AVDASI3
Initial sizing process for helicopter design

1 INTRODUCTION

This guide is an amended version of "Guidance for sizing helicopters", written by Leonardo Helicopters (former AgustaWestland).

The first steps in the project design are all about selecting a start point for the aircraft weight and major dimensions so that a synthesis of the performance (**a power model**) can be created. Once a power calculation can be performed a better assessment can be made for the **mission fuel** and in addition the required capabilities of the various systems **(transmission, engines, rotors etc.)** become apparent and some assessment can be made for their **size** and **weight**. From this point onwards the aircraft will be "built up" by defining and summing its component parts. Therefore to start the search for a solution the major requirements of the specification (weight of payload, mission fuel, specific performance points required) must be extracted and used together with very simple logic, data or experience to lay out this starting point. **There is no right answer at this point but an enlightened selection of parameters will save time in iterating to a compliant solution.**

2 PARAMETER SIZING

The first and most important dimension that needs to be derived is the likely **aircraft All-Up-Weight *(AUW)*** associated with the conduct of the primary mission. From this the other component dimensions can be pieced together using **rules of thumb** and inspection of other successful rotorcraft solutions.

The following notes are provided as some guidance to the initial sizing process described above. They also reflect the logical order for selecting the key parameters.

1. Use the aircraft specification to determine the primary missions and therefore the weight of their relevant payloads.

2. Either use knowledge of other products with a similar role to judge a mission fuel load or if necessary use cruder approaches to estimate a fuel load. A possible approach could be to assign a weight of fuel which is some proportion of the mission payload weight. While this is very imprecise it provides a reasonable scaling to the fuel load. No guess or estimate will take the place of a proper mission calculation which clearly requires the use of a power required analysis. At this stage of the evaluation it is suggested that the fuel weight is initially set as **100%** of the payload weight.

3. The weight of the whole aircraft can be roughly estimated from the mission disposable weight (payload plus fuel). A **disposable load fraction** in the range **25% to 30%** of the AUW for small helicopters and **35% to 40%** of the AUW for large helicopters can be expected, with the lower figure reflecting an aircraft with more ancillary equipment installed to do a more specialised job. This could be an aircraft with say a search and rescue role or paramilitary tasks and equipment. The weight of the crew, carry-on equipment and mission furnishings will increase this net weight by some **10%** for small helicopters and about **5% to 10%** for large helicopters.

4. With an *AUW* selected it is necessary to consider the dimensions of the rotors. Whether considering a single main rotor aircraft or a tandem, the dominant dimension is the main rotor diameter which is usually driven by considerations of disc loading (*AUW*/Disc area) since this parameter has a leading influence on hover capability. Disc loading has a traditional relationship with both the size of the helicopter and the configuration with generally higher values being observed on large single main rotor aircraft. While some highly specialised or stretched larger helicopters have disc loadings greater than **12 lb/ft^2**, a new design for a large transport type of helicopter would favour a lower disc loading to provide margins for in-service weight growth. Certainly numbers as low as **10lb/ft^2** might be expected. A similar class of tandem helicopter would probably have a disc loading some **2lb/ft^2** lower. At the small end of the range of helicopter sizes typical disc loading values can be found well below **10 lb/ft^2** *(typical values* **4-10 lb/ft^2***).*

   Since a large rotor diameter will have a beneficial influence on hover power it is tempting to settle on a solution with an extremely large disc area for the weight of helicopter. The counter to this solution is the need to manage the dimensions of the aircraft to fit in a transport aeroplane. Other practical considerations exist. For a single rotor aircraft a large main rotor will lead to inefficiencies in the aircraft structural weight, caused by the excessive length of the tail-boom and the need to extend the cockpit and forward structure to retain mass balance. If the blades are to have a reasonable aspect ratio and satisfactory torsional stiffness the solution for the rotor blade number will also be unsatisfactory with the rotor design being driven towards a low blade number with strong vibration penalties. These issues would be encountered during the detail design stage for the aircraft systems late in the design iteration loop during the detailed analysis of the system designs.

For a tandem configuration the use of very large rotor diameters would lead to a large disc overlap if the airframe size is to be contained and this would introduce enlarged lift inefficiencies and potential problems with blade clearances between the two discs.

