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CONSTANT ALTITUDE HELICOPTER FLIGHT TESTING

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Abstract

Normalized W/σ performance data for a helicopter can be obtained during a constant altitude flight with the same accuracy as the ratio can be obtained during a variable density flight. The only bound on this method is that the data acquisition be as rapid as possible to minimize the weight change during the data acquisition phase.

Background

In order to have any utility, the data obtained during the performance testing of helicopters must be normalized. This not only permits the development of meaningful relationships between the performance parameters of different helicopters, but normalization of the data is essential when a single helicopter is tested under different conditions.

The parameters that are the most likelely to vary during a series of tests, or even during an individual test, are gross weight, ambient air conditions and rotor rotational velocity. Even though it is necessary to make corrections to the to the raw data for changes in these factors from some established reference, some of these changes can be used to an advantage in flight testing.

Variation of Parameters

It is virtually impossible to find two times with exactly the same ambient conditions of temperature, pressure and density. However, since the ambient density of the air is the principal factor of these three in determining the power requirements of a helicopter, even though it might not be convenient, it is usually possible to duplicate the ambient air density of a given flight by flying at a different selected altitude.

For most helicopters, and in particular for the military helicopters, the rotational velocity of the rotor system is highly stabilized. Even for those models that do not have this feature, it is usually possible for the test pilot to duplicate the rotor velocity from a previous flight.

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This leaves the aircraft weight as the most troublesome variable in performance flight testing. Even though a test vehicle may be loaded exactly the same for different flights, the weight will vary throughout the flight duew to the consumption of fuel. Thus it is that the data obtained near the end of a flight may be significantly different from that obtained early in the test.

Weight to Pressure Ratio

For fixed-wing jet aircraft, where the thrust required (F) is a function of true airspeed (V), ambient air temperature (T_a), ambient air pressure (P_a), engine speed (N) and engine size (D), it can be shown[1], using the Buckingham Pi therom that,

$$F/\{P_a D^2\} = f\{V/\sqrt{T_a},\ ND/\sqrt{T_a}\} \tag{1}$$

Inasmuch as sea level values of pressure and temperature are constants, as is the engine size, equation (1) may be rewritten as,

$$F/\delta = f\{M,\ N/\sqrt{\theta}\} \tag{2}$$

where δ is the ratio of ambient to sea level pressures, M is the Mach number and θ is the ratio of ambient to sea level temperatures.

Equation (2) indicates that F/δ, a pressure corrected thrust parameter, is a function of Mach number and the temperature-corrected engine RPM parameter.

If the dynamic air pressure is expressed in terms of Mach number and pressure ratio, vice velocity and density ratio, it can be shown that the aircraft drag to pressure ratio parameter is a function of Mach number and the weight to pressure ratio,

$$D/\delta = f\{W/\delta,\ M\} \tag{3}$$

Thus, for the fixed-wing jet aircraft in equilibrium flight, when drag is equal to the thrust required, a normalized drag (drag divided by pressure ratio) or a normalized thrust (thrust required divided by pressure ratio) can be expressed in terms of a normalized weight divided by pressure ratio,

$$F/\delta = f\{W/\delta,\ M,\ N/\sqrt{\theta}\} \tag{4}$$

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Not only is this normalization handy from a mathematical point of view, but it has great utility in the flight testing. Equation (4) indicates that, with a constant engine speed, at each Mach number the thrust divided by the pressure ratio is a constant if the ratio of aircraft weight to pressure ratio is held constant. As fuel is consumed (weight decreased), the factor W/δ can be maintained at a constant value by decreasing the pressure ratio, i.e., climbing to a higher altitude.

Inasmuch as pressure ratio is read out directly in the aircraft cockpit as pressure altitude, the Test Pilot and/or the Test Engineer can develop during the preflight planning a schedule of altitude as a function of fuel consumed in order to maintain a constant value of W/δ.

Such a schema can be (and frequently is) used in the performance testing of helicopters, but due to the nature of the power required equation, this method does not provide as accurate results as other methods.

Helicopter Power Required

The total power required for a single main rotor helicopter in level, unaccelerated forward flight may be broken single main rotor down into the sum of four components: Induced power, Profile Power, Parasite Power and Tail Rotor Power[2].

$$P_T = P_i + P_o + P_p + P_{tr} \tag{5}$$

Let us now consider each of the power components.

