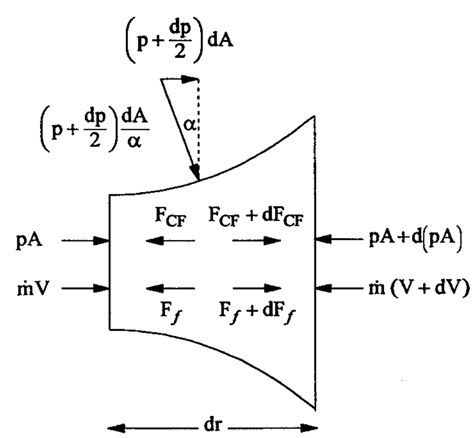

Fig. 3 Forces acting on an arbitrarily small segment of flow.

warm-cycle system is envisioned, the high-temperature material required for the ducts may present manufacturing difficulties. The Reynolds number of the flow also affects the friction factor; typically higher Reynolds numbers will correspond to lower friction factors.

Significantly elevated temperatures may characterize the flow in the rotor ducts, especially in a hot-cycle rotor system where exhaust gas from a turbofan engine can exceed 1200°F. Structural design concerns in the rotor blade will demand both high-temperature materials and some sort of insulation to surround the gas ducts. This insulation determines the amount of heat transfer from the gas in the rotor duct to the surroundings, thereby determining the change in flow total temperature. An adiabatic assumption for the rotor duct can be made by assuming perfect insulation so that $dT/dr = 0$. Temperature measurements in the hub and at the tip jet during the whirl test of the XV-9A rotor indicates this to be a reasonable assumption since the temperature differential was negligible.[18]

The solution of the differential equation [Eq. (2)] provides the flow properties up to the location where the duct turns to the blade tip nozzle. Losses are calculated for the 90-deg turn and the flow is then assumed to meet the exit boundary conditions for a converging nozzle.[19] This specifies the required nozzle exit area. From this information, the exit velocity is calculated using isentropic expansion equations; a discharge coefficient is applied to reflect the actual nonisentropic and non-one-dimensional characteristic of the flow. The thrust produced at the tip nozzle can then be calculated using Eq. (3). Typical values for converging nozzle C_D fall in the range of 0.93–0.99:

$$F_N = (C_D\dot{m})V_e + (p_e - p_a)A_e \tag{3}$$

A converging nozzle provides the best choice of nozzle for two reasons: 1) acoustics and 2) operating range. The acoustic signature of a reaction-driven rotorcraft will be much more significant if the nozzle has supersonic exit velocities; a converging nozzle ensures that the maximum exit Mach number is 1. A converging nozzle will also operate satisfactorily with subsonic exit velocities. In a practical reaction-drive stopped-rotor concept, the rotor shares engine exit flow with the forward propulsion system, particularly during conversion between flight modes. Because the rotor is designed to ensure vertical takeoff and landing ability, certain flight regimes will require the tip nozzles to operate with significantly less mass flow and total pressure than the design conditions. A converging–diverging nozzle suffers a significant loss in efficiency for off-design conditions, while a converging nozzle experiences far less degradation in performance. Of additional note, reaction-driven rotors enjoy their best operating conditions when the ratio of rotor tip speed to the nozzle exit velocity of the gas V_T/V_e is nearly 0.5. This ratio is most easily accommodated by subsonic exit velocities.

The torque supplied by the blade tip nozzles is calculated from the magnitude of the thrust multiplied by the radial distance of the nozzle from the center of rotation. This torque can easily be converted to power, but in this methodology this power is not defined as "power available," instead it has been called "nozzle power":

$$Q_{F_N} = bF_N r \tag{4}$$

$$HP_{F_N} = \frac{Q_{F_N}\Omega}{550} = \frac{bF_N\Omega r}{550} \tag{5}$$

Ducting the gas through the rotor blade requires that the mass flow accelerate in rotation as it passes through the blade. This rotational acceleration produces a "Coriolis power" that must be overcome in addition to the aerodynamic power requirements to rotate the blades:

$$Q_{co} = b\dot{m}(\Omega r^2) \tag{6}$$

$$HP_{co} = \frac{Q_{co}\Omega}{550} = \frac{b\dot{m}(\Omega r)^2}{550} \tag{7}$$

