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HELICOPTER FLIGHT TEST INSTRUMENTATION,

by

10 Kenneth R. Ferrell

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HELICOPTER FLIGHT TEST INSTRUMENTATION

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SUMMARY

This document discusses the helicopter characteristics with which the instrumentation must contend and outlines typical tests that are conducted. Major aircraft components and systems which may be instrumented are listed and suggestions are made for sensors, locations, and installation. Details are provided for instruments peculiar to helicopters. Interface of the test instrumentation with data recording systems and ground support facilities are also considered.

A summary of instrumentation requirements is provided along with recommended range, accuracy and resolution. A sample instrumentation management technique is also included.

1. INTRODUCTION

The definition of flight test will vary as widely as the activity of those who are pursuing the subject. Perhaps the only consensus is that the vehicle be in free atmosphere as opposed to a wind tunnel or an enclosure. Scale models, unpiloted vehicles, tethered, or constrained vehicles are flight tested. The type of power or even the absence of power is not a decisive factor. Flight test may involve measurements, rely upon opinions, or simply be a demonstration of success or failure. This document will deal with flight testing of helicopters where it is necessary to record data which describe the vehicle operation and response to specified conditions and maneuvers.

Individual sensors, systems, and recording devices must be incorporated into the test vehicles in a manner that will meet the data requirements within the known constraints. The total instrumentation system is best designed by starting with the data requirements. Accuracy, data quantity, reliability, and physical characteristics of the test vehicle are first considerations. Sensors can then be selected within cost, availability, and installation limitations. Characteristics of the sensor are evaluated to determine recording system requirements. Calibration schedules for sensors and systems can then be established.

A helicopter flight test installation requires consideration of some parameters that are unique to the helicopter. In other instances the instrumentation is common to other airborne vehicles; however, special attention must be given to items such as recording range or sensitivity. The instrumentation must provide data which will allow the flight crew to establish flight conditions as well as record data for engineering applications.

1.1 Test Vehicles

The instrumentation discussed is directly applicable to helicopters. In a more general sense it is intended for any vehicle which operates in a low speed omnidirectional flight regime. The helicopters flight envelope introduces a need for special instrumentation. For example, it may be necessary to determine airspeed in all directions and the altimeter must be capable of operating over a large height band while retaining high accuracy for operations near the ground. Mechanical complexity is necessary to integrate engines, rotors, and control systems. Losses in the power transfer system are often small and are difficult to define accurately. The mechanical problems associated with rotating masses introduces a need to measure many angular motions, positions, and torques. The rotating members cover the range from engine speeds to very low shaft speeds. Transfer of information from rotating to stationary members is a particular problem. In addition, rotating parts experience various types of loads and quickly amass a high cyclic count. Testing must be accomplished to establish a fatigue life for each part. Vibrations in all directions are prevalent and encompass a wide range of frequencies and amplitudes. The aerodynamics of the helicopter produce stability and control characteristics that often require improvement through use of mechanical, hydraulic, or electronic systems. Complex control systems require that actuator motions and electronic inputs be measured. Instrumentation must consider each system relative to the basic flight controls. Rotor blade information can include static and dynamic pressures, positions, angles, and stresses. The blade tip may be in the transonic regime while reverse flow may exist at the hub. Blade instrumentation must have minimum influence on the lift and drag characteristics. The large volume of air displaced by the rotor at hover and low speed can have a strong influence on weapons firing or personnel working near the helicopter. In this flight regime the helicopter is usually near the ground and the downwash can introduce environmental problems associated with high velocity and hot engine exhaust gases. Combined rotor wash and flight in any direction may require measurement of unusually large angle airflows into the engine inlets and on lifting surfaces. The rotor ground effect is usually quite strong which mandates that heights within one rotor diameter above the ground be determined very accurately. Considerable noise is generated by the engines, power transfer mechanisms, and other rotating parts. In addition, the rotors contribute significant aerodynamic noise that may include a wide range of frequencies and magnitudes.

1.2 Types of Tests

The type of test being conducted will significantly influence the instrumentation requirements. A typical listing of tests is shown in Table 1.2-1.

TABLE 1.2-1

Typical Helicopter Flight Tests

Performance
- Hover Performance
- Take-off Performance
- Climb Performance
  - Vertical
  - Forward Flight
- Level Flight Performance
- Maneuvering Performance
  - Acceleration and Deceleration
  - Turning
  - Dive Recovery
  - Return to Target
  - Terrain Following
- Autorotational Descent Performance
- Landing Performance

Handling Qualities
- Control System Characteristics
- Control Positions in Trimmed Forward Flight
- Static Longitudinal Stability
- Static Lateral-Directional Stability
- Maneuvering Stability
- Dynamic Stability
- Controllability
- Ground or Deck Handling Characteristics
- Takeoff and Landing Characteristics
- Slope Landing Characteristics
- High and Low-Speed Flight Characteristics
- Power Management
- Mission Maneuvering Characteristics
- Effects of Weapons Firing
- Stores Jettison Envelope
- Instrument Flight Capability Aircraft Systems Failures
  - Simulated Engine Failure
  - Automatic Flight Control System Failure
  - Hydraulic System Failure
  - Tail Rotor Failure
- Autorotational Entries
- Autorotational Landings

Structural Dynamics
- Vibration
- Structural

Human Factors
- Cockpit Evaluation
- Night Evaluation
- Internal Noise
- Temperature
- Toxicity

Reliability and Maintainability

Subsystem Tests
- Engine Performance
- Aircraft Pitot-Static System
- Weapons System
- Electronic Equipment and Antennas
- Hydraulics

Environmental Aspects
- External Noise
- Radar Reflectivity
- Infra Red Radiation
- Downwash Effects

Types of instruments, ranges, accuracies and environmental aspects must all be considered. The optimum situation is to have a fully instrumented aircraft capable of recording all parameters. However, for some tests, satisfactory results can be obtained with limited instruments at considerable time and cost savings. The most exacting instrumentation requirements are for the performance tests. In these tests quantitative data are the primary results and subjective opinions are used to evaluate pilot ability and machine capability relationships. Power measurement is the most difficult and most important. Small helicopters often have engines in the range of 150 to 225 KW (200 to 250 SHP) and a one percent error is most difficult to measure. A limited amount of stability and control or user data can be obtained during the performance tests. Stability and control tests are a combined quantitative and qualitative effort. For these tests, emphasis is placed on flight control systems, aircraft motions, and positions. Power and atmospheric conditions are not as critical as for the performance tests. The data provide design information and establish flight capability and flight envelopes. Qualitative pilot comments are used to assess pilot workload and man/machine compatibility. Test pilots must relate their experiences with the test vehicle to the expected ability of the operational pilots. A very important part of these tests is the failure mode tests. Characteristics of the control system are evaluated in great detail and all possible combinations of failures are considered. Appropriate caution or warning notes are generated and placed in the pilots operating manuals. User tests will be peculiar to the mission of the organization or dictated by the aircraft characteristics. Those tests may be quantitative or qualitative. Operators can be either test pilots or user pilots. The instrumentation may be special test equipment or it can be the standard aircraft equipment. A common approach is to have combinations of the variations mentioned above. The user tests cannot be done with the quantitative accuracy that is possible in the performance or stability and control tests. The greatest difference is in the atmospheric conditions. Performance and stability tests are normally conducted in a stable air mass while the user tests are conducted in operational conditions. Turbulence, wind, snow, ice, rain, and dust are ever changing and create complex effects that are presently beyond our ability to account for or measure. Thus, the suitability of the machine is largely determined by the pilot comments or the capability to accomplish a specific task at a general set of conditions. While inexact from an engineering viewpoint, these tests are a good measure of the ability of the men to live with the machines and of the capability of the machine to accomplish the mission.

1.3 Instrumentation Environment

Helicopter instrumentation often must survive in conditions more adverse than are generally present during flight tests of fixed wing aircraft. Small helicopters have limited space available and various compartments may be used. The instrumentation system may have components separated, which can cause many electrical problems. Electrical power may be limited, and, in the case of transmission driven alternators, power may be interrupted at low rotor speeds. Throughout, the helicopter high vibrations should be expected. The amplitudes and frequencies can vary widely. Basic frequencies will be multiples of the main rotor speed. Superimposed will be the tail rotor frequencies as well as those from structural components and other rotating parts. The main rotor will generate in-plane and vertical vibrations. Fuselage vibration absorbers may be used. These absorbers are usually effective only within a certain frequency range. During operation at other frequencies they may amplify the basic vibration. Aircraft compartments usually have no environmental control and instrumentation placed there will experience a variety of conditions. Where there is no heating the compartment temperature will vary from 40°C in desert conditions to -25°C during high altitude tests. When instruments are placed in compartments near engines or transmissions special care must be used to determine compartment temperatures prior to installation. A marine environment leads to consideration of any salt spray that may occur. Tests in a desert situation generate dust and debris from the rotor wash. This dust can be very fine and dense and will probably enter any compartments not specially sealed. In addition to the cold temperatures, ice on the rotor and airframe can significantly change the vibration environment of sensors and recorders. Weapons firing tests generate severe local pressure variations and alter the aircraft vibration characteristics. Special landing tests such as minimum distance over an obstacle, shipboard landings, or autorotational landings may generate significant normal acceleration loads.

1.4 Systems Planning

The instrumentation system must be carefully planned to insure that the necessary data will be recorded in the best manner within the physical and cost limitations. The recorded data may be used in different forms or may be processed in several ways which require consideration of the data processing facility. The test objectives must be carefully analyzed to determine the required number of measurements. These results determine the size of the installation and the recording device. They also have an impact on the method of recording. Volume or weight conflicts may arise which dictate priorities among the desired information. Essential, desirable, and non-essential items can then be determined accordingly.

Test requirements set the initial data accuracy goal and then appropriate system characteristics are established. Required data accuracy must be considered for all system components. Transducer requirements are established and signal conditioning is designed. Throughout this effort the magnitude of the expected error must be known. Close coordination must be maintained between flight test and instrumentation engineers to insure that accuracy requirements are not overly stringent. Compromise or relaxation of the requirements may be needed to prevent escalating complexity or cost.

Helicopter flight tests usually require an instrumentation system with a wide range of dynamic response. Design emphasis in this area can have a most significant impact on the system. The parameters can be divided into low frequency atmospheric conditions and aircraft operation, medium frequency aircraft motion and response, and high frequency vibration and structural loads measurements. The data parameters should be grouped by dynamic response and maximum use should be made of electrical filters and multiplexing.

Decisions must be made regarding the form of the data recording and the data processing methods to be used (Ref 1). The minimum requirement will be dictated by the tests. The most common method is recording electrical signals on magnetic tape. The tape may be on board the aircraft or the data may be transmitted to a ground station. It is not unusual for both methods to be used. Documentation of the data can be accomplished by use of a voice track in the recording system or with written notations by the instrumentation operator. Provision should be made for automatic data numbering and data event markers. Event markers are extremely important for the flight crew to note significant data points or unusual occurrences during the test. Cockpit and/or ground playback and monitoring capability contributes to data validity and assurance that desired test data is being recorded. When feasible and cost effective the data should be machine processed. In most modern facilities the instrumentation and data processing systems are difficult to separate. Thus, it is mandatory to consider this interface when designing the instrumentation system.

1.5 Installation

The instrumentation installation must be designed to be compatible with the test vehicle, facilitate pre-flight inspection and maintenance, and to minimize crew workload during the testing. Access to the test vehicle or scale drawings are necessary to establish the location of instrumentation. The instrumentation buildup can usually be accomplished more easily in the shop than in the aircraft. The instrumentation layout must consider:

a) Accessibility for check-out and maintenance,
b) Structural integrity for flight safety, and crash worthiness,
c) Mass locations for aircraft weight and balance considerations,
d) Possible influence on vibration characteristics,
e) Convenience for flight crew operation.

An effective test program requires that the pre and post flight instrumentation activities can be accomplished in a minimum time. Centralized location of the necessary equipment reduces time and eases checkout or correction procedures. Maximum accessibility is gained by placing racks away from the sides of the compartments and by using a minimum of closed panels. Routing of cables should consider electro-magnetic interference as well as allow visual and electrical inspection.

The racks containing the instrumentation and instrument mountings must be designed to withstand specified loadings. The instrumentation should be able to withstand forces greater than the occupant seats or restraints to insure safety during an accident. Typical design is for impact forces of ±20 g's in each axis. Wires, cables, or other restraining devices should not present hazards during normal operations around or with the equipment.

The weight and location of each piece of equipment must be known. The instrumentation engineer should coordinate with the flight test engineer to consider the total weight of the instrumentation with respect to performance capability of the aircraft, and location of instruments or components with respect to the center of gravity and inertia. Weight and locations are often critical for small test vehicles. Common practice is to write the weight on the larger pieces of equipment. This provides a rough accounting during the installation. When the installation is complete an aircraft weight and balance is required to account for wiring and small miscellaneous items.

2. ATMOSPHERIC

2.1 Air Data Instrumentation

All flight tests require some measurements of atmospheric data. Measurements include pressure, temperature, liquid water content, dust or debris concentrations, humidity, and flow angles. The measurements may be devoted to the far field, relative to the aircraft, or local conditions at a component or surface. Special problems arise during climb and descent or dynamic maneuvers near the ground in various surface winds. In the latter case a ground station is often used to define the far field environment.

2.1.1 Free Air Temperature

Free air temperature systems must be installed so that they will receive a minimum influence from the aircraft. The sensor should be shielded from heat generating sources or from hot airflow. Solar radiation should also be considered. Common practice is to install a calibrated test system to record the data. This test system is then used as the standard to evaluate the basic aircraft temperature sensing installation.

The test sensor is usually mounted on the airspeed boom. When a boom is not available the sensor is often mounted on the underside of the nose of the aircraft. Many test sensors have a de-ice capability, however care must be used to insure that the de-ice is on only at the specified conditions. Typically, operation above 0°C or below 30 m/s (59 Kn) will introduce a 2°C error. The activation of the de-ice may be manual or automatic. The system must include a cockpit indicator for use in establishing flight test conditions. This indicator should have at least 1°C increments.

Helicopter flight test temperature conditions may vary from climatic hangar or arctic tests at -55°C to a desert condition where the temperature is 55°C. For other than extreme environmental tests a commonly used instrumentation range is from -35°C to 50°C. A platinum element resistance probe is generally used to sense the free air temperature. Pure platinum has been selected as the international standard temperature measurement from -182.97°C to 630.5°C, and when properly used and calibrated, accuracy to 0.1°C can be realized in field operation. To achieve accuracies of this magnitude, care must be taken in both calibration and system integration of the probe. For a calibration covering the entire range of the platinum probe, measurements of probe resistance are made at four specific temperatures and these values are employed to generate values of resistance for any other desired temperature. This is accomplished through the resistance-temperature relationship for platinum which is given by the Callendar-Van Dusen equation:

$$\frac{R_T}{R_o} = 1 + \alpha \left[T - \delta \left(\frac{T}{100} - 1\right)\left(\frac{T}{100}\right) - \beta \left(\frac{T}{100} - 1\right)\left(\frac{T}{100}\right)^3 \right]$$

where R_T is the element resistance at T°C, R_o is the element resistance at 0°C, and α, δ, and β are constants for each individual platinum element. A platinum probe system provides greater output voltage, therefore has greater tolerance to noise than thermocouples and does not require a reference junction temperature or other compensating device. The platinum probe is superior to most other methods of on-board temperature measurements, but care must be taken in signal conditioning to insure that effects such as self heating do not occur. Signal conditioners specifically designed for platinum probes are available and can produce excellent results. For instrumentation systems requiring both cockpit display and data recording, dual element probes are available to prevent undesirable interaction of electronics.

2.1.2 Altitude

Altitude measurement is accomplished in terms of atmospheric pressure and height above the ground. The data is required in the cockpit so the pilot can stabilize at an altitude or maintain a prescribed path relative to the ground. The information is also made available to the instrumentation system. Typically, pressure altitude measurements will range from 60 m (200 ft) below sea level to 7500 m (25,000 ft) during climbs to service ceiling. Radar altimeters are often required for operations less than 300 m (1000 ft) above ground level.

Test system static and pitot sources are placed in a location which will minimize effects from aircraft and best reflect the true atmospheric conditions. When possible these sensors are placed on a nose boom. The static pressure is connected to both cockpit indicators and instrumentation transducers. For most altitude applications, a standard indicator is used with 6 m (20 ft) resolution. This type of indicator is generally acceptable for pilot information. The instrumentation system altimeter can be a strain gage pressure transducer, capacitive transducer, or other suitable transducer. The capacitive transducers come in both analog and digital output formats. Temperature effects can be sizeable, and therefore should be quantified for necessary correction by appropriate circuitry, enclosure of the transducer in a temperature controlled oven, or data manipulation during analysis. Care should be taken to provide any necessary preflight warmup time for these transducers. From one to thirty minutes may be required for proper stability. With proper installation and appropriate data correction, accuracies of better than ±34 Pa (0.01 in of mercury) can be realized which meets the requirements of most applications.

