

DYNAMICS OF HELICOPTER FLIGHT

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we can talk about a pressure altitude of 10,000 ft. The pressure altitude has nothing to do with the physical altitude of the aircraft, either above ground or above sea level. It is only an alternate and very useful way of expressing pressure. Meteorologists will quote the pressure either directly (usually in millibars or inches of mercury) or by pressure altitude. All altimeters are calibrated on the basis of a sea level pressure of 29.92 in. Hg. If the altimeter indicator is set to this value, the instrument reads pressure altitude directly.

Density altitude is that altitude in the standard atmosphere which corresponds to a particular density. A density of 0.001545 slug/ft³ can also be expressed as a density altitude of 14,000 ft. Density altitude greatly affects helicopter performance, as will be reflected in performance charts. If the pressure and temperature were known, we could calculate density using the universal gas law. Another method is to find the pressure *altitude* and temperature, and then determine the density *altitude*. Figure 2.2 illustrates the relationship between pressure altitude, density altitude, and temperature. For example, a pressure altitude of 4000 ft and a temperature of +24°C result in a density altitude of 6000 ft. It is the density altitude that the helicopter will "feel" and will respond to. Aviation computers can produce the same result, though perhaps less accurately.

Note that as temperature drops with increasing altitude, so does the absolute viscosity μ . However, the density is decreasing at an even faster rate; hence kinematic viscosity ($\nu = \mu/\rho$) increases with altitude.

2.3 BERNOULLI'S PRINCIPLE AND AIRSPEED MEASUREMENT

All aerodynamic forces bearing on a surface are the result of air pressure (perpendicular to the surface) or air friction (parallel to the surface). Friction effects, associated with the property of viscosity, are generally confined to a thin boundary layer and usually are not dominant. As a first approximation we can therefore neglect the effects of viscosity and concentrate on pressure distribution.

Consider the uniform flow of a fluid through the converging-diverging channel of Figure 2.3. Represent the conditions upstream by p_1 , T_1 , ρ_1 , and the velocity and cross-sectional area by V_1 and A_1 respectively. Assign the subscript 2 to the parameters at the throat. We do not inject any fluid into the flow between sections 1 and 2, nor do we allow any to escape.

An important initial concept is that of conservation of mass: what goes in must come out. To put it more exactly, the rate of mass flow by section 1 must be equal to that of section 2 if no fluid is being injected or extracted in between. The volume of air per unit time is simply the product of the cross-

