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ENGINEERING DESIGN HANDBOOK, HELICOPTER PERFORMANCE TESTING

Army Materiel Command
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ENGINEERING DESIGN HANDBOOK

HELICOPTER PERFORMANCE TESTING

HEADQUARTERS, U S ARMY MATERIEL COMMAND AUGUST 1974

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ENGINEERING DESIGN HANDBOOK
HELICOPTER PERFORMANCE TESTING

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FOREWORD

The helicopter industry is a highly competitive field of endeavor which by its very nature requires both practical and theoretical engineering of the highest caliber. Research and advance design progress are such that the existing models are obsolete as soon as, and often before, they can become operational. Further time lags often are introduced by production schedules, modification requirements, and retrofit actions. Flight testing is an integral part of conceiving, designing, and producing a useful flight vehicle. This production cycle with its great cost and effort obviously places a great demand on the resources of any particular company. Performance, safety, cost, time, and expediency are some of the many factors which influence any aircraft development.

The flight test effort often is considered by many to be the demonstration of the aircraft and not a great deal of anything else. This view is far from the truth. Test results should be incorporated throughout the development cycle, and failure to accomplish this may compromise or jeopardize the end product. Pilot considerations should be included in the initial design phases to insure man/machine compatibility. Pilot effort and opinion during the simulator tests provide the first indication of conceptual feasibility and any significant problems. Production and product improvement tests define operating characteristics and allow for the development of improvements or solutions to deficiencies as they are discovered.

As the state-of-the-art advances, the aircraft become more complex with accompanying difficulties in the development cycle. Also, the customer organizations become larger, more efficient, demand more reliability and accuracy, and in general, refine their capability in all areas. The manufacturers and Government in turn are forced to provide more comprehensive and accurate information about the product, thus requiring an ever increasing flight test effort. This handbook discusses flight testing as it relates to helicopter performance determinations.

This handbook was written by Kenneth R. Ferrell, Chief Advanced Methodology and Analysis Office, US Army Aviation Test Activity.

PREFACE

The realm of engineering flight test as it exists today developed primarily out of the necessity for defining the performance, stability, and control of military and commercial aircraft. The requirement for accurate, repeatable data is of particular importance for these applications since many people may be affected by failure to accomplish a given flight plan.

Practical operations make it obvious that theoretical data do not provide sufficient accuracy for predicting actual operating characteristics. Also as the aircraft become more complex, the number of undefined parameters becomes larger and the necessary information becomes increasingly difficult to obtain through the usual wind tunnel and model testing. Flight testing thus was from inception, and is today, a significant and essential part of any development and production effort.

Flight testing may be classified into various categories such as research, development, production, functional system, and handbook testing. Each of these types of testing has their unique objectives, techniques, and worths. Research and experimental testing is being accomplished by both military, educational, and commercial organizations. This testing usually is conducted on test beds, modified versions of current aircraft, and on unique experimental vehicles. Research testing has objectives concerned with advancement of the overall state-of-the-art and in obtaining information relative to specific areas. Developmental testing is generally considered to be defining and improving an established concept or aircraft. Production testing is devoted primarily to evaluating the production aircraft to insure that production tolerances and changes have not introduced variations that will violate delivery standards. The majority of this development and production testing is accomplished during the early stages of the production cycle. Product improvement testing normally continues as long as the aircraft is in production.

Improvements, modifications, and new equipment are evaluated as required by the customer or by the manufacturer when considered advisable. Functional and handbook testing is of primary concern to the using organizations. For the military, this testing consists of operational and suitability evaluations relative to mission accomplishment and logistic requirements. The testing includes different environmental conditions and is conducted at various locations which provide the necessary variables. Some of these tests are conducted with operational units under actual operating conditions.

Handbook testing is defined as that testing necessary to produce the pilot's handbook of operating instructions. Two sections of the handbook are of primary concern. One is definition of the flight characteristics with particular emphasis on precautionary items and emergency procedures. Another is the performance section that provides data for the entire operational envelope of the aircraft. Because of the critical nature and great importance of the handbook data, emphasis is placed on accuracy and completeness. A part of the tests is to determine the capability of the aircraft as well as to establish the performance penalties for improper operating procedures and flight techniques. Atmospheric conditions and the configurations available are also factors that are considered.

Many enter the helicopter field of endeavor with the conception that little or no difference exists between helicopter and fixed wing flight test. Needless to say, the fallacy of this assumption is soon discovered. True, the overall objectives are the same and the general approach is similar, but there the paths diverge. The intimate details concerning nearly every facet are greatly different, and it is necessary to become accustomed to these differences. The vibration characteristics, low speed, and autorotation flight regimes are especially disconcerting to one familiar with high speed fixed-wing aircraft.

This writing is an attempt to provide some basic testing considerations that perhaps can be used to consolidate, stimulate, and further the profession. No attempt has been made to present theory beyond that required to accomplish the particular test being discussed. This theoretical information can be found in many references and it is not necessary that it be repeated here. This is, rather than theory, some considerations relative to *practical helicopter flight testing.*

The Engineering Design Handbooks fall into two basic categories, those approved for release and sale, and those classified for security reasons. The Army Materiel Command policy is to release these Engineering Design Handbooks to other DOD activities and their contractors and other Government agencies in accordance with current Army Regulation 70-31, dated 9 September 1966. It will be noted that the majority of these Handbooks can be obtained from the National Technical Information Services (NTIS). Procedures for acquiring these Handbooks follow:

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Comments and suggestions on this Handbook are welcome and should be addressed to:

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DA Forms 2028 (Recommended Changes to Publications), which are available through normal publications supply channels, may be used for comments/suggestions.

LIST OF SYMBOLS

a	= speed of sound; kt, ft/sec
A	= main rotor area, ft^2
	= acceleration, g's
A_s	= engine inlet area, ft^2
ΔA	= change in acceleration, g's
b	= number of blades, dimensionless
BVP	= bleed valve position, % from full open
c	= equivalent blade chord (on thrust basis), ft
C_L	= average lift coefficient, dimensionless
C_P	= power coefficient, dimensionless
C_T	= thrust coefficient, dimensionless
C_{T_S}	= aim thrust coefficient, dimensionless
CG	= helicopter center of gravity
ΔC_P	= differential power coefficient, power coefficient correction; dimensionless
ΔC_T	= differential thrust coefficient, thrust coefficient correction; dimensionless
D	= distance traveled, ft
D_{ACC}	= acceleration distance, ft
D_G	= ground distance, ft
D_{LEVEL}	= horizontal projection of D_{SLOPE}, ft
D_{SLOPE}	= distance along runway to takeoff, ft
E	= endurance, hr
EGT	= exhaust gas temperature, °C
$ESGW$	= engine start gross weight, lb

LIST OF SYMBOLS (Cont'd)

F = force, lb

F_g = engine thrust, lb

F_G = gear load at touchdown, lb

FC = fuel counter reading, counts (ct)

ΔFC = fuel counter difference, ct

FM = figure of merit, dimensionless

FS = fuel specific weight, lb/gal

FSW = engine fuel specific weight, lb/gal

FU = fuel used; counts, lb, gal

g = acceleration due to gravity, ft/sec^2

GW = gross weight, lb

ΔGW = gross weight deviation, lb

H = height above ground, ft

HP_{id} = ideal horsepower, HP

H_D = density altitude, ft

H_G = gear height above ground, ft

H_p = pressure altitude, ft

ΔH = height difference, correction, error; ft

ΔH_p = altimeter instrument difference, correction, error; ft

$\Delta H_{p_{PE}}$ = altimeter position error correction, ft

IGE = in-ground effect

IGV = inlet guide vane position, % from full open

K = total pressure distortion, dimensionless

LIST OF SYMBOLS (Cont'd.)

K_{c_P}	= power coefficient constant, slug/ft^3-HP
K_{c_T}	= thrust coefficient constant, lb/slug-ft^3
K_F	= fuel counter to fuel volume ratio, counts/gallon (ct/gal)
K_{FC}	= fuel counter constant, gallons/count (gal/ct)
K_{GR}	= gear ratio, dimensionless
K_{H_P}	= altimeter temperature correction factor, dimensionless
K_P	= power correction factor, dimensionless
K_{TM}	= torquemeter conversion factor, in.-lb/psi
K_{T_e}	= temperature recovery factor, dimensionless
K_W	= gross weight correction factor, dimensionless
KTAS	= knots true airspeed
L	= static pressure distortion, dimensionless
LT	= left
M	= Mach number, dimensionless
M_B	= advancing blade Mach number, dimensionless
M_{TIP}	= advancing blade tip Mach number, dimensionless
N	= revolutions, rpm
ND	= nose down
NU	= nose up
N_e	= engine speed, rpm
N_R	= rotor speed, rpm or rps
N_{II}	= power turbine speed, rpm
NAMPP	= nautical air miles per pound of fuel (specific range), air-n mi/lb
NAMT	= nautical air miles traveled, n mi

LIST OF SYMBOLS (Cont'd.)

ΔN_R	= change in rotor speed, rpm
OGE	= out-of-ground effect
P	= power, HP
	= pressure; psi, in. H_2O, in. Hg
P_{s_g}	= exhaust gas static pressure, psi
PR_e	= engine (turbine) pressure ratio, dimensionless
ΔP	= pressure difference, correction, error; psi, in. H_2O, in. Hg indicated torque (see Eqs. 14-6 and 14-8), psi
Q	= actual torque, lb-in.
R	= range, n mi
	= rotor radius, ft
RT	= right
R/C	= tapeline rate of climb, ft/min
$(R/C)_{\Delta H}$	= average rate of climb from H_{p_1} to H_{p_2}, ft/min
$\Delta(R/C)$	= rate of climb correction, ft/min
R/D	= touchdown rate of descent, ft/min
S	= ground distance traveled after contact, ft
SHP	= power required, HP
t	= time, elapsed time; sec
Δt	= time change, required, differential; sec
T	= temperature, °C
	= total thrust, lb
TAS	= true airspeed, kt

LIST OF SYMBOLS (Cont'd.)

T_C	= restraining thrust, lb
T/C	= time to climb, min
TR_e	= engine (turbine) temperature ratio, dimensionless
T/W	= takeoff thrust to weight ratio, dimensionless
ΔT	= temperature difference, correction, error; °C
	= restraining thrust correction, lb
V	= airspeed, wind velocity; kt
V_{TIP}	=advancing blade tip speed; kt, ft/sec
ΔV	= airspeed deviation, loss, correction; kt
W_a	= air flow, lb/sec
W_{a_i}	= ideal mass flow, lb/sec
W_B	= weight of ballast, lb
W_f	= fuel flow; lb/hr, lb/sec
W_{INSTR}	= weight of load cells and cable, lb
WF	= fuel available, lb
ΔW	= incremental fuel, lb
X	= course speed length, horizontal distance; ft
α	= angle of attack, deg
	= azimuth, deg from north
β	= angle of sideslip, deg
γ	= flight path angle
δ	= ambient air pressure ratio, dimensionless
δ_s	= stick position, in.

LIST OF SYMBOLS (Cont'd.)

$\bar{\delta}_s$	= average stick position, in.
δ_T	= throttle position, % from full open
$\Delta\delta_s$	= change in stick position, in.
η_R	= total pressure recovery factor, dimensionless
θ	= runway gradient, theodolite angle, pitch attitude; deg
	= air temperature ratio, dimensionless
θ_B	= maximum retreating blade angle, deg
θ_{c_p}	= blade collective pitch, deg
$\Delta\theta$	= change in pitch angle, deg
$\dot{\theta}$	= pitch rate, deg/sec
$\ddot{\theta}$	= pitch acceleration, deg/sec²
μ	= advance ratio, dimensionless
$\dot{\mu}$	= mass flow, slug/sec
ρ	= air density, slug/ft³
σ	= air density ratio, dimensionless
σ_m	= rotor solidity, dimensionless
ϕ	= roll attitude, deg
$\dot{\phi}$	= roll rate, deg/sec
$\ddot{\phi}$	= roll acceleration, deg/sec²
ψ	= azimuth for maximum blade angle, yaw attitude; deg
$\dot{\psi}$	= yaw rate, deg/sec
$\ddot{\psi}$	= yaw acceleration, deg/sec²
Ω	= rotor angular velocity, rad/sec

SUBSCRIPTS

a	= ambient, free stream, lateral

LIST OF SYMBOLS (Cont'd.)