Very accurate height above ground information is often needed during hover and take-off and landing tests, and a radar altimeter is used to supplement the pressure altitude measurement. The antenna for the radar altimeter is mounted to insure no return signals from the airframe. This is a particular problem for aircraft with fixed gear or skids. The cockpit indication of radar altitude is of special importance during tests near the ground and in some cases a .3 m (1 ft) resolution is required. In all cases the pilot must know height within 3 m (10 ft). A typical radar altimeter is the Honeywell model APN-171. This altimeter has an accuracy of ±.5 m (1.5 ft) plus one percent plus five percent of the average range rate, and offers a test mode switch for system checkout and preflight. Auxillary outputs are used on the radar altimeters to provide inputs to the instrumentation system for recording both absolute height and rate of change of height. These signals are most often in analog format.

2.1.3 Humidity

At a given atmospheric pressure and temperature, humidity can affect helicopter performance relative to dry air by a decrease in power available or an increase in power required. In the first instance, the humidity decreases air density and thus mass flow through the engine; and in the second case, the rotor will experience an effective increase in density altitude. The effect of humidity can cause several percent error in the density which can have a significant impact on helicopter performance. Humidity effect, while large in theory, have not adequately been measured in flight. However measurements of humidity should be made in order to build up a data bank for further analysis. The criticality of density changes increase with higher temperature and higher relative humidity.

The density can be measured directly with nuclear radiation devices (Refs 2 and 3). The accuracy of the referenced devices are 1 to 2% as they existed at the time. Increased accuracy can be obtained by increasing the signal strength. However, extreme care must be used relative to the radiation hazards. Electronic hygrometer equipment is also available to measure the relative humidity directly. Quoted accuracy is ±1.5%. When engine power is being corrected for humidity, the measurement must be recorded for each test condition.

Independent measurements of free air temperature and dew point allow calculation of relative humidity and the effect of air density (Refs 4 and 5).

2.1.4 Icing

Helicopter icing tests require that the water characteristics of the cloud be measured for correlation with ice accretion and effects on the performance or handling qualities of the helicopter. Measurements include droplet size and distribution as well as liquid water content. The airflow characteristics around the helicopter are extremely complex for other than high speed flight, and it is difficult to find a sensor location which is free from aircraft disturbance or contamination. While it is expected that ice will accrete on all parts of the aircraft, it is not practical to measure ice thickness on blades or other rotating parts. It is common practice to paint or tape the blades in a grid which identifies span and chord locations. Photographs are then taken to establish patterns and amount of ice accreted. Those determinations are correlated to the atmospheric conditions and accretion measured on the fuselage or other non-rotating parts.

Droplet size can be determined by various types of impact measuring devices. Slides coated with oil, gelatin, or carbon are exposed to the airstream for a short period of time. The droplets are either captured by the surface or leave marks representative of their size. Examination under a microscope allows determination of size and distribution. Another technique involves a water sensitive tape or paper which is continuously moving behind a slot exposed to the cloud (Ref 6). This provides a time history of the droplets being encountered. Droplet size can also be determined by the rotating cylinder method. This method exposes cylinders of various diameters to the airstream with their axis perpendicular to the airflow. The cylinders are rotated slowly so that the ice build up is uniform. The collection efficiency of each cylinder is different and thus accretes ice from different droplet sizes. From the amount of ice on the different cylinders a profile of droplet size and distribution can be constructed. For other than conditions of cold temperature and low liquid water content the cylinders have limitations which can cause significant errors (Refs 7 and 8). A newer method uses a laser driven spectrometer (Knollenberg probe). This instrument operates on the principle that the laser light will be scattered by the droplets as they pass the light beam. An optical system collects the scattered light and through electronic means the practical size and distribution is determined (Refs 9, 10, and 11). Each probe is designed for a range of droplet sizes and care must be taken to insure that a sufficient number is used to encompass all the droplet sizes. The output from the laser system can be recorded on magnetic tape, or with proper equipment, can be viewed in real time.

Liquid water content can be calculated from accreted ice or measured directly in the atmosphere (Ref 12). The previously discussed rotating cylinders accrete ice which can be removed and, in conjunction with the collection efficiency can be used to calculate the liquid water content. The visual ice detector probe has a small airfoil with a steel rod protruding forward of the leading edge. The protruding rod is marked or color coded in increments for visual or photographic documentation of ice accretion. The buildup on the rod gives an indication of ice accretion on non-aerodynamic surfaces, while the airfoil is indicative of conditions on lifting surfaces and may correlate with main or tail rotor conditions.

The Rosemount ice detector uses magnetostriction to drive a sensing probe at its natural frequency. As the probe accretes ice, the natural frequency changes due to the increased mass. The change is calibrated in terms of ice accretion rate. The calibration of such a system must take into consideration factors such as airspeed which affect ice accretion. When the ice thickness reaches a predetermined value, the probe is deiced and the cycle repeated. Cycle counting can be used to obtain total ice accretion. The probe is housed in an electrically heated aspirator shroud which uses engine bleed air to induce ambient airflow over the probe during hover and low airspeed.

The Leigh ice detector consists of a light emitting diode/photo transistor assembly which provides an optical path that is partially occluded by accretion of ice on the ice detector probe. The assembly is encased in an annular duct and ejector nozzle which is supplied with bleed air to induce high velocity airflow over the ice collecting probe and provide anti-icing. When the ice accumulation reaches a pre-set level the probe is electrically deiced and the cycle is repeated. The icing signal is displayed on cockpit indicators and recorded by the data system. Cycle counting is used to establish total accumulation. Electronic circuitry is incorporated which calculates rate of accretion during each cycle.

The hot film anemometer is an electrically heated surface which is one leg of a wheatstone bridge network powered by the output of a high-frequency, high-gain, differential amplifier where bridge unbalance determines the amplifier output. When a water droplet impinges on the sensor it is abruptly cooled. The resistance of the sensor is highly temperature dependent and the cooling causes a bridge unbalance which is sensed by the differential amplifier. The amplifier applies sufficient power to the bridge network to return the sensor to equilibrium temperature. The number of cycles indicates the droplet distribution and the applied voltage shows the droplet size. Calibration data are then applied to calculate droplet information and liquid water content. The frequency response of the system is critical with respect to the distortion and attenuation of the droplet data signal in the processing and recording portions of the system. Large droplets or multiple droplet strikes may cause data loss if the temperature does not recover before the next strike occurs. Network noise must be minimized in order for the output from small droplets to be recognizable.

2.2 Airspeed and Relative Wind Direction

Standard airspeed systems are oriented for conventional level flight. These systems are fixed pitot-static differential pressure probes. They are usually designed and arranged to show a minimum error at cruise airspeed. The threshold is relatively high and the systems are often unuseable at indicated airspeeds below 15 m/s (30 Kn). In level flight, sideslip effects on position error normally increase with airspeed. Sideslip angles up to 5 degrees will not usually introduce noticeable errors. This is an important consideration since most helicopters have sideforce cues which allow the pilot to stay within these limits. Very steep climbs or descents will cause noticeable shifts in the position error for angle of attack or sideslip, and separate calibrations are accomplished for those flight regimes (Refs 13 and 14). Standard cockpit indicators normally have 2.5 m/s (5 Kn) increments and the instruments are not calibrated but are accepted with a specification accuracy tolerance. Thus, the accuracy of a particular instrument in a given aircraft is not known. When the standard sensors are used for test data it is common practice to mount a test indicator in the pilot's instrument panel. The test indicator is calibrated and can be read to 0.5 m/s (1 Kn) increments. During the test, data will also be taken from the copilot's standard instrument to obtain information concerning what the operational pilot will be seeing. This standard instrument is often satisfactory for user or flying qualities evaluations.

2.2.1 Swivel-Head Test System

To minimize effects of angles of attack and sideslip or pressure distortions near the fuselage, a swiveling pitot-static probe is mounted on a boom extending forward from the helicopter. Ideally, the boom length should place the sensor beyond the rotor downwash in hover. However, this is usually not practical. When the boom cannot be placed on the nose, other possibilities include wing, fuselage, or vertical stabilizer mountings. Unusual airflow and vibration conditions should be expected in these locations. Consideration must be given to the natural frequency of the boom to prevent excessive rotor induced vibrations. The boom is attached to the airframe on one end and supported by cables from the other end. A typical boom is 2.5 to 3 m (8 to 10 ft) long and is constructed of 2024-T3 aluminum with a 4.45 cm (1-3/4 in) outside diameter with a .32 cm (1/8 in) wall thickness.

The boom and pitot-static probe combination can be precisely aligned with the aircraft axes by use of survey equipment. The center line of the aircraft as determined from known airframe reference points is projected approximately 15 m (50 ft) in front of the vehicle. The offset distance from the boom mounting point on the fuselage and the aircraft center line is then determined. A transit is placed at the end of the center line projection and offset laterally the same as the boom mount is from the aircraft center line. By sighting the length of the boom through the transit, adjustments are made to the boom until it is parallel to the line of sight. This insures the boom is aligned with the aircraft center line. A similar method is used with a plane through a water line to assure proper angle of attack alignment. Free stream angle of attack and sideslip are obtained by mounting movable vanes on the airspeed boom. A vertical vane gives sideslip and a horizontal vane provides angle of attack. Suitable fixtures are constructed to calibrate the vanes relative to the boom. A typical swivel-head probe and vane installation is shown in Figure 2.2.1-1.

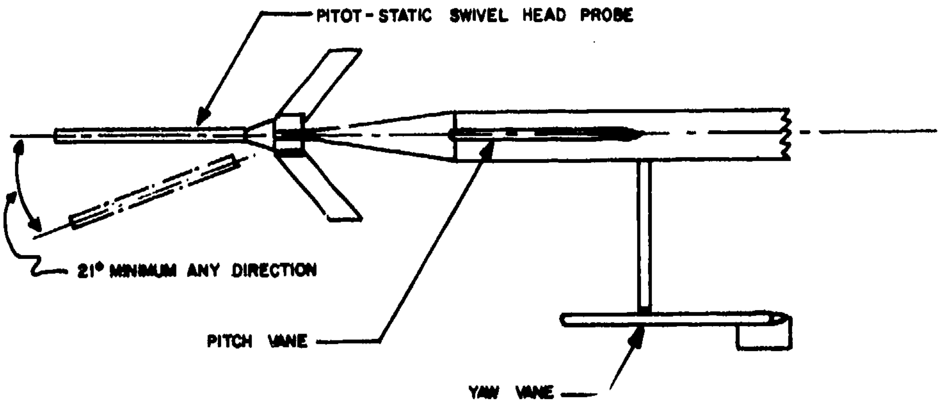

Figure 2.2.1-1
Boom Installation of Airspeed, Angle of Attack, and Sideslip System

A suitable pitot-static probe has a swivel head gimbal freedom of at least 20 degrees in all directions, airspeed capability up to 100 m/s (200 Kn), angle of attack and sideslip motion of ±45 degrees. Airspeed error should be less than 0.5% at angles of attack and sideslip to 30 degrees. The vanes are mass balanced and airfree to align with the relative airflow. Since the positioning torque is low, the sensor must have low friction bearings and extremely good balance. Potentiometers and bending beam strain gages are the most common methods of sensing angle of attack and sideslip vane positions.

Pitot-static systems are usually volume balanced to eliminate airspeed indication errors caused by differential pressure lags in the two circuits during climb or descent. The pitot and static circuits are trial and error balanced by applying a pressure or vacuum to both the pitot and static sources simultaneously. The pressure or vacuum is then bled to ambient pressure at a constant rate ∿20 m/s (4000 ft/min) and the pressure differential read on the installed sensitive airspeed indicator. Care should be taken to avoid over pressuring the airspeed indicator, particularly in the negative direction. A known volume (∿160 cc (10 cu in)) is then added, usually to the pitot side and the process repeated. The final volume to be added can then be calculated by linearly extrapolating/interpolating the change in differential pressure caused by the known change in volume. Usually two iterations are sufficient to balance the systems within 5 m/s (10 Kn) at 20 m/s (4000 ft/min) vertical rate.

2.2.2 Omni-Directional Airspeed Systems

Many helicopter tests require airspeed information at low airspeeds and in various directions. Several systems have been developed which provide data in hover, vertical climb and descent, and during sideward or rearward flight. These systems are also operable in high speed conventional flight. Hover performance is very sensitive to relative wind which must be measured within 0.5 m/s (1 Kn). The wind direction can also affect the power required or critical directional control margin and should be measured with an accuracy of ±2 degrees.

Location of the sensor is critical since it is desired to measure aircraft velocity and not local flow conditions. Rotor wash is the largest single factor, although disturbed flow from the fuselage, wings, or stores must also be considered. It is expected that each installation on a particular aircraft model will be unique and the system will require a flight calibration to determine the position error. Typical changes in position error with sensor location are shown by Ref 15. Most of the low airspeed systems have been developed further since the referenced tests were completed. In fairness to all manufacturers and to avoid misinforming the reader, resolution, threshold, and accuracy numbers will not be presented here. Capabilities of the various systems are summarized in Table 2.2.2-1. Performance and special characteristics of the systems as they were tested may be obtained from references 16 through 21.

TABLE 2.2.2-1

SUMMARY OF OMNI-DIRECTIONAL AIRSPEED SYSTEM CAPABILITIES

System	Longitudinal Airspeed ∿ Kn	Lateral Airspeed ∿ Kn	Vertical Airspeed ∿ Ft/Min	Angle of Attack ∿ Deg	Angle of Sideslip ∿ Deg
Aeroflex	0 Rearward to 250 forward	50 left to 50 right	None	None	± 180
Elliott	40 rearward 150 forward	40 left to 40 right	0 to 4000 up	± 180	± 180
J-TEC	30 rearward to 130 forward	40 left to 40 right	None	None	± 180
Loras	50 rearward to 150 forward	50 left to 50 right	None	None	± 180
Rosemount	40 rearward to 60 forward	50 left to 50 right	None	None	± 180
Honeywell	50 rearward to 200 forward	50 left to 50 right	0 to 5000 up	0 to 90 up	± 180

NOTES: 1. Data shown are for the sensor mounted vertically. Forward or lateral mounting will change the capability in the various axes.
2. With the exception of the Elliott, rotor downwash will adversely affect performance.

a. Aeroflex

Aeroflex Laboratories, Inc., Plainview, Long Island, New York, was responsible for development of the true airspeed vector system (TAVS). The TAVS consists of an airstream direction sensor, a true airspeed sensor, a visual indicator, and the associated electronics. The corresponding sideslip angle in degrees and true airspeed in knots are available as DC signals suitable for recording on an oscillograph or magnetic tape system.

The airstream direction sensor consists of four hot-wire sensors (bolometers) mounted on top of the airspeed stream tube. The bolometers form an error-sensing bridge of the airflow direction with respect to the longitudinal axis of the stream tube. Airflow at an angle to the turbine duct causes the right and left bolometers to be cooled unequally, which unbalances the error-sensing bridge. A servo system then rotates the pylon until the bridge is balanced and the stream tube is parallel with the airflow. A vertical stabilizer is externally mounted on the aft portion of the stream tube to provide directional stability at high speeds.

The true airspeed sensing unit is a hollow tube mounted on a pylon base. The forward portion of the tube contains a honeycomb structure, which assures axial flow at the inlet and also creates turbulent flow through the stream tube throughout the speed range of the sensor. The rear portion of the tube contains a 16-blade turbine and two "V" bolometer assemblies aft of the turbine. The principle of operation is based on the premise that for a given airflow through the turbine duct, the turbine can be rotated at a speed (synchronous speed) which will permit undisturbed axial flow. An illustration of the sensor and a vector representation of its operation is shown in Figure 2.2.2-1. If the turbine is not at synchronous speed with the airstream ($V_{TR} \neq V_{TP}$), the two "V" bolometer assemblies sense the resultant airflow, V_R, as a deviation from axial flow, and a servomotor adjusts the turbine speed until the error signal is nulled. Thus, the component of the airstream parallel to the turbine axis is synchronous with the turbine speed.