5. The parameter which, perhaps, has the most significant influence on all of the mechanical and dynamic systems is the main rotor tipspeed. It has a strong scaling influence on the blade area, defines the overall gear ratio between the engine output shaft speed and the rotor turning speed and together with the diameter selection defines the critical frequencies that will drive the dynamic design of both the rotors and the structure. Ideally, for overall weight efficiency considerations, the rotor tipspeed would be as high as possible to minimise total blade area but it is in fact limited by two major considerations. These are concerns for the maximum advancing tip Mach number that can be sustained and the noise levels that the aircraft must respect.

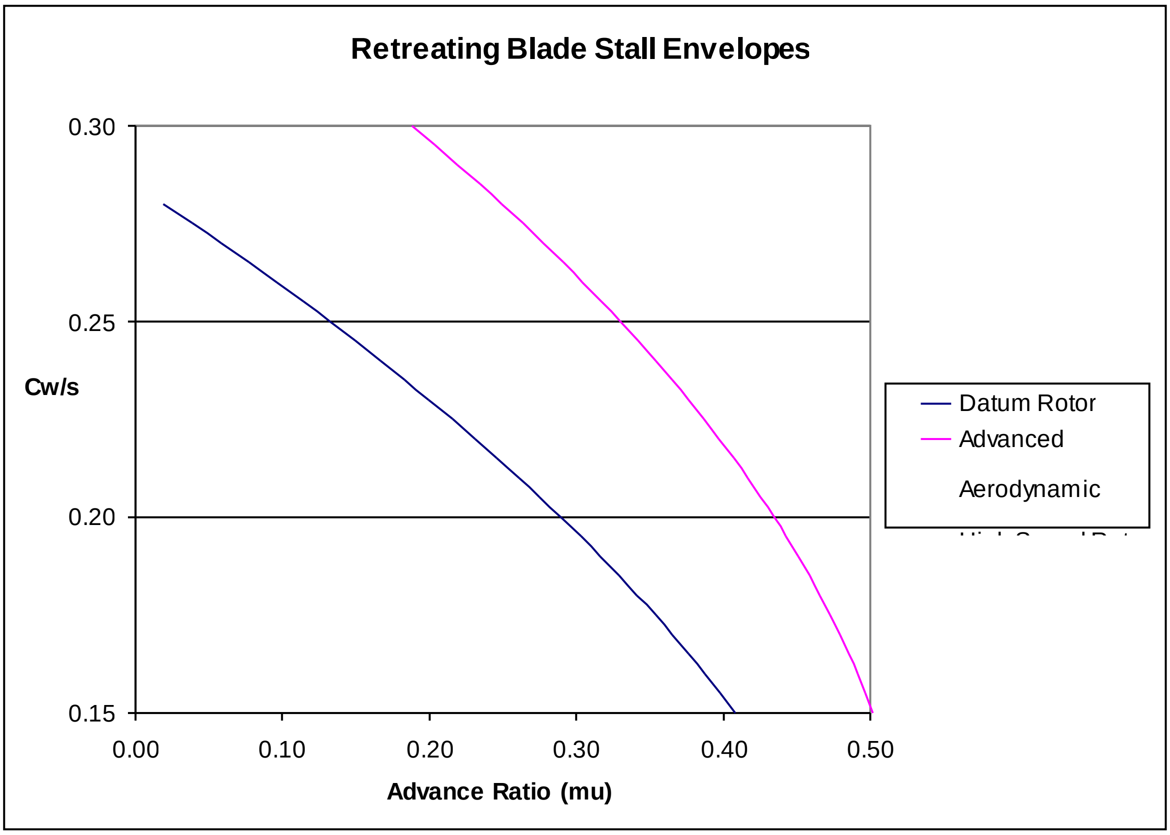
   The specification for the aircraft will normally define a number of conditions for cruise speed. The speed requirement which leads to the highest advancing tip Mach number should be identified to select the tipspeed. This is likely to be a flight case with very low air temperature (possibly very high on a cold day) such that the speed of sound is low. It may not be associated with a very high forward speed but still lead to the highest Mach number. The aerodynamic technology that will be used for the blades will determine the Mach number characteristics of the outer aerofoil sections including the critical and drag rise Mach numbers at zero incidence. It is normal practise at the aircraft layout stage to ensure that the effective Mach number at **95%** radius on the advancing blade always stays below the drag rise Mach number. Thus the critical cruise speed condition and the drag rise Mach number together define the maximum tipspeed that can be used. Later in the design the tipspeed will be checked to ensure that the noise character of the aircraft would be compliant with appropriate regulations.

For a tandem helicopter the two rotors must obviously turn at the same rotational speed. If one rotor is chosen to have a different radius to the other then the critical tipspeed must clearly apply to the larger of the two discs.

6. The tail rotor tipspeed (for a traditional single main rotor helicopter configuration) is not necessarily the same as the main rotor tipspeed. Noise considerations usually require the tail rotor to have a lower tipspeed value than the main rotor to ensure that it does not dominate the noise signature of the aircraft. It is usually found for noise reasons that the tail rotor needs to have a tipspeed some **20-30ft/sec** lower than that of the main rotor.

7. The main rotor parameters that determine the cruise speed capability are rotor tipspeed, blade aerodynamic technology and the total blade area ***(NCR).*** For this starting point it would be sufficient to inspect the normalised relationship between main rotor blade area and equivalent airspeed in cruise for other similar duty helicopters. By mapping this information as blade loading ***(C_w/S = AUW/NCR)*** against equivalent airspeed it should be possible to settle on a typical modern blade loading value. Given the usual wish to fly a new design of aircraft to higher speeds than the current generation it would be wise to start this design with a slightly lower blade loading than the current trend and consider the selection of better blade technologies when the design iterations have settled down. Thus it is suggested that a main rotor aerodynamic technology level of around **5** is used for the sizing. This lies midway between the technology of old helicopter types designed forty years ago and the very latest rotor aerodynamic designs of today.

8. In the preliminary design phase, rotor forward flight envelopes have been defined to cover a range of blade aerodynamic technologies, from relatively basic aerofoil and planform designs to advanced aerodynamic configurations. The designer is able to select the level of aerodynamic technology between these 2 flight envelope limits. For a specific blade aerodynamic technology these flight envelopes can be non-dimensionally expressed as rotor loading (CW/s $=T/(0.5*rho*N*c*R*(\Omega R)^2)$ and advance ratio (mu=$V/\Omega R$). These boundaries define the rotor stall envelope at VNE. The VNO boundary is defined as having at least a speed margin of **10%** to VNE with the capability at VNO of performing a **30** degree bank turn whilst remaining inside the rotor stall envelope.



9. To meet the aircraft performance requirements of a VNO for a certain aircraft mass when operating at the required altitude and temperature conditions, the total blade area (NcR) will be defined. To define this total blade area it will first be necessary to fix the rotor tip speed. The higher the tip speed the more rotor noise will increase. Too low a tip speed will, among other things, increase the rotor blade area and reduce the forward speed capability for a given advance ratio. Knowing the blade radius dimension will define the parameter which is the product of chord and number of blades.

10. A sufficiently stiff blade in torsion must be maintained, to avoid cyclic stick gradient reversals in high speed flight as a result of excessive blade torsional motion. To achieve this, then small values of blade chord must be avoided and "an absolute maximum blade aspect ratio (R/c) will be defined as **21**". This will provide a lower limit on blade chord for a specified radius and may require a re-assessment of blade chord and blade number, particularly where a high value has been selected for the number of blades.

11. If a too low a value for the number of rotor blades is selected then there will be a need for a larger blade chord which will increase the blade mass and associated hub mass. A guideline limit of rotor mass as a percentage of aircraft all up mass is defined as **10%**. It will become apparent later in the design that a

selection of a low number of blades will also induce higher vibration, so initial assumptions of blade chord and blade number may need to be reviewed throughout the design process, if inappropriate decisions are made at this stage.

12. The tail rotor diameter is likely to be something between **1/6th and 1/5th** of the main rotor diameter. There is some hint that larger helicopters, through scale issues, have proportionally larger tail rotors but this could also be driven by a variety of needs including aircraft compactness and manoeuvrability. The tail rotor and fin are often penalised in cruise because of their interaction with the turbulent wake of the main rotor hub and mast. In an attempt to get the tail rotor and fin away from this bad airflow they are often set high relative to the main rotor hub. This normally gives rise to a tall fin configuration with a small fin chord (to keep tail rotor blockage to a minimal level). This high positioning of the tail rotor allows concerns of ground obstacle clearance to be dealt with whilst still having a large diameter, efficient solution. It must be recognised that the height of the tail rotor centre relative to the main rotor plane must be chosen to ensure that in any flight state it does not produce an excessive roll couple with consequent extreme roll attitudes.

13. The tail rotor blade area would be set to provide a reasonable level of yaw manoeuvrability in the hover. A simple rule of thumb, which was originally used to size the tail rotor blade area for the EH101, was to ensure that at tip stall in hover the tail rotor could provide sufficient thrust to overcome the main rotor torque and produce a **1 *rad/sec^2*** instantaneous yaw acceleration with the tail moving to starboard. On the EH101 this gave rise to a tail rotor with a rotor solidity (*NCR*/Disc area) of **0.195**, with the tail rotor diameter of 4 metres.

14. Some assessment of the likely drag of the helicopter is needed. Based on a UK definition of dynamic head being $0.5\rho V^2$, larger transport helicopters could be expected to have a flat-plate-drag of **40-45** square feet (based on a drag coefficient of 1.0) while smaller transport helicopters fall into the **25-35** square feet range. A tolerable rule of thumb to provide a design starting point could be to assume that the ratio of helicopter AUW to the flat-plate-drag area is generally about **600-800 *lb/ft^2*** for small to medium helicopters rising to **1000*lb/ft^2*** for large helicopters.

    Synthesis of a believable airframe drag is not easy, it will include a number of subtle components examples of which might be the drag induced by cruise download or rotor interference effects on the body. Once the dimensions of the aircraft begin to stabilise it is important that the overall drag is evaluated from first principles using a source such as Hoerner (Fluid dynamic drag) or reference to previous wind-tunnel studies. The aerodynamic characteristics of the aircraft are usually measured in a wind tunnel for a full range of pitch attitudes and a wide yaw range.

15. In hover the rotor downwash produces a vertical drag on the fuselage. The precise value depends on the cross-sectional shape and the area distribution of the fuselage under the lifting portions of the rotor. In the early stages of the project a realistic typical value for this hover download needs to be identified. Good configurations are unlikely to be less than **5%** of the *AUW* and some squat designs perhaps with large quantities of external equipment could have a hover download of more than **10%** of the *AUW*.

16. The tail rotor suffers from a similar form of loss to the download effects of the fuselage. In the case of the tail rotor its induced flow over the fin creates cross flow drag which opposes the direction of the tail rotor thrust. When estimating the tail rotor power it is important that account is taken of this fin blockage. Typically the tail rotor must provide up to **10%** more thrust than an unimpeded rotor to overcome the fin blockage.

17. The aircraft uses a number of systems to provide various forms of internal power (electrical, hydraulic, and pneumatic). These power sources are generated from the mechanical power train of the helicopter and therefore represent a further part of the power balance of the aircraft. A typical small helicopter operating in benign atmospheric conditions would expend up to **20hp** to support these internal systems. In extreme conditions where specialised systems (like rotor anti-icing) are deployed or when the aircraft is conducting specialised operational tasks like sonar dunking or winching the internal power loading can climb to values in excess of **120hp** for a medium sized transport or naval helicopter. Very large helicopters would have larger internal power demands since a number of the tasks are dependent on size (rotor electrical de-icing, power to control the pitch of the rotor blades and others). A large helicopter can consume up to **200hp** dealing with internal power demands.

18. The power delivered by the engine into the helicopter gearbox does not all get through to the rotors. Each meshing stage within the gearbox is a source of friction and displaces and stirs the lubricant, further creating heat. A simple

rule of thumb is to assume that each mesh or gear step loses **1%** of transmitted power. Thus the gearbox would probably pass **96%** of the power output of the engine.

19. Similar to the internal power demands within the helicopter described above, an increasingly important power budget that should be accounted for is the provision of an Environmental Control System (ECS) powerful enough to manage extreme cold and extreme hot conditions. If the environment of the whole aircraft is to be tightly managed by the ECS the power loading can become considerable (up to **40hp** for a medium sized transport aircraft). If the major task of the ECS is to control the cockpit environment and manage the cooling of internal systems like the avionic boxes then the power consumed is less extreme. The detailed effect of ECS on the sizing of the engine installation goes beyond the increment of power that it demands. The way that the power is supplied (either as mechanical/electrical power or as compressed air drawn from the engine compressor stages) has a significant impact on the aircraft solution. In the early stages of the sizing of the solution it is usually sufficient to assume that any ECS power demands are achieved without the complicated efficiency considerations that come from the use of engine bleed air.

20. Engine capability is defined by the engine manufacturer on the assumption that the flow conditions both in and out of the engine are near perfect. These conditions reflect the environment of a static test-bed. The engines do not deliver power so freely or efficiently when installed in the aircraft. Because the aircraft intakes do not deliver the air symmetrically and without swirl to the engine and lose some pressure in the entry there is a loss of power and some increase in fuel consumption. Aircraft exhaust conditions also increase backpressure which diminishes the engine power output. On top of these realities the engines can also ingest hot gasses from the forward cowling area where cooling is ducted for the transmission and avionic bays. The overall effect is that an engine may lose up to **5%** of its rated power when installed in the aircraft and use up to **2%** more fuel in delivering that power.

With values assigned for the above parameters it is now possible to piece together the components of a representative power model for the helicopter. Use of such a model will allow an evaluation of the system parameters and provide the tool to make the aircraft self consistent with capabilities, weights and dimensions that fit together as a solution.

3 THE COMPONENTS OF THE BASIC POWER MODEL.

The following principles apply to both single main rotor helicopters and tandem helicopters. In creating a power model from these basic principles each helicopter type will have extra features (for instance a tail rotor) or features which are not ideal (for instance an overlap of the main rotor discs) and these therefore lead to type specific forms the relationships. For the most part these performance models are consistent with “Rotary-Wing Aerodynamics” by Stepniewski and Keys.

The power consumed by the aircraft is the sum of the work done in performing a number of duties.

First, whether in hover or cruise, the main rotor(s) must continuously do work in moving the air to create a thrust force. This flow of work, known as the induced power, is fed by the application of torque at the main rotor shaft. The size of this thrust vector would ideally be equal to the weight of the aircraft. However once the crude sizing process has indicated the architecture of a solution, a more refined power calculation is needed which must account for the download effects and ensure that all forces on the aircraft are in balance. The simple momentum based power model also assumes that the rotor has a uniform influence on the air and produces a downwash velocity field which is constant across the disc. This is the helicopter version of the fixed-wing elliptic loading. The downwash is not uniform and therefore the induced power is never as low as the ideal value. For this reason an “Induced power factor” is introduced, normally taking a value around 1.1 to 1.15 for main rotor blades, to empirically correct this error. The value of the induced power factor can be assessed explicitly as the project progresses using numerical blade element and vortex wake methods.

The area of overlap on the rotor discs of a tandem helicopter represents an additional loss of ideal efficiency and a further empirical factor is introduced to the induced power equation to account for this. Stepniewski and Keys provide information on the magnitude of this factor, K(ov), and show its relationship with the overlap geometry of the discs. For a typical tandem rotor configuration with an overlap of about **33%** the induced power is increased by some **13%** over the ideal value for two completely separated discs.

In forward flight the main rotor thrust vector is tilted forward to counteract the so-called “parasite” drag force on the fuselage and thus propel the aircraft. This additional force balance of course will also affect the size of the thrust vector.

This forward component of thrust gives rise to a second power term, known as the parasite power. Main rotor torque must also be supplied to overcome this power drain.

The main rotor blades themselves have profile drag and thus need some level of torque to be turned at the normal rotational speed. Thus the main rotor applied torque must also overcome this term. Again as the solution becomes better defined the non-linear effects of blade loading and the onset of drag rise and local stall need to be included in the profile power.

The shaft power to turn the main rotor(s) is the product of the rotational speed and the turning torque in the shaft.

For a tandem helicopter with rotors turning in opposite directions the net torque reaction applied to the aircraft is largely cancelled. Any residual torque due to differences in the power of the front and rear rotors is countered by a couple achieved through differential lateral tilt of the two discs.

The main rotor torque of a single main rotor helicopter is reacted by the tail rotor thrust in any steady flight state. Therefore the tail rotor will have induced power and profile power components that will determine the torque required to power it. The detailed definition of the aircraft systems requires a correctly balanced aircraft to ensure all forces are properly sized and therefore like the main rotor considerations, the tail rotor thrust will need to include any blockage effects to create a net force sufficiently large to counteract the main rotor torque.

The power required by the various aircraft internal systems must be evaluated and included as part of the energy account within the aircraft to ensure that its impact in terms of engine size and fuel used is included.

3.1 POWER REQUIRED.

Algebraically, the power components of a simple sizing model are therefore composed as follows.

3.1.1 Main rotor power

The main rotor power is comprised of;

$P_{(main)}$ = *Induced power + Profile power + Parasite power*

So, in units of horse power and where necessary feet and pounds,

$$P_{(main)} = \frac{K_m T_{(main)} Vi_m}{550} + \frac{1}{550} \cdot \frac{1}{8} \rho N_m C_m R_m Vt_m^3 Cd_m \left(1 + 4.7 \left(\frac{V}{Vt_m} \right)^2 \right) + \frac{1}{550} \cdot \frac{1}{2} \rho V^2 A_f Cd_f V$$

As mentioned above, a tandem helicopter would have a further penalising factor K(ov) applied to the first term (induced power) to reflect the inefficiency caused by rotor overlap. The value of K(ov) would be about 1.13.

Note that the thrust in hover must overcome both the AUW and the download and so is normally written as

$T_{(main)} = AUW(1+DL)$

Away from hover a simple analysis would assume that download became small.

Early sizing calculations would overlook the contribution of airframe parasite drag to the size of the thrust vector. However as the definition of the aircraft becomes more refined it is important that the thrust vector should be derived from a trim and balance analysis of the aircraft in forward flight such that both airframe download and drag forces are included.

The induced velocity of a rotor, based on momentum considerations and assuming the rotor produces a uniform pressure rise, takes the following form in hover;

$$Vi_h = \sqrt{\frac{T_{(main)}}{2 \rho \pi R_m^2}}$$

In cruise the momentum model can be developed as a guide to the way that induced velocity is related to forward speed. A general expression used for induced velocity in forward flight is

$$Vi_m = Vi_h \sqrt{-\tfrac{1}{2} \left(\frac{V}{Vi_h} \right)^2 + \sqrt{\left[\tfrac{1}{4} \left(\frac{V}{Vi_h} \right)^4 + 1 \right]}}$$

Where T(main) is the thrust of the main rotor

DL is the fuselage download increment

K_m is the induced power factor

K(ov) is the tandem rotor overlap factor

Vi_m is the induced velocity at the main rotor disc

Vi_h is the induced velocity (hover) at the main rotor disc

N_m is the number of main rotor blades

C_m is the main rotor blade mean chord

R_m is the main rotor radius

Vt_m is the main rotor tip-speed

Cd_m is the main rotor blade mean profile drag coefficient

V is the flight speed

ρ is the air density

A_f is the reference area for fuselage aerodynamic forces

Cd_f is the fuselage drag coefficient based on A_f

3.1.2 Aircraft torque balance.

The main rotor torque (which must be reacted by the tail rotor thrust) is therefore;

$$Q = P_{(main)} \cdot \frac{550}{\Omega}$$

Where Ω is the main rotor angular velocity in radians per second.

Clearly $\Omega = Vt/R$

3.1.3 Tail rotor power.

Assuming that the main and tail rotor centres are separated by a distance equal to the sum of the two radii plus a small clearance space (say **0.5 feet**) then the torque reaction force becomes;

$F_{(tail)} = Q/(R_m + R_t + 0.5)$

This resultant tail rotor shaft thrust must account for the presence of fin blockage and therefore becomes

$T_{(tail)} = F_{(tail)}\,(1 + B_t)$

The power consumed by the tail rotor therefore, like the main rotor, contains an induced power component and a profile power component and can be written

$$P_{(tail)} = \frac{K_t T_{(tail)} Vi_t}{550} + \frac{1}{550}.\frac{1}{8}\rho N_t C_t R_t Vt_t^3 Cd_t \left(1 + 4.7\left(\frac{V}{Vt_t}\right)^2\right)$$

Where, $T_{(tail)}$ is the tail rotor shaft thrust

B_t is the fin blockage increment

K_t is the tail rotor induced power factor

Vi_t is the tail rotor induced velocity at the rotor disc

N_t is the number of tail rotor blades

C_t is the tail rotor blade chord

R_t is the tail rotor radius

Vt_t is the tail rotor tip-speed

Cd_t is the tail rotor blade mean profile drag coefficient

3.2 NET AIRCRAFT POWER REQUIRED.

The power demanded by the aircraft and effectively supplied as the output of the transmission is therefore;

$P_{(transmission)} = P_{(main\ rotor)} + P_{(tail\ rotor)} + P_{(auxiliaries)}$

3.3 POWER DELIVERED BY THE ENGINES.

The power provided by the engines into the gearbox must reflect the losses through the gear-train and is therefore;

$P_{(engines)} = P_{(transmission)} / (1 - TL)$

where TL is the transmission power loss

3.4 USE OF THE CALCULATED ENGINE POWER.

The calculated helicopter power required does not yet account for the performance losses sustained by the engines because they are installed in airframe.

The use of fuel will probably be less efficient than the engine specification would indicate and it is usual to increase the fuel flow based on the calculated power required by **1-2%.**

The limiting performance of the aircraft as engine rated power is reached must also be treated with care. It is usual practice in assessing limiting performance to increase the calculated power required by the installation losses which usually amount to about **5%** for a good installation. This then represents the "thermodynamic" power state of the engine in delivering the required power level and can be compared to the rated power levels derived from the test bed or engine performance manual.