Induced Power

The Induced power, the power required to produce the thrust that balances the weight of the helicopter, is equal to the product of the thrust (weight) and the velocity induced by the power input to the rotor system. At zero forward velocity, hover, the induced velocity can be estimated very closely from Momentum Theory, and at forward velocities (V_f) in excess of about forty (40) knots, the induced velocity for a helicopter with main rotor radius of R feet may be estimated as,

$$v_i = W/\{2\pi\rho R^2 V_f\} \tag{6}$$

so that the Induced power in forward flight may be written as,

$$P_i = W^2/\{2\rho\pi R^2 V_f\} \tag{7}$$

Profile Power

At hover, the power required to turn the rotor, Profile power, is a function of the coefficient of profile drag of the rotor blade, the total area of the blades, the ambient air density and the cube of the tip velocity of the rotor. The Profile power required in forward flight is equal to the value at hover times a correction factor that is a function of the advance ratio (μ), the ratio of forward velocity to the rotor tip velocity.

$$P_o = P_{o_h}\{1 + 4.3\mu^2\} \tag{8}$$

Parasite Power

The power required to overcome the drag of the fuselage and rotor hub in forward flight is called the Parasite power and is a function of the ambient density, the equivalent flat plate area (f_f) and the cube of the forward velocity. The equivalent flat plate area is an area that, with a coefficient of drag equal to 1.0, would provide the same drag as is experienced by the helicopter with its actual frontal area and actual coefficient of drag.

$$P_p = \{{}^1/_2\rho V_f^3 f_f\} \tag{9}$$

Tail Rotor Power

The tail rotor power is the sum of the tail rotor Profile power and the tail rotor Induced power, where the latter is the power required to generate a tail rotor thrust to balance the main rotor torque. Inasmuch as the tail rotor power is usually but about 3 to 10 percent of the main rotor power, and inasmuch as the change in tail rotor power is very small across the velocity range, it will be assumed that the variation in tail rotor power with velocity is negligible.

Coefficient of Power

In order to non-dimensionalize the power functions, a coefficient of power is defined as the power divided by the ambient air density, the rotor disc area and the cube of the rotor tip velocity.

$$C_p = P/\{\rho\pi R^2 V_T^3\} \tag{10}$$

In order to convert the total power required, Equation (5) to coefficient form, it is therefore necessary to divide each term by the conversion factor $\{\rho\pi R^2 V_T^3\}$.

Coefficient of Induced Power

Dividing equation (7) by the factor produces,

$$C_{p_i} = W^2/\{2\rho^2(\pi R^2)^2 V_T^3 V_f\} \tag{11}$$

With a constant tip velocity, and by converting density to density ratio ($\sigma = \rho/\rho_{ssl}$), Equation (11) may be written as,

$$C_{p_i} = K_1\{W^2/\sigma^2\} \tag{12}$$

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Coefficient of Profile Power

By splitting Equation (8) after dividing by the conversion factor, then collecting all of the constant terms, the coefficient of profile power in forward flight may be written as the sum of the coefficient of profile power in hover (a constant) and a term that is a function of the forward velocity.

$$C_{P_o} = K_2 + K_3\{V_f^2\} \qquad (13)$$

Coefficient of Parasite Power

Dividing the Parasite power term by the conversion factor and collecting constant terms,

$$C_{P_o} = K_4\{V_f^3\} \qquad (14)$$

Coefficient of Tail Rotor Power

This term, with the previously made assumptions, is simply,

$$C_{P_{tr}} = K_5 \qquad (15)$$

Combined Terms

Combining all of the total power terms in coefficient form produces,

$$C_{P_T} = K_1(W^2/\sigma^2)/V_f + K_2 + K_3V_f^2 + K_4V_f^3 + K_5 \qquad (16)$$

From Equation (16) it may be seen that in order to maintain a constant coefficient of power at each airspeed, it is necessary to have a constant ratio of weight to density ratio. This has been the traditional method for helicopter performance flight testing[3]. As fuel is consumed during a flight, a change in density altitude is made so that the W/σ ratio remain a constant.

The result of a series of W/σ tests is a family of power (or coefficient of power) versus airspeed curves. Interpolation between test value curves can be accomplished to provide intermediate values. And, of course, any W/σ curve is also a sea level plot for a weight equal to the W/σ ratio. It is to be noted that with high values of density ratio, i.e., high pressure altitude and/or high ambient temperature, it is possible to achieve test W/σ values far in excess of the maximum permissible gross weight when the ratio is normalized to sea level density ratio.