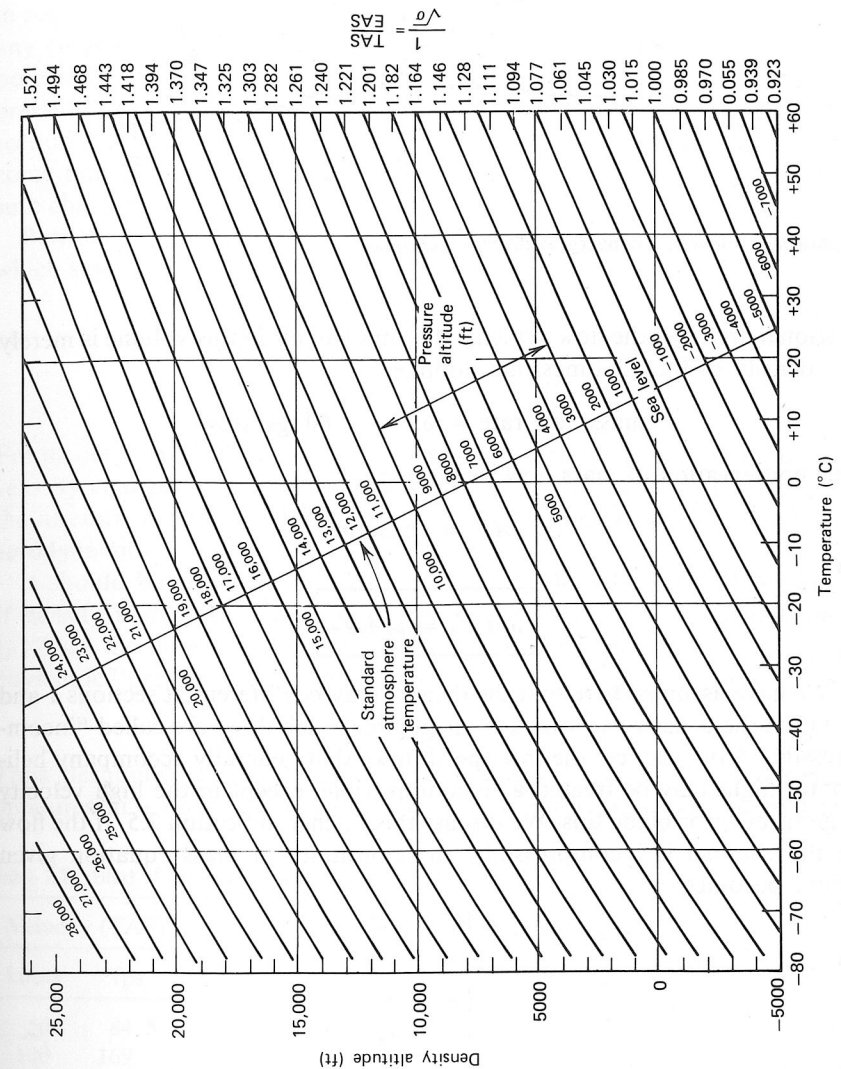


Figure 2.2 Density altitude, pressure altitude, and temperature.

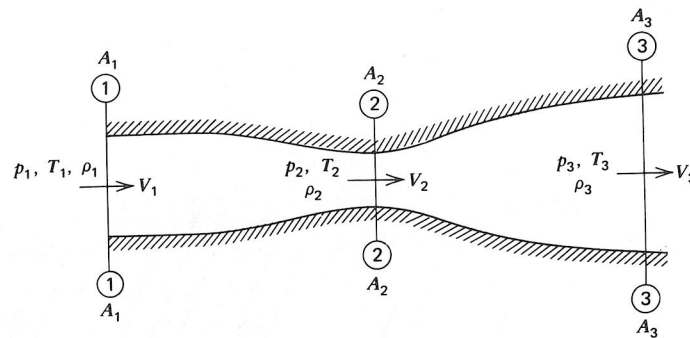


Figure 2.3 Flow in a convergent-divergent channel.

sectional area and the flow velocity. The mass of air in this volume is merely the density of the air times that volume:

$$\text{mass flow rate} = \rho A V \quad (\text{slugs/sec})$$

The conservation of mass flow then says:

$$\rho A V = \text{constant}$$

or

$$\rho_1 A_1 V_1 = \rho_2 A_2 V_2$$

We have assumed here that the density may be different at sections 1 and 2. Often the density variations are negligible. Such flows are called "incompressible." In practice, the low speed flows that generally accompany helicopter flight may be treated as incompressible, except in the high velocity regions of rotor blade tips. We discuss this further in Section 2.5. If the flow in the channel is incompressible, the continuity of mass equation given above becomes:

$$A_1 V_1 = A_2 V_2$$

or

$$\frac{V_2}{V_1} = \frac{A_1}{A_2}$$

Thus if A_1 is greater than A_2 (as it is in our example) V_2 must be greater than V_1 in incompressible flow. This is also known as the Venturi effect and simply says that the flow of an incompressible fluid speeds up through a restriction in direct proportion to the reduction in area.

Bernoulli's Principle

If the flow velocity increases, its kinetic energy level has increased. We know from the concept of conservation of energy that a corresponding decrease in some other kind of energy must have taken place (since we did not add any energy between the two sections). In aerodynamics, the equivalent of potential energy is static pressure. If the kinetic energy increases, the static pressure decreases. Viewed another way, the flow was accelerated from section 1 to section 2. According to Newton's second law this could have come only from an unbalanced force. This force is provided by the pressure change.

Bernoulli's equation is essentially an energy conservation statement, which says:

$$p + \frac{1}{2}\rho V^2 = \text{constant}$$

or

$$p_1 + \frac{1}{2}\rho_1 V_1^2 = p_2 + \frac{1}{2}\rho_2 V_2^2$$

Only correct if $\rho_1 = \rho_2$.

For incompressible flow $\rho_1 = \rho_2$. Bernoulli's equation relates, quantitatively, velocity changes to pressure changes. Since pressure, acting on an area of the aircraft, produces a force, Bernoulli's equation is extremely useful in aerodynamics.