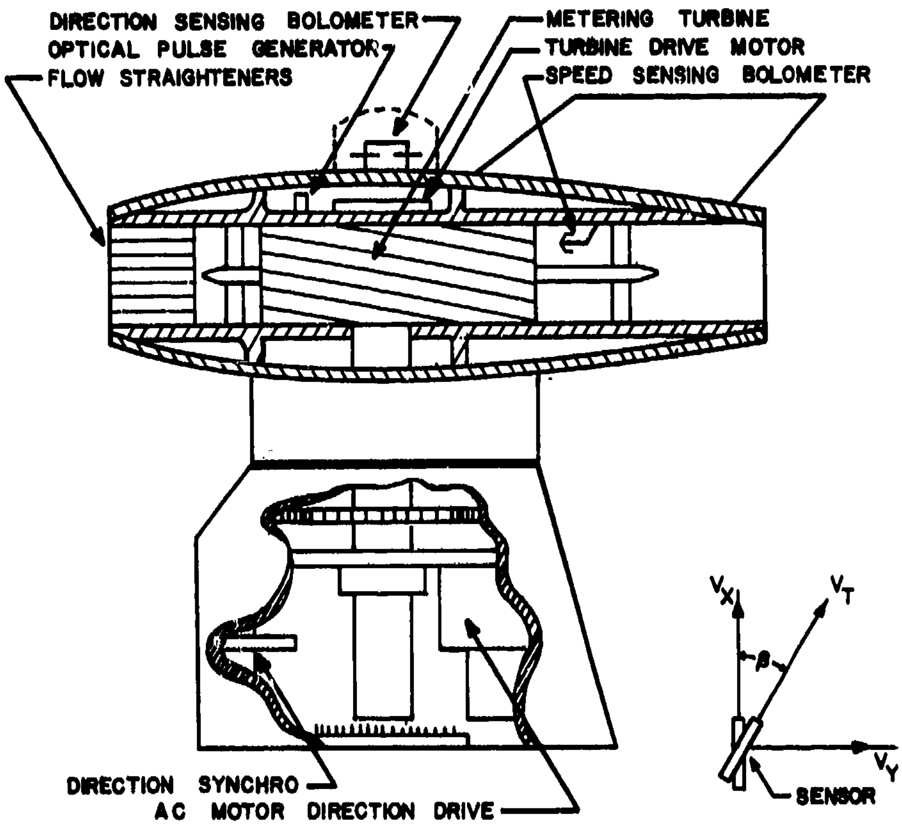

Figure 2.2.2-1
Aeroflex Airspeed Sensor

The airspeed and direction sensors drive a visual indicator and provide DC voltage outputs. The indicator contains a roller-suspended, servo-driven tape, marked in .5 m/s (1 Kn) increments at its center, to display airspeed in the range of zero to 180 m/s (350 Kn). At the perimeter of the indicator face, a servo-driven ring continuously displays the sensor head position relative to the sensor base, through 360 degrees of rotation. The DC output has three separate output recording terminals. Each output consists of four buffered channels and can drive as many as four oscillograph galvanometers or similar recorders. The DC outputs consist of a coarse signal for airspeed (zero to 130 m/s (250 Kn)), a coarse signal for direction (zero to 360 degrees), and a fine signal for airspeed and direction which cycles every 13 m/s (25 Kn) and 36 degrees, respectively.

b. <u>Elliott</u>

The Elliott low airspeed system is manufactured by Elliott Flight Automation Ltd, Airport Works, Rochester, Kent, England. In the United States the equipment is the responsibility of E-A Industrial Corporation, Chamblee, Georgia, the associate company. The sensor and vector resolution is shown in Figure 2.2.2-2.

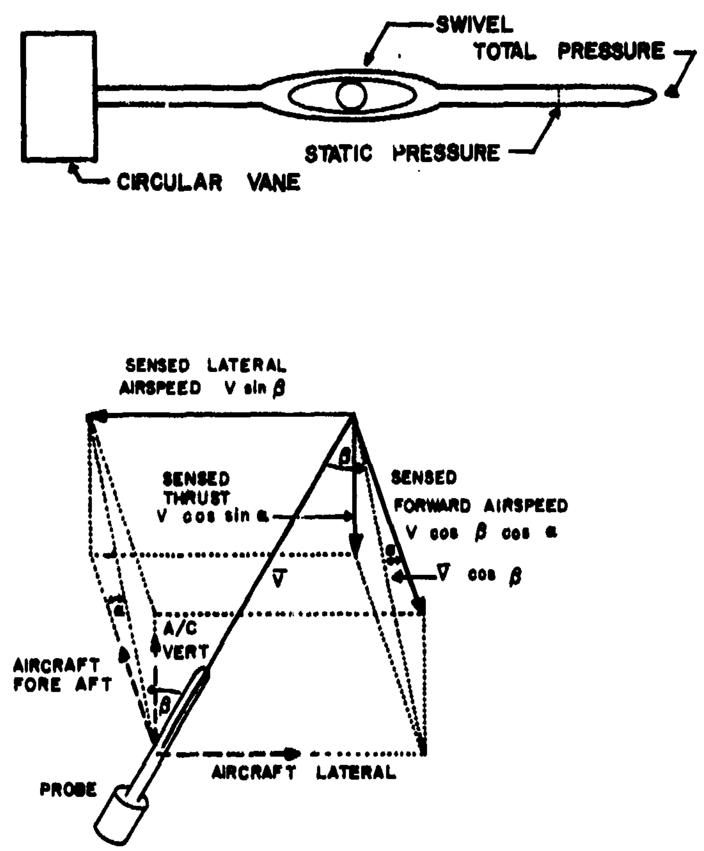

Figure 2.2.2-2
Elliott Airspeed Sensor

The system includes a swiveling total and static pressure sensing probe, a computer, and airspeed indicators for three axes. The resultant downwash aligns the probe with local relative wind (vector sum of aircraft velocity and rotor induced velocity) and provides adequate dynamic pressure at all airspeeds. The angle of the probe and the differential pressure are used to calculate aircraft speed and relative wind direction. Static pressure is measured and rate of change is calculated to provide rate-of-climb information. The airspeeds presented to the pilot are longitudinal, lateral, and vertical components. A resultant is not presented, nor is angle of attack or angle of sideslip calculated. Free air temperature is measured and the computer calculates true airspeed.

Individual longitudinal and lateral airspeed indicators (type 71-011-01) consist of a stepper motor and a feedback potentiometer. This provides an indicator rate signal and position signal which is fed back to the airspeed computer. The signals are summed with the computer longitudinal airspeed and are checked by the servo monitor. Detected failures are indicated by a warning flag on the indicator.

c. J-TEC

The VT-1003 vector airspeed sensing system is manufactured by J-TEC Associates, Inc. of Cedar Rapids, Iowa. The J-TEC vector airspeed sensing system measures relative wind speed and direction with no moving parts. The VT-1003 consists of a sensor head, an electronic processor, and an airspeed and direction indicator. The sensor is illustrated in Figure 2.2.2-3.

The sensor head consists of six identical tubes 6.67 cm (2-5/8 in) long, mounted radially on a 13.65 cm (5-3/8 in) diameter hub. It is mounted on the aircraft so that one pair of tubes is aligned with the lateral axis of the aircraft and the other tubes are 30 degrees either side of the longitudinal axis. The sensor weighs approximately 1.6 kg (3-1/2 lb).

Regardless of wind direction, flow exists in at least two adjacent tubes at any time, allowing two equations to be solved simultaneously for the two unknowns.

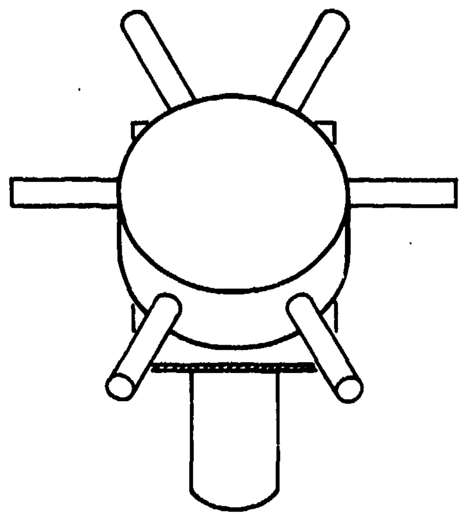

Figure 2.2.2-3
J-TEC Airspeed Sensor

At the inboard end of each tube, near the hub, is a vortex strut (a wire of known diameter) located just ahead of an ultrasonic transducer. As air moves through the tube and across the strut, a series of alternating vortices is created. The frequency of these vortices is directly proportional to true air velocity, and is independent of density. The vortices pass through an ultrasonic beam transmitter, modulating it. The modulation frequency is detected and is sent to its receiver where it is converted to an audio frequency signal.

The electronic processor and its case, a box 12.7 cm (5 in) wide by 20.32 cm (8 in) high by 50.8 cm (20 in) long, weighs 3.18 kg (7 lb). It converts the input audio frequency signals from the sensor to voltages, and determines which two adjacent tubes have the greatest velocities. The processor outputs two voltages proportional to longitudinal and lateral true airspeed. Typically, the calibration is approximately 100 mv/m/s (50 mv/kt). Airspeeds are calculated within the processor.

The cross-pointer indicator in the cockpit has a fixed display in the form of concentric circles 10 Kn (5 m/s) apart with zero located at the geometric center and 50 Kn (25 m/s) at the outer ring. The horizontal pointer moves up with increasing forward airspeed; the vertical pointer moves in the direction of lateral aircraft motion. The intersection of the two pointers indicates resultant vector airspeed.

d. LORAS 1000

The LORAS 1000, made by Pacer Systems Inc. of Arlington, Virginia, consists of a sensor unit, air data converter, omni-directional airspeed/density altitude indicator, and a control panel. The sensor consists of two venturi tubes mounted on opposite ends of a tubular rotor. The venturis are connected to opposite sides of a differential pressure transducer. A motor drives the rotor at a constant speed of 720 rpm in the horizontal plane to assure adequate dynamic pressure in the venturis, independent of aircraft motion. The air data converter combines the sensor unit outputs (differential pressure and the corresponding angular position of the venturis) with temperature and static pressure and outputs longitudinal, lateral, and resultant true airspeed. Density altitude is also an optional output of the computer. The system was designed to operate over an airspeed range of 25 m/s (50 Kn) true airspeed (KTAS) rearward to 100 m/s (200 Kn) forward and to 25 m/s (50 Kn) in lateral flight. The system was also designed to be insensitive to vertical motion and its method of operation is shown by Figure 2.2.2-4.

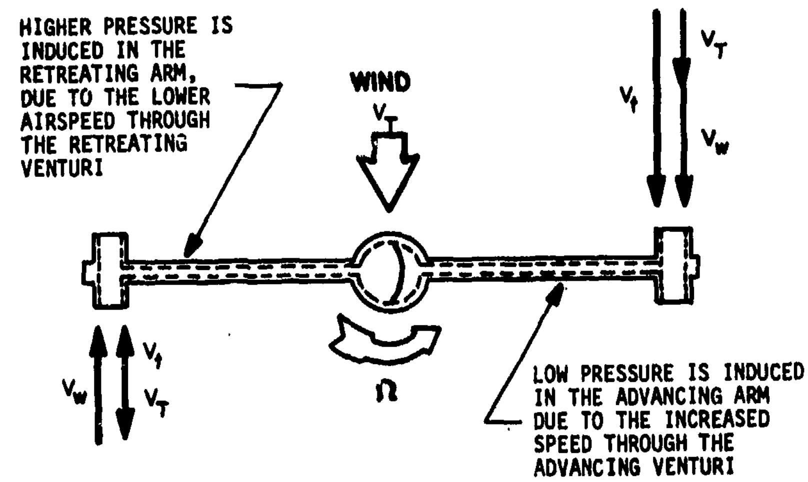

Figure 2.2.2-4
LORAS Airspeed Sensor

e. Rosemount

The Rosemount orthogonal airspeed sensor is manufactured by Rosemount Engineering Company, Minneapolis, Minnesota. The system includes a sensor, airspeed indicator, transducer/analog multiplier unit and tubing. Sensor dimensions and operation is outlined in Figure 2.2.2-5.

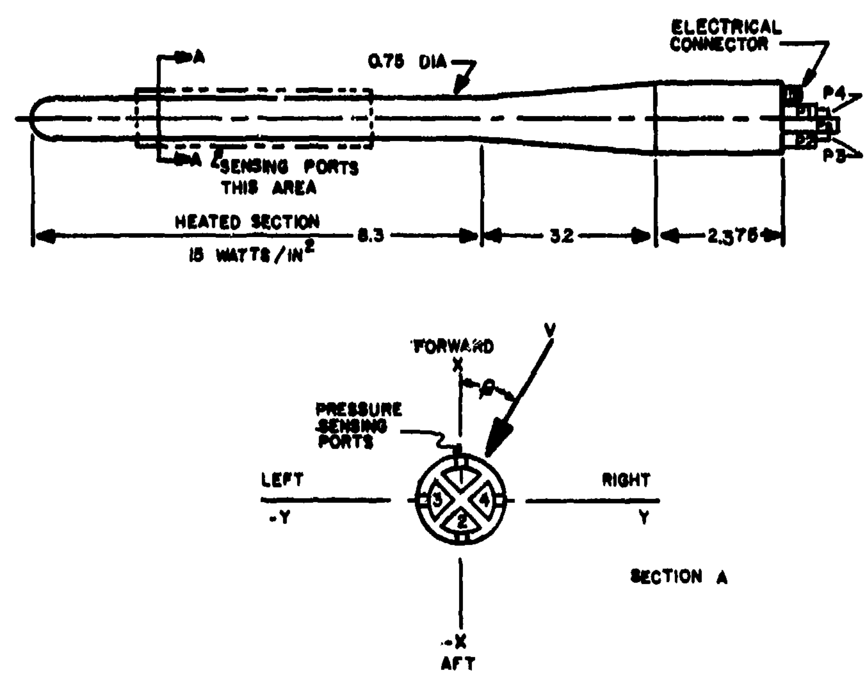

Figure 2.2.2-5
Rosemount Airspeed Sensor

The sensor contains internal electrical wiring for deicing. Power consumption with deicing operations is 250 watts in flight and 150 watts in still air.

The Rosemount orthogonal airspeed indicator is a dual pointer instrument, with both pointers moving rectilinearly. The dual traversing pointers are driven by DC signals from the Rosemount transducer and move the pointers through scales representing 60 Kn (30 m/s) forward to 40 Kn (20 m/s) aft, and 50 Kn (25 m/s) left to 50 Kn (25 m/s) right, respectively, when the airspeed sensor is mounted parallel to the aircraft's vertical axis. Those indicator limits were chosen to provide maximum sensitivity while encompassing the expected range of helicopter operation.

The indicator scale is presented in the form of concentric rings located at 10 Kn (5 m/s) circle increments, with zero located at the geometric center of the indicator and a 40 Kn (20 m/s) circle being the most distant ring. The horizontal pointer reflects forward velocity by moving upward, and rearward by moving downward. The vertical pointer indicates transverse velocity (right, left). A left vertical pointer deflection indicates flow coming from the left and, similarly for right deflection, a flow from the right. Viewing the intersection of the horizontal and vertical pointers will depict the vector resultant of airspeed.

f. <u>Honeywell</u>

The Ultrasonic Wind Vector Sensor (UWVS) is designed and manufactured by the Government and Aeronautical Products Division of Honeywell Inc., St Louis Park, Minnesota. The system was designed to provide an accurate measure of the relative wind while using no moving parts, giving linear sensitivity over the airspeed range, and responding to rapid changes in wind magnitude and direction. The UWVS operates on a principle involving ultrasonic signal transmissions through the moving air mass. The sensor and relative wind vectors are shown in Figure 2.2.2-6.

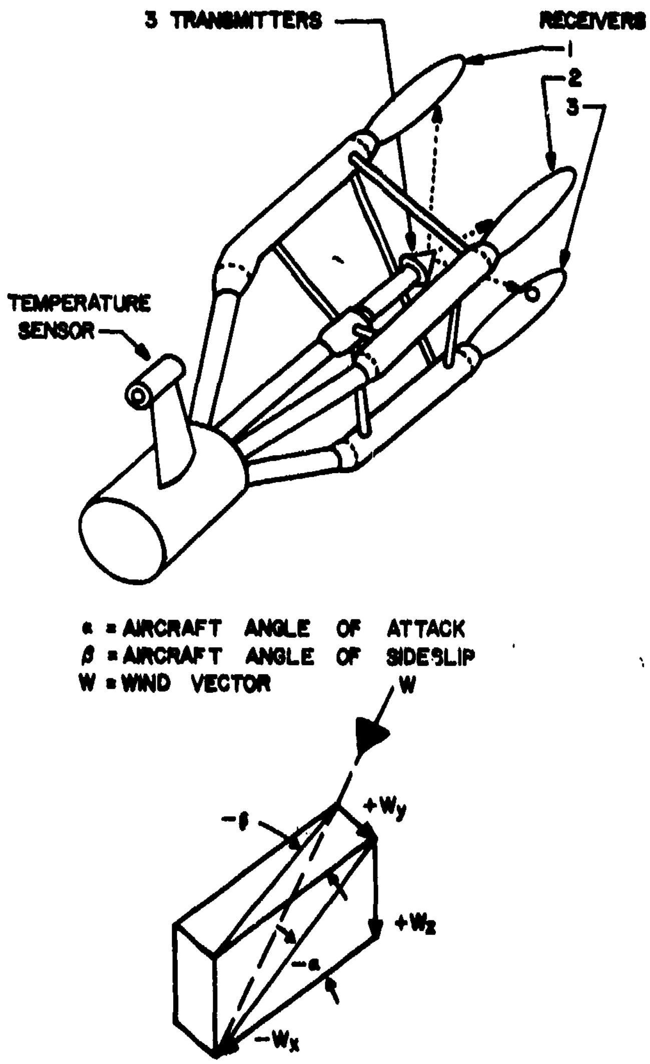

Figure 2.2.2-6
Honeywell Airspeed Sensor

The system is a sensor head and an associated electronics package. The sensor head has three receiver probes spaced at 120° intervals around a transmitter and a temperature sensor. The transmitter is a piezoelectric transducer resonant at 75 kilohertz and the receivers are wide band-width ceramic microphones with response to 400 kilohertz. The temperature sensor is a platinum element, thermally isolated from the structure. The sensor unit also contains a temperature sensor amplifier and three receiver preamplifiers. The transmitter drive, timing logic, pulse detection circuitry and electronics used to solve the equations are contained in the electronics unit.