A	= available, advancing
AR	= airflow ratio
AVG	= average
B	= boom
BR	= best range
c	= counts, collective
C	= corrected
CAL	= calibrated
e	= engine, longitudinal
f	= fuel
F	= flare
FC	= fuel counter
g	= gas
G	= ground, ground station, gear
GR	= ground roll
H	= hover, horizontal
HW	= head wind component
IB	= interstage bleed
IC	= instrument correction
IND	= indicated
MAX	= maximum
MIN	= minimum
N_R	= rotor speed
P	= power, pressure, pacer
PE	= position error

LIST OF SYMBOLS (Cont'd.)

r	= pedal
R	= resultant, rotational, rotor
REF	= referred
REQ	= required
RW	= runway
s	= static droop, static, stick
S	= standard, standard system, standard day
SC	= system correction
$SCHED$	= scheduled
SL	= sea level
t	= test, test day, test system
T	= true, transient droop, total, turbine
TC	= test cell
TIP	= tip
TL	= tapeline
TO	= takeoff, tower
TR	= tail rotor
V	= vertical, airspeed
VOL	= volumetric
W	= wind, wheel/skid, hover, weight
WC	= wind component
WS	= wind shear
x	= horizontal component

LIST OF SYMBOLS (Cont'd.)

z	= normal component
o	= initial
1	= start, entry
2	= final stop
50	= 50-ft obstacle

CHAPTER 1

PLANNING

1-1 GENERAL

The same basic technical approach that is applied to testing should be applied to the planning, management, and organizational phases of the program. All aspects should be considered with emphasis being placed on management principles that will be compatible with the technical requirements. The most effective planning can accomplished only when both the technical and support groups are knowledgeable and sympathetic with the peculiar requirements of the other.

1-2 PRELIMINARY PLANNING

The preliminary planning is the first action that should be taken by a test agency on a subject program and should begin as soon as the program is known to exist. In the event of a cancellation at the last moment, this early planning effort may be lost; however, in view of the potential gains, the risk should be taken. A thorough effort at the onset should result in logical, timely program progress and should minimize the risk of lost productivity caused by unpreparedness at a later date. The preliminary planning includes an evaluation of the total scope of the program; a definition of the objectives, authorities, and responsibilities; an assessment of the time and resources necessary to accomplish the tests; a schedule establishment and milestone defintion; establishment of a suitable work flow channel; and establishing the appropriate follow-up action.

The request or direction to accomplish a test program should include the objectives in sufficient detail to allow a test plan to be prepared. The objectives should be studied carefully in light of the existing test capabilities and then a determination made as how best to achieve the desired results. The objectives should be clear and concise. Any item that is not so stated is open to misinterpretation and should be clarified immediately. In a similar manner the authorities and responsibilities of the program should be as equally clear as the objectives. Most organizations are adequately governed in these areas; however, when the organization is not adequate or when unusual circumstances arise, a clarification should be made immediately.

Upon receipt of the test objectives and authority, every possible effort should be made to obtain all information concerning the test vehicle. The most important information is that relative to its physical nature and operating characteristics. These will greatly influence all aspects of the testing to be done. The most fertile area from which to obtain information is usually the technical library. Most libraries have the capability to do research and provide references on any desired subject. The subjects given the library to research should include the name or designation of the test vehicle and appropriate words to describe unusual aerodynamic, propulsive, or operating characteristics. In all cases, it is advisable to consult with the aircraft manufacturer. By necessity the manufacturer must have extensive knowledge concerning all general aspects of the test item and may be the only source of information concerning its unique characteristics. This is invariably the situation for the newer VTOL and advanced helicopter designs.

An important portion of the preliminary planning is the establishment of a schedule and the fixing of milestone events. Some events to be considered as milestones are test plan publication, instrumentation specification, arrival of the test vehicle, first flight, program phase completion, completion of the total flight program, completion of data reduction, completion of report writing, and report publication. When the calendar dates cannot be established, the schedule should be arranged in terms of time from arrival of the test item. This will provide a general schedule and allow continuation of the planning effort. The program logically should be separated into various phases within the whole. The nature of these phases will depend upon the individual program, the schedule, and the test objectives. Most performance programs can be divided into phases similar to the subjects discussed in Chapters 4 through 14. Other tests such as environmental, reliability, and maintainability will be susceptible to a similar type of definition. These phases in turn can be separated into subdivisions. It may be advisable to define the phases according to configuration (clean and dirty), gross weight (light and heavy), atmospheric conditions (high and low altitude), temperature (hot and cold), and performance regimes (high and low speed). In other cases it may be judicious to separate the phases by test area such as sea level, desert, mountains, or tropics. Regardless of the phase definition method used, accomplishing this will aid greatly in organizing, implementing, and reporting on the program from inception to conclusion. One note of caution here is to be sure that the definition procedure is logical and that it is not so detailed as to become meaningless.

The next item to be established is the personnel and resources required to accomplish the test effort. Defining the personnel may include creating an organizational chart and assigning responsibilities. The complexity of this effort will depend upon the magnitude and scope of the program. Arrangements should be made to insure that support will be available at the proper time and manner. The person responsible for the preliminary planning should monitor the program and ascertain that the initial action is accomplished in a timely and proper manner.

1-3 TEST PLANS

The test plan may be written as soon as the preliminary planning has been adequately accomplished and sufficient information is available concerning the program and test vehicle. In some cases the test plan may be written prior to the scheduled submission date. When this is possible, it will provide more opportunity for correction, revision, and improvement. It is also desirable that the plan be written by the party responsible for the execution of the tests. When this is accomplished, there will be a minimum of communication and interpretation problems at a later date. It is also possible to make the plan more general in nature, thus providing more flexibility.

The test plan is the proposed plan of action and as such should contain information pertinent to the test objectives, schedule, locations, and necessary resources. The exact sequence of the tests is not a requirement; however, a general indication of the procedure to be used will enable the reader to understand more clearly the intent of the test plan. When possible, the plan should be written in a general rather than a detailed format. In the case of unusual tests it may be necessary to provide detailed information in order to allow a clear understanding of the work to be accomplished. Good writing practice is required, however, it must be remembered that the most important aspect is to disseminate information and to establish clearly in the reader's mind how the testing is to be accomplished and how this will achieve the test objectives. A clear understanding of the overall program must be transmitted as well as the necessary details.

The scope and nature of the program are the primary factors that influence the test plan. An outline should be prepared first to insure that all the necessary areas are being considered. From this general outline, it is possible then to proceed with confidence to a more detailed outline where the specific tests are noted. With this in hand, the next step is to develop the first draft of the detailed test plan. Each test must be clearly stated relative to method, objective, and anticipated results. A minimum of detail should be presented in each of these categories. When a large amount of detail is unavoidable, references should be given or an appendix should be added. A good technique is to use general statements and refer to other reports for details.

An important part of most test plans is the detailed schedule. Some milestones have been established during the preliminary planning phase. These must be modified to reflect all changes and include the most current information. The instrumentation phase must be coordinated to prevent any slippage caused by workload, schedule, or logistics. The allocated time for the instrumentation installation should be verified to insure completeness and adequacy. Any aircraft maintenance work prior to flight also should be included and coordinated with the appropriate groups. The flight portion of the program should be established. Each test can be examined to determine the flight time required. The total productive time then will be the total of all the individual tests specified. Some factor may be applied to determine the number of flights required to produce the required productive total time. This factor primarily will be a function of the type of aircraft and the testing being done. The average time per flight will be the most important single consideration. The next item to determine is the number of flights per day that can be expected. This, again, is dependent upon the aircraft and the type of tests. All these factors then are summed to give an estimate of the calendar time required to accomplish the flight portion of the program. The wise scheduler now will rely on knowledge and experience to include a margin for some of the inevitable delays that will be encountered during various phases of the program. To estimate best the advisable margin, a review of similar programs will provide a rule of thumb for the percentage of flights aborted, weather considerations, instrumentation requirements, and nonavailability of the test aircraft. Allowance also should be made for any necessary travel to additional test sites, "down time" for scheduled aircraft maintenance, configuration changes, and any instrumentation recalibrations. All support groups should be consulted to insure that no omissions exist in the contingency considerations.

Another significant item is the time required for the data reduction effort, data analysis, and report writing efforts. These estimates are the ones most often in error. The estimates are determined by consideration of the personnel available and the anticipated workload. The engineering effort required per flight varies greatly with aircraft and the type of data being obtained. A most pessimistic estimate is advisable. There are many items that dictate this, however, in the main they characteristically come from these sources. The necessary personnel and facilities planned and scheduled invariably do not materialize. Test progress usually uncovers areas that require additional efforts both in testing and data reduction, and analysis. It is surprising the amount of time these items can add to this portion of the program. Although the report writing effort is usually concurrent with the data reduction effort, for scheduling purposes they should be separated. Here, again, it usually takes longer to write the report than one would suppose. Some of the delaying obstacles encountered are interruptions, concurrent efforts on other programs, lack of clerical and editing support, and lack of author inspiration. The latter item is occasionally the most important consideration. It is also the most difficult to account for, and is invariably the least understood by all, including the author. Increased magnitude of the program introduces extensive cross referencing, comparisons with other results, and requirements for detailed discussion of the data. An accelerated program or a high productivity test will reduce the amount of time available for writing the report. A highly complex test vehicle or test procedure will require more technical research and original work. All these factors are difficult to plan and schedule, though they should be considered. The coordination cycle for the report must be analyzed carefully to determine the number of personnel and offices involved, and what has been the past performance relative to coordination. A reliable estimate of the publication time usually can be obtained from the graphic arts group. The total of all these items then will yield an estimate of the total program time.

The publication of the test plan changes the program from an abstract idea into a reality. The project engineering office now can assign personnel to the project for preliminary calculations, construction of data reduction forms, and preparation for the anticipated workload. Maintenance and flight personnel are able to enroll in the necessary training schools, and the appropriate logistic actions can be taken. From the specified tests, it is now possible to determine the instrumentation requirements and prepare the instrumentation specification. Each group can establish cost estimates and start providing equipment and personnel.

1-4 FLIGHT CARDS

The flight cards should be prepared as soon as the test plan is written. This procedure will prevent the occurrence of a high workload at an inopportune time during the program. They also will provide detailed information on how each test is to be performed and will allow time for improvements or corrections to be made at the earliest date. It must be remembered that making the cards in advance probably will result in certain changes in light of unexpected test results and additional requirements based on further knowledge. In any event, it is easier to modify an existing card than to originate one, and there will be a net reduction in the total effort.

Flight cards are necessary for the pilot and flight test engineer to conduct each test effectively. The card must contain the necessary information relative to the pilot effort as well as the data to be recorded. It is desirable to have the pilot concentrate on the operation of the aircraft with a minimum of other duties. To accomplish this, most effectively an observer should be used to record all necessary data, calculate changes in the test conditions, and to plot the data as they are obtained during the flight. When specifying conditions for the pilot, the instrument error should be accounted for to reduce the pilot effort and to insure the proper flight conditions.

In cases where the pilot is required to fly solo, he should be briefed thoroughly on the data requirements and any deviations that are anticipated. This briefing also enables the pilot more accurately to plan the flight so that the maximum productivity may be realized. Radio contact should be maintained at all times, and ground personnel should record the data transmitted by the pilot. The pilot also may have questions that can be answered by the ground personnel. This aspect is doubly important when conditions are such that the primary test mission cannot be accomplished and it is necessary to adopt an alternate test. The pilot may not have sufficient intimate knowledge of the data requirements to plan and execute adequately an alternate mission without ground support. The ground personnel should be prepared at all times for this situation. Also, additional tests should be planned prior to flight in the event the test progress is more rapid than expected. Giving the pilot more than he can accomplish is the general practice to insure that no portion of the flight time is wasted. However, this practice may be detrimental in that the pilot will try to accomplish the entire card regardless of the magnitude and in his haste may obtain questionable data. Perhaps a better technique is to give the pilot a realistic card and keep the other tests in reserve until they are required. This is somewhat more of a workload on the ground personnel but may increase the flight productivity.

Detailed flight cards for each type of test are discussed in the appropriate chapters.