It should be appreciated that the term $\frac{1}{2}\rho V^2$ has the units of pressure. If density is in slugs per cubic feet and velocity in feet per second, then $\frac{1}{2}\rho V^2$ has the units of pounds per square foot. This term is also called the "dynamic" pressure and is denoted by q :

$$\text{dynamic pressure } q = \frac{1}{2}\rho V^2$$

Thus Bernoulli's equation states that the sum of the static and dynamic pressures in a flow is constant. Table 2.2 gives typical values of dynamic

Table 2.2 Effect of Speed and Altitude on Dynamic Pressure

Velocity (TAS)		Dynamic Pressure, q (psf)					
knots	fps	SL	5000	10,000	15,000	20,000	30,000
50	84.5	8.48	7.30	6.25	5.34	4.51	3.17
100	169	33.9	29.2	25.0	21.3	18.1	12.7
150	254	76.2	65.7	56.1	47.9	41.0	28.5
200	338	135.6	116.8	100	85.2	72.4	50.8
400	676	542.0	467	400	341	289	203
600	1013	1221	1051	902	769	651	457

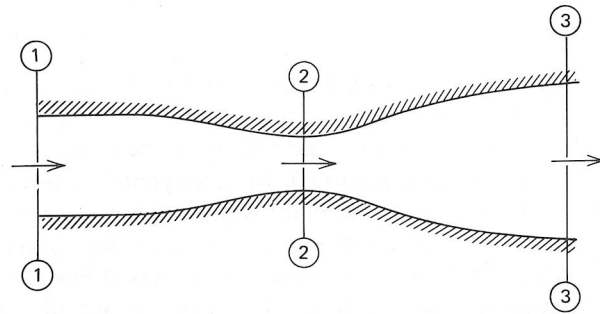


Figure 2.4 Variation of flow parameters for incompressible fluid.

$A_1 = 10 \text{ ft}^2$	$A_2 = 5 \text{ ft}^2$	$A_3 = 15 \text{ ft}^2$
$V_1 = 100 \text{ knots}$	$V_2 = 200 \text{ knots}$	$V_3 = 66.6 \text{ knots}$
$\rho_1 = 0.00238 \text{ slug/ft}^3$	$\rho_2 = 0.00238 \text{ slug/ft}^3$	$\rho_3 = 0.00238 \text{ slug/ft}^3$
$p_1 = 2116 \text{ psf}$	$p_2 = 2014 \text{ psf}$	$p_3 = 2135 \text{ psf}$
$q_1 = 34 \text{ psf}$	$q_2 = 136 \text{ psf}$	$q_3 = 15 \text{ psf}$

pressure for various true airspeeds and altitudes in the standard atmosphere. Notice that the dynamic pressure at some fixed velocity varies directly with the density ratio at any altitude. Also, appreciate the fact that at an altitude of 40,000 ft (where $\sigma = 0.25$) twice the airspeed is needed to produce the same dynamic pressure.

Figure 2.4 is a numerical illustration of flow in a channel.

Airspeed Measurement

If a symmetrically shaped object is placed in a moving airstream, a flow pattern such as shown in Figure 2.5 results. There is a point at the very nose of the object at which the flow is brought to rest. This point is called the stagnation point, and the streamline ending in this point is the stagnation streamline. Streamlines above this line go over the top of the object; all

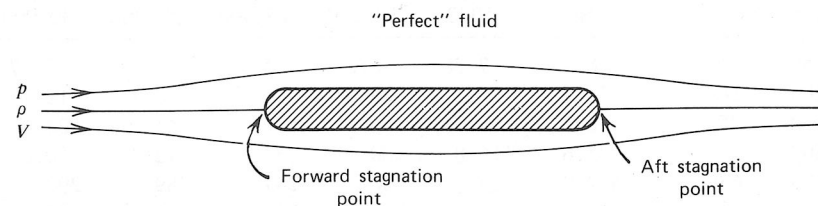


Figure 2.5 Flow pattern around a symmetrical object. Stagnation pressure $p_s = p + \frac{1}{2}\rho V^2$.

others pass below. Since the velocity at this point is zero, the stagnation pressure must be

$$p_s = p + \frac{1}{2}\rho V^2$$

where p and V are the upstream conditions. A pitot tube, such as shown in Figure 2.6, has a stagnation point at its entrance (at zero angle of attack); there is no flow through the tube. A pitot tube alone, hooked up to pressure sensitive bellows and an indicator, would therefore measure the sum of the upstream static and dynamic pressure. If a source of static pressure can be found on the aircraft, equal in magnitude to the upstream static pressure (which implies that the local velocity past the static port is equal to the upstream velocity), it can be fed to the other side of the bellows. The indicator then measures only dynamic pressure, since static pressures are balanced on either side of the diaphragm.