For a reaction-drive rotor the power available can be obtained by subtracting Coriolis power from the nozzle power, as in Eq. (8):

$$HP_{avail} = (b/550)[F_N\Omega r - \dot{m}(\Omega r)^2] \tag{8}$$

The second task of the rotor-sizing methodology involves computation of the power required by the rotor system. The power available as calculated previously is then compared to the power required for hover. Blades of a reaction-driven, stopped-rotor aircraft will vary considerably from conventional helicopter rotor blades. They may be large in an area with relatively low AR and no twist. This results in a low blade-loading and a large tip-loss factor uncharacteristic of conventional rotor blades. Using the stopped rotor as a fixed-wing requires a midchord symmetric airfoil section with blunt leading and trailing edges. This allows the trailing edge of the retreating rotor blade to become the leading edge of the fixed wing upon stopping the rotation, and vice versa. The methodology to predict the power required must be flexible enough to account for this. Circulation control could also be used, but adds considerable complexity and will not be addressed.

The third task of the sizing methodology requires matching power available to power required. To do this, the designer can change engine operating conditions, rotor blade geometry, or both. While performing this task, the design engineer may encounter several unique design issues. These issues relate to the interrelationship of the internal gas flow and the external aerodynamics.

Changing engine conditions depends on the engine design and selection approach used by the designers. As in many aircraft design studies, the engines used in the exercise can either be "fixed" engines or "rubber" engines. To size a rubber engine for hover, the actual size and characteristics of the engine can be changed to meet hover requirements. For example, if the power available at maximum engine operating conditions is below the required power, a larger engine size can be adopted. Conversely, if a fixed engine is used, engine output is varied by adjusting the engine throttle setting. In this fixed engine scenario, it is possible that the engine will be too small to hover the aircraft, if maximum power available does not meet the power required.

Changing the blade geometry to match the power required to the power available can provide a solution; however, the external aerodynamics and geometry of the blade are affected by and affect the internal gasdynamics of the blade ducts. During rotary-wing flight, the blade must be able to vary pitch for control of the aircraft. In order for this pitch motion to occur a feathering hinge at the blade root is required. Additionally, the gases used for the reaction-drive system must pass from the nonfeathering hub to the feathering blade. To transition the ducting carrying the gas, the logical arrangement is a circular duct with a rotating seal at the blade root. With this circular duct section at the blade root, a minimum wing or blade thickness is achieved if the duct lies along the midchord of the fore and aft symmetrical blade. Figure 4 shows a simple cross section of the blade root. The internal duct diameter, along with structural requirements, determines the blade's minimum thickness. For a specified thickness-to-chord ratio, this also determines the root chord length:

$$c = (\mathcal{D} + k)/(t/c)_{root} \tag{9}$$

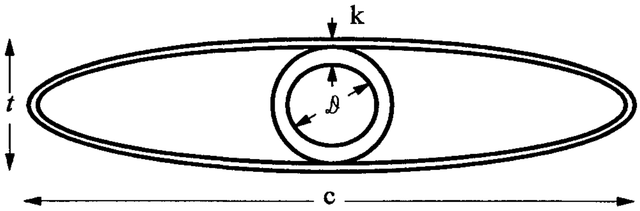

Fig. 4 Cross section at blade root.

Minimizing the thickness-to-chord ratio is desired for low profile power in rotary-wing flight and low profile drag in fixed-wing flight. The duct across the feathering hinge is likely to become a choke point in the flow because a minimum blade thickness corresponds to a minimum internal duct area. If the engine exhaust mass flow exceeds the maximum mass flow allowed in the ducting, defined by a choked flow condition, the equations describing flow properties will either not have a solution, or the solution will be invalid. To circumvent this, choked conditions must be recognized and lower mass flow output from the engine requested. Then, if the maximum, nonchoked engine flow produces less power than is required by the rotor system, the rotor system geometry must be altered. Increasing blade planform area increases the blade chord length; which, for a given thickness-to-chord ratio, provides a greater internal duct diameter and area. Using this method of resizing keeps the desired blade thickness-to-chord ratio to maintain the profile drag coefficient, but it increases the planform area of the rotor blades, increasing drag. Alternatively, maintaining a given planform area and increasing the thickness-to-chord ratio provides for a greater internal duct area. This latter approach will slightly increase the wetted area of the blades which, along with the increase in thickness-to-chord ratio, could significantly add to blade profile drag.

In the reaction-drive rotor, total pressure losses from wall friction and duct turns have the net effect of reducing the velocity of the gas discharged at the blade tips. To minimize losses in sections where the duct turns and to lower the impact of friction, the Mach number should be small. Duct area is a significant factor here; the larger the duct, the lower the Mach number for a given mass flow rate. It is desirable from an internal flow standpoint to increase the duct areas, but this conflicts with the desired external aerodynamic characteristics for thin blades with small planform area.