Defining the wind velocity components is done with a geometric arrangement of three ultrasonic transmission paths deployed in the airflow. From this, three equations can be derived to express the velocity components as functions of the measured transmission times along the paths. The temperature sensor is needed to compute the wave velocity in air as a function of temperature. The transmitters are simultaneously pulsed and at a later time, typically 200 to 300 microseconds, the wave arrives at the receivers.

By writing the equations for the other two transmitter/receiver pairs as a function of their respective transit times, three equations with the three unknown vectors result. There are not of a closed form because the vectors are a function of total vector and therefore must be solved by iterative or feedback methods. With zero relative wind velocity, the three transit times will be identical and equal to the ultrasonic wave transit time at the particular temperature.

With a relative wind along the X axis only, the times are equal, but increase in value for forward aircraft motion. For relative wind in an arbitrary direction, the three times will be different in value. In general the times can be considered as quantities which vary by a percentage around the still air value. The total vector can be used to provide three airspeed components as well as angles of attack and sideslip.

3. PROPULSION SYSTEM

The propulsion system data is critical for all performance tests. The system includes engines, transmissions and drive train components. Emphasis is placed on parameters which are used to determine power required and power available. Power measurements vary considerably with test objectives. For engines which have been previously defined it may be necessary to only measure power input to the rotors. With new engines or new installations it may be required to measure every element in the propulsion system. The most direct method to determine power is measurement of torque and speed which are then used to calculate power. Other methods include use of fuel flow and temperature in conjunction with engine charts and engine characteristics data. The engine/airframe interface must be established in terms of inlet and exhaust characteristics. Engine cooling and vibration can also have a significant impact on suitability.

When system losses must be determined, each component will be instrumented to provide input and output data. Accessory power must be determined for any power extracted to operate aircraft systems. The instrumentation may include electrical, hydraulic, or pneumatic measurements. For tests of the dynamic compatibility of new or modified engine-airframe combinations and tests to evaluate engine/rotor response characteristics the accuracy may need to be compromised to obtain satisfactory dynamic response from the instrumentation. In some cases redundant instrumentation will be necessary to meet both steady state accuracy and dynamic response requirements.

3.1 Shaft Speed Measurements

Contained within the propulsion system are a wide variety of rotating components; and measurements of the rotational velocities of these components are often of critical interest to the test being conducted. Evaluation of the methods now available to measure these rotational velocities centers around the magnitude and transitory nature of the velocity. Those of high or low speed with little short term variation are easily measured; but rapid changes in velocity must be given special attention. In general, less transient parameters are handled by measuring the frequency of rotation in a rather direct fashion. As an example, a constant rotor speed is often measured by outputting the rotor tachometer generator to a frequency to D.C. voltage converter. This provides a D.C. voltage level proportional to rotor speed. While this technique provides good results with little or no transient rotor conditions, large rotor speed variations can result in sizeable measurement errors. By providing high resolution, high sample rate period measurements of variable low speed shaft parameters, problems of response to rapid frequency changes and/or invalid data averaging associated with frequency to D.C. measurement techniques, can be eliminated. Accuracies greater than ±1% are possible. Consideration should be given to measurement repetition rate, master clock frequency, etc. required for the particular transient conditions present.

3.1.1 Engine Speed

Engine speeds usually vary from high compressor or turbine speeds to lower shaft speeds following gear reductions. Measuring internal engine speeds with a test system is difficult at best, and may not be possible in some cases. Most engines have an integral rotational speed sensor which provides an electrical signal whose frequency is proportional to the speed. This signal is used to drive standard cockpit instruments and can also be input to a test recording system. The standard instrument is usually not suitable for recording test data from the cockpit and is often replaced or paralleled with a high resolution test instrument. This method is most useful for the pilot when conducting tests or for use by observers monitoring test progress for completeness or quality of test results.

Power turbine speed measurement is often a tachometer generator system similar to that described for the compressor speed and comparable methods are used. The power turbine shaft may be directly available or may have an integral gear reduction transmission. The most convenient shaft is fitted with speed measurement instruments and if necessary gear ratios are used to calculate the turbine speed. A typical test instrument to measure engine output shaft speed is a frequency to D.C. converter. The output of the converter is a voltage proportional to the speed and is recorded by the instrumentation system.

3.1.2 Drive Shaft Speed

The drive shaft speeds which must be measured will depend on the test requirements and the physical nature of the test vehicle. In some instances it may be necessary to know the speed, while in other instances the power being transmitted is of prime importance. An example of the first case is determination of rotor speed by measurement of input shaft speed to a transmission. The second case arises when power must be known for each component in the drive system. Transmission losses can only be established by measuring input and output power. This necessitates a shaft speed measurement. A magnetic sensor and recording system similar to that used for the engine output shaft is the most common method.

3.2 Engine Torque

Engines commonly have a torquemeter which can be incorporated into the test instrumentation system. The wide variety of aircraft types requires that the instrumentation system have great flexibility for interfacing with engine torque sensors. Rather than measuring torque directly, it is more common to sense some characteristic which is proportional to torque. The sensing devices in use include monitoring electrical permeability of the shaft, optical measurement of the shaft twist, and strain gages for torsional measurements. Appropriate electrical circuits must be developed to provide signals to cockpit indicators and aircraft systems. These circuits are normally used as input to the instrumentation system and care must be taken not to alter the operation of the standard torquemeter system. Isolation amplifiers may be required to insure separation of the aircraft torque system and the instrumentation system. In most cases the signal level of the torque system will be less than one volt and noise reduction techniques should be included. The engine is placed in a test cell and the torque is measured directly with a dynamometer and the indicator reading is noted. From this calibration, torque can be determined for any indicated reading. Engine torquemeters have an accuracy on the order of ±5% although, in one instance, an accuracy of ±1% is claimed. The engine torquemeter output must be recorded during the test since the operators manual will be developed in terms of the power indication to the pilot.

During development of new engines, or for standard torquemeters that provide inadequate data, it may be necessary to install a test torque measuring system. The test power measurement system is usually placed on the engine output shaft. The torque must be measured on the shaft for which the speed measurement was taken. Extreme care and close coordination with the flight test engineer is needed to determine what power is being measured and that it is the correct power for the data requirements. Note should be made of the power being measured relative to transmissions and power extraction sources. Resistance type strain gages are commonly used to sense the torsion in the shaft. Temperature compensation must be adequate for the installation and consideration must be given to shaft bending moments.

The strain gages are connected to a slip ring brush assembly which transmits the signal. The electrical and mechanical properties of the slip ring assembly must be compatible with the strain gages being used. After the strain gage installation the shaft is calibrated in terms of force and deflection. Checks are made for adequacy of compensation efforts and if necessary those influences are included in the calibration. It may also be necessary to dynamically balance the shaft to compensate for the added instrumentation. In some cases slip rings have been replaced with telemetering systems which transmit torque data from the rotating shaft to a stationary receiver.

3.3 Shaft Torque

Torque measurements on the individual drive train shafts are made at the same place as the speed measurements. Shaft speeds, diameters, and environments will vary considerably and special care must be taken to compensate for mechanical or environmental effects. A strain gage system similar to the engine torque is most commonly used. Mechanical, optical and electromagnetic systems have been used to measure shaft deflection with applied torque.

3.4 Inlet

Inlet conditions are a critical item in the analysis of the propulsion system and for determination of all aircraft performance. Data must be obtained which will show the nature of the flow into the compressor, establish the mass into the engine and establish the starting point for a thermodynamic analysis of the engine. The inlet may be all or any part of the total ducting, shaping, guiding or holding apparatus between the free air stream and the compressor face. Consideration must be given to the extreme range of conditions generated by the helicopter flight regime. Vertical, forward, rearward and lateral flight produce the full range in terms of sideslip and angle of attack. Rotor downwash is usually present and there may be engine exhaust gas ingestion caused by circulation of the rotor wash. The inlet performance is usually defined in terms of pressure and temperature conditions at the engine compressor face. Test requirements may dictate establishing the turbulence or distortion in the inlet flow. During the instrumentation design phase, special care should be given to obtaining data compatible with any previous engine calibrations or any data needed to operate the engine computer program. Any instrumentation must be fully certified for the expected dynamic pressures, temperatures, and vibrations before it is placed in the inlet. Cockpit instruments are usually provided for in-flight recording of pressure and temperature and should there be multiple sensors a switch should be provided to allow the flight crew to monitor the data.

3.4.1 Inlet Pressure

Most engines are delivered with at least one total inlet pressure sensor installed. For a well defined engine or for a cursory performance evaluation, this may provide sufficient information. A single sensor is not satisfactory for rigorous performance tests or for dealing with a new engine or installation. To obtain data which will show distortions and provide construction of pressure profiles, it is necessary to use several sensors mounted on a rake and placed in a suitable location in the inlet. The rake will best show engine inlet conditions when it is placed near the compressor face. The number of sensors on the rake will depend on the data requirements, physical nature of the inlet, and the recording system capability. Accuracy of the profile and distortion information is very sensitive to the number of probes and the probe array (Ref 22). Struts or any other physical characteristics of the inlet will influence the flow and may change engine performance. A typical inlet rake is shown in Figure 3.4.1-1.

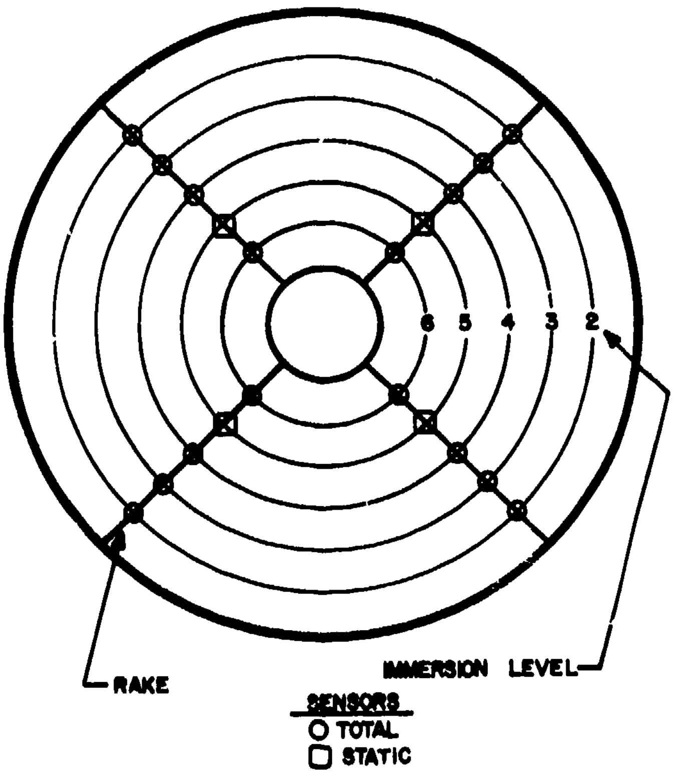

Figure 3.4.1-1
Inlet Rake

Both static and total pressure sensors are required and should be located at the same engine station. The sensors must have a sensitivity, range, and response compatible with the data required. For a large number of sensors it is common practice to use a scani-valve arrangement where each sensor is switched to single pressure transducer in sequence rather than being continually input to the transducer. The time increment between samplings must be carefully considered. A good technique is to measure selected sensors continuously while still including them in the sampling sequence. This provides a check on data validity and aids correlation of data from all the sensors. Total pressure ranges from ambient to dynamic pressure at maximum airspeed. Pressure should be measured very accurately since a small change can result in significant differences in calculated engine power available or power required. When scani-valve arrangements are used, dynamic response of the pressure transducer must be considered to insure proper performance with the multiplexed inputs.

Some inlets have filters, particle separators, and flow control or by-pass devices which may require evaluation. In most cases a single upstream and downstream pressure differential across the device will be adequate; however, a rake similar to that for the compressor face may be necessary to provide the needed data. It is only possible to generalize here and let specific decisions be made for each individual situation.

3.4.2 Inlet Temperature

As in the case for pressure, an inlet temperature sensor is often standard with the engine and may be used as a suitable data source in some cases. During hover or in low speed omni-directional flight, the engine exhaust may be trapped in the down wash and be re-circulated into the engine. Heat from transmission systems can also find its way to the engine inlet. These heat sources can cause large inlet temperature rises with dramatic effects on engine or aircraft performance (Ref 23). For re-ingested gas the temperature may be uniform across the inlet while for radiated heat it may be concentrated in a particular sector of the inlet. Complete inlet temperature data require a rake with probes spaced at different levels and azimuths. Temperature sensors can also be placed on the pressure rake as previously discussed. The number of sensors is established by the degree to which the profile must be determined. Hot gas re-ingestion may cause temperature rises of 50°C and the flow is very turbulent which causes large rapid fluctuations and dictates a high response characteristic for the sensor. A suitable inlet temperature sensor is a chromal-constantan thermocouple. With a large number of sensors, it may be necessary to use a time dependent sampling technique. A large time increment between samples will restrict capability to establish variations in the temperature and a continuous record of selected sensors may be necessary.

3.4.3 Inlet Devices

The status of inlet devices such as guide vanes, by-pass doors or variable geometry equipment must be recorded. This information is needed to evaluate and correlate inlet flow characteristics and calculate airflow. In addition there may be drag considerations. These inlet devices are usually mechanical and position sensors such as potentiometers or micro switches are used to record their motion. There may also be inlet ram air bleed devices and it may be necessary to measure the flow taken from the inlet.

3.5 Engine Temperature

Engine temperature requirements can vary from a single parameter to detailed measurements at various engine stations. Engine stations and nomenclature usually varies with each engine, however, the system to be used must be defined prior to instrumentation system design. A typical engine layout and definition is shown in Figure 3.5-1.

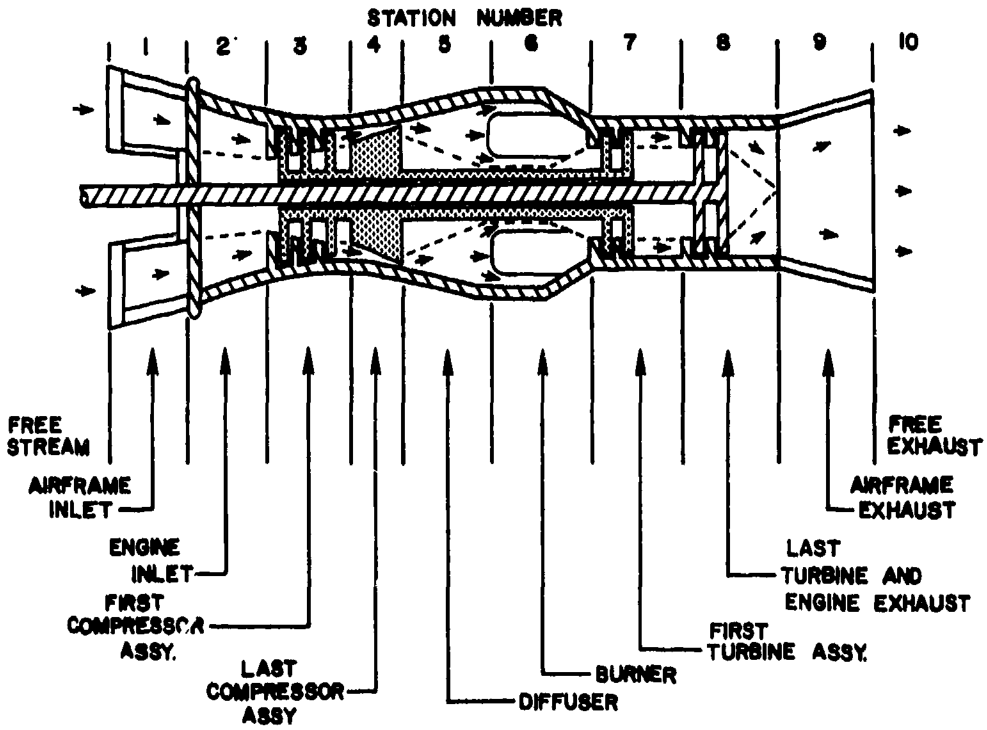

Figure 3.5-1
Engine Station Definition

Instrumenting for a detailed engine temperature evaluation is an extremely difficult task and should only be conducted with the assistance of the engine manufacturer. This is advisable because of potential engine performance changes or structural implications caused by the test installation. The temperature evaluation may be for internal gas flow parameters or for external surface and compartment conditions.

The internal engine temperatures are ambient at the inlet and increase to a maximum in the burner section. Engine design information can be used to select sensors which are suitable for the range and response requirements. Most flight tests require measurements of only compressor discharge, power turbine, and exhaust gas temperatures. These measurements are commonly taken from standard engine sensors which are usually available. Additional temperature information is obtained by placing thermocouples at selected stations shown in Figure 3.5-1.