If we assume, as most airspeed indicators do, that the density is constant at 0.00238 slugs, the indicator displays true airspeed (note that doubling true airspeed quadruples dynamic pressure so that the scale on the face of the indicator is nonlinear). There are a number of reasons why the airspeed so indicated may not be an accurate measure of the true airspeed of the helicopter:

1. Position and/or installation error, ΔV_i : the instrument is calibrated for greatest accuracy at a particular flight condition, usually for straight and level flight at some usual cruise speed. At other conditions the stagnation point may migrate, lowering the pressure in the pitot tube and, often

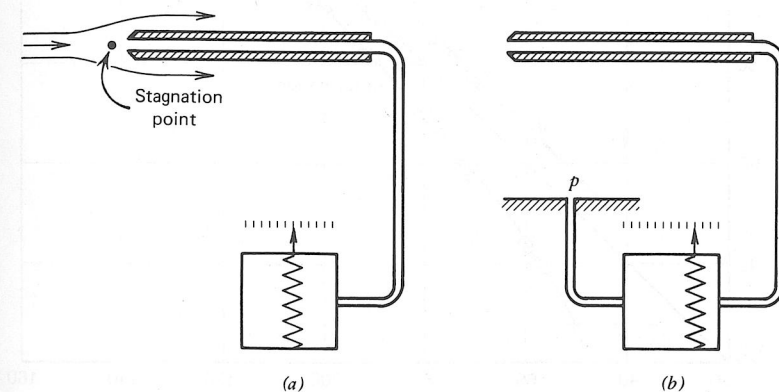


Figure 2.6 Airspeed measurement. (a) Pitot tube measures stagnation pressure $p_s = p + \frac{1}{2}\rho V^2$. (b) Pitot/static system measures $p_s - p = \frac{1}{2}\rho V^2$ (i.e., dynamic pressure).

more important, changing the pressure at the static ports because of local flow condition changes. Good engineering can minimize these errors. Note that the error in the installation shown in Figure 2.7a is essentially zero at 100 knots straight and level, but increases to as much as 20 knots in autorotation at 50 knots. In this installation the indicator reads too high in climbs and too low in descents. Other installations, such as that shown in Figure 2.7b, have opposite effects. Calibrated airspeed (CAS) is the indicated airspeed (IAS) corrected for position error.

2. Compressibility effects, ΔV_c : a further source of error comes about at high speed (greater than about six tenths the speed of sound) where density changes occur that affect dynamic pressure independent of velocity. Within the speed and altitude range of conventional helicopters this effect is small, and will be ignored for computational purposes. Equivalent airspeed (EAS) is the result of corrections in CAS for compressibility errors.