For a stopped-rotor concept, the rotor system sizing defines, to a great extent, the drag of the entire aircraft. Any increase in external drag due to increasing blade area results in higher engine thrust and fuel requirements for fixed-wing flight, increasing the aircraft's gross weight and requiring more power for hover. If the rotor system is "oversized," i.e., the blades and associated ducts can produce an extreme excess of power in hover, it will be desirable to reduce the thickness-to-chord ratio and/or the planform area of the blades to reduce drag. However, the blade loading C_T/σ of the rotor is directly related to the average lift coefficient of the blades. Reducing the blade planform area increases the blade loading; and so the area of the blades can only be reduced until either maximum blade loading or internal flow limits are reached.

Lifting Surface Sizing

To minimize the adverse dynamic effects of stopping or starting a rotor in flight, the use of lifting surfaces allows for unloading the rotor at the appropriate conversion speed. With the rotor unloaded, its degrees of freedom can be reduced by mechanical means to maintain stability during the stop/start process, while minimizing the forces and moments transferred from the rotor to the air vehicle. Past concepts, such as the Hughes rotor/wing, used a large rotating centerbody as the lifting surface. The CRW uses a combination of canard and lifting horizontal tail. Since these additional lifting surfaces add to the total lifting area for cruise, it is desirable to minimize their size to reduce drag. The centerbody approach denies the use of high-lift devices, requiring it to be relatively large, thus reducing the overall aerodynamic efficiency in both rotary- and fixed-wing flight. Locating surfaces with high-lift devices in the nonrotating system (on the fuselage) provides less "additional" lifting surface area in cruise. This approach could be implemented in a manner similar to the CRW, or a conventional wing placed beneath the rotor could be used depending on the severity of the penalty associated with the download caused by the rotor downwash impinging on the wing.

Because of the function of the lifting surfaces conventional tail volume coefficients usually used for sizing horizontal tail and/or canard surfaces during the conceptual design process are not applicable for a three-surface aircraft such as the CRW. These surfaces must provide all of the lift for the vehicle during conversion at relatively low speeds and provide trim in cruise at significantly higher speeds. In cruise, the lift can be shared by all surfaces to minimize induced drag. Both the canard and horizontal tail are all moveable providing several advantages. It allows the fuselage to maintain a level attitude during conversion, efficient trimming of the surfaces in cruise, and feathering the surfaces into the rotor downwash in low-speed rotary-wing flight. A variety of models are available to assist in sizing these surfaces. A simple lifting-line-based method provides the major influences between surfaces and can be implemented into a design sizing code without a major penalty in computational time, which a more rigorous method would encompass. Figure 5 shows the horseshoe vortex model used for the CRW lifting surface system in cruise. The same system, minus the rotor contribution, models conversion.

The previously discussed rotor sizing methodology determines the size of the rotor/wing based on hovering and vertical flight considerations. If the mission for which the aircraft is designed requires high-speed flight, the canard would be sized to minimize its area. A mission requiring low-speed loiter might dictate a larger canard area to minimize low-speed induced drag, thus minimizing fuel expended during the loiter segment of the mission. In this case, a strategy of varying "conversion wing-loading" and examining the resulting aircraft gross weights would determine the optimum canard area. Either way, the mission dictates the strategy taken to sizing the canard. For the high-speed mission, the canard, with the aid of flaps, operates at $C_{L_{\max}}$ with some stall margin during conversion; this minimizes its area. Using an initial guess at the surface areas and fuselage attitude, the relative locations and strengths of the horseshoe vortex circulation is determined. Using the Biot–Savart law, the influence of the horseshoe vortices on each surface can be assessed and ε can be determined through iteration. Summing moments about the

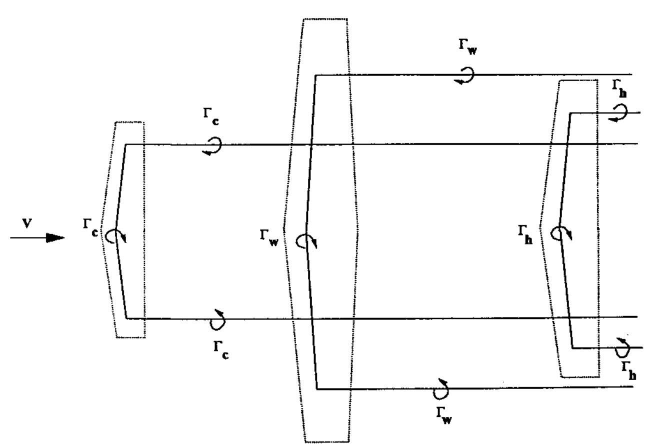

Fig. 5 Horseshoe vortex model for CRW.