Engine surface and compartment temperature measurements are necessary to establish the engine environment and assess the heat being transmitted to the surrounding structure. Surface temperatures can be measured within ±1% by use of color coded temperature sensitive material. A change in color shows that a specified temperature has been exceeded. This material is available as a paint or as a template. A wide range and choice of increments can be selected to meet the expected requirement. More accurate data can be obtained from thermocouples bonded to the test surface. Compartment temperature is measured with a resistance thermometer using the techniques discussed for free air temperature measurements.

3.6 Engine Pressure

Engine power and compressor performance information requires measurement of compressor discharge pressure. Engine specifications will provide the compressor ratio data which is used to select a sensor with the proper range. Exhaust gas pressure is measured with individual sensors or with an array of sensors on a rake as discussed for the engine inlet. Care should be taken to insure that the sensor chosen is operationally compatible with the installation as well as suitable for the gas temperature being measured.

3.7 Fuel

Fuel measurements include flow rate, temperature, and quantity used. Flow rate and temperature are necessary for engine or aircraft performance tests while quantity used is needed to determine aircraft weight. A careful study of the fuel system should be made prior to designing the flow measurement instrumentation. Special attention must be given to relative flow from various tanks, fuel transfer for CG control, fuel by-pass valves or vapor return lines. Fuel tank level devices must also consider aircraft attitudes and flight conditions.

3.7.1 Fuel Flow

The fuel mass being used by the engine can be measured directly with a mass flowmeter or calculated from volume and temperature measurements. Sensors are available for various flow rates and the appropriate one should be selected for each engine installation. It should be noted that small helicopters have flow rates as low as 9 kg/hr (20 lb/hr) at idle while large machines use up to 1130 kg/hr (2500 lb/hr) at rated power.

In flight test, the flow is most commonly determined by measuring volume flow and temperature and then calculating the mass flow. The volume flow is obtained from a turbine flowmeter. Prior to installation the flowmeter is calibrated so that for a given turbine rotational speed the flow can be determined. Calibration of the flowmeter is critical and requires precise control of calibration fuel parameters to insure definition of the number of cycles of output from the turbine flowmeter for a specific unit of flow. Most test turbine fuel systems use the cyclic output to determine total fuel used information. Both cockpit display and digital output for instrumentation system recording of fuel total are generally used. Fuel flow rate is computed by interfacing a frequency to D.C. voltage level converter to the turbine output. Again, both cockpit and instrumentation system recording of this data is normally used. The flowmeter creates a small resistance to flow which may affect engine operation and should be considered prior to installation. The sensor is placed in the fuel line as near the engine as practical. Extreme care must be used to insure that the installation plumbing is the same as that used during the calibration. Variations can cause flow effects which will invalidate the calibration. The specified flowmeter installation requirements should be met. These usually specify the diameter of the input/output plumbing, length of straight line tubing required at input/output and may provide limits of vibration exposure.

3.7.2 Fuel Temperature

To obtain a true mass flow the fuel temperature must be measured at the same place as the volume flow was measured. When the measurements are made very near the engine, the system will usually be enclosed by the cowling and fuel temperatures up to 50°C are not uncommon. Consideration must be made for any oil-fuel heat exchanges in the system. Fuel flow measurements are usually made at relatively steady flow conditions and for maneuver situations, dynamic response of the sensor must be considered. The temperature sensor is placed directly upstream of the flowmeter. A suitable type of sensor for this application would be a platinum element probe designed for immersion in fluids. Any number of platinum element probes can be obtained for this application, but attention must be given to proper design of the installation utilizing them. Frictional heating of the probe due to fluid motion, stem conduction errors, and flow obstruction must be considered in the design. Some relief from these requirements for engineering can be obtained by implementing a platinum probe/tubing combination sometimes called an in-line sensor. These sensors install as a short piece of tubing with the platinum probe integrated into the tubing by the sensor vendor and can be ordered to meet the specific application at hand.

3.7.3 Fuel Quantity

Prior to flight, the fuel mass and aircraft weight are precisely known. Therefore, an accurate in-flight gross weight requires determination of fuel used from engine start to the time test data is recorded. Fuel used is determined by the flowmeter cycles and the fuel density in the fuel tank. Specific gravity of the fuel in the tank is established in the laboratory for samples taken before and after each flight.

Any flow through fuel return or by-pass lines is measured and used to correct fuel flow and fuel used data.

3.8 Power Extraction

Power may be extracted from the engine to operate electrical systems, hydraulic systems, and environmental control systems for the occupants. For exacting aircraft performance tests these powers must be considered in the power required terms. During a propulsion system analysis these powers affect the total engine power available, and in the case of airflow taken from a particular engine station, may influence the thermodynamics of engine operation. The systems are often complex and redundant and each must be carefully analyzed to determine sensor location so that the proper power is being measured.

Electrical power from the alternator is determined by measuring power directly or indirectly from measured values of voltage, current, and the phase angle between voltage and current. Standard engine sensors normally are provided and the signal can usually be recorded by the test data system. High accuracy requirements for power usage by any component will dictate installation of test sensors.

Hydraulic system power may require measurement of power to drive the pumps or the flow of hydraulic fluid. The pump may be driven by engine gear box, rotor transmission, electrical power, or bleed air. Gear trains or drive shafts require a speed and torque measurement as previously discussed. Provided the density of the hydraulic fluid is known, the hydraulic mass flow may be determined from the measurement of volume using a method similar to that for determination of fuel mass flow as discussed in paragraph 3.7.1. Electrical pumps need voltage and current measurements. Bleed air flow is measured by pressure and temperature sensors. Another determination of power loss to bleed air can often be obtained from engine test cell data with bleed air on and off.

Cockpit or cabin environmental control systems are often combined electrical/airflow devices. Cooling may be obtained by circulating outside or cabin air through a refrigeration unit. This unit may be electrically driven or may use engine bleed air. Previously discussed instruments can be used to obtain power and airflow data.

3.9 Power Plant Controls

Power plant controls include those in the cockpit and at the engine. Reciprocating engines usually have cockpit controls whose positions need to be measured while turbine controls are normally on-off or three position devices. Automatic engine controls are often instrumented to provide correlation data and insight to engine performance characteristics. Engine and cockpit control relationships are needed to evaluate dynamic response and engine/rotor capabilities. Potentiometers or microswitches are generally used as sensors for these applications.

The amount of motion can be expressed in degrees or linear measurement around the arc for a control which moves about a fulcrum. Calibration of control motion is generally done by using an inclinometer to measure the angular motion in degrees and then by measuring the radius of the control arm. Attention must be given to properly identifying the radius of the control arm. The distance from the center of hand contact to the fulcrum is most often used.

3.9.1 Cockpit Controls

Power levers or twist grip throttles are instrumented with position transducers to show position from closed to full open. For on/off controls, microswitches are used to record the positions. Microswitches or position transducers can also be used to show activation of engine trim devices. Strain gages are placed on the controls to measure forces applied by the pilot.

3.9.2 Engine Controls

To determine lost motion and delay in activation, the controls at the engine are instrumented and results are compared to control motions in the cockpit. Fuel control levers can be measured in terms of angular or linear displacement. A very sensitive sensor must be used since the displacements may be very small. Engine controls such as droop compensation are not controlled directly by the pilot and motion must be correlated with items other than cockpit engine controls. Engine governor inputs to these can be instrumented as previously discussed.

3.10 Engine Vibration

It may be necessary to measure either vibration generated by the engine or the vibration from the airframe to the engine. Transducers are placed on the engine to measure motion in all axes. The sensors must be located relative to any absorbers or dampers so that the desired vibration is being measured. In the case of more than one engine, each must be instrumented because of potentially large changes with asymmetric power. The sensors are usually placed on the engine mounts.

Transducer types employed can vary greatly due not only to environmental and parameter requirements, but also as a result of the analysis philosophy employed. In most instances, accelerometers are used to assess vibration levels, but some analysts use velocity sensors instead. No attempt will be made in this document to influence the reader in favor of one method or the other, but a brief discussion of transducer types is presented.

Vibration levels at engine stations are due to a broad frequency spectrum of input. The rotor system excites the area with low frequency, while rapidly rotating devices including the engine, provide high frequency excitation. Most often, the full range of inputs can be sensed using piezoelectric accelerometers. These have a flat frequency response of approximately 3 to 30,000 HZ. Mounting of the accelerometer should be accomplished without changing the vibrational characteristics of the test article. That is, the mass of the accelerometer and the mounting device or material should be carefully chosen. Often, glue is used to attach the accelerometer. If this method is employed, the temperature of the surface should be used as one criteria for selection of the glue. Further comment on accelerometer useage will be given in the discussion of airframe vibration.

Velocity transducers are generally of two types, piezoelectric or a permanent magnet/coil combination. The piezoelectric transducer is actually a piezoelectric accelerometer with an integral amplifier/integrator and has a frequency response range of approximately 1 HZ to 2000 HZ. The permanent magnet/coil combination transducer has a lesser frequency range (typically 10 HZ to 1000 HZ), but is self generating and therefore does not require a regulated power supply as does the piezoelectric transducer. Both provide a millivolt output proportional to velocity.

4. AIRFRAME

The airframe measurements are needed for a variety of tests. Performance testing includes drag determination, which requires attitude and relative wind data. In some cases, the data must be corrected for linear or angular accelerations. Attitudes, rates, and accelerations are critical for stability and control tests. In these tests the structural and dynamic maneuvers require measurement of loads or stresses in various components to establish component life of flight envelope limitations. Airframe vibration information is needed to evaluate occupant environment and conditions experienced by instruments and aircraft sub-systems.

4.1 Attitude

The aircraft attitude is measured relative to an earth axes system. Usually it is necessary to measure pitch, roll, and yaw attitudes. Sensor location relative to the aircraft body axes must be precisely known. For tests near the ground, photographs showing flight path and an earth reference can be used to measure the attitude. Most tests are sufficiently above the ground that ground mounted cameras cannot be used. Photographs from chase aircraft seldom show true angles and thus cannot be used for engineering data. With the proper equipment, celestial bodies in conjunction with earth position can be used to determine the aircraft attitude. These photographic and optical techniques require special equipment. The data are difficult to process and cannot be used in certain atmospheric conditions. For steady state condition, pendulum type sensors can use earth gravity to measure the relative position of the aircraft. Acceleration effects render such a system unuseable for most flight test purposes. Gyroscopes mounted on each of the aircraft axes provide the best approach to obtaining the aircraft attitude. Certain helicopter characteristics alter the requirements somewhat from those needed in a fixed wing application. While some helicopters are fully aerobatic, pitch attitude will seldom be more than ±60° and roll is usually within ±100°. In several tests the helicopter is required to yaw 360° at hover and low speeds. The gyroscope data is usually recorded continuously for times of less than one minute, which should be taken into consideration when selecting suitable sensors.

4.1.1 Pitch and Roll Attitude

Attitudes are measured with a 2 axes gyroscope. For most engineering flight tests a nominal range is ±45° in pitch and ±60° in roll with a 0.5° resolution. The resolution is dependent on the signal conditioners and encoders. Electrical conversion of pitch and roll position is accomplished, in most instances, by one of two methods. Synchro signals are, in some cases used for position information output, but this method requires conversion of the synchro data into an electrical form compatible with the data recording systems signal conditioners or encoder. More direct input of position data to the recording system can be achieved by using gyros with potentiometers for position encoding. These are most often directly compatible with recording system signal conditioners. The attitudes are measured for steady state conditions as well as deviations from the trim during maneuvers.

4.1.2 Yaw Attitude

Yaw attitude must be measured from some reference point. The gyroscope is similar to those for pitch and roll with the addition of a caging feature. At trim or some desired starting condition the gyroscope is uncaged and then will show deviations from initial aircraft heading. The range is then ±180° from that heading with a linearity of 0.5% of full range. For some instruments, yawing more than 180° will cause the gyro to tumble.

Certain tests of navigational systems or earth referenced maneuvers require magnetic headings. This data is obtained by use of the aircraft gyroscope with signal conditioning such as a synchro to D.C. or binary converter which tracks the compass heading. Yaw attitude changes can be computed using Euler angle transformations.

4.2 Angular Rate

Angular rate data can be calculated from the attitude gyroscope output. In addition to the computation requirements, a portion of the data is lost with this approach and helicopter flight tests usually require installation of rate gyroscopes. The rate will be measured from an initial test condition which can be either static or dynamic. Following a control input, the maximum rate will usually occur within two seconds, and appropriate sensor characteristics should be selected. The rates are measured for each aircraft axis, and the location of the sensor must be accurately established. The rates generated during the tests are considerably less than for high performance fixed wing aircraft, and sensor selection should be influenced accordingly. The range must be sufficient to encompass aircraft motions which are generated by 2.54 cm (1 in) control inputs that are held for one second. Typically a rate gyro with ±30°/sec in pitch, ±100°/sec in roll and ±60°/sec in yaw is used. The format of the electrical output of the rate gyro motion will vary but a high level single-ended output is most often used.

4.3 Angular Acceleration

Angular acceleration data is needed to assess the helicopter controllability and for use in aircraft energy calculations. Angular accelerations are often computed by differentiating the rate gyro output. When necessary, accelerometers are usually mounted at the aircraft center of gravity and are aligned with the pitch, roll, and yaw axes. The alignment and location of each sensor must be accurately established and recorded. The helicopter angular acceleration can be up to 200°/sec^2. Stability and control systems input must be considered and the sensor is usually sized for systems "off" which produces the highest acceleration. The pitch, roll, and yaw accelerations are inversely proportional to aircraft moments of inertia and therefore are usually highest in the roll axis. Presently available angular accelerometers are highly susceptible to vibration contamination, making mounting critical.

4.4 Linear Acceleration

Linear accelerations are required for energy analysis in performance testing and for certain stability and control tests. When the data are to be used for power corrections due to accelerations in the various axes, the sensors are placed as near the center of gravity as possible to minimize the effects of angular motion. During systems testing or handling qualities evaluations, it may be necessary to measure the total acceleration (linear plus angular) at a component or at the pilot's station. In these specialized cases, it may not be required that all axes be instrumented. The omni-directional flight capability of the helicopter imposes requirements somewhat different than fixed-wing aircraft. Sideward and forward acceleration capability are nearly equal and may be up to 1 G. Normal acceleration seldom exceeds 3 G. The accelerometer data should be very accurate since small errors can introduce large variations in the performance calculations. In addition to high accuracy, the sensitivity and frequency response must be sufficient to record rapidly changing conditions during maneuvers. A large variety of accelerometers are available for measuring this range of acceleration. The requirement to measure acceleration in a frequency band which includes static accelerations can be satisfied by the use of strain gage or piezoresistive accelerometers. Although both types provide a lower frequency response of D.C., the upper limit can vary from 600 to 8000 HZ and requires excitation. Careful selection of pre-sampling or signal conditioner low-pass filtering will eliminate the acceleration inputs from the sensor above a predetermined maximum frequency of interest. The general frequency band desired is from D.C. to main rotor frequency. Selection of the proper accelerometer should consider ambient conditions, size constraints, the acceleration range present, and the frequency range desired. As with attitude gyros, the accelerometer orientation should be exactly defined by the aircraft axis. Accelerometers are usually calibrated to the standard gravitational acceleration value. For some tests, output may be corrected for local gravity.

4.5 Vibration

Airframe vibration frequencies are predominantly multiples of the main and tail rotor speeds. An out of balance or out of trim blade will generate a vibration with a frequency equal to the rotor speed. Helicopter main rotor speeds may produce vibration with a frequency as low as 3 HZ. Other vibration sources usually have frequencies greater than those from the main rotor. It is necessary to define both frequency and amplitude in order to evaluate the effect on structures, components, and occupants. The testing often will include ground operation to insure that natural frequencies of the airframe are different than rotor excitation frequencies. These types of test are mainly concerned with structural integrity. The flight tests are generally concerned with crew or passenger environment or determination of conditions to be experienced by avionics, aircraft systems, or cargo.

4.5.1 Sensor Locations

Sensors should be located in all areas where vibrations can be transmitted to the crew members. Typical location are the seats, flight controls, instrument panels, foot rests, and consoles in the cockpit. Each potential location must be analyzed to determine the number and orientation of the required sensors. Consideration should also be given to instrumenting external stores, pylons, doors, horizontal and vertical tail surfaces. It may also be necessary to obtain data for wings and landing gear. Vibration of aircraft components is measured and compared with excitations to evaluate performance of the mounting or damping mechanisms. Mounting of the sensors will vary according to the physical makeup of the accelerometer and the mounting location. In all cases, however, the vibrational characteristics of the structure under test should be altered as little as possible. Care should be taken to insure that the natural frequency of the accelerometer is not shifted into the frequency range of interest by the mounting technique. This can happen quite easily if poor surface contact results from improper mounting. Alignment of the sensors is critical and should be traceable to a known reference.