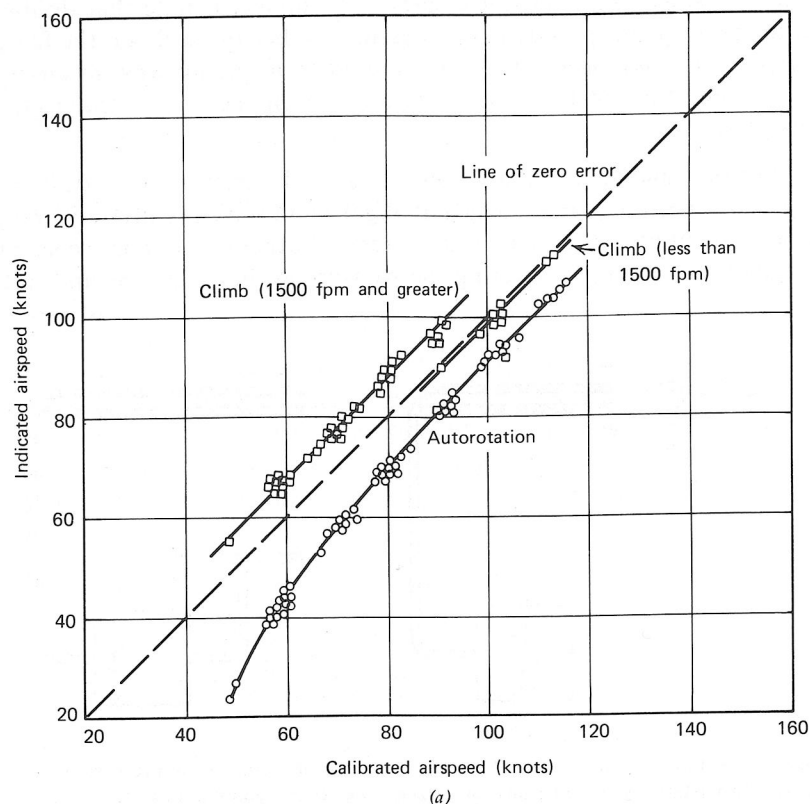


Figure 2.7 (a) Airspeed calibration in tandem rotor cargo helicopter. (U.S. Army data.)

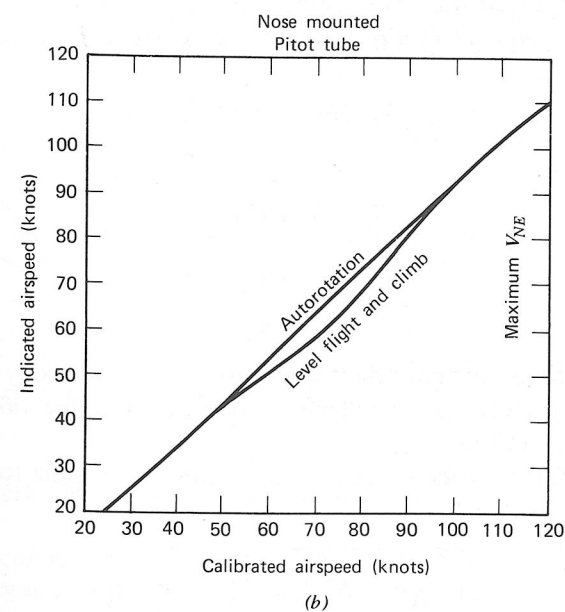
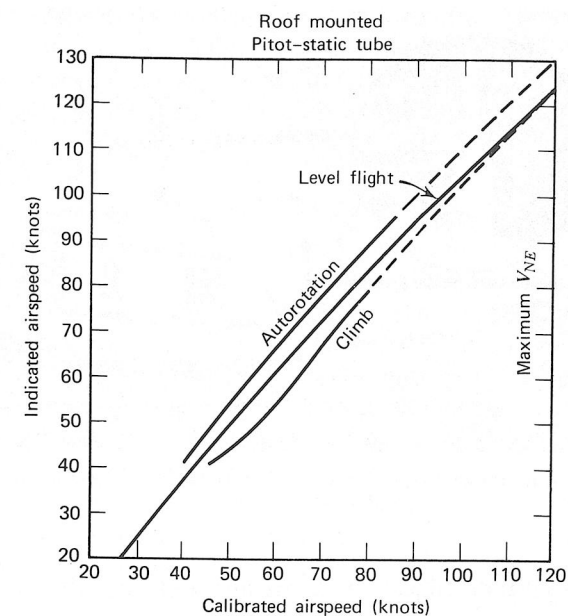


Figure 2.7 (b) Airspeed calibration in utility helicopter. (U.S. Army data.)