horizontal tail location during conversion (no rotor lift assumed) determines the canard area:

$$S_c = \frac{(n_{\text{con}} W l_h - M_{\text{con}} - C_{M_h} q S_h \bar{c}_h)}{[C_{L_{\max_c}}(l_c + l_h) + C_{M_c}\bar{c}_c]q} \tag{10}$$

For the purposes of conceptual design, the terms containing the pitching moment coefficients can be conservatively neglected since their resolution also requires knowledge about the geometry that is being determined. Likewise, the M_{con} term consists of moments not created by the canard or tail, principally thrust and drag, and can be neglected if these are assumed to act on a line of action passing through the horizontal tail location. Otherwise, initial layouts and drag estimates at conversion speeds provide the necessary values of these moments. The term n_{con} represents a stall or maneuver margin capability for the canard. Longitudinal stability in cruise, with aerodynamic centers of both the rotor/wing and the canard ahead of the c.g. (the assumed c.g. location is under the rotor/wing hub center) represents the sizing criteria for the horizontal tail. For a specified static margin requirement and using the previously calculated canard area and rotor/wing area, the following cruise trim equation determines the horizontal tail area:

$$S_h = \frac{\left[S_c\left(1 - \frac{d\varepsilon_c}{d\alpha}\right)(C_{L_{\alpha_c}} l_c - C_{M_{\alpha_c}}\bar{c}_c) + \eta_w S_w\left(1 - \frac{d\varepsilon_w}{d\alpha}\right)(C_{L_{\alpha_w}} l_w - C_{M_{\alpha_w}}\bar{c}_w)\right]}{\eta_h\left(1 - \frac{d\varepsilon_h}{d\alpha}\right)(C_{L_{\alpha_h}} l_h - C_{M_{\alpha_h}}\bar{c}_h)} \tag{11}$$

Once again, the moment coefficient terms may be conservatively neglected for the sizing exercise, without a significant effect. The canard and the horizontal tail areas are then compared to the initial guess for convergence. If they have not converged, the process is repeated. The same approach can be applied to any configuration using lifting surfaces in the nonrotating system. Portions of the fixed-wing sizing methodology have been validated in wind-tunnel testing of a CRW aircraft model.[20]

Engine Design Implications

One of the significant features enjoyed by the reaction-drive, stopped-rotor concept is the use of the same propulsion unit for both powered-lift and forward flight. Prop-rotor aircraft can make this same claim, since turboshaft engines typically provide power in both flight regimes. However, prop-rotor propulsive efficiency decreases with increasing flight speed beyond about 400 kn. Stopped-rotor concepts do not experience this same decline in propulsive efficiency because in cruise the engines perform as a turbofan, turbojet, or a tip-driven cruise fan. The reaction-drive, stopped-rotor can use the exhaust of an existing technology gas generator(s) (i.e., turbojet, turbofan, or load compressor) to produce both rotor power and forward thrust, averting loss of propulsive efficiency and the need for new engine development. To perform this task satisfactorily, the engine(s) must be able to operate in a different range of operating conditions than normally experienced by these engines.