4.5.2 Sensors

Velocity pickups or accelerometers are suitable, with the latter being in most common usage. It may be necessary to sense vibration in a single axis or in 3 axes. A suitable single axis sensor is a piezoelectric with a frequency response of 5 to 2000 HZ. Piezoelectric accelerometers are a good choice because they have self generating output, wide frequency response, small size and are easily mounted. For dynamic acceleration, which is of interest in airframe assessment, the piezoelectric sensor also has the advantage of not responding to input frequencies much below 3 HZ. These devices do require some care in application, however. By employing the piezoelectric effect, the sensor produces a charge that is proportional to the acceleration level. It is then necessary to convert this charge to a voltage for input to the instrumentation signal conditioning. The voltage produced will be proportional to the charge and the capacitance of the sensor, cable and signal conditioner input combination. Unfortunately, the capacitance of the cable will vary with length, which hampers interchangeability and by flexing the cable, charge noise can be generated and is indistinguishable from the sensor charge output. These drawbacks can be overcome by using a well placed charge amplifier rather than a voltage or source follower amplifier and by using low noise cable. The output of a charge amplifier is strictly a function of the sensor charge output, the amplifier feedback capacitor and charge noise generated by the cable. By placing a low gain miniature charge amplifier near the sensor and then amplifying the resultant voltage output with the instrumentation signal conditioning, low noise measurements can be made. This low gain configuration helps in suppressing the triboelectric noise (cable charge noise) and eliminates cable capacitance effects. A number of manufacturers produce suitable charge amplifiers.

4.6 Loads

The structural loads demonstration is conducted in conjunction with the early vibration testing. During the performance, or stability and control testing, the limits of the envelope will be reached and new conditions or maneuvers may be attained. Loads data can be used to evaluate the hardware suitability under mission operating conditions, allow comparison with design information, and contribute to fatigue life calculations. Loads instrumentation is extremely critical with respect to sensor location and number of sensors. Analysis of design information, bench or component testing results, and previous flight tests will suggest critical locations. Models can also be constructed of materials which will visually show stress concentrations. Comparisons of data from these different sources are most accurate when sensors are in exactly the same location on the structure.

4.6.1 Sensor Location

Sensors should be placed on all components expected to be fatigue critical. Special consideration should be given to structures directly transmitting or receiving thrust or lift forces. Examples would be tail boom mounting structures, transmission mounts, and wing or stabilizer attachments. Specific guidance on sensor location is not possible, and the instrumentation must follow the directions of the stress analyst who will consider local stress concentrations, operating environment, and inter-relations with other components or structure.

4.6.2 Sensors

Load measurement is best accomplished with a bonded strain gage bridge. Particular attention must be given to the gage factor, type of material being tested, environmental temperature, and conditions at the mounting location. Bonding must be of the highest quality. Calibrations can be calculated on the basis of sensor specifications and verified dynamically.

4.7 Cockpit and Cabin Environment

Tests may be conducted to measure the operating environment of the crew compartments to insure that occupants can function adequately throughout the helicopter flight envelope and during the mission requirements. The data is obtained during ground tests, climatic hangar tests, and flight tests. In addition to vibration, which has previously been discussed, the compartment temperature, quality of air and noise environment are of primary concern. Ground tests and climatic hangar tests often generate absolute data which is used in assessment of basic design or hardware modifications. Flight test data is usually evaluated in terms of how the conditions affect the occupants.

4.7.1 Air Temperature and Airflow

The stabilized temperature within the compartment is dependent upon the outside ambient conditions, the quantity of heating or cooling added, the efficiency of the distribution, and the heat transferred from the compartment to the outside. The vertical and lateral temperature gradients should be measured at the crew stations. Solar radiation or extraneous heat sources should be considered when selecting sensors. Thermocouples, shielded or unshielded as required, provide satisfactory results. Depending on the accuracy desired and temperatures to be measured, the systems in use range from iron-constantan thermocouples to platinum element probes with very exacting wiring practices and signal conditioning. The number of sensors and the distribution within the space can be based on a human factors evaluation, analysis of the airflow pattern, or qualitative judgement of occupants. A similar procedure is used for avionics and cargo compartments.

The quantity of airflow and heat being provided to the compartment is measured at the duct outlet or the heater. Outlet air temperature is measured with a thermocouple as discussed above. Total and static pressure sensors are also placed at the outlet to determine airflow. Selection of sensors must consider the very low velocities to be expected. Planning information can be obtained from design specifications or from systems test results.

Humidity in the compartment can be measured with any suitable hygrometer.

The airflow patterns within the compartment can be measured with a hot wire anemometer. The anemometers can be mounted on a rack and moved to different locations or the sensors can be placed at the position where the temperature profile is being determined.

Air quality can be measured with various instruments to monitor and sample different types of gases and toxicity levels. In addition it is common practice to obtain air samples in suitable containers and then to perform a laboratory analysis.

4.7.2 Surface Temperature

The temperatures of interior compartment surfaces and any exposed ducting are measured with thermocouples. Calculation of heat loss through windows requires that both interior and exterior surface temperatures be measured.

4.7.3 Internal Noise

The internal noise level in the helicopter must be measured to evaluate crew comfort, performance, communication, and safety aspects. Consideration must be given to measurement at point locations such as the pilot's ear or to obtaining data needed to create noise profiles. Noise data for the passenger section are of particular interest since these personnel do not normally wear helmets or protective gear. Selection of the sensors and recording equipment must accommodate a frequency range from 20 to 10,000 HZ and overall decibel levels up to 120. This will be influenced by windows, rotor and engine speed variations and any weapons firing. Sensor and recording requirements will be further addressed in the far field measurements, Section 8.2.

5. ROTORS AND PROPELLERS

Rotors and conventional fixed wing aircraft propellers have a great deal in common, however; rotors have several features which render them considerably more difficult to instrument and test. Significant differences are:

a) Rotors have blades which are longer and thinner with less rigidity.
b) Rotors are controlled with both collective and cyclic pitch inputs.
c) Blades may be attached to the hub with various hinge arrangements.
d) Rotors have different and complex axial and in-plane flow relations.

The critical nature of the rotor system dictates that a great deal of data and analysis be considered before conducting flight tests. Ground vibration tests are conducted to determine the blade natural frequency and mode shapes. The blades and hubs (all possible actual hardware) are then placed in a whirl tower to confirm the ground tests. Further vibration and stress data are then obtained from a restrained aircraft. These tests provide information concerning stress distributions, magnitudes of loads, and boundaries for blade compressibility or stall.

As a rule, the preliminary tests do not accurately simulate flight conditions and it is necessary to obtain flight test data to accurately assess rotor performance, stability and structural capabilities. In some cases the instrumented rotor components from the ground test are available for the flight tests. Such equipment will reduce the instrumentation needed for the flight test and will produce the best comparative data for determination of effects of actual flight conditions. New instrumentation must make maximum use of all test results to insure that the proper sensors are placed in the correct locations. Occasionally a blade will be fully instrumented. More common practice is to instrument the most critical locations for comparisons with design expectations and previous test results. Most rotor systems have symmetrical parts so that it is only necessary to instrument typical components, such as one blade, one hub attachment, or one control linkage. When this is done, consideration must be given to any mass imbalance that may result.

5.1 Blades

The sensors placed on the blade must consider aerodynamics as well as structures. The sensors must not create extra drag or reduce lift. An aerodynamicist should provide guidance as to the best locations. A stress analyst should be consulted to insure that the desired loads are being measured. Equal consideration must be given to any wiring on the blade from the sensor to the recording system. Significant aerodynamic effects can result from wires placed incorrectly on the lifting surface.

Vibratory stresses are best measured with resistance gages. The strain gages are bonded to the blade using the proper technique and the greatest possible care. The gages are oriented to provide blade measurements of torsion in pitch, flapping, and in-plane bending.

5.1.1 Airflow

In certain cases it is necessary to determine the nature of the airflow around the blade. Visual displays such as tufts, smoke, or oil films provide qualitative information and are most useful as a guide to the best location for the sensors. The sensor location and data to be measured are provided by an aerodynamicist. A common technique is a matrix of small holes drilled in the blade and tubing is then used to duct the static pressure to differential pressure transducers. Physical alteration of the blade must be accomplished under the direction of structural engineers. The tubing should be as short as possible to reduce lag in the system and to provide the best response to rapid pressure changes. In order to determine the pressure distribution accurately, such a large number of pressures must be sensed that the recording capability is often overloaded. Commutation is used to reduce the number of recording channels. The speed of commutation and the type of data must be such that interruptions do not invalidate the results. The static pressure changes are usually small values and high sensitivity is needed.

5.1.2 Blade Positions

The blade positions are controlled by inputs of collective pitch to all blades and cyclic pitch which varies as the blade azimuth changes. Aerodynamic forces cause vertical blade motions (flapping), in-plane motions (lag), and torsional pitching motions. The sum of those motions combine with rotational speed and free stream air to produce a local blade angle of attack. The blade angle of attack is different at each blade section and is changing very rapidly. Blade angle of attack is not measured directly but can be calculated from the pressure distribution data. For articulated or teetering rotors, blade flapping is measured at the blade hub attachment with a position transducer. Lag angle is also measured at the blade root in a similar manner. These transducers are generally potentiometers or linear variable differential transformers.

Blade azimuth during each revolution must be measured to evaluate significance of blade behavior. The main rotor speed can be measured with a tachometer, however, this does not give the azimuth of a particular blade at a point in time. One way to record blade azimuth is with stationary receivers which sense passage of a magnetic or optical device attached to the blade. The number of sensors needed per revolution will depend on the accuracy with which the azimuth must be established. The sensor signals can also be correlated with the rotational speed measurement from the tachometer. Optical and acoustic devices may also be used for blade position measurements.

5.2 Hubs

The rotor hub experiences large tension loads caused by centrifugal forces on the blade, bending moments from in-plane motion, vertical moments from blade flapping and torsion caused by blade pitching moments. Rotor hubs are usually complex forms and stress analysis is required to determine the location for the strain gages. The number and location of the sensors will be unique to each installation and must be established on a case by case basis. Resistance strain gages are used in a manner similar to that discussed for the blades.

5.3 Pitch Links

The collective and cyclic control is transmitted to the blades by pitch links. Aerodynamic and blade dynamic loads from the blades are also fed into the pitch links as are static loads when the rotor is not turning. Stress analysis will determine number and location of strain gages required. The gages are mounted to measure vibratory tension and compression. Strain gage specifications and mounting considerations have been previously discussed. In most designs the pitch link loads are transferred directly to the swash plate and can be measured at that location.

5.4 Data Transfer

The rotor data measured on the rotating parts must be transferred to the stationary data recording system in the aircraft. Mechanical slip ring and brush devices are the most common method. The strain gage signals are very low voltages and the slip ring assembly must not generate noise which will influence the data. Factors which must be considered in the slip ring design are:

a) Shaft speed and diameter
b) Slip ring material and surface condition (hardness, finish, eccentricity)
c) Brush material, contact pressure and number of brushes.

The slip rings can be mounted on the shaft and the brushes mounted on the stationary airframe. This arrangement usually gives poor performance because the main rotor shaft may move independently from the airframe which will affect the brush slip ring performance. This can be avoided by mounting the brushes on a stationary standpipe mounted inside the rotor shaft. With this installation, there is no relative movement between the brushes and the slip rings. Wiring is then routed inside the standpipe through the transmission and then to the data recorder. A typical standpipe installation is shown in Figure 5.4-1.

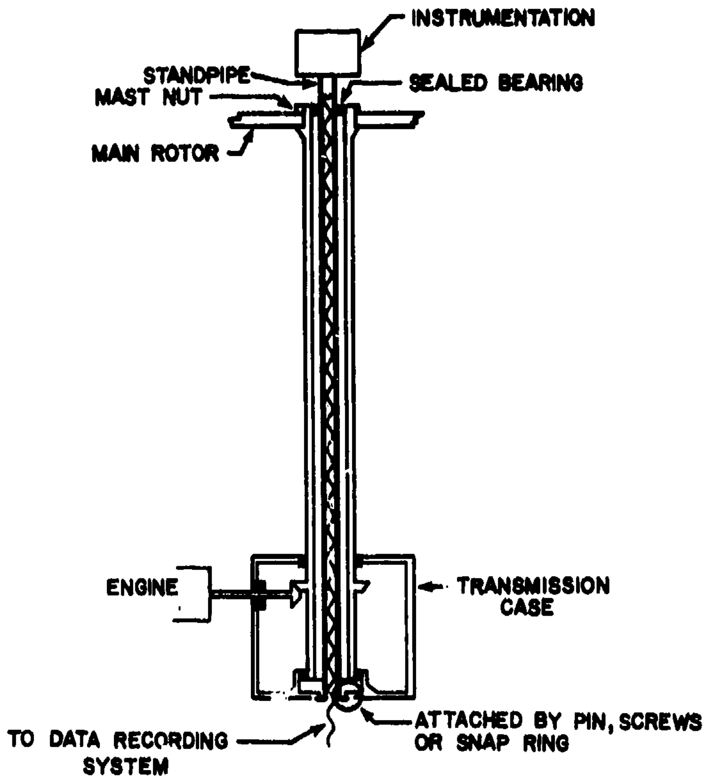

Figure 5.4-1
Typical Rotor Shaft Standpipe Installation

An alternate method to transfer the data is to use a telemetry system. A transmitter, receiving power inductively from stationary coils, is mounted on the shaft and a receiver is stationary on the airframe. The distance between the two should be minimized to reduce transmitter size and power requirements. Extreme care must be used so that shaft balance problems are not introduced by the transmitter installation. Some advantages of telemetry compared to slip rings are:

a) System can be designed for variety of installations whereas a slip ring is usually unique to one application.
b) Less noise and better quality data.
c) Less maintenance and more reliability

During development testing, testing of new rotor systems or flight in extreme aerodynamic conditions where rotor behavior must be closely monitored, a telemetering system on the rotor hub may be used with recording and analysis equipment on the ground.

5.5 Non-Rotating Surfaces

Many helicopters use horizontal and vertical stabilizers for stability and control and wing lift to augment rotor lift. The stabilizers can be fixed, connected to the flight controls, driven by electronic controls systems or dynamic pressure. The operation of the system will determine the instrumentation required. Instrumentation for stress and pressure distribution is accomplished as discussed for the blades. Surface position is usually measured as control and displacement or angular deflection from a specified zero or trim point. Transducers for those applications are usually potentiometers or linear variable displacement transformers.

6. FLIGHT CONTROL SYSTEM

Helicopter flight control systems vary from the most elementary mechanical arrangements to very complex systems which include mechanical, hydraulic, and electronic components. Control inputs in the cockpit are transferred through the system to the rotor and lifting surfaces. The input may be modified or shaped during the process. Aerodynamics of the rotor are continually providing feedback throughout the system. Aircraft motions provide pilot cues and those motions may also cause stability systems to respond. The instrumentation may simply measure the pilot actions or it may be required to measure each input and motion of every system component. When instrumenting the control system, special care must be used to insure that the instruments do not introduce forces, cause interference, or change the characteristics even if the test system should fail.

6.1 Cockpit Controls

Basic helicopter cockpit controls are cyclic stick for longitudinal and lateral control, pedals for directional control, and a collective lever for thrust control. When free, those controls are essentially cantilever beams and will vibrate as driven by the airframe or control system feedback forces. Linear accelerometers are placed at the top of the cyclic control and oriented to measure longitudinal and lateral vibrations. An accelerometer is placed on the collective to measure vertical vibrations. Similar accelerometers are placed on the pedals to measure longitudinal accelerations. Cockpit control forces are usually not measured directly unless a specially instrumented hand grip or hand held force gage is used. Normal practice is to mount strain gages on the control rod attached to the end opposite the hand grip or foot pedal. The cockpit control is usually a lever; and from the mechanical advantage and the measured force, the actual pilot input force can be calculated. Cockpit control motions are measured with position transducers. Those transducers are placed on the control rods attached to the cockpit controls. The transducer output is calibrated by moving the controls through full range of travel.

6.2 Mechanical Linkages

In structural tests it may be necessary to instrument all linkages from the cockpit controls to the stationary swash plate. This instrumentation will usually be strain gages or position transducers which are treated as discussed under cockpit controls.

6.3 Actuators

Control systems may have hydraulic or electrical actuators which transfer pilot inputs to the rotor. The actuators may also be driven by any stability and control devices. The resultant inputs are thus a sum of all inputs and must be measured to obtain the net control input. Electrical inputs are measured and the voltages are then used to determine equivalent linear deflection of the control input. The actuator stroke may be measured, or more commonly, the motion of the control member connected to the actuator is measured. These motions are sensed with a position transducer. Any electrical signals within the control system must be measured in a way which insures no change in the signal characteristics. This may require an isolation type of amplifier to eliminate any detrimental effect caused by tying signal conditioners to the control signals. In control system development it may be necessary to instrument for hydraulic fluid pressure or flow.

6.4 Swash Plate

The resultant of all control inputs culminate at the non-rotating swash plate where they are transferred to the rotary control system. In certain stability and control analysis the swash plate angle is a requirement. The longitudinal and lateral angles relative to the shaft or the airframe are measured with position transducers.

6.5 Stability Augmentation Systemss

Most helicopters have systems to improve the stability and control characteristics. Increasing use is also being made of flight director systems which assist the pilot in normal flight and increase the capability to operate in adverse weather conditions. These systems may be self contained or may use parts of the standard helicopter systems. The systems are essentially a computing device which receives information and on the basis of calculations or predetermined logic, makes an input to the flight control system. The instrumentation needed for the system varies greatly with type of test being conducted. During control system development and optimization tests, each input and response must be measured. For conventional flight tests, the system input to the flight control system is measured and correlated to aircraft behavior as recorded by the test instrumentation system. As was the case with actuator signal measurement, stability system electrical signals should be carefully assessed for requirement of electrical isolation of the measurement.