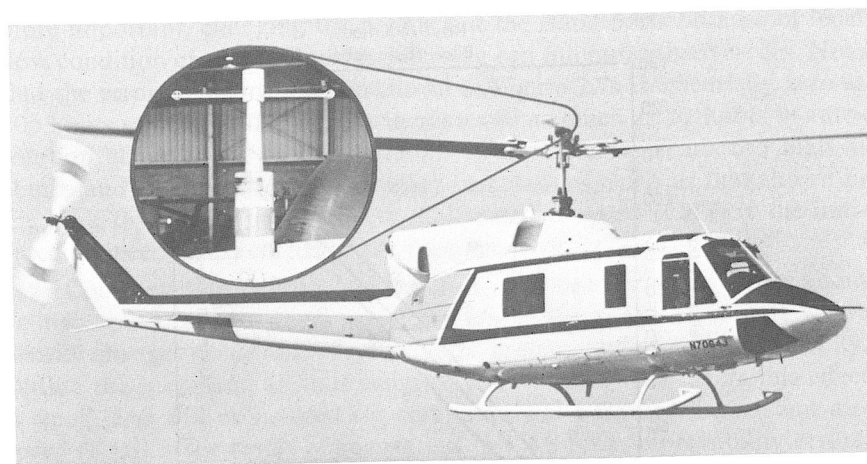


Figure 2.7 (c) Low range airspeed system. (Courtesy Pacer Systems, Inc.)

3. Density altitude effects: since the airspeed indicator is calibrated for the dynamic pressures corresponding to airspeeds at standard sea level conditions, variations in air density must be accounted for. The true airspeed (TAS) coupled with the existing density, produces the same dynamic pressure as the equivalent airspeed (EAS) coupled with standard sea level density. That is,

$$(\text{TAS})^2 \rho = (\text{EAS})^2 \rho_0$$

or

$$\text{TAS} = \text{EAS} \sqrt{\frac{\rho_0}{\rho}}$$

$$\text{TAS} = \frac{\text{EAS}}{\sqrt{\sigma}}$$

This latter effect is very important and must be taken correctly into account in helicopter performance problems. Figure 2.2 gives the value of $1/\sqrt{\sigma}$ for any density altitude.

Combining all these effects results in the following formula for conversion from IAS to TAS:

$$\text{TAS} = \frac{\text{IAS} + \Delta V_i + \Delta V_c}{\sqrt{\sigma}}$$

Example. The helicopter shown in Figure 2.7a is in autorotation at 60 knots IAS. The pressure altitude is 4000 ft and the temperature is $+24^\circ\text{C}$. Ignoring compressibility, what is the aircraft's TAS (knots)? From Figure 2.7a, $\Delta V_i = +11$ knots, and from Figure 2.2 $1/\sqrt{\sigma} = 1.094$

$$\therefore \text{TAS} = (60 + 11)1.094 = 71(1.094) = 77.9 \text{ knots TAS}$$

The accuracy and sensitivity of conventional airspeed systems deteriorates rapidly at speeds below 30 knots, and new methods must be devised for this situation if IFR flight and/or speed stability augmentation is contemplated. Various new ultrasonic, electronic, and laser systems demonstrate improved accuracy, but often at high cost and complexity. It must be remembered that any measuring device measures conditions at the location of the sensor, which may or may not be the quantity sought. For helicopters, the main rotor wake greatly influences local flow conditions around the aircraft, and the airspeed of the helicopter itself, relative to the undisturbed airmass, may be very difficult to measure. We cannot here discuss the relative merits of all the different engineering approaches to the problem. We examine only one system, which has shown promise in helicopter application. The device is shown installed above a rotor in Figure 2.7c. It consists of a small constant speed rotor with shielded pitot tubes on each end. The difference in pressure between the two is sensed by a transducer in the hub. When the helicopter velocity is zero, the pressures are identical and no signal is produced. If the vehicle is moving (or the wind is blowing past the helicopter) the relative velocity is increased on the "advancing" side and decreased on the "retreating" side; hence a difference in dynamic pressure is measured, proportional to the square of the airspeed. Notice also that by knowing the rotor azimuth when it experiences the maximum pressure difference, the *direction* as well as the magnitude of the airspeed are obtained. Thus an accurate measurement of sideslip is possible. Measurement of airspeed, both forward and sideward components, have been measured with an average error less than 1 knot down to zero airspeed. Backward velocities are measured with equal precision. A system of this type, with its plane of rotation mounted vertically, measures forward and downward velocity components, hence aircraft angle of attack.

2.4 DEVELOPMENT OF LIFT AND DRAG ON AIRFOILS

The cross-sectional size and shape of a helicopter blade is carefully chosen to give the best overall lift, drag, and pitching moment that are possible within given structural, aeroelastic, and manufacturing constraints. Sym-