The selection of the type of reaction-drive system and gas generator can be made from three alternatives: 1) hot, 2) warm, and 3) cold cycles. A turbojet engine (or engines) provides the gases for a hot-cycle reaction-driven rotor system. The hot cycle has the advantages of high energy flow with a fairly large pressure ratio p_c/p_a; this combination can generally provide enough pressure to overcome the losses in the system. However, the hot-cycle system requires the structure of all ducting and surrounding components to withstand very high temperatures; some turbojets have exhaust temperatures in excess of 1200°F. The turbofan approach allows for the use of a warm-cycle system, which possesses the highest theoretical operating efficiency. This suggests that high bypass ratio turbofans provide the best engine option. Unfortunately, the operating conditions required of reaction-drive rotor systems include high static pressures at the exit plane of the fan that can induce fan surge or stall. During conversion between flight modes when exhaust flow is shared between the rotor system and the forward propulsion nozzles, the engine can also experience fairly large fluctuation of back pressures, ranging from very high back pressures that can surge the fan, to very low back pressures that may place the fan at a rotational speed limit. For a warm-cycle system to be successful, especially in a stopped-rotor system, the fan must be very insensitive to changes in back pressure. This suggests that high-bypass turbofans may not be practical for a warm-cycle system. The third alternative, a cold-cycle system, has seen more application to helicopters than the other two approaches. The cold-cycle system can be driven by a load compressor, which is essentially a turboshaft or reciprocating shaft engine that powers a separate compressor stage. These systems are often used as auxiliary power units in aircraft. For the cold-cycle system, the load compressor must be able to provide a significant pressure ratio and mass flow to the rotor system. The drawback of this approach is the physical size of the load compressor. To meet the desired conditions, the load compressor may be quite large in dimension and heavy in weight, or the use of tip-burning may be employed. Tip-burning requires fuel to be carried to the blade tips and ignited to add energy to the flow exiting the tip nozzle. This approach may save weight by reducing the demands on the load compressor, but it requires a system that can operate in the high centrifugal force present at the blade tips. Also, for cruise flight, a system for forward propulsion-like tip-driven cruise fans is needed for a cold-cycle system. For all three of the reaction-drive rotor systems, these design trades must be examined and reconciled to result in a functional system. If the use of existing engines constrains the design, the hot cycle, using a turbojet or the core flow of a bypass engine, represents the only viable alternative for this type of vehicle. The use of other cycles requires the development of an engine specifically for this application.

Engine sizing is also an important consideration when examining a reaction-driven concept. The engine must be of appropriate size to not only hover the aircraft, but also to fly at a desired cruise speed and to overcome the drag present during conversion between modes of flight. These three regimes must be accommodated by the same engine(s). The largest of these requirements defines the engine to be used. As discussed previously, either "fixed" engines or "rubber" engines can be used in a design study. Usually during conceptual design, aircraft designers require only information about the propulsive force (shaft power or thrust) and fuel flow of the installed engine(s). To examine engines for the reaction-drive rotor system, additional engine information beyond that normally required for sizing more traditional rotorcraft is needed. The mass flow rate, the total temperature and the total pressure of the exhaust gasses provide the information necessary to calculate flow properties in the ducts of the reaction-drive rotor system, and hence, the power available.

The rubber engine approach to aircraft design provides a great deal of flexibility, but the concerns of sizing an engine for a reaction-drive system can be complex. To begin, the rubber engine is based on an existing engine, and the size of this baseline engine is represented by its maximum sea-level static thrust rating. As mentioned in the rotor system sizing section, the hovering requirements of the aircraft determine not the thrust required, but the necessary flow conditions in the engine exhaust. These necessary flow conditions do correspond to an engine size (maximum sea-level static thrust) requirement. Care must also be taken not to scale a rubber engine too far from the original baseline engine, as engine characteristics and output conditions vary in a nonlinear manner with the engine's size and thrust rating.

The fixed engine approach is less complicated. In this mode of operation, the known engine performance provides constraints to the design space. For example, a fixed engine may not provide the necessary flow for the reaction-drive rotor to hover, resulting in an infeasible aircraft design. Similarly, the conversion and cruise requirements demand thrust levels that the fixed engine must provide. If the engine cannot provide the thrust, the rotor configuration's size and/or geometry must be changed to find a feasible combination. Failing this, a different engine(s) for the aircraft must be chosen.

Sizing Aircraft Using the Methodology

The methodology presented in this article has been customized for conceptual design of the canard rotor/wing aircraft in a computer sizing code, RWSIZE.[21] This code has been used to size different CRW aircraft, ranging from manned attack type aircraft to unmanned aerial vehicles (UAV). The code has proven to be very helpful in examining different combinations of design variables with quick speed and satisfactory reliability. To show the utility of the methodology, a hypothetical UAV mission, shown in Fig. 6, was used to size a CRW aircraft.

An unmanned CRW for this mission was evaluated for several combinations of hover disk loading and tip speed to produce the carpet plot shown in Fig. 7; from this plot, the minimum weight solution is chosen for further, more detailed evaluation. During this exercise the "rubber" engine option was used, allowing the engine to be sized to the exact needs of each aircraft design point. It can been seen in the carpet plot that the minimum weight point has a disk loading of 20 lb/ft² and a rotor tip speed of 700 ft/s. Corresponding vehicle parameters are also determined by the RWSIZE code including: lifting surface geometry, rotor solidity; equivalent flat plate drag area, weight estimates; etc., which are listed in Table 1.