The stability system output is usually an electrical signal to an actuator placed in the flight control system. The voltage can be measured directly and through calibration or specification data the actuator motion can be determined. A more common practice is to measure the actuator motion with a position transducer.

7. WEAPONS SYSTEMS

Current helicopter practice is to use weapon systems that are an integral part of the design as opposed to earlier vehicles where the systems were added in the field as required. This development allows installation of more powerful systems and at the same time, gives an opportunity to minimize the weapons effects on the flight vehicle. However, greater complexity usually accompanies the increased capability and, in turn, more instrumentation is needed to measure and record the data. The total weapon system includes sighting and aiming controls and the weapon. The weapon system is usually developed and tested independently of the aircraft. This effort is concerned with assuring the system will deliver the specified ordnance without electrical, mechanical, or explosive deficiencies. The flight test of a weapon system then becomes a task of determining the weapon compatibility with the helicopter and crew. The weapon system will usually add drag, introduce loads into the airframe, and alter the stability and control characteristics.

7.1 Forces and Motions

Externally mounted weapons generate drag which transmits forces through the attachment hardware into the airframe. Recoil forces will be added during firing. Strain gages are placed on the airframe or on the weapon structure attached to the airframe. In the non-firing mode the weapon will react to rotor induced vibration through the airframe and, when firing, vibrations will be generated by the weapon. Instrumentation design must consider that rotor vibrations are usually low frequency while weapon firing rates can be up to 2000 rounds per minute which generates high frequency reactions. Accelerometers are usually oriented along the recoil axis of the weapon. Traversing or elevating weapons may require accelerometers in three axes.

7.2 Firing Effects

Gun type weapons generate significant overpressures which can cause structural damage. A pressure transducer is attached to the airframe where air pressures are the greatest. The number and location of sensors will depend on each particular installation. Missiles create exhausts which can impinge on airframe structure, stabilizers, and tail rotors. Those exhausts may cause surface heating which can be measured with thermocouples. In addition, ingestion of gun-gas products or missile propellant products can have a severe impact on engine operation. The hot gases can cause airflow disturbances which affect stabilizer lift or tail rotor thrust and cause a change in stability and control. Aerodynamics studies will determine the necessity of using anemometers to measure the exhaust.

7.3 Ejected Material

Gun systems usually eject shell casings, links, or cartridges into the free stream. Still air patterns from ground tests and expected airflow are used to estimate the in-flight dispersion patterns. Cameras are mounted to photograph the ejected material. The camera installation must not disturb the airflow or change flutter or structural characteristics of the member to which they are attached. Camera speeds of 400 frames per second will provide data suitable for a pattern analysis.

7.4 Fire Control Systems

Complex fire control systems utilize data inputs concerning the atmosphere, target, and aircraft conditions. These inputs may be from standard aircraft sensors or they may be an integral part of the weapon system. The inputs are fed to a computer which aims the weapon or makes corrections prior to firing or during the missile flight. In either case there may be a requirement to compare the system input to the computer with measurements from an independent test instrumentation system. Typical comparative parameters include airspeed, angle of attack, aircraft attitude, or acceleration. A completely instrumented test aircraft as previously discussed will provide adequate comparative data. In other cases the data requirements must be carefully studied and instrumentation added as required. Generally, measuring the fire control system is required to insure that the proper functions are being recorded. .

7.5 External Noise

Noise from weapon systems must be considered from crew exposure and from aircraft detection by ground personnel. Crew stations are instrumented to measure noise levels. Ground recording stations are established in a grid around the helicopter to record the noise for different flight and firing conditions. During forward flight, the helicopter is flown over a prescribed course through an instrumented range. Additional noise measurement discussion is presented in section 8.2.

7.6 Gas Contamination

Toxic or explosive gas from the weapon may enter the crew compartments and engine inlet or may impinge upon the airframe or rotors. Measurement of crew compartment contamination was discussed in the airframe section. Significant amounts of gas in any external area may require obtaining samples for laboratory analysis. Gas temperature is measured with thermocouples.

8 GROUND SUPPORT INSTRUMENTATION

Helicopter missions involve a relatively large amount of time spent near the ground at hover and low speeds; and it is necessary to accomplish much of the testing in a similar environment. In hover, the helicopter may contaminate the near field atmosphere such that airborne measurements are inaccurate or unreliable. Hover performance is most easily and safely conducted by measuring thrust rather than loading the aircraft with weight. During takeoff and landing maneuvers or near ground maneuvers the flight path must be corrected to a zero wind condition which requires measurement of ground distances and wind velocities. Independent data recording systems may be used and can cause extremely difficult data correlation problems.

8.1 Atmosphere

Atmospheric measurements include wind speed and direction, ambient air temperature, pressure, and humidity. The measurements should be taken as near to the helicopter as possible while ensuring that the air mass is undisturbed by the downwash. Temperature and wind conditions usually vary with height above the ground so that a ground surface measurement does not describe the helicopter operating environment. Best results are obtained when the sensors are mounted on a tower at various heights above the ground. The data then gives a profile of the gradients, inversions or shears. More than one tower allows comparison of the air upstream and downstream from the helicopter.

8.1.1 Wind Speed

Accurate measurement of small, rapid wind speed changes or local air velocities are best accomplished with a hot film anemometer. An ultrasonic sensor could also be adapted to this application. Lower response or time averaged data is usually obtained from a vane mounted pressure transducer or a cup anemometer.

Rotating cup anemometers carefuly constructed to minimize weight and friction, can be sensitive to speeds down to .25 m/s (.5 Kn). The greater the sensitivity, the more the instrument will respond to speed variation. However, these instruments are usually fragile and pose reliability problems. They are usually designed for a low speed range and can be damaged by gusty, turbulent air.

8.1.2 Wind Direction

Tri-axial hot film or ultra sonic sensors will provide the total vector in space (wind speed and direction). These high response instruments will show rapid changes.

For most applications a wind vane will provide acceptable data. With careful attention to design and construction details, vanes can provide data accurate to 0.5 degrees.

8.1.3 Ambient Air Temperature

The temperature sensors are usually placed at the same location on the tower as the wind instruments. This eases the equipment installation and aids the data correlation. The sensor must be shielded from solar radiation. Sensitivity and response are the most important criteria in selecting a sensor which will meet the data requirements. Time averaged data suitable for most requirements is provided by a low response sensor such as a resistance probe. Small rapid variations such as required for quality of airflow are obtained with hot wire or hot film anemometers.

8.1.4 Ambient Air Pressure

The ambient air pressure is usually measured at ground level and a standard lapse rate decrease for height above the ground is subtracted to obtain the pressure at the test vehicle.

8.2 External Noise

The helicopter noise is generated by the engines, transmissions and rotors. The helicopter internal noise measurements were discussed in section 4.7.3. The external noise is measured with fixed or portable ground located sensors. The noise is evaluated for tactical suitability for the military and in terms of environmental impact when operating in the public sector. Helicopters are inherently noisy despite efforts to reduce noise levels. These efforts include transmission design and manufacture noise isolation techniques, insulation application as well as number of blades and airfoil design. The sensor performance will be influenced by the atmospheric environment and these parameters must be measured. In addition, the ambient noise level must be established prior to taking measurements of the test vehicle noise.

The noise measurements are usually taken at different azimuths at specified distances from the helicopter while it is on the ground or hovering. Piezoelectric type microphones are commonly used. An amplifier system is interfaced to the microphone with preamplifiers used in some cases. The circuitry can be designed to meet program specific frequency response characteristics and provide the desired output by incorporating the necessary filters. The amplifier output data is recorded by magnetic tape or oscillographs for later analysis. Whether this is recorded in a direct or FM format on tape, care should be taken to insure that the frequency response is not degraded by the magnetic tape recorder. With oscillographs, the galvanometers must be selected and set up to insure that desired signal information is not degraded. Quick look capability can be provided in a fixed base system by use of a real time narrow band spectrum analyzer in parallel with the recorder. With the spectrum analysis, the number of samples averaged and therefore the degrees of freedom must be carefully chosen to obtain a desired confidence limit. This scheme allows the driving inputs at different frequencies to be quickly assessed for contribution to overall noise production.

Noise measurements for flight regimes other than hover may be necessary to evaluate speed effects on noise propagation. The most important effect is the impulse noise generated by blade tip-vortex interactions. The most common method is to fly over or near the hovering ground matrix of microphones. An alternate in-flight technique has been developed (ref 25). With this technique, microphones are placed on a pacer aircraft. The impulse noise from the rotor may be directionally sensitive and a lateral displacement of the microphones may be advisable. The pacer aircraft is fitted with an automatic recording device or equipment that will transmit the signals to a ground station. Provision should be made to adjust the instrumentation gain to optimize the signal to noise ratio. The typical peak pressure will vary from 10 to 500 Pa(1.45×10^{-3} to .07 PSI) with the maximum occurring at high advancing tip mach numbers in forward flight. The noise of the pacer aircraft is obtained prior to the test and taken into account either through instrument adjustments or later in the data analysis.

8.3 Thrust

Hover performance requires measurement of the thrust that the helicopter generates for a given power setting. This is usually accomplished by restraining the aircraft to the ground and measuring the various forces generated as a function of power or variation of thrust devices. To be effective the thrust stand must have the capability to change height above the ground and vary heading through 360° (Ref 24). Another system involves suspending the aircraft, and sensors are used to measure the changes in the forces (Ref 26). The sensors are usually an integral part of the thrust stand, and the data is recorded on the ground. Since the aircraft is restrained, the easiest way to transfer data is by electric cabling.

For other than thrust stand operations, a suitable ground restraint system with great flexibility can be constructed with instrumented cargo hooks or load cells. Load cells may be constructed by the test instrumentation group, or a commercial sensor may be used. The commercial equipment comes with various ranges and the sensor selected should be compatible with the expected thrust of the particular helicopter. The load cell is placed in series with cables attached to a ground restraint. It is important that the installation does not allow the load cell to drop and be damaged when the emergency cable release mechanism is activated. When the load cell is ground restrained, an electrical quick release must be placed between the aircraft and the load cell. The sensors are usually strain gages and the output is wired into the airborne data system and when possible is ground recorded.

The longitudinal and lateral deviation angles are measured with linear accelerometers mounted on the load cell. When vertical, the accelerometers read zero G and when horizontal, the output is 1 G. The angle is calculated from the measured G recorded. The accelerometer output is also displayed on the pilot's instrument panel to assist in establishing a hover that minimizes the deviation angle.

8.4 Space Positioning

Many tests require precise measurement of the helicopter position in space at a given time. These tests include take-off, landing, acceleration in three axes, and various agility maneuvers. In all cases, the distances are relatively short and the aircraft is near the ground. The space positioning system includes data acquisition, range support, and atmospheric measurements, and provisions for subsequent data processing.

The initial instrumentation planning should consider: (1) Test Site -- whether the tests will be conducted on an instrumented range or at a remote site; (2) Equipment location -- whether the equipment will be located in the aircraft or on the ground; (3) data recording -- whether the data will be ground recorded or recorded on the test aircraft. In all cases, provision must be made for items needed to control and conduct the test as well as document the data for later correlation, merging and processing.

With rare exceptions the space positioning system ground station layout will be needed. This will normally be precisely determined and readily available from an instrumented range. For temporary installations a survey is required. The space position systems provide motion relative to the ground while the aircraft is moving within the air mass. Atmospheric data is necessary to obtain correlation of air distance and ground distance. In either case the aircraft instruments must be able to measure the atmospheric conditions and any necessary performance or stability and control parameters.

8.4.1 Instrumented Range Operations

A permanent range installation will provide a space position data acquisition and recording system. Details concerning the equipment specifications should be obtained from the manufacturer or the range facility (Ref 27) . The range support instrumentation must provide means for conducting the test, measurement and recording of all data, data processing unique to the range equipment and any necessary interface with the test aircraft. The range timing system must be available to control the test sequence and provide correlation of data recorded at various locations. The range timing method may be different from the airborne time code generator and provision must be made to integrate the systems. Various methods of synchronizing timing systems include a physical connection between ground station and the test aircraft, using an R.F. link to impose time at one location on the other recording location, and recording a common standard source at each location. Communications for test control are usually voice transmissions over radio or telephonic instruments. Precise data information is communicated over wire lines or radio or telemetry links. The atmospheric measurements at the range will usually include information necessary to correct the raw data obtained from the acquisition system. Parameters such as noise, humidity, pressure, and temperature are necessary for assessment of aircraft performance and are usually measured by aircraft sensors. These data may be telemetered to the ground station and recorded with the space position data. Range facilities have various data processing capabilities. Typical equipment provides data readout, translation, format-conversion and automatic display.

Commonly used space position systems are optical, radar, doppler, and laser. A limited understanding of how the different systems operate is necessary to assess the instrumentation requirements. Ballistic plate cameras record aircraft images on a glass emulsion plate. These cameras may be fixed or may be tracking devices such as the Fairchild Flight Analyzer. Askania cinetheodolites track the aircraft and make a film record of the azimuth and elevation relative to a known set of coordinates. Ribbon film cameras, such as the Bowen-Knapp, track the aircraft and record the image along with fiducial markers which are projected onto the film. Recording optical tracking instruments use a telescope to track the helicopter and record the data on film. These systems have different accuracies and capabilities. A principal instrumentation consideration is that the data recorded outside the aircraft is difficult to correlate or merge with airborne recordings. A typical solution is to photograph external event lights on the aircraft while recording time or electronic identification data at other locations. In addition, the sample rates are inadequate to obtain accurate acceleration data.

The radar range systems may use either pulse or continuous-wave (cw) equipment. The most frequently used system is a pulse type with high peak power, wide band-width signal transmission, and a highly directive electro-mechanically steered antenna. The apparent radar range is derived from the time needed for the pulse to reach the target and return. Tracking system electronics maintain the antenna parallel to the returning wave front and the bearing is measured by tracking system. The data output is range, angles, and rate of change. The operation of the system and the many corrections which must be applied to the raw data are usually beyond the capability of the flight test personnel and must be accomplished within the range facility. Various equipment or tracking problems can be expected because of low angle multi-path and refraction conditions, high target accelerations and target-radar geometry. The cw radar determines distance by phase comparison of the transmitted and return signal.

With a doppler system, a signal is radiated by a transmitter on the ground. This signal is received on board the aircraft and retransmitted at a different frequency by a transponder. At least three receiver stations on the ground are needed to receive the reference frequency and the retransmitted frequency. These two frequencies are electronically subtracted and the difference is the doppler frequency at the particular receiver. The doppler frequencies are used to calculate the three dimensional position in space. Additional receivers allow statistical techniques to be used in accuracy analysis.

The most recent development in space position equipment utilizes a laser tracking system. The system radiates a short wave length signal which is highly collimated and power is adjusted as a function of range. The system operation is much like the tracking pulse radar system. The accuracy is extremely good with capability for high density data that can be transmitted in real time or recorded on magnetic tape. Tracking error is minimized by use of a retroreflector element, installed on the aircraft which provides a strong return signal that is tracked automatically. In addition to improved tracking the apparent range is independent of the many atmospheric variables which must be corrected for in radar systems. The Sylvania Electronics System has developed a self contained unit which is van mounted and can be transported to road accessible sites. This system incorporates a mini-computer and assorted data handling equipment.

8.4.2 Remote Site Operations

The versatility of the helicopter allows testing in remote sites where ground support and perhaps even a runway do not exist. All test equipment must be portable, use a minimum of power, and be able to operate in an adverse environment. Systems which are crude in comparison with range instrumentation can produce surprisingly good data when used with care. In most cases, some of the previously described optical systems are used because of simplicity and low power requirements. When optical systems are used to record data, visual theodolites can produce quick look data; and, in extreme situations, may be the source for final data. The greatest flexibility in remote site operations are achieved with airborne acquisition systems. The simplest system is a camera mounted on the aircraft which records terrain or markers during the test maneuver. The markers are usually runway lights or distances along the flight path. These markers must be carefully surveyed and the data accuracy is highly dependent upon interpretation of the film. Correlation of the film with other recorded data is also very difficult.

An aircraft portable radio ranging system has been developed by Del Norte Inc., Euless, Texas (Ref 28, 29, and 30). The system is solid state, compact and has a range of 4.8 km (3 miles). This system has a distance measuring unit (DMU) which controls all operations, a master unit (MU) which transmits and receives all signals, and remote units (RU) which receive signals from the master unit and retransmits a signal. The airborne equipment operates on a 24 volt D.C. power while the ground units are powered with 12 volt D.C. battery power. The system is line of sight and operates on radio frequency signals. The DMU and MU are located in the aircraft and the remote units are placed along the test area. As many as eight ground units may be used. The remote stations require no ground support personnel and for simple runway distance a single remote unit can be used without a site survey. The time for the signals to travel to the remote units and return is measured and provides slant range distance. Data from individual ground stations can be examined for random points, dropouts, or multi-path interference. Stations which provide best data quality can be used to calculate horizontal and lateral displacement to within ±1 m (3 ft). However, at relatively low heights above the ground, small errors in range create extremely large height errors in the computations and the data is not useable. Height above ground is best obtained from a radar altimeter. Extreme maneuvers may cause radar altimeter problems, in which case, a precision pressure altimeter should be considered. The system should be field calibrated before each test. The RU is placed 1000 m (3300 ft) from the aircraft and the range calibrate screws on the DMU are adjusted until the correct range is displayed on the unit. The system has an adjustable measuring rate.