At first glance, the W/σ testing appears to be exactly like the fixed-wing W/δ testing. The problem in helicopter testing, however, lies in the fact that one must not simply change altitude as fuel is consumed, but one must find a new pressure altitude and ambient temperature to provide the desired value of density ratio. Without a precise and detailed knowledge of the ambient pressure conditions and temperature lapse rates, one can not derive a weight versus altitude schedule *prior to launch* that will ensure a constant W/σ ratio.

One method that is used is the development of a set of curves such as are shown in Figure 1.

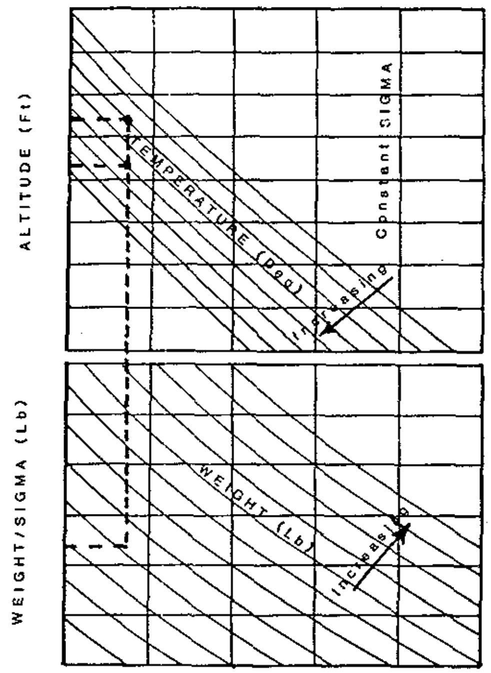

Fig. 1 - W/σ Chart

One enters the lower, left portion of the chart with the desired value of W/σ and moves horizontally to the right until the actual weight value is intercepted. Then one proceeds vertically (along lines of constant density ratio) until an intersection of actual ambient pressure (read as pressure altitude from the altimeter) and outside air temperature is reached. It is at this pressure altitude *that one flies at this weight to obtain* the desired W/σ value.

It can be seen that in order to maintain constant values of the W/σ ratio, one must be able predict the weight of the aircraft by the time the aircraft has been positioned to take the next observation. One must also be able to predict the ambient temperature at the test point altitude. The weight prediction is not difficult, but the temperature prediction is quite difficult unless there is a very constant temperature lapse rate. An error in temperature of about two degrees (2^o) Celsius may mean as much as a three hundred (300) feet difference in desired altitude.

In addition, a fuel-efficient helicopter may be required to change altitude only fifty (50) to eight (80) feet between test points. As a result, more time is spent in (a) determining the conditions for the next test point and (b) positioning the aircraft at that altitude/temperature combination than is required to conduct the test.

Constant Altitude Technique

In an effort to make the performance data acquisition for a helicopter less troublesome, a hypothesis was formulated that, particularly for a light helicopter, the amount of fuel consumed during a single series of test points is so small that the test could be conducted at a constant altitude, i.e., one density ratio, and an average weight could be used for the input to the W/σ value.

The Test

To start the test, the aircraft is positioned at an altitude that provides the pressure and temperature conditions for the desired test value of density ratio. It is to be noted that this does not have to be the precise pre-flight planned point. Just as a velocity point of sixty eight (68) knots is accepted for a desired seventy (70) knots, a W/σ value of fourteen thousand nine hundred and fifty (14,950) is as good a value as a fifteen thousand (15,000) value.

An initial weight determination is made (from fuel remaining) as are ambient temperature and pressure observations. The aircraft is then flown through the desired velocity range at a constant altitude, stabilizing at velocity points only long enought to (a) ensure that the point is indeed stable and (b) to take a power, torque and/or fuel flow reading for the determination of power required.

As soon as a reading is obtained, the airspeed is changed to that for the next test point. At the conclusion of the run, the final weight is noted, as are the ambient temperature and pressure conditions.

The average weight of the helicopter during the test run (final weight plus one-half the difference between the initial and final weights) is divided by the density ratio (which is obtained from the pressure and temperature readings). This number is used as the test W/σ value.

Test Results

Flight tests were conducted on a series of helicopters to validate the hypothesis. At first, tests were conducted only on light helicopters whose fuel flow rates were so low that it was felt that there would be little adverse effect from applying the Constant Altitude Technique. Initial tests were made on a Westland Wasp and an Aerospatiale Alouette III. The former aircraft has a maximum gross weight of about four thousand (4,000) pounds and the latter grosses out at about four thousand eight hundred (4,800) pounds. Both are single-engine aircraft.