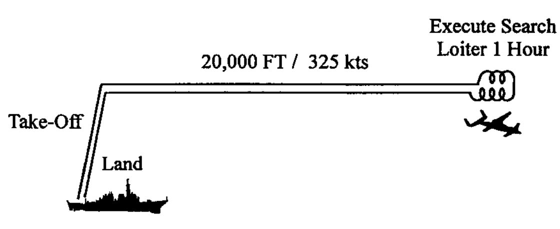

Fig. 6 Unmanned aerial vehicle design mission.

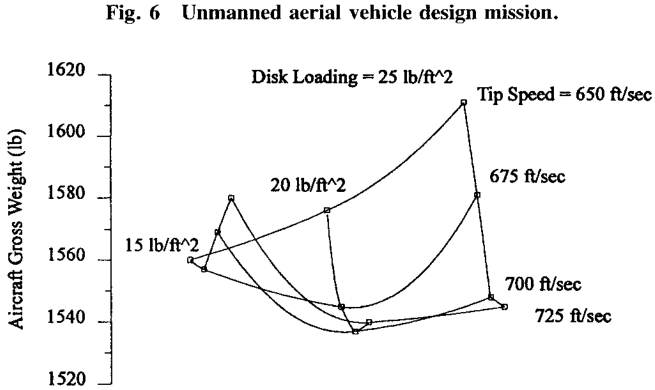

Fig. 7 UAV sizing carpet plot.

Table 1 Design point vehicle parameters

Parameter	Value
Gross weight	1536 lb
Engine maximum sea level thrust	422 lb
Blade loading, C_T/σ	0.10
Rotor solidity, σ	0.184
Disk loading	20.0 lb/ft²
Hover tip speed	700 ft/s
Rotor diameter	9.86 ft
Rotor root chord	2.56 ft
Rotor tip chord	1.53 ft
t/c at root	0.22
t/c at tip	0.118
Flat plate equivalent drag area	2.29 ft²
Rotor/wing area	23.9 ft²
Canard area	5.9 ft²
Horizontal tail area	10.0 ft²

Conclusions

Sizing a reaction-driven, stopped-rotor VTOL concept calls for using a wide range of different engineering disciplines to cover the various features of the aircraft and its propulsion system. Several important conclusions about sizing this type of high-speed rotorcraft can be made.

1) Existing design codes normally used for sizing rotary-wing aircraft are incapable of sizing reaction-drive concepts due to their inherent complex design interrelations. The internal gasdynamics and the external aerodynamics are highly interrelated and must be included in any design model for these types of vehicles. Unlike shaft-driven concepts, determining power available requires knowledge of the engine exit flow parameters of temperature, pressure, and mass flow rate. Through internal gasdynamic analysis, this then allows matching power available to power required. The conditions into which the engine exhausts, typically relatively high back pressures, favor the use of a hot cycle approach (low bypass ratio or turbojet) if existing engines are to be considered for this application. Engines must possess sufficient surge margin to operate in this manner. The drawback is the high temperature experienced by the hub and ducts.

2) The effect of rotation on the gas as it travels down the ducts results in both a positive and negative contribution to the power available. The positive effect occurs due to the centrifugal pumping that acts to recover the total pressure loss due to friction and minor losses. The increased power due to Coriolis forces acting on elemental mass of the flow results in a "Coriolis power" penalty. Both of these effects are significant and must be included.

3) Unloading the rotor during conversion through the use of auxiliary surfaces is most efficiently accomplished through their placement on the fuselage; the use of high lift devices minimizes their size. More attention must be given to aircraft stability during the sizing of these surfaces. It requires some rudimentary stability analysis due to the shift in the aircraft center of lift location from rotary- to fixed-wing flight modes. Use of tail volume coefficients is an inappropriate sizing criteria for these concepts. Once sized, trimming the aircraft for minimum induced drag during the various mission segments reduces vehicle gross weight by saving fuel.

4) Reaction-driven, stopped-rotor concepts offer significant advantages for some types of missions over their shaft-driven counterparts. They should not be neglected during concept

selection simply because of the lack of availability or understanding of how to size them.

5) Finally, the RWSIZE code, which uses this methodology, has demonstrated its capability to size CRW aircraft for a variety of manned and unmanned VTOL applications. Portions of this methodology have been validated through wind-tunnel testing and internal flow tests.

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