1-5 DATA REDUCTION FORMS

The test plan and the flight cards specify what data will be obtained and how the tests will be conducted. The instrumentation specification will determine the form of the data when the data are available for reduction. With this information at hand, it is now possible to establish the data reduction methods and procedures. Accomplishing this task at this time will alleviate the backlog of data that normally occurs when the flying starts and the reduction procedure is not at full capability.

The data may be either automatic or manually processed. In either case the initial procedure is the same. The test to be conducted should first be placed in similar categories. The test plan is normally in this format which simplifies the task. Then for each category a data reduction procedure should be established and a data flow chart constructed. This will serve to clarify the process and to provide the most effective means for accomplishing the effort. Items to be included are data handling, data reading, data processing, data presentation, and data analysis. The data handling term is meant to cover the overall processing, transportation, and the data management. The data reading is intended to encompass the processes of converting the recorded data to usable engineering information. The most common forms of this are reading of oscillograph traces, reading of photo panel film, and obtaining a computer print out. The data processing refers to the calculations or corrections that must be applied to put the data in a form suitable for presentation. This may involve either manual or computer techniques. In either case, the engineer must develop a detailed step by step procedure in engineering terminology of all the computations that are required. This form serves two purposes. First, it provides the information for the programmer to develop into computer language, and, second, it provides a tool for checking the final computer program. Failure to accomplish this program checking can lead to disastrous results, since the conversion is not always accomplished literally. The data presentation required is determined by the test objectives and the anticipated test results. The data may be in either tabulated or plot form. In either case, the format can be established and this will save a great deal of hurried effort later in the program. Generally, a review of contemporary reports will provide a guide as to the appropriate plots necessary to present the data.

Detailed data reduction forms and graphical presentations of the data are presented in the chapters that follow.

CHAPTER 2

INSTRUMENTATION

2-1 GENERAL

Instrumentation as discussed here will pertain to that required for performance, stability, and control testing. However, a similar approach will provide adequate results for other test programs such as structural, developmental, and research. Each test may have particular requirements, however, the objectives are similar; i.e., to provide quantitative data to aid in future design, to correct existing deficiencies, or to increase the overall knowledge of a specific area. The test data are used primarily to provide information relative to operating and performance characteristics as well as to correct deficiencies that resulted from insufficient or inadequate design criteria and assumption. The importance of the instrumentation cannot be overemphasized, since the entire effort will be for naught if the data are not accurate, valid, and complete.

2-2 PLANNING

The initial instrumentation requirements should have been considered in the preliminary planning phase. The detailed planning should have been accomplished at the earliest possible date to insure an adequate and timely progression of the instrumentation effort. An unforeseen delay may cost valuable flight and calendar time. However, in some cases it may be more expedient to plan a progressive installation based on the test requirements and sequence. So long as adequate lead time and planning are provided for the instrumentation personnel to build, install, and calibrate the additional equipment a minimum or no delay will be incurred by this approach. An unscheduled instrumentation effort is to be guarded against since in most instances an installation and calibration time is required and, unless properly planned for, will result in an excessive delay in the test program. Generally speaking, even the most elaborate and complete instrumentation arrangement will prove to be inadequate in some area and revisions or additions may be necessary. This primarily is caused by new requirements as the test program progresses and the data are analyzed. In view of this, the necessity for careful analysis and planning of the instrumentation requirements and installation is readily apparent.

2-3 REQUIREMENT GUIDELINES

The instrumentation requirements can be determined from the test objectives and the test plan. From these general requirements it is necessary to prepare a detailed instrumentation specification. This specification must contain a complete listing of all equipment required, work to be accomplished, time and manpower available, as well as sequence and general cost analysis. Sufficient detail must be given so that equipment may be ordered, wiring diagrams be prepared, and drawing be made for fabricated parts and instrument packages to be installed. The instrument specifications should be prepared by the instrumentation engineers. The flight test engineer should coordinate to insure adequate coverage as well as compatibility with the existing facilities and the general suitability of the installation. Also, it is necessary for the flight test personnel to be intimately familiar with the instrumentation so that malfunctions during the testing may be detected quickly and corrected.

Since the instrumentation facilities must operate within the overall guidelines established by the flight test engineer, he should specifically consider the following items:

1. In many cases, more than one method of recording the data may be used.

2. In general, the most important parameters should be recorded by the most reliable method with items of lesser importance being recorded as appropriate.

3. For critical items it may be desirable to have duplicate recording. This duplication also may be useful in data reduction and correlation procedures. This method will reduce the loss of data from individual instrumentation malfunctions and the number of repeat or abort flights.

2-4 RECORDING OF DATA

The data should be recorded in a manner that will allow the reduction to be accomplished most easily. Many variables can be recorded conveniently in only one manner and there is little choice. The degree of accuracy which can be obtained will vary with different recording methods and consideration should be given in this area. The most satisfactory data recording method may be determined by consideration of the following.

1. Number of parameters to be recorded

2. Type of data reduction to be employed

3. Time allowance for reduced data to be presented

4. Cost and time available for instrumentation installation

5. Reliability required

6. Type of data presentation to be made

7. Weight and space available.

The number of parameters to be recorded should determine immediately the requirement for automation. Whenever more than twenty parameters are recorded, an automatic system is desirable. Some of the automatic data recording systems to be considered are magnetic tape, telemetry, photo panel, and oscillographs. These systems are invaluable for particular types of tests and in some cases must be used regardless of individual desires. Tape and telemetry easily can record a large number of variables. Oscillographs and photo panels are limited individually in scope and several may be required. For a small number of variables it is possible for either the pilot or an observer manually to record visual data. This method is most effective for a performance test and, when possible, is preferred over any other method. In many cases the manual system is supported by an oscillograph or photo panel which is a highly desirable arrangement whenever feasible.

For some types of tests, such as structural demonstrations and critical stability and control test, it is necessary for real time during the progress of the tests. The only method that can accomplish this is telemetry or visual indicators in the aircraft. Since stress and dynamic stability data cannot be presented easily with indicators, this leaves telemetry as the only realistic method. Most telemetry layouts have the capability of visually portraying the data while simultaneously recording a permanent record. This permanent recording device may be in the aircraft or located in the ground station. All methods other than telemetry require that the aircraft return prior to accomplishing any data reduction effort. Photo panels and oscillographs require a time interval for the records to be developed and read prior to starting the data reduction. Hand recorded data are immediately available for processing after the flight and can be analyzed between flights. This may be an important consideration when consecutive flights are dependent upon each other. Visual hand recorded data are also very useful for in-flight plots which will insure completeness and that erratic points are repeated at the appropriate time.

The type of data presentation to be made will somewhat restrict some of the recording methods. The recording systems all may have a similar capability but the amount of data reduction effort may vary greatly. When high density time histories are desired, the photo panel and hand recorded data are inadequate and difficult to process. The values must be plotted and then curves faired through the points. Telemetry, magnetic tape, and oscillographs can produce this type of data directly.

The weight and space allowance is usually a function of the aircraft dimension and performance. The number of parameters is the single determining factor, and the instrumentation system must be arranged accordingly. The number of parameters per unit weight or volume is greatest for the magnetic tape system of which there are many variations. For small, limited space vehicles, where a large number of variables are of such interest, a magnetic tape system is invaluable. When space or weight is not at a premium, then it is desirable to use several oscillographs or photo panels. The biggest problem with this type of arrangement is the time to correlate the data and accomplish the data reduction. The time correlation must be accomplished very accurately or the data may be invalid or unusable. To best achieve this, each of the recording devices should be identified or coded simultaneously in order to fix the events in time and sequence.

2-5 COST CONSIDERATIONS

The funds available for the instrumentation and the emphasis placed on the expenditures may affect greatly the final instrumentation arrangement. Good instrumentation is seldom inexpensive, however, quality should never be sacrificed for quantity. It is far better to obtain a small amount of accurate data than to obtain a large quantity of inaccurate or questionable data. In fact, inaccurate data may be worse than no data at all. Inaccurate data may damage the reputation of the flight test department and the organization as a whole and, in some cases, result in crashed aircraft and loss of life. Also, great financial losses may result from erroneous data being incorporated into future designs that may result in an undesirable or unusable product. For the Government agency, there is a dual responsibility, an accurate and fair evaluation of the contractors' products as well as an obligation to the country and the men who may be using the vehicle for national defense. In the event financial resources are not available for an adequate instrumentation installation, the appropriate group should be notified immediately so that action may be taken. All personnel concerned should be aware of the situation so that, if necessary, the program objectives can be altered or more resources can be made available to accomplish the existing objectives.

The most costly data system is a magnetic tape recording system in the aircraft combined with telemetry of the data to a ground station. Photo panels and oscillographs are relatively inexpensive and, of course, visually recording instrumentation is the least expensive of the systems previously discussed. Unless cost is of prime importance, the choice of data recording methods should be determined by accuracy requirements and overall suitability. The time for the installation must be scheduled commensurate with the type of installation required. The more complex systems generally require considerably more time than the simpler systems.

The reliability aspect is associated most closely with the anticipated productivity required. For aircraft with very limited time per flight or flights per day, it is essential that no data be lost. This also may be the case for aircraft which are of a very complex or expensive nature. Reliability is also very important during high risk tests where it is dangerous and perhaps impossible to repeat the test. Reliability of the various systems is a highly controversial subject; however, the consensus is that complexity contributes to low reliability. Flight test experience with the various instrumentation arrangements bears out this consensus.

2-6 SENSOR DEFINITION AND CALIBRATION

The most important part of the instrumentation specification is usually the one most neglected by the planning and technical groups. This is the sensor definition, the range and accuracy capability, and the calibration procedure. Many times the instrumentation groups encounter what to them are unrealistic or impossible requirements. This can be caused by very stringent test requirements or simply from the requestor not being familiar with the instrumentation capabilities. In some cases the instrumentation personnel will assume the latter and freely substitute a lesser capability. This, of course, should not be done without first determining that this change will be satisfactory. Conversely, the requestor should never impose a requirement without follow up action to be sure the desired effort will be accomplished. The proper sensor must be chosen in order for the rest of the recording system to obtain the necessary input data. The sensor also must be compatible with the remainder of the system. The calibration schedules must be specific with respect to procedure, range, sensitivity, and increments. This is necessary to insure that the calibration is accurate and complete. The validity and accuracy of the calibration standards should be compatible with the calibration requirements. Environmental and operating conditions should be considered during the calibration. When the laboratory calibrations differ greatly from the test conditions, significant variations may result. The most potentially accurate system available will not yield correct data if the calibration is not accomplished properly. Too often the lack of communication between the flight test and instrumentation personnel causes the calibration to be done in an undesirable manner. The flight test engineer should be thoroughly familiar with the calibration procedure and the requirements, and should participate in the calibration to insure a satisfactory result.

CHAPTER 3

GROUND EQUIPMENT

3-1 GENERAL

The helicopter is associated more closely with ground operation than is the conventional fixed-wing aircraft. This includes scope as well as percentage of time spent in this environment. As a consequence, the number of tests necessary in this area is greater, and it is more important to define the aircraft condition relative to the ground and existing ground environment. This task is complicated by the rotor influence on both the atmosphere and ground conditions. These influences become greater during the more important in-ground effect operations.

Some of the tests that are conducted in ground proximity are hover, takeoff, translation, air taxi, powered landing, and autorotational landing. These tests are discussed in detail in Chapters 5 through 14.

3-2 HORIZONTAL THRUST STAND

The horizontal thrust stand is necessary to determine the static installed longitudinal thrust of the test aircraft. This device has limited use for most helicopter tests. Most modern thrust stands are equipped with self-centering or null-balancing devices so that the load cells can be zeroed prior to beginning a test. This zero should be checked to be sure there is no excessive drift with either running time or thrust load. All tare values prior to and after the test should be recorded. The geometry of the stand should be noted with emphasis placed on the aircraft/stand compatibility. This can be significant when testing aircraft with other than conventional propulsion systems, heat or hot gas impingement being the most frequently encountered offender. The instrumentation should be inspected carefully. The calibration of the stand should be current and if any doubt exists, the calibrations should be repeated. The recording range should be noted and incorporated into the test procedures to prevent exceeding the limits of the equipment. The aircraft mounting requirements should be noted, and any special equipment should be constructed. The installation should provide a blockhouse or other suitable protection from blast and noise for the recording equipment and personnel. Any available communication and atmospheric equipment should be utilized to the maximum. When not available, arrangements should be made to provide the necessary items.

3-3 VERTICAL THRUST STAND

The vertical thrust stand is usually far more complex than the horizontal thrust stand. The geometry of the stand is of vital importance. The various forces and moments usually will have to be resolved to obtain the necessary data. The distances or moment arms for the sensors are required to accomplish this. The sensor positions and dimensions relative to the aircraft also must be known in order to correlate the thrust data with the aircraft free flight performance. A careful study should be made of the instrumentation to insure that all the parameters are clearly defined and that there are no unknown force interactions.

Normally, there is a multitude of data being recorded at each test point from the environmental stations, the thrust stand, and from the aircraft. The most difficult task is to correlate accurately all the recordings. The surest way to accomplish this is to record the same identification mark on all the records at precisely the same time. This can be accomplished by manual marking on a signal over a common communication network or by having the data systems mark automatically at the activation of a switch. The latter method is preferable but when properly done, either system should yield satisfactory results. The signal to record and identify data should come from the aircraft rather than the ground since that is where the specified test conditions are being controlled. Since most thrust stands have height as well as attitude variables, great care must be taken to establish and record the precise stand configuration for each test

The ground equipment on most vertical thrust stands contains some provision for measuring the external environment—the most common measurements being pressure and temperature. Velocity and direction of airflow also may be included. There are many different arrangements that may be utilized and no attempt is made here to discuss them all in detail. One thing common to all thrust stand tests is the communication problem. Normally, the environment personnel are in the open with no access to a radio. For their data to be meaningful, they must be able to correlate these data with the aircraft operating conditions. One good way to accomplish this correlation is through a ground intercommunication network. In fact, if the facilities permit, elimination of all radio equipment will simplify the entire operation. This is particularly true for aircraft that have poor radio operation near the ground.

The preparation relative to safety, operation, instrumentation, calibration, and stand suitability has been discussed previously for the horizontal thrust stand. The mounting problem is far more difficult in general, and in particular when attitude changes introduce side forces on the gear or other restraining points.

3-4 ATMOSPHERIC EQUIPMENT

For most ground tests it is necessary to define the condition of the external environment. During the thrust stand, hovering, takeoff, and landing tests, the relative wind velocity and direction are required to correct the data for wind effects on thrust, distance, and airspeed. For these conditions, the aircraft recorded atmospheric data are usually unreliable because of rotor downwash and re-ingestion effects. The aircraft performance is usually sensitive to very low wind conditions and an appropriate anemometer is necessary. The wind condition and direction relative to the aircraft also should be determined. The best method is to mount the anemometer so that it will be possible to note direction and velocity simultaneously. The wind vane seeks the direction of the wind and insures that the wind velocity is measured at the same time in the same direction. The assembly should be portable so that it can be positioned in the proper location at the test site. There are several commercial models available, however, a suitable instrument can be constructed easily.

The ambient pressure and temperature also should be recorded by a source independent of the aircraft. Most tests are conducted at an aircraft installation, and the tower usually has atmospheric data available. This type of data may be unsatisfactory for several reasons. The accuracy is normally less than that required by test standards, and the recording location is not in proper relation to the test site. A better method is to procure an installation that is portable and has the desired accuracy. Commercial equipment is available, or a suitable installation can be constructed. Both the thermometer and the altimeter should be sensitive calibrated test instruments. Caution should be used so that the equipment is located in a manner that will yield true atmospheric data. Free air should be flowing through the devices and care should be used to prevent direct exposure to the sun.

The previously discussed wind measuring techniques record the conditions relatively near the surface and may not indicate any shear effects with altitude. This knowledge usually is required during landing tests. An indication of the wind characteristics may be obtained by observing the behavior of ascending smoke. This method will not yield quantitative data and usually is not effective in any case for altitudes above 50 ft. An elevated anemometer device will provide the necessary data, however, this type of device usually is not available when operating at remote sites.

Another method for estimating the wind at altitudes consists of a neon filled balloon tied to the ground. The deviation of the string from the vertical is an indication of the wind velocity at the balloon elevation. By setting the balloon at the same height as the aircraft, the wind at the test altitude can be evaluated.

3-5 THEODOLITES

During takeoff and landing tests, visual theodolites should be used to determine both horizontal distance and height above the ground. These theodolites may be in both vertical and horizontal forms, and can be constructed easily from metal and Plexiglas. Using the relationships of Fig. 3-1, one may construct a theodolite for any desired height or offset distance:

$$\frac{x}{w} = \frac{d}{\ell} ; \frac{x}{y} = \frac{d}{h} \tag{3-1}$$

(by similar triangles)

where

d = offset distance from the runway

ℓ = runway horizontal flight path

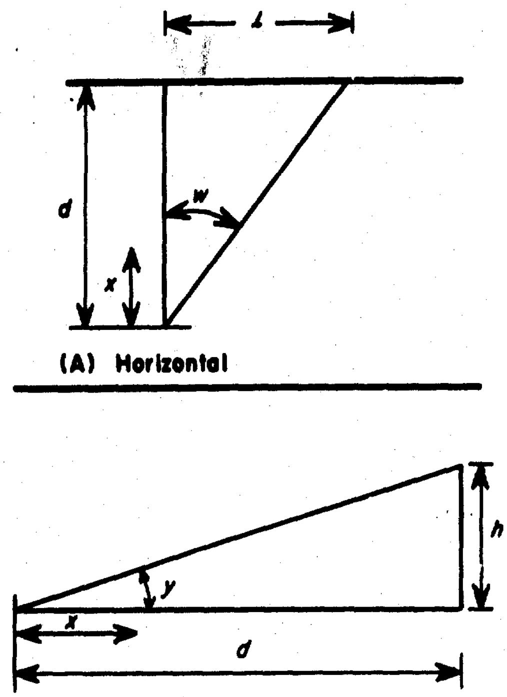
h = height above the runway

x = distance from sight to the theodolite scale

y = vertical length of scale

w = horizontal length of the scale

The scale may be constructed for any desired distance, and employment of multiplication factors can make it useful for more than one offset distance. The accuracy of the instrument decreases as the offset distance becomes greater. The theodolite should be aligned accurately with the runway. The vertical theodolite is of particular use during autorotation landing tests where height above the ground must be known precisely in real time.



(A) Horizontal

(B) Vertical

Figure 3-1. Theodolites

3-6 CAMERAS

Variable speed cameras presently are used to record visually presented data in the cockpit and on the photo panel. The camera speed is adjusted according to the density desired. A desirable installation is one which has more than one speed available to the recording personnel in the cockpit. This will allow selection of the speed most suitable for the type of data being recorded. Film also can be conserved when a high speed is not required. This type of data recording system, requiring a great deal of data reduction effort to read each frame, is generally a poor method of recording other than static conditions. Other uses of photographic recordings are primarily to provide qualitative information with respect to tuft studies, flutter and other component motions, ordnance and debris patterns, and recording pilot visual phenomena. These cameras are located on the aircraft in an appropriate position or sometimes carried in a chase aircraft. In some instances, the film is used to provide quantitative data by scaling the dimensions from the picture. This is extremely difficult, since the mounting must be precisely right to show the true dimensions. When this is not done, reducing the data may become a descriptive geometry problem of some magnitude. When more than one camera is being used, the time correlation becomes a factor. A frequently used technique to accomplish this is to have an interrupter that will expose a frame at the same time on each camera. Placing a clock in the field of view of each camera is another technique that sometimes is used.

Ground cameras are used to record the behavior while in ground effect or during takeoff and landing tests. During hover, a camera is a useful tool for establishing the aircraft height above the ground. This can be accomplished using a known dimension to determine a scale for the photograph. This dimension may be a part of the aircraft or may have been added prior to the test. The reference dimension should appear in true size in the photograph. This requires that the photographer be in the proper position when the exposure is made. The best way to insure this is to designate the spot over which the aircraft is to hover, and then at some desirable distance inscribe a circle. The photographer then should be instructed to stay on the circle and at right angles to the reference mark or dimension. This will insure that the reference is viewed in full dimensions and also that the distance from the photographer to the object is a known value.

During the takeoff and landing tests, a special camera known as a Fairchild Flight Analyzer is used to record the aircraft motion. This camera has the capability of taking a series of pictures while simultaneously recording relative lapsed time. The camera carefully is located a known distance from the runway and is in a level position. Included in each series of pictures are two targets that are at a known distance and geometric pattern. Normally, they are parallel to the runway and 200 ft in front of the camera location. One target usually is directly in front of the camera and the other one is offset laterally 100 ft from the first. A typical installation is shown in Fig. 3-2.

The flight path distance between the targets can be calculated by similar triangles as previously discussed under visual theodolites. The offset distance and target distances may be changed as desired. Increasing the offset distance will provide a greater flight path distance coverage but will reduce the size of the aircraft image and a loss in reading accuracy usually will result. The relative distances between the different pieces of equipment must be measured accurately and recorded for each installation. A surveyor's equipment should be used to assure that the angles and lines are correct. The timing device in the camera is a critical item and should be checked periodically with a stop watch. The accuracy of the plates will be influenced by the tracking precision which is a function of the operator proficiency.

3-7 HOVERING EQUIPMENT

For obtaining hovering data, a free flight or tethered technique may be used. Atmospheric measuring equipment at the ground station is needed for both methods. For the free flight technique, two conditions should be considered which require special equipment. The in-ground effect hovering requires some means for stabilizing the helicopter and determining the height above the ground. The equipment most often used is weighted cables attached to the helicopter. The cables are of the proper length to yield the desired height above the ground. For out-of-ground effect hovering, a line is lowered from the helicopter. This line should be approximately 200 ft long and mounted on a hand reel. A 10-15 lb weight is attached to provide stability and tension. Tufts are attached at intervals to indicate airflow relative to the line. The equipment for tethered hovering consists of a load cell, arresting cables, and a position sensing device. The load cells commonly in use are hydraulic or strain gage types. A suitable readout instrument is assumed to be a

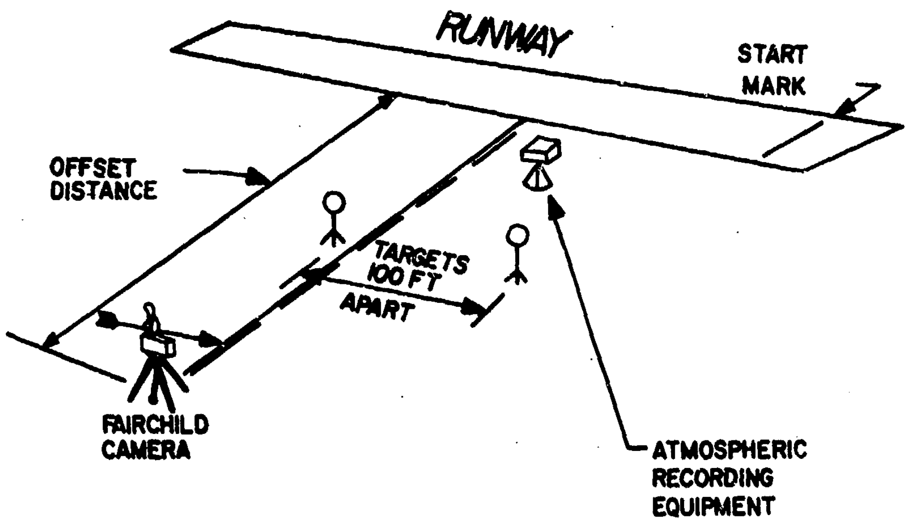


Figure 3-2.
Fairchild Camera Layout

part of the load cell device. The most suitable instrument consists of a direct readout in pounds in addition to a permanent recording. Direct readout of the data will allow a preliminary plot to be constructed during the test and provide an evaluation of the results in real time. The pilot will indicate questionable points, and these will be reflected in the recorded data. The permanent record will provide data that will show the thrust variation with power for nonstabilized conditions. This record also will provide information concerning the validity of the instantaneous test data. These are values usually determined by averaging or choosing a most representative value. This may be changed after reviewing the permanent data recording. The data readout may be located in the aircraft or be a part of the ground station. There are merits to both methods and the most convenient should be used. The load cell should be attached to the ground tether point or to the aircraft so that when a quick release is made the load cell will not be dropped to the ground and be damaged.

The arresting cables should be in sections equivalent to the desired heights. This will facilitate the installation and insure that the data are obtained at the desired height. Care must be used to measure the lengths of the connecting clevises, the load cell, and the aircraft hoist as well as the cables. In a similar manner, the weight of each item must be known. The cables must be of sufficient strength to withstand the anticipated steady-state load in addition to some dynamic loading. For convenience and ease of handling the cable diameter should be no larger than necessary. An integral part of the cable system is a method for a quick release in the event the pilot has difficulties and desires a free flight condition. For aircraft equipped with a cargo hook this requirement poses no problem. In other aircraft a similar device should be installed with the actuator switch located in a premium cockpit position.

It is very difficult for the pilot to maintain a vertical position over the tethering location without guidance in some manner. This may be accomplished by visual references to the front and to the side of the aircraft or by radio directions given to the pilot. Visual references require the pilot continually to observe out of the cockpit, and verbal directions may distract him somewhat. The best way to accomplish this directing is to provide a cockpit instrument that will inform the pilot of his position relative to the tethering point. An instrument to accomplish this can be devised by presenting the output from accelerometers located on the load cell element. These accelerometers should be attached to the load cell in such a manner as to give an indication of when the load cell is away from the vertical either longitudinally or laterally. The display need not include the exact magnitude of the deviation since data will be recorded only during a vertical condition. This arrangement will allow the pilot to focus his attention within the cockpit and still not be distracted by excessive radio communication.

3-8 COMMUNICATION EQUIPMENT

During hovering, takeoff and landing, and translational tests, it will be necessary to have communications between the aircraft and the ground stations. The magnitude of the communication requirement increases considerably when there are individual ground stations or teams measuring height, thrust, atmospheric conditions, downwash factors, noise, and temperature conditions.

CHAPTER 4

GROUND TESTS

4-1 GENERAL

The ground tests are the final aircraft preparation that is necessary prior to conducting any flight tests. These tests should encompass the total aircraft and/or individual subsystems. The tests most commonly conducted include weight and balance, fuel tank calibration, control rigging checks, and instrumentation system calibrations.

Since the performance will be influenced by gross weight and CG location, the weight and balance must be known precisely for any test. The longitudinal CG is normally more critical than the lateral or vertical CG location. The CG location for each test point is obtained from the empty weight, the fuel aboard at engine start, and the fuel used. The aircraft control rigging must be known to insure that control limitations will not restrict inadvertently the performance in any way. The instrumentation system may introduce significant changes that will not be included in the individual instrument laboratory calibrations. The system calibrations usually accomplished are for the pressure, temperature, and torque measuring systems.

Detailed instructions and procedures for instrument calibrations should be obtained from the appropriate literature or personnel.

4-2 WEIGHT AND BALANCE

The basic or empty weight is important from a performance viewpoint and may have contractual implications when it is a guaranteed value. This requirement can be determined from the model specification. A thorough research effort must be made to establish the exact configuration of the aircraft relative to this guarantee. Some items to be considered are internal fuel and oil, trapped fuel and oil, internal equipment, miscellaneous or mission essential equipment, and external stores. These items should be as required in the model specification prior to any weighing. The initial weighing should be accomplished immediately after receipt of the aircraft and prior to any instrument installation. The documentation must be complete relative to the previously discussed items and conditions. The actual weighing procedure will vary with the aircraft and the weighing facilities. The scale calibration should be current and the scale should have the required accuracy. After the instrumentation installation, the weight and balance should be repeated to determine the resultant changes.

The most common weighing method for small aircraft is by use of platform scales. For larger aircraft, electronic load cells or fixed floor scales are used. The scale chosen should be compatible with the gross weight of the aircraft. This is particularly important for small aircraft since most large scales are inaccurate and unreliable at the extreme of their ranges. For example, a scale designed for aircraft with weights to 500,000 lb should not be expected to possess sufficient sensitivity to weigh accurately an aircraft of 1000 lb. The scale calibration data and design criteria may not indicate this; however, considerable experience has demonstrated that such is the case.

When the aircraft is in the proper configuration, it is placed on the scales and leveled in the specified manner. Usually this is a level attitude. When weighing out of doors there should be no significant winds to impose side loads or lift on the aircraft. A closed hangar should be used when feasible. Rotor blade or control surface position will not influence the total weight but may change the center of gravity data. Unless specified, these should be in a symmetrical position. The weighing should be repeated a sufficient number of times to rotate the load cells or scales to each position. For fixed scales, three weighings will suffice. All moment arms to the scales should be measured and recorded. Dimensions from drawings, decals, or specifications should be verified by actual measurement prior to being used in any calculations. The tare values and serial numbers of the load devices should be recorded for each weighing. A sketch of the equipment arrangement and the aircraft positioning is also an advisable item. This will assist later in the center of gravity calculation.

When a fuel tank calibration has been conducted previously, the aircraft may be weighed with fuel aboard. A fuel specific weight and fuel temperature reading is necessary to correct the data to an empty weight condition. This weighing procedure should not be used when the empty weight is a guarantee value.

The weight information is entered on the Weight and Balance Clearance Form (DD Form 365F).

4-3 CENTER OF GRAVITY DETERMINATION

The longitudinal center of gravity location can be calculated from the weight determination data. When using these data, the internal and external configuration should be checked to determine the exact location of any loose or optional equipment. Rotor blade and control surface positions may change significantly the CG location and should be placed in a symmetrical position unless otherwise specified in the weighing procedures. All moment arms between the scale reaction points should be accurately measured and recorded. The moment arm of the fuel usually will change with the fuel quantity and should be determined from actual test weighings rather than from calculated values. This is accomplished by adding a known fuel increment, determining the CG change, then calculating the fuel moment arm. A sufficient number of points should be obtained to establish a curve showing the fuel moment arm as a function of fuel quantity. The fuel becomes a smaller percentage of the total weight as the gross weight increases, and the CG change with fuel quantity will be smaller. This test should be conducted at minimum and maximum gross weights. The number of fuel increments necessary at each conditon will depend largely upon the shape and size of the fuel tank. Generally, four values are sufficient. For tests where the CG location may be highly critical, it is prudent to determine the CG with the actual test loading rather than to rely on calculated values. This may include conducting a weighing with the flight personnel at their crew stations. The longitudinal and lateral CG data are calculated on the Weight and Balance Clearance Form (DD Form 365F).

The lateral CG location also can be determined from the total weight and longitudinal CG determination data. However, these data do not provide information relative to the vertical CG location. Vertical CG data are obtained by tilting the aircraft laterally while recording the angle of tilt θ and the load reactions W as shown in Fig. 4-1 (A).

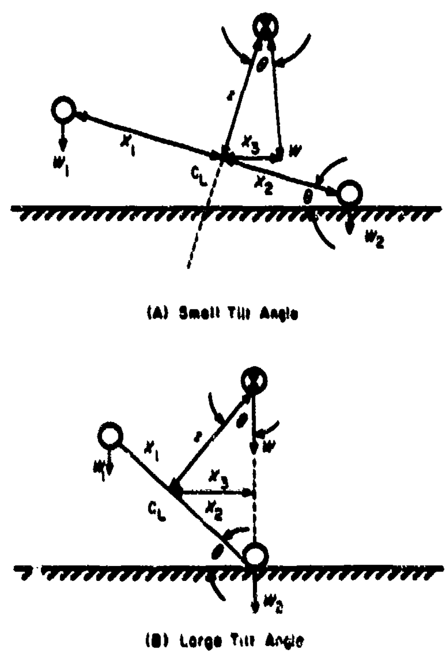


Figure 4-1.
Vertical Center of Gravity Determination

For the level condition as determined previously,

$$W_1 X_1 = W_2 X_2 \tag{4-1}$$

and summing moments M about the centerline yields

$$\Sigma M = W_1 X_1 \cos\theta - W X_3 - W_2 X_2 \cos\theta = 0 \tag{4-2}$$

$$\sin\theta = \frac{X_3}{Z}, \quad Z = \frac{X_3}{\sin\theta} \tag{4-3}$$

where X_3 represents the lateral movement of the CG. The W_2 value increases as the tilt angle increases and at some angle becomes equal to W as illustrated in Fig. 4-1(B).

When these tests are being conducted, the tilt angle introduces a side load on the lower gear which increases with the angle. Care must be exercised to insure that the loads do not exceed the limits. The gear must not be allowed to slide. Also, the aircraft should be restrained to prevent a tip over in the event the critical angle is exceeded. The vertical CG will change with internal loadings and with fuel level. When these variations may be large, it is desirable to conduct additional tests to determine the magnitude of the vertical CG movement.

4-4 FUEL TANK CALIBRATION

The standard fuel system normally indicates the fuel quantity aboard in pounds. This may be of questionable accuracy depending upon the particular system. There is usually no compensation for variation in specific fuel weight, fuel temperature, or aircraft attitude. The increments on the gage are designed for normal flight operations and may not be of sufficient resolution for flight test purposes.

The two most common test methods used for determining the fuel aboard are calibrated dipsticks and sight gages. These provide information as to the fuel quantity prior to engine start from which is subtracted the fuel used at any given test condition to yield fuel aboard at that time. The dipstick is shaped to fit into the tank and is inscribed with appropriate fuel quantity marks. This stick is usually a metal rod of suitable size and strength. Unusual fillers and tank shapes render this method awkward. In addition, it is also difficult to add a specific quantity of fuel when the stick must be removed periodically to check the level during the adding of fuel. A fixture should be mounted to the aircraft so as to guide the dipstick to the same position each time a reading is taken. Consistent, accurate readings will be obtained only when the dipstick is in the same position for which the calibration was taken. Visual sight gages are designed to show the actual level of the fuel in the tank. The tank is tapped at the sump or fuel drain with a clear flexible tubing. This tubing then is fastened to a rigid support that is equipped with suitable quantity indicators to encompass the vertical range of fuel levels. This support then is attached to the aircraft. The visual indicator is much easier to use, is more accurate, and should be given preference over a dipstick when possible.

The calibration is accomplished by incrementally adding known quantities of fuel and recording the indicated values on the sight gage or dipstick. The standard aircraft fuel system should be calibrated as the test system is calibrated. The magnitude of the fuel increments will be determined by the size and shape of the fuel tank. Most calibrations will be nonlinear at the capacity extremes where sump and filler irregularities may be encountered. In these areas increments as small as one gallon may be necessary. When the calibration data indicate the fuel added is linear with the fuel indicated, the increments should be increased to 5- or 10-gallon quantities. The fuel added should be taken from calibrated containers or from a calibrated flow device. After the fuel increment has been added, the indicated reading should not be taken until the level has had sufficient time to reach a stable condition. The time required for this to occur will vary with the tank complexity and shape. Particular care should be used when multiple or series tanks are being calibrated.

During the flight program, the aircraft attitude on the ground will not be the same for all preflights. Thus, to determine an accurate fuel level, the aircraft must be leveled to the fuel tank calibration conditions prior to taking a sight gage reading. This is tedious and time consuming since the aircraft must be jacked and a plumb bob installed to indicate the position. This procedure can be eliminated by calibrating the fuel tank as a function of pitch attitude. The aircraft is fueled to approximately one-fourth full in a level attitude, and the fuel indication is recorded. The aircraft then is pitched down incrementally and for each increment the pitch angle and fuel indication are recorded. The procedure then is repeated for nose-up conditions. For nonsymmetrical tanks the variation will not be the same for all fuel levels, and a calibration should be conducted for a three-fourths full tank condition. With these data a plot is constructed as illustrated in Fig. 4-2.

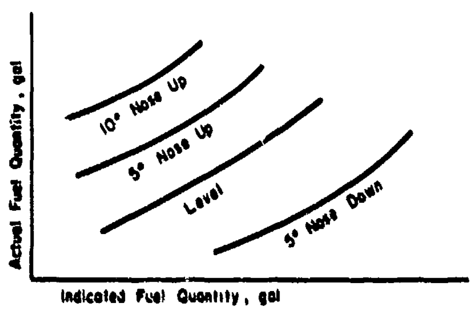


Figure 4-2.
Fuel Tank Calibration With Pitch Attitude

From this plot it is not necessary to level the aircraft since the fuel quantity can be obtained from the sight gage indicator and the aircraft attitude. It is best to use an inclinometer to measure the attitude. The measurement should be taken at the same position used during the calibration.

4-5 PRESSURE SYSTEM TESTS

The pressure systems normally installed are the inlet pressure, airspeed, and altitude systems. The inlet pressure and airspeed systems must be leak checked throughout. The systems should be tested by applying a known dynamic pressure and then observing the indicated reading. The indication should not change with time when the pressure is applied. The difference between the indicated and true values is the system error. The pressure systems also should be tested for lag. This lag results from pressure drops in the tubing from friction, orifices, and restrictions as well as the inertia of the air and the time required for the pressure change to travel the length of the tube.

The lag of the system can be determined by applying a pressure, releasing the pressure, and then measuring the time required to return to a predetermined percentage of the initial value. The airspeed systems should be balanced by measuring the lags of the total and static systems, then adjusting the volumes to provide equal lags. This will reduce the error of the system during maneuvers, climbs, and accelerations.

4-6 TEMPERATURE CALIBRATIONS

Most temperature systems are electrical systems. The temperature indicator calibrations do not include any system losses. The wire, connections, and soldering may introduce resistances. This will result in a different voltage reaching the indicator for a given output from the sensor. A system calibration will show the effect of the total installation. This is accomplished by applying a known temperature to the sensor and recording the indicated value. The range calibrated should be greater than that anticipated during the program. This range normally will be from a hot day hover of 40°C to cold high altitude temperatures of −25°C.

The temperature source should be applied to the sensor. In order to obtain the total system error, the temperature source must be an accurate stabilized input. Liquid baths with ice or carbon dioxide are not stabilized easily, and a great deal of effort must be spent to be sure the liquid is well mixed. This is the least accurate calibration standard. A more common method is to apply an electric voltage calibration to the system as near to the sensor as possible.

4-7 CONTROL SYSTEM RIGGING AND CALIBRATION

The control system rigging is checked with the maintenance procedures and tolerances prior to testing. Appropriate changes are made to achieve the proper values. In addition to the normal rigging procedures, any linked or programmed control surfaces such as elevators also should be calibrated. These devices usually are connected to the longitudinal cyclic or collective sticks. Some devices may be operated by dynamic pressure, in which case, a pressure is applied and the motion is recorded. The performance analysis also requires information concerning the blade motions as a function of control positions. The depth of the analysis will establish the detail required in the calibration. The minimum is one blade azimuth and one location. An inclinometer mounted to the blade is used to give the angle of deflection. The control motion should be in the same units as those to be recorded during the tests. Blade tracking is included in the control rigging tests.

The engine rigging also should be checked to insure proper acceleration schedules and biasing for any pressure or termperature sensors.

CHAPTER 5

HOVERING PERFORMANCE

5-1 GENERAL

The hovering flight regime was for many years the exclusive domain of the helicopter, but now is shared by other forms of VTOL vehicles. During the steady-state hovering condition the rotor is in the simplest aerodynamic mode that will be encountered in the flight envelope. This condition affords the best opportunity for evaluating and comparing the rotor with appropriate contemporaries. The cyclic control inputs by the pilot are limited to those required to balance the pitch and roll moments of the aircraft, while the collective pitch requirement is that necessary to produce a net thrust equal to the gross weight. The directional control input is that required to oppose the torque generated by the rotor. The data usually are recorded during low wind conditions which contribute a minimum of random moment upsets. These upsets, of course, must be countered by pilot control inputs and generally will influence the resultant hovering performance values and repeatability to some degree.

The purpose of the hovering performance tests is to measure the total power required to produce a given amount of hovering capability and to evaluate the rotor performance. The end result will be to provide sufficient information to predict the hovering performance over a wide range of operating conditions and mission profiles. The nature of the testing and the data recorded also should be such that the design engineers can utilize the results to validate the design assumptions, compromises, and the basic equations.

The scope of the tests should be sufficiently comprehensive to minimize the extrapolation necessary to define any operating condition within the hover envelope. This requires that the aircraft configuration and operational parameters be varied over the widest possible range. Since the helicopter is highly sensitive to ground proximity, hovering tests within the ground effect region are necessary at various ground elevations. High field elevations usually involve remote test sites with the attending support and logistical problems.

The testing is normally accomplished in relatively calm wind conditions (less than 2 kt) to reduce disturbances since an acceptable wind correction procedure is not currently available. Several factors complicate tests conducted during windy conditions. First, most rotors are very sensitive to small cross flow velocities and significant changes in thrust are immediately apparent. This situation becomes more complex with observance of a change in relative wind velocity as well as direction. The nature of wind is such that velocity distributions (gust spreads) normally are encountered. The probability of finding similar wind conditions on successive tests to check for repeatability of data is unlikely. The second significant problem is to stabilize the aircraft in order that representative performance data can be recorded. The pilot must apply controls to compensate for the moment and thrust changes caused by the wind and other disturbance factors. Thus, a dynamic condition is created with respect to helicopter control, engine power, thrust, and height above the ground. Depending on wind velocity, gust spread, hovering height, and aircraft characteristics, these changes may be very small or quite large. The total power required in hover is then the sum of the power required to produce an effective thrust to weight ratio of one and the control power requirements generated by the gust upsets.

The flight planning for a successful hover operation is no small task, and should be thorough. The complexity generally results from the number of ground personnel, management and coordination of the ground operation, and the remote site operation. Any omission in the planning will result in a decreased productivity, increased safety hazards, and perhaps a delay during the test operation. Even a small delay assumes major proportions when considering the critical nature of the atmospheric requirements.

The technique used to obtain the data should be the most suitable for the particular test vehicle and the recording equipment, and which will result in the highest productivity. For machines not normally equipped with a cargo hook, restraining cables or the load cell may be attached to the appropriate structural points. When this cannot be accomplished, a free flight technique must be used. The pilot technique should not be a significant factor in the validity of the quantitative data. It is, however, a prime factor in the productivity of the test. The pilot continually should develop his technique for quickly stabilizing the aircraft and, when stabilized, making a minimum number of control inputs during the recording period. A thorough knowledge of the flight control parameters, aircraft behavior changes, and the test sequence will contribute significantly toward increased pilot proficiency in the testing. The data recording procedures should be arranged to maximize the recording efficiency during premium atmospheric conditions.

The hovering data reduction effort is easier and less complex than for other tests encountered during the course of a flight test program. Care must be used not to overlook any "fallout" test results from the primary hovering tests. Some of these include stability and control, environmental, ground proximity, noise, vibration, and propulsion performance data. The data reduction procedure is simplified greatly by the nondimensional method, and the data are most adaptable to an automatic data processing procedure.

5-2 PLANNING

The hovering performance tests must be planned carefully since there are ground station personnel as well as the flight crew involved. The tethered hovering site must be reserved for the proper time. The site must be inspected for restricting obstacles and to insure that the equipment is properly placed and operable. The load cell must be checked for satisfactory range, operation, and current calibration. The tethering cables must be available in the proper lengths. An often neglected item is a sufficient number of connectors to accomplish all the necessary cable connections. The quick release mechanism must be checked to insure a satisfactory and safe operation. Arrangements with the support groups should be made to provide the necessary photographic, fire suppression, refueling, rescue, and maintenance support. The proper amount of ballast and number of handling personnel must be available to achieve quickly the desired gross weight changes. Radio and other communications must be planned carefully to insure that sufficient equipment will be available with compatible frequencies assigned to the test operation. The atmospheric equipment must be available to measure wind velocity, wind direction, free air temperature, and atmospheric pressure.

The planning best can be accomplished by first obtaining the data requirements from the test plan, then determining what data must be recorded, and, finally, insuring that the personnel and equipment are available to accomplish the various tasks. When support groups are involved, it is wise to inform them of the requirements and then generally to discuss with them how they will fulfill their responsibility. From this discussion, it will be possible to determine their state of preparedness, how much checking is needed, and how much assistance will be required. The important thing is to make the specific requirements clearly understood so far as they are known. In the event they are not known, general requirements should be given and the support group then should provide the best response within their capability.

Data cards should be prepared for the flight crew as well as the ground crews recording data. These cards should be prepared in as much detail as possible to allow the most accurate and efficient recording of the data. A typical flight card for the air crew is presented in Table 5-1.

The personnel recording the atmospheric data should record free air temperature, ambient pressure altitude, wind velocity, wind direction, and the time of day for data correlation purposes. When sufficient communications are available, they also should record point number, gear height, and thrust load. These parameters will insure exact data correlation with the data taken at other recording stations. When there are several groups participating, it may be advisable to have each prepare their own cards and then coordinate them with the control group. This will allow each group to provide their specialized input and to reduce the workload for the control group.

When the flight cards have been prepared and coordinated, a briefing should be conducted. The test detail covered in the briefing will be dependent upon the number of personnel involved and the complexity of the test. The most effective way to accomplish the briefing is through a series of preliminary briefings prior to the formal group briefing. The preliminary briefings may be with individuals or with the test groups. Specific items should be covered in great detail so that action, timing, sequence, and data requirements are clearly understood by all concerned. Following the preliminary briefings, the entire participating group should be briefed collectively. This briefing should be general in nature and should stress the operational interfacing of the groups since the pertinent details have been discussed previously. Each group should be presented a general knowledge of what the other groups will be doing and the internal workings of the total team. Safety is of prime importance for personnel working near the aircraft. It is important to have one individual in charge of the ground support team. The aircraft crew chief is the logical choice since he is the individual most familiar with the individual test aircraft. This individual should designate participants in such tasks as reballasting and instrumentation support. This will minimize the possibility of personnel walking into tail rotors, yaw booms and intakes, etc. Every effort should be made to avoid lengthy briefings. Time is valuable and when more than one flight per day is involved, a great deal of time can be consumed by briefings. In addition, deviating from the significant points can introduce extraneous material and confuse the pertinent issues to the extent that personnel may leave the briefings lacking the information they require. Adequately controlling a large briefing is a difficult task and must not be treated lightly. The amount of briefing required will depend on the experience level of the test team. It is reasonable to assume that the proficiency level of the team will be dependent on the least proficient member of the team at some time during the tests.

TABLE 5-1

HOVERING PERFORMANCE FLIGHT CARD

Point number	Free air temperature
Gear height	Pressure altitude
Rotor speed	Engine speed
Ballast aboard	Engine torque
Engine start gross weight	Inlet temperature
Fuel counter number	Thrust load

5-3 INSTRUMENTATION

The instrumentation required for hovering performance will vary somewhat with the method being used to obtain the data. For all methods where the helicopter is close to the ground, it is necessary to record the ambient atmospheric conditions quite accurately since the helicopter is very sensitive in this area. The ground instrumentation includes measurement of wind velocity, wind direction, free air temperature, and static pressure. For the tethered hovering method, a load cell device should be used to record the thrust generated by the helicopter. These data should be presented visually and permanently recorded by an automatic recording device. The visual data are suitable for average values and for monitoring the test progress. The automatic permanent recording is necessary to validate the manually recorded visual data and to provide transient information that can be correlated with other aircraft parameters. Detailed discussions of the ground equipment necessary to measure gear height and other items have been presented in Chapter 3.

An airborne instrumentation system is required to record steady-state and dynamic conditions. The steady-state performance parameters presented in the cockpit or the recorder's panel should include the items shown in Table 5-2.

Additional parameters such as ballast aboard, thrust load, height above the ground, free air temperature, wind velocity, and wind direction will be obtained from external ground sources. When an automatic analog recording device is available, the additional parameters shown in Table 5-3 should be recorded.

These aircraft position and motion parameters will be required to correct the power and thrust data for aircraft or pilot transient movements.

Every effort should be made to correlate all the automatic data recording devices to include the ground equipment.

5-4 TEST METHODS

5-4.1 GENERAL

The hovering performance tests may be accomplished on a vertical thrust stand, in tethered flight, or in free flight. The vertical thrust stand tests provide the greatest pro-

TABLE 5-2

VISUAL COCKPIT INSTRUMENTATION FOR HOVERING PERFORMANCE

Fuel used	Event marker
Engine torque	Counter number
Rotor speed	Inlet temperature
Engine speed	Free air temperature
Pressure altitude	

TABLE 5-3

AUTOMATIC RECORDING INSTRUMENTATION FOR HOVERING PERFORMANCE

Event marker	Longitudinal stick position
Counter number	Lateral stick position
Engine torque	Collective stick position
Engine speed	Pedal position
Rotor speed	
Angular acceleration (all axes)	Linear acceleration (all axes)
Attitude (all axes)	Rate (all axes)

ductivity but are very difficult to perform. For all these methods it is necessary to record the ambient atmospheric conditions. The aircraft temperature and pressure readings do not provide accurate ambient data when the helicopter is near the ground. Recirculation flow and engine gases entering the downwash can cause significant changes. For most helicopters, the downwash under the center of the rotor disk is very small, and direct impingement of the rays of the sun on the fuselage near the hub can result in a heat soaking effect. The weather station should be established as close to the test site as possible which will avoid the downwash influence. The wind velocity, wind direction, free air temperature, and pressure altitude should be recorded at a minimum of every 5 min during the progress of the tests. More frequent readings generally are desirable and are imperative when unstable or unusual conditions are observed. The weather station personnel should have predetermined values at which to inform the control group that unfavorable conditions are being recorded. Significant test events and any unusual occurrences observed should be recorded by the weather station personnel.

5-4.2 VERTICAL THRUST STAND

The vertical thrust stand hovering performance technique requires the greatest amount of preparation. The mounting fixtures must be prepared, and the stand must have a current calibration and be in operable condition. The data recording procedures must be thoroughly established and personnel briefed since the productivity is extremely high and there is limited time available during the test for other than data acquisition. The vertical thrust stand normally has a capability of recording both forces and moments. There also may be an attitude and height variation feature. Prior to the tests—when the aircraft is mounted in the desired condition—tare readings should be taken for all parameters, and, if possible, all forces and moments should be zeroed. The best test procedure is to obtain data for one variable at a time. The baseline performance thrust should be measured as a function of engine power at a minimum height above the ground with all controls centered and the aircraft in a level attitude. Changing height or aircraft attitude is time consuming, and it is usually best to accomplish all possible tests prior to changing these conditons. Following the power sweep, a test should be made at full power while varying the collective pitch or other existing thrust modulating devices. Since most helicopters achieve control largely through thrust vectoring, there is usually a vertical thrust loss during any control input. Each axis should be explored to determine the magnitude of the loss as well as the control moment generated. When testing one axis, great care must be taken to insure there is no input on other axes. When possible, a flight control fixture should be installed to aid the pilot in this task. A minimum number of points on these tests are four in each direction from neutral. Exteme caution should be used that the moments are of insufficient magnitude to damage the thrust stand or the aircraft. This can be a formidable task when the aircraft capability is not well known. The moments obtainable may be so great as to cause excessive blade flapping or fuselage bending with resultant permanent damage. Following the individual control axes evaluation, it may be desirable to investigate the case of static and dynamic simultaneous control inputs. The hazards are increased during this test, and a careful build-up technique should be used with the static conditions being tested prior to entering any dynamic conditions. These tests should be repeated at different heights above the ground to evaluate the effects of ground proximity and at different attitudes to evaluate the rotor vector directional effects.

Each static point should be maintained a minimum of 30 sec to allow the engines, rotor wash, and other parameters to stabilize. A greater stabilization period should be used when feasible. Data should be recorded at several intervals during the test run. The pilot should set the required conditions and inform the ground personnel when the specified test conditions are attained. During this set up time the ground equipment should be monitored with a view toward establishing a stable indicated data condition. When the stable condition is achieved, data should then be recorded simultaneously by the recording devices and personnel. A countdown may be useful when automatic devices are not being used. Time and counter correlation must be given special attention to aid in the data reduction, presentation, and analysis. An "event" device is of great assistance to correlate the information. After each test has been completed, a repeat tare reading should be obtained at the initial condition to determine any bias of variation that may have occurred during the test.

The vertical thrust stand provides a unique opportunity to obtain environmental data. The major difficulty is to record the data in the time period available at each test condition. When the environmental equipment must be moved from station to station, the primary test may be delayed to provide sufficient time to obtain these data. Unless the data can be recorded simultaneously at several locations, it is best that the primary performance test not be delayed for secondary objectives. In this situation, it is more advantageous to define completely one area than to measure inadequately the data for several locations. Refueling is a significant problem for most aircraft and, if possible, fuel should be used from a ground source. This will reduce delays caused by frequent refueling operations. The necessary instrumentation work also must be carefully scheduled and swiftly accomplished to prevent excessive loss of valuable test time.

5-4.3 TETHERED HOVERING

The preparations required for the tethered hovering technique are less difficult but the test operations are more difficult than the vertical thrust stand technique. The most advisable way to accomplish the tethered hovering tests is to start at an intermediate height and then to progress to the high and low extremes. The pilot effort is highest at the out-of-ground effect (OGE) position because of orientation and position difficulties, and at the minimum height because of the ground effect influence. The quick release mechanism should be checked with the aircraft on the ground initially and in the air with a moderate load to familiarize the pilot with the aircraft reaction in the event the cable breaks, the cargo hook fails, or a deliberate disengagement is accomplished. The aircraft should be positioned with the relative wind in a manner which will minimize the directional and lateral control requirements.

At a given gear height, the power should be varied incrementally from a minimum required to hover at the height to the maximum power available. Again, from an operation and safety standpoint, it is advisable to start at a low power and then to advance in power increments to the higher power conditions. A range of rotor speeds also should be used to give the maximum available variation in thrust coefficient. The sequence of rotor speeds is not of significance but care must be taken to insure that a complete curve is obtained for each rotor speed used. Normally, three rotor speeds are sufficient and a minimum of six points for each rotor speed is necessary. Additional points will better define the curve and increase the confidence level in the data. Prior to recording the data, the aircraft should be stabilized with respect to vertical alignment, power, and control positions. Thrust indications should be monitored to insure a reasonably steady value and a representative static condition. Data should be recorded over a minimum time interval of 30 sec with a greater period being desirable. The pilot usually must make small control inputs during the data recording time and these should not invalidate the data. The collective stick or thrust control and, for tail rotor helicopters, the directional controls should not be moved. The cyclic control movements should be held to an absolute minimum. In the event power is changed, the point should be repeated. During the recording interval, the indicated thrust level will be fluctuating to some degree. The pilot or engineer should use an event marker to identify the stabilized values. A signal, such as a landing light, can be incorporated into the event marker switch. This will enable the ground personnel to identify the exact time the data are being recorded. It is advisable to record the maximum and minimum values observed as well. A representative value should be plotted against some power variable during the test in order to monitor the test progress and the data validity. This plot also can be used to control the power increments which will insure a suitable distribution of the data points through the available performance range. The tests should be repeated at various heights from the minimum in-ground effect (IGE) to a height of 1.5 rotor diameters. The height increments should be 5 ft to a height of 20 ft and then 10-ft increments at the greater heights. The size of the helicopter and the nature of the ground effect will indicate any advisable changes to the suggested increments. When possible, the gear heights should be measured from the ground. Another convenient way to measure the height is to drop a weighted measure from the aircraft to the ground. The attitude should be considered, and it is usually best to take readings both forward and aft on the gear. This is of special importance for large helicopters where the height differential may be several feet.

The effects of any existing wind can be evaluated to some degree by hovering with relative winds from each quadrant. Normally the tests are conducted in winds less than 2 kt. Winds in excess of this value result in an excessive pilot workload, large thrust variations, and a generally unstable data gathering condition. At the present time there is no adequate method to correct the data for these types of wind effects.

5-4.4 FREE FLIGHT HOVERING

The free flight hovering method normally is used only to check specific performance guarantees and to insure that the tethering or thrust stand restraints did not cause a performance loss. This method is used as a primary data gathering method only when a tethered technique is not feasible.

The free flight hover technique may be accomplished near the ground or at altitude. Near the ground the technique most commonly used is to ballast the aircraft for an OGE hover capability and then to set the power to obtain a low height IGE hover. Power then is increased incrementally to obtain data at several heights until the OGE height is reached. Rotor speed is maintained constant during the series of data points, and the height is measured for each stable condition. The test then is repeated for various gross weights and rotor speeds. The primary difficulty is to determine the hover height since fuel constantly is being consumed and the helicopter is climbing on each data point during the recording period. Another variation of this method is to vary power and rotor speed while maintaining a constant height. When varying the rotor speed, time can be saved by utilizing the previous rpm setting for the next time point. For example, if low, medium, and high rpm settings are utilized for a test condition, the test following could utilize a high, medium, and low rpm sequence. Care must be observed when varying rpm rapidly at the critical hover ceiling condition.

Gear height then is varied incrementally and the test is repeated for a range of gross weights. The major problem with this method is establishing and maintaining the desired height since the pilot continually must be provided height information during the stabilizing period and must change the thrust controls accordingly. This reduces the test productivity since it is both tiring to the pilot and time consuming. When stabilized, the thrust control should not be moved. The cyclic control movements should be kept to an absolute minimum. An event marker is of assistance in identifying the exact stablized condition. In addition, the data must be recorded very rapidly to reduce the effect of the aircraft climbing as fuel is consumed.

In some instances, it is desired to obtain hover ceiling performance data at particular gross weight, altitude, and power conditions which may be at a considerable height above the ground. The problem with this technique is that an indicated hover, which is a true zero airspeed condition, is difficult to achieve without a ground or artificial reference for the pilot. A true reference can be obtained from a weighted, tufted string suspended from the helicopter. The string should be approximately 200 ft long with tufts attached at intervals to aid in airflow identification and reference. The hover height that can be attained should be calculated prior to the test to avoid spending an excessive amount of time searching for the proper altitude. The hover condition should be approached from below and climbed into rather than approached from above where high rates of descent or settling with power may be encountered. When the aircraft is in a position slightly below the anticipated hover altitude, the pilot should slowly decrease airspeed and approach the hovering condition. The airspeed indicator will become inoperative below approximately 20 kt and the pilot then must proceed on aircraft feel, control positions, altitude, and attitude references. As airspeed is reduced further, the helicopter will pass through translation which usually is evident from the vibration, trim, and power required changes. The reference string should now be lowered. As the string is lowered to the required length, a bow will be knotted in a direction opposite to the relative wind velocity. The tufts will react in a similar manner. The pilot should be advised to move the aircraft by longitudinal and lateral control inputs in a direction which will remove the bow from the string and the directional aspects from the tufts. The string now may exhibit a characteristic ripple or wave motion, which indicates the downwash is traveling vertically along the string. A true hover now exists although the pilot should verify the condition by moving the aircraft while observing the string for anticipated results. Usually this procedure is done only once to determine the aircraft and string characteristics at the hover condition. When the hover condition is attained, the data should be recorded as rapidly as possible, while monitoring the string to insure that the hover condition is maintained. The aircraft will be climbing slowly during the stabilizing and recording period. This should not influence significantly the data validity unless the fuel consumption is very high and the data are not recorded rapidly.

When at the hover condition at altitude, it is advisable to obtain data for several different rotor speeds. The pilot should slowly change the rotor speed. In the event a high rate of descent develops, the test should be abandoned immediately since a power settling condition can develop. The string operator should be prepared to jettison it should the pilot be required to increase forward speed. In this situation, the relative wind will carry the weighted string aft in the direction of the tail rotor. By the same token, the pilot should give the string operator as much warning as possible prior to leaving the hover condition. When the data have been recorded, the string should be retrieved prior to increasing airspeed. Under no circumstances should the string be securely fastened to the aircraft or a passenger in the aircraft. Since there is a possibility of losing the weight or jettisoning the string, the test should be conducted over an uninhabited area.

This method is the most difficult to perform from a pilot workload standpoint due to the lack of adequate visual references or instruments. Particular care should be taken when varying rpm under these conditions.

5-5 DATA REDUCTION

The data reduction should be conducted as soon as possible after the testing has been accomplished. This will allow prompt rescheduling of the test in the event the data are unsuitable and also will prevent the accumulation of a backlog of data processing. The data must first be obtained from the various recording devices and personnel, and converted to engineering units if applicable. The data cards should be checked carefully for legibility and completeness. The recording personnel should review the card and explain each item and be questioned to insure that no important events are committed to memory and not subsequently recorded. The latter is a common occurrence when data are being acquired rapidly. Following the assembly of the card, the next effort is to correlate them both individually and collectively. This must be done carefully and completely to insure that the reduced test results will be usable in all respects. Many times the exact data presentation is not known at this time and, as such, no effort can be left undone with respect to correlation.

If data were recorded on magnetic tape and then processed through a ground station to produce the engineering data, instrument and position error corrections must be applied before the data can be processed further. Many ground stations have the capability of applying instrument corrections, and the printout from this facility then is suitable for additional processing directly. It also may be desirable to leave the data in tape form for additional automatic processing.

The oscillograph roll must be developed and read to obtain engineering data. Prior to reading any data from the roll, all possible calibration factors must be checked, as well as the voltage monitor, to determine how the calibrations should be biased. Following this, the trace deflections may be read, calibrations are applied, and the corrected data are then available for plotting for additional analysis.

The photo panel film is developed and the indicated values are recorded. Instrument and position error corrections are applied, and the data are in a form suitable for additional engineering effort.

The hand recorded data are corrected and processed as soon as the aircraft has returned from the test flight. This will provide information in the shortest time with a minimum reduction effort. Additional time may be saved by transmitting the data to the ground for immediate reduction as the test progresses.

The preliminary planning should have established the data reduction procedures, and the effort should proceed smoothly. However, there usually is insufficient time, knowledge, and manpower available to accomplish all planning prior to the start of the program. The equations to be used must be established and arranged in a logical order with respect to entry of the test data. It is best to record the data in the sequence of the readings taken from the primary recording instrument. Any necessary corrections are made then in adjacent areas on the data reduction form rather than on a separate document. Then test data are entered and the required calculations are performed in the order that will be most efficient and useful.

As soon as the initial data reduction form is completed, a known test point should be calculated to check for omissions, errors, continuity, and suitability of the arrangement. Usually it is necessary to revise the form. In some cases, it may be desirable to divide the form into various areas such as power, dimensional coefficients, and correction procedures. This often is done on complex programs to reduce the complexity of each individual form or to permit several groups to process data simultaneously.

If a computer will be used to reduce the data, the hand reduction form may now be given to the computer programming personnel as a guide in developing a computer program. The resulting program then is checked by a hand calculation with an identical point to insure that there has been no programming or translation error. These checks then become an integral part of the data reduction records and may be used at any time to accomplish a confirmatory check. Some engineering input should be applied to the program to facilitate the presentation and analysis of the data. It is useful to have the computer output contain the flight number, flight date, type of test, and any other pertinent identification data. The machine time used also may be noted on the printout since this information can be used for both current and future planning purposes. The program should be constructed so that the input data may be printed or plotted prior to performing any calculations. This printout can be examined to verify the validity of the input and to detect errors introduced by the data handling procedures to this point. It is also advisable to have a printout of the current and effective date of all the calibrations to be used in the calculations. All too often instruments are recalibrated or changed, and the change is not reflected in the automatic data processing (ADP) procedures. The computer output data may be in a physical form that is difficult to work with. It may be helpful to transfer the computer values to the hand reduction form or to prepare another summary form. This new summary form should contain all the data that are to be plotted or that require further processing. The unwieldy, fragile computer sheets may then be filed for reference while working and plotting from the summary form.

A manual data reduction form, for reducing hovering performance data for a typical helicopter, is presented in Table 5-4*.

The instrumentation, position, and system error correction procedures are included on the form primarily for illustration purposes. It

*The data reduction forms (tables) are located at the end of each chapter.

is usually better and more convenient to accomplish these steps elsewhere and enter only corrected values on the reduction form. The fuel specific gravity topic is discussed in detail in Chapter 14. The inlet performance was included on this form for completeness although in general practice it is accomplished as shown in Chapter 13.

The reduction form, of course, will vary with the test vehicle depending upon the rotor configuration, aircraft configuration, and the propulsion system. Changes also will be necessary for variations in the data acquisition systems and correction methods. The data entered on the form should be obtained from the most accurate data source, regardless of the type of recording system being used. During the data reduction process, the engineer should monitor the critical parameters to insure accuracy and personally should make random calculation checks. It is helpful also to make preliminary plots which will show general trends and orders of magnitude.

5-6 DATA PRESENTATION

When the data reduction has been completed, consideration must be given to how the data are to be presented and reported. The most useful and common method is the graphical form. Plots can be used to show the greatest amount of information with the smallest volume. Clear, concise, and useful plots are not constructed as easily as one might imagine. There are several cardinal rules that must be followed namely:

1. The plot must be clear with respect to intent and content.

2. Too many parameters tend to confuse the reader and the desired meaning is no longer clear.

3. The scales, symbols, and curves must be labeled clearly and identified.

4. The reader should not be required to research the entire report, perhaps in vain, to determine what the conditions were on a particular curve or point.

5. All the items on the plot should be logical and the format should be obvious and consistent. These rules should apply consistently for format, symbology, abbreviations, and definitions. The artistic aspects are important. A plot that is attractive will be more useful than one that is otherwise, all other factors being equal.

Some typical plots are illustrated in Figs. 5-1 through 5-8 and discussed in the paragraphs that follow. For the free flight, constant power technique, a plot is constructed from the data contained in steps 30, 55, 59, and 63 of the data reduction form, Table 5-4. The resultant plot is shown in Fig. 5-2. There will be a similar plot for each constant power setting tested. The various rotor speeds and density altitudes are denoted by the various

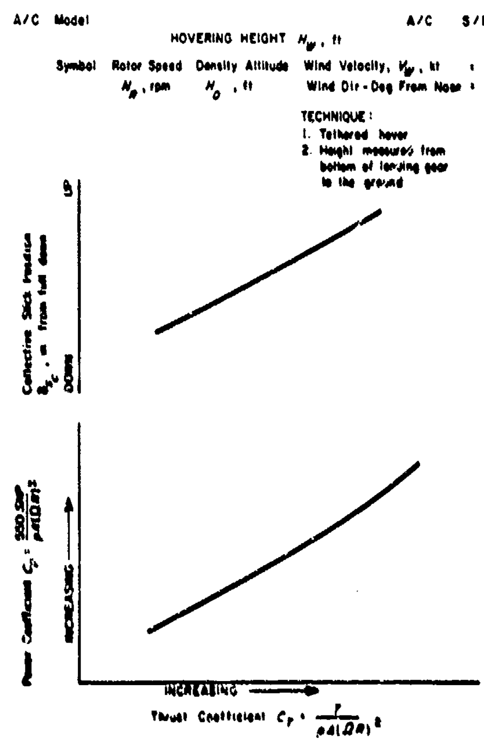


Figure 5-1. Nondimensional Hovering Performance and Collective Stick Positions

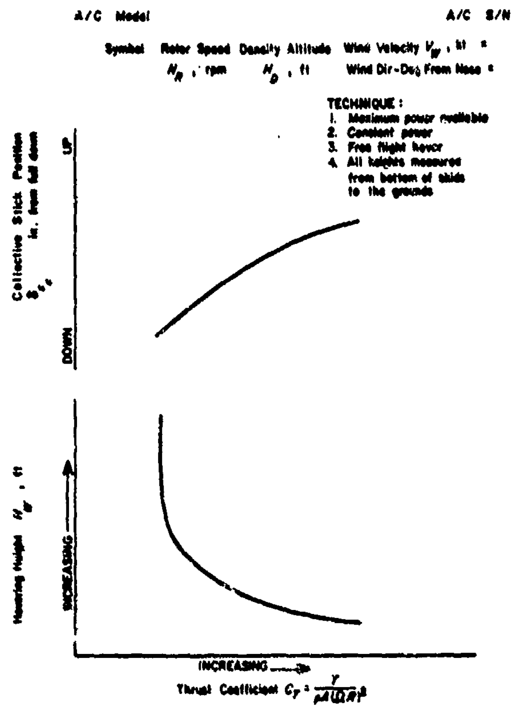


Figure 5-2. Ground Effect Influence on Hovering Performance

symbols drawn around the data points. These plots then are entered at the desired hovering heights to obtain the precise C_p and C_T values required to hover at that height. In order to insure completeness, accuracy, and adequate documentation, it is advisable to use a summary calculation data sheet as shown in Table 5-5.

TABLE 5-5

SUMMARY HOVER PERFORMANCE

H_W	C_T	N_R	δ_{s_c}	C_p

These values are then plotted in the format illustrated by Fig. 5-4.

For the tethered, thrust stand, or constant altitude free flight technique, the nondimensional data form steps 30, 55, 59, and 63 of the data reduction form, Table 5-4, are plotted to obtain Fig. 5-3. As previously discussed in par. 5-4, there will be one of these plots for each hovering height tested. Various symbols are used to show the different altitudes and rotor speeds. All the data obtained for a given hovering height are plotted on the same figure.

The series of nondimensional plots illustrated by Fig. 5-3 is used to construct cross plots showing collective pitch as a function of thrust and hovering height (see Fig. 5-4). The plot is constructed by entering Fig. 5-3 at an even thrust coefficient value and reading the corresponding collective stick position at a given rotor speed. A summary data reduction sheet is shown in Table 5-6.

TABLE 5-6

SUMMARY COLLECTIVE STICK REQUIRED

H_W	C_T	N_R	δ_{s_c}	C_p	θ_{cp}

The plotted values then yield Fig. 5-4.

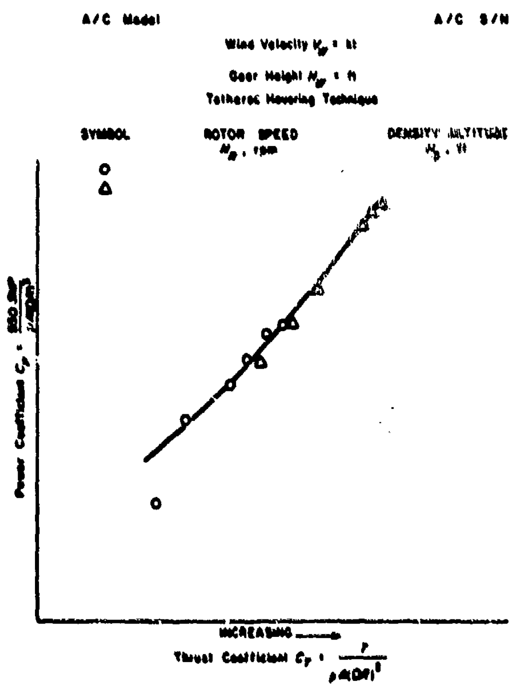


Figure 5-3. Nondimensional Hovering Performance

Further summary calculations are facilitated by summarizing the nondimensional performance as shown in Fig. 5-5. This is accomplished by entering the curves illustrated by Fig. 5-3 and reading the faired line values of C_p for incremental C_T values. A summary data reduction form is shown in Table 5-7.

TABLE 5-7

NONDIMENSIONAL HOVERING SUMMARY

C_T	C_p	H_W

The C_p scale is staggered to give a reasonable spacing for the values of the various hovering heights. Symbols normally are not used on this type of plot since there are no test points.

The resultant curves should make a family with characteristic trends. Curves now are faired in a manner that will show continuity and characteristic shapes. The modified fairings are not transferred to the data points on the original nondimensional curve. Generally, there is sufficient scatter in the test points to allow curve fitting which satisfactorily matches the composite plot (Fig. 5-5) and still passes reasonably well through the test data (Fig. 5-3). Failure to meet these criteria is an indication that the test values are invalid or that the data reduction was in error.

From the nondimensional summary plot it is possible to calculate performance in most any desired manner. Performance at a given altitude may be presented as shown in Fig. 5-6. The altitude may be presented as shown in Fig. 5-6. The altitude, temperature, and gross weights selected may be any desired

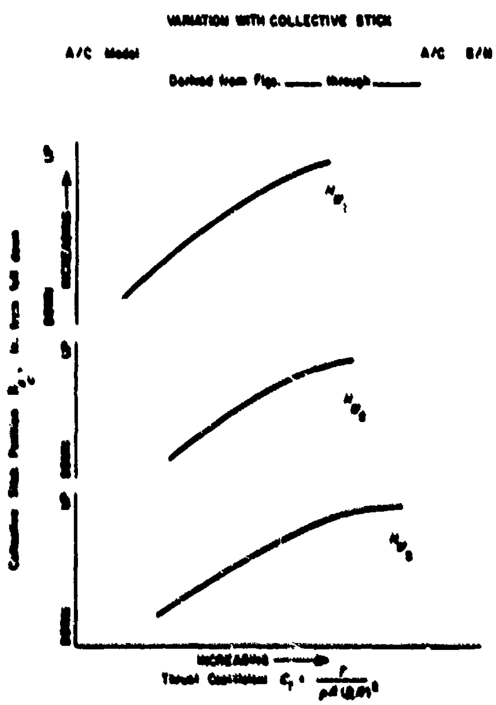


Figure 5-4. Hovering Performance Variation With Collective Stick