These tests were conducted on a not-to-interfere basis during a Test Pilot instruction course, so tests were limited to two (2) W/σ values for each aircraft. Inasmuch as data points were not obtained at exactly the same airspeeds for both the constant altitude tests and the variable altitude tests, the results of each comparison for a given value of W/σ were plotted on a plot of power per density ratio (P/σ) versus airspeed. However, the correlation between the two methods was so close in both tests for both helicopters that only a single curve could be shown unless an extremely magnified scale were used. Therefore, in order to demonstrate the outcome, each set of data was separately plotted to a large scale, a curve was faired through the points and readings were made at discrete values of velocity.

Table 1 shows a comparison for one set of readings of the constant altitude data and the variable altitude data for the Wasp, and Table 2 shows the same type of relationship for the Alouette III. Although two (2) W/σ values were checked for each aircraft, only one set of data is presented for each due to the similaritry of information.

Table 1 - Wasp Aircraft Data
W/σ = 4,520 lbs+
{W = 3,500 lbs, H_ρ = 8,500 ft}

Velocity Kts	Power/σ SHP Constant	Power/σ SHP Variable
30	459	464
40	433	437
50	425	426
60	437	437
70	464	460
80	502	498
90	554	547

Table 2 - Alouette III
W/σ = 4,820 lbs
{W= 3,800 lbs, H_ρ = 7,900 ft}

Velocity Kts	Power/σ SHP Constant	Power/σ SHP Variable
30	302	298
40	268	268
50	253	252
60	254	252
70	t68	264
80	292	294
90	327	332

Heavy Aircraft Tests

It was not surprising that such close correlation could be obtained from the test results for light single-engine helicopters with their low specific fuel

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consumption. There was also a question raised whether or not the good results, as shown in Tables 1 and 2, might be due in part to the pilot's experience in the aircraft that permitted the rapid stabilization and data acquisition.

It was therefore decided to test the hypothesis on a heavier weight helicopter with a pilot who, although highly qualified in rotary wing aircraft, had not previously flown either this model aircraft or a helicopter with this much gross weight. A series of tests were therefore conducted using an Aerospatiale Super Frelon with the first test run being on the pilot's first flight in the aircraft.

After an initial lift-off into a hover-in-ground-effect, the pilot landed and then made a rolling takeoff and climbed to six hundred (600) feet above the terrain to transit to the operating area. As soon as the aircraft was level, the pilot ran through a series of airspeeds commencing at forty (40) knots and accelerating at ten (10) knot increments to one hundred thirty (130) knots. Because this aircraft was not specifically instrumented for flight test, it had neither torque nor fuel flow indicators, either of which could have been used to determine the power required. As a result collective angle, which was indicated on a cockpit gage, was used as a measure of power required.

Whereas the Wasp and the Alouette III (and later in the program the Aerospatiale Alouette II and Puma) had constant altitude tests performed in order to compare with the variable altitude runs, with the Super Frelon the constant altitude tests were performed first. Table 3 is the result of the first of two constant altitude tests with the Super Frelon.

Table 3 - Super Frelon
W/σ = 25,985 lbs
{W = 21,000 lbs, H_ρ = 7,100 ft}

Velocity Kts	Collective Pitch Constant	Collective Pitch Variable
40	14.3°	14.4°
50	12.9°	13.0°
60	12.1°	12.1°
70	11.6°	11.7°
80	11.5°	11.6°
90	11.7°	11.8°
100	12.1°	12.3°
110	12.9°	13.0°
120	13.8°	14.0°
130	15.0°	15.1°

Conclusions

From the flight tests of five different models of helicopters it was observed that the values of W/σ obtained from constant altitude tests were of the same order of accuracy as those obtained from the conventional variable altitude tests, provided that due care is taken in the data acquisition and that the data acquisition is accomplished in an expeditious manner.

Because of the considerable reduction in the time required, both pre-flight and in-flight, the constant altitude method is time and cost efficient.

Ackowledgement

Considerable credit must be given to Mr. Sean C. Roberts, President, Flight Research, Inc., Mojave, California for both the development of the hypothesis and the making available of aircraft and flight test time through his National Test Pilot School.

The flight tests referred to in this paper were conducted while the author was a consultant to National Test Pilot Schools.

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