The test aircraft may have a doppler or Inertial Navigation System (INS) which can be used to obtain accurate space position information (Ref 31, 32, 33, 34, and 35). Such a system may be used in conjunction with an instrumented range; and since it is self contained within the aircraft, may be useful for remote site operation. The navigation system may be a stand alone system, however, it is more common to have a central unit which interfaces with other aircraft systems. The system must be carefully analyzed to determine what information is available from the system and how this data can be obtained without altering the system operation. The navigation system measures component velocity in the aircraft axes and in conjunction with a heading input computes ground speed and direction. Inputs from the airspeed system are then compared with ground speed to obtain wind information. Some systems also make navigation corrections for attitudes, angles of attack and sideslip. With the doppler system, four signals are transmitted and speed is determined from the doppler shift in the return signal. Inertial systems use accelerometers in the aircraft axes to provide data for speed calculations. Depending on the system, it may be desirable to record either the accelerometer data or the differentiated output. Navigation systems are intended for trimmed flight and the true airspeed calculations may be affected by sideslip angle. This is of particular importance in helicopters which frequently have large sideslip angles at low speed or during crosswind maneuvers. Complete space positioning data is obtained from the navigation system ground speed components combined with a low airspeed omni-directional airspeed system and ground atmospheric measurements.

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APPENDIX I

TYPICAL HELICOPTER INSTRUMENTATION REQUIREMENTS

This appendix provides typical requirements for a helicopter instrumentation installation. While certain characteristics can be specified for each parameter, it is usually necessary to make adjustments dependent upon the nature of the test. Some common variances have been noted in the remarks section. The accuracy stated is based primarily on the data requirements and resolution should be adjusted as necessary. GREAT CARE SHOULD BE USED TO SPECIFY NO MORE ACCURACY OR RESOLUTION THAN IS ESSENTIAL. Conversely, instrumentation should comply with the specification if feasible and of course, any additional capability will enhance the quality of the results. The helicopter can be expected to generate vibrations which will affect each sensor used. The vibration frequency will vary from a one per main rotor revolution to high speed jet engine frequencies. Each sensor should be evaluated with respect to the driving frequency it should experience. For a particular location this value is then used to establish the maximum frequency of interest for electronic filtering and signal conditioning. In many cases a careful study of the sensor characteristics will greatly reduce the amount of electronics needed.

REFERENCE	PARAMETER	UNITS	RANGE	ACCURACY	RESOLUTION	REMARKS
2.1.1	Free Air Temperature	°C	35 to 55	0.5	0.1	Expand Range to -65°C for cold weather testing
2.1.2	Pressure Altitude	ft	-1000 to 20,000	5 ft at SL	0.1	Quartz capsule with microcomputer
2.1.2	Radar Altitude	ft	0 to 1000	1.5	1.0	100 ft vernier with 5 ft increments for cockpit
2.1.2	Pressure Rate of Climb	ft/min	±5000	5.0	2.5	
2.1.2	Radar Rate of Climb	ft/sec	±100	1.0	±1.0	
2.1.3	Dew Point Temperature	°C	-35 to 50	0.5	0.1	
2.1.4	Liquid Water Content	G/M^3	0 to 3	0.1	0.05	
2.1.4	Droplet Size	microns	0 to 400	1.0	1.0	Laser nephelometer
2.2.1	Pitot-static Airspeed	Kn	20 to 250	1.0 or .5%	0.5%	Swivel head with 20° freedom in all directions
2.2.1	Angles of Attack and Sideslip	deg	±180	1.0	0.5	
2.2.2	Relative Wind Vertical	deg	±180	1.0	0.5	
2.2.2	Relative Wind Azimuth	deg	±180	1.0	0.5	
2.2.2	Omni-directional Airspeed	Kn	50 in all directions	2.0	0.5	Longitudinal may be extended to 250
3.1.1	Gas Generator Speed	%	50 to 110	0.1	0.05	Frequency or period counting digital tach with 10μ sec quartz clock reference
3.1.2	Shaft Speed	rpm	0 to 100%	0.1	0.05	
3.3	Shaft Torque	ft-lb	50 to 100%	5%	0.1	Transducer used is often the normal aircraft torque system
3.4.1	Inlet Pressure	lb/in^2	±1 psid	±3.5%	0.1	Accuracy includes error of pressure rake.

REFERENCE	PARAMETER	UNITS	RANGE	ACCURACY	RESOLUTION	REMARKS
3.4.2	Inlet Temperature	°C	±30°C	±1	0.1	
3.5	Engine Temperature	°C	900°C	±3.0	0.5	Over normal operating range
3.6	Engine Pressure	lb/in^2	150 psig	±2%	0.5	Bleed Air
3.7.1	Fuel Flow	gal/hr	Variable by acft type	1%	0.1	Volume measurement used primarily with turbine sensor
3.7.2	Fuel Temperature	°C	0 to 50	1	0.1	Platinum probe
3.7.3	Fuel Used	Gal	Variable by acft type	1%	0.1 gal	Volume measurement used primarily with turbine sensor
3.8	Electromotive Force	Volts	0 to 100%	±1.0	0.1	A/C power; source generally 28VDC or 115VAC
3.8	Electric Current	Amperes	0 to 100%	±1.0	0.1	A/C power; source generally 28VDC or 115VAC
3.9.1	Cockpit Power Controls	deg	0 to 100%	±2%	0.5	Generally repeatable to ±1%
3.10	Engine Vibration	G	Varies by location and axis	±3%	0.1 G typical	Piezoelectric accelerometer
4.1.1	Pitch Attitude	deg	±45	±1.0	0.5°	
4.1.1	Roll Attitude	deg	±60	±1.0	0.5	
4.1.2	Yaw Attitude	deg	±180 max	±1.0	0.5° min	*180 max, but can be reduced for increased resolution
4.2	Angular Pitch Rate	deg/sec	±30	±1	0.1	
4.2	Angular Roll Rate	deg/sec	±100	±1	0.1	
4.2	Angular Yaw Rate	deg/sec	±60	±1	0.1	
4.4	Linear Acceleration	G	Z +4, -1 X and Y ±2	±0.1	0.001	
4.5	Airframe Vibration	G	2	±3%	0.001	Can include both low and high frequency accelerometers
4.6.2	Structural Loads	μ"/"	0 to 100%	±5%	0.5%	100% should be 10% above expected yield point
5.1.3	Blade Positions	deg	0 to 100%	±2%	0.5%	100% should be 10% above the maximum calculated
6.1	Flight Control Positions	in	0 to 100%	±2%	0.5%	
6.2	Flight Control Forces	lb	Long, Lat, & Dir = 10 Pedal = 25	±5%	0.5%	Range dependent on aircraft and specification require-
6.3	Actuator Position	in	0 to 100%	±1%	0.5%	
6.4	Swash Plate Angle	deg	0 to 100%	±2%	0.5%	100% is maximum flight Control input

APPENDIX II

TYPICAL INSTRUCTIONS FOR DEVELOPING AND MAINTAINING RECORDED INSTRUMENT PARAMETER LIST

This appendix provides procedures and instructions for preparation and use of an instrumentation form during formulation and conduct of an engineering flight test program. Emphasis must be placed on the bookkeeping to insure that the instrumentation configuration and status is correct for any proposed test. This information generally typifies the approach of the flight test community, however, it should be modified to accommodate specific procedures/instructions that may vary widely. These instructions are for a pulse code modulation (PCM) data system which is in most common use today. The purpose of the form is to consolidate and standardize all of the airborne recorded instrumentation project information to eliminate common coordination errors. The sample form shown and these instructions should be modified to meet individual requirements. A form should be completed for each project using airborne recorded instrumentation.

The instrumentation or data systems office is the proponent for the form and is responsible for maintaining the status and instructions current.

Chronologically, the following actions are taken by the following responsible individuals to prepare and maintain the Recorded Instrument Parameter List(s):

Flight Test Engineer (F.T.E.)

The F.T.E. requests airborne instrumentation from the instrumentation or data systems office. The request will contain schedules, controls, displays, instruments, and other information. The majority of information will be transmitted by attaching a draft form with the first 9 columns (except PCM/FM CHAN) completed. This will be the basis for completing the master list. Polarities should be conventional with the possible exception of vertical acceleration which is positive downward in some systems. Use of these polarities is mandatory. Provisions should be made for multi-engine/rotor helicopters and FM data. Nominal ranges and gains (zero count and max count or per count) are used on the draft. Actual ranges and gains will be obtained from the calibrations.

Instrumentation Engineer (I.E.)

The instrumentation engineer will arrange for calibration and installation of the requested parameters. In coordination with the project engineer or programmer, he will complete PCM/FM CHANNEL assignments and SIGNAL CONDITIONER information. The instrumentation engineer and flight test engineer will review and approve all calibrations after they have been completed and plotted. Calibrations are then provided to the instrumentation technician.

Instrumentation Technician (I.T.)

The instrumentation technician is responsible for the physical installation of all requested parameters in coordination with the flight test and instrumentation engineers. He is also responsible for performing required on board calibrations. The I.T. will provide all information required in the transducer section for each parameter. This will generally be done via the calibration data sheets.

Data Systems Technician

The data systems technician has overall responsibility for completion, maintaining currency, and distributing copies of the Recorded Instrument Parameter List. He is specifically responsible for providing information in the CALIBRATION section after he has run the calibration and it has been approved by the engineers. The specific procedures used to complete the initial list follow:

(1) As calibration data sheets are received, calibrations are run and plotted. They are checked for obvious errors.

(2) The master plot and data sheet are then given to the I.E. and a plot copy to the F.T.E. for approval.

(3) After approval (or recalibration) CALIBRATION information and TRANSDUCER information are inserted in the master list draft. The calibration deck is assembled simultaneously.

(4) Master plots and Calibration data sheets are then filed in the aircraft instrumentation file in the I.E. office.

(5) After receipt and approval of the last calibration, the draft form is typed for the initial master list.

(6) A listing is made of the completed calibration deck. The original is filed in the aircraft instrumentation file and a copy given to the F.T.E.

UNITS: Engineering units used in calibration and data reduction software.

WORD NO: Software calibration channel assigned by F.T.E. or programmer in coordination with instrumentation engineer.

PCM/FM CHAN: PCM main frame location or FM band center frequency. D suffix indicates digital channel. Decimal notation indicates bits used in split digital channels. Assigned by instrumentation engineer in coordination with flight test engineer.

CALIBRATION

POSITIVE POLARITY: Direction or sense of physical input that corresponds to increasing PCM counts or FM frequency.

Engineering value at:

ZERO COUNT: Engineering unit value that corresponds to zero (000) PCM counts or lower FM band edge.

PER COUNT: Slope of relation between engineering units and PCM counts or FM subcarrier (gain). Note; This also corresponds to the resolution of the parameter. For non-linear calibrations, calibration function coefficients can be substituted.

MAX COUNT: Engineering unit value that corresponds to PCM full scale or FM upper band edge.

PCM counts at:

ZERO VAL: PCM count or subcarrier frequency that corresponds to an engineering value of "zero" if applicable (i.e., zero roll rate).

MAX VAL: PCM count or subcarrier frequency that corresponds to a maximum positive or other specified engineering unit value, (i.e., "full" 100 percent control deflection). Note: This does not necessarily correspond to the engineering value at Max Counts.

R-CAL: PCM count or subcarrier frequency that results from activation of a system self test feature. An acceptance tolerance can also be specified for preflight purposes.

DATE: Latest parameter calibration Julian date (last digit of year + day of year, i.e. 5052 = 21 February 1975.)

SIGNAL CONDITIONER

NO: Index number assigned to signal conditioning equipment used for parameter.

CHAN: Subdivision number in multiple channel signal conditioning unit.

FILTER: Basic filter characteristic and corner of pre-sample filters used in analog signal conditioning, i.e., BLP 10 = Butterworth low pass filter with 10 hertz cutoff.

TRANSDUCER

NO: Arbitrary index of transducers

TYPE/MODEL: Manufacturers (name plate) type, model, or part number or transducer.

SERIAL NO: Manufacturers (name plate) serial number of transducer.

SIGNAL: Code identifying the form of the signal generated by the transducer. The code is as follows:

Analog Signals
AHL = (DC) Analog High Level, Bipolar
AHLP = (DC) Analog High Level, Unipolar Positive
AHLN = (DC) Analog High Level, Unipolar Negative
ALL = (DC) Analog Low Level, Bipolar
ALLP = (DC) Analog Low Level, Unipolar Positive
ALLN = (DC) Analog Low Level, Unipolar Negative

AC Signals
CX = Control Transmitter
CT = Control Transformer
CR = Control Receiver
Y1 = X1 = Synchro Transmitter/Receiver Line Polarity
Y2 = Y1 = Synchro Transmitter/Receiver Line Polarity
Y3 = Z1 = Synchro Transmitter/Receiver Line Polarity
TG2 = Tachometer generator - number of poles

(7) Copies of the typed Recorded Instrumentation Parameter List are distributed after the last calibration. The master list is filed in the aircraft instrumentation file.

(8) Normally six (6) copies are made and distributed; two to the F.T.E., one in the aircraft, one in the Data Processingfile, one to the instrumentation technician and one retained by the data systems technician as the master correction draft. The F.T.E. may specify other distribution.

(9) All copies will be made in reduced size (8½ x 11) and on both sides (if two sheets are required) so that the complete information is on one piece of paper.

Normally a few minor changes can be made by hand on the copies. After several changes have accumulated, the list will be revised using the following procedure:

(1) Last effective flight for that list version will be filled in and a full size copy made and filed for historical purposes.

(2) New applicable first flight, last change date, and date, flight, and reason for change of each affected parameter will be completed and retyped.

(3) Copies and distribution will be made as on the initial list.

(4) If applicable, a new calibration deck listing will be made, copies distributed, and filed. There should be a direct correlation between the Recorded Instrument Parameter List versions and calibration deck listings.

(5) A brief chronological log will be maintained of all changes, their effective date and flight and the reason for the change.

Detailed instructions for completing the form. This contains detailed instructions for completing each entry on the form. If a particular entry is not required for a parameter, enter NA (not applicable) in that block. For some parameters it may be desirable to substitute information other than that described here. In that case, inform the instrumentation engineer so that future instructions can be updated.

<u>Typing Information</u>

This form is arranged for standard elite type spacing (12 characters per inch, 6 lines per inch with 1½ space vertical spacing except for heading information. The maximum number of characters per column is listed by each block title in the following instructions. The form size is 11 x 17 inches and can be typed in standard size typewriters by folding it on the line between the major headings of "CALIBRATION" and "SIGNAL CONDITIONER". The entire form does not need to be retyped each time corrections or changes are made. Correction tape can be applied over previous entries. The only requirement is that clear, reduced size copies can be made.

NOTE: Great care should be taken in typing and proofreading this form as it is used by several people for a variety of purposes. A single mistyped or misplaced character can have a significant impact.

Instructions for completing each blank are detailed below.

<u>PROJECT</u> Project Number

<u>AIRCRAFT</u> Test aircraft designation; status, mission, type, model and series.

<u>S/N</u> Serial number (tail number or other designation). Note: Programs, calibration decks, tapes, etc., will be filed by S/N.

<u>SHEET _ of _</u>. Number in sequence of sheets required for complete list (normally two will be required, i.e., 1 of 2 and 2 of 2.

<u>EFFECTIVE FLIGHTS __ THROUGH __</u>. Series of flights for which this version of list is applicable, i.e., flights 114 thru 149. Last effective flight will be filled in when list is updated and before new copies are made.

<u>ORIGINAL LIST DATE</u> Date of completion of original basic list.

<u>LAST CHANGE DATE</u> Date when list was updated for recent changes and new copies were made and distributed.

COLUMN ENTRIES

<u>PARAMETER</u>: Measurement name

DATA PROCESSING

<u>NAME</u>: Mnemonic name assigned to the measurement in data reduction software.

Digital Signals
PD NB 08 example

First two characters: PD = Parallel digital
SD = Serial digital

Second two characters: NB = Natural Binary
BC = Binary coded decimal
2C = 2's complement
3C = 3's complement
OB = Offset Binary
GR = Grey code

Third two characters: Number of bits, decades, or octaves.

LOCATION: Location of transducer measurement using aircraft reference system numbers.

FS Fuselage station (longitudinal)
BL Buttline (lateral)
WL Waterline (vertical)
WT Weight of transducer and associated equipment in pounds (to the nearest pound)

CHANGE

DATE: Effective Julian date of last change to each parameter. Note: Not required on original list.

FLT: First effective flight number for which the change is applicable.

REMARKS

REMARKS: Reason for latest change or any other pertinent information relating to that particular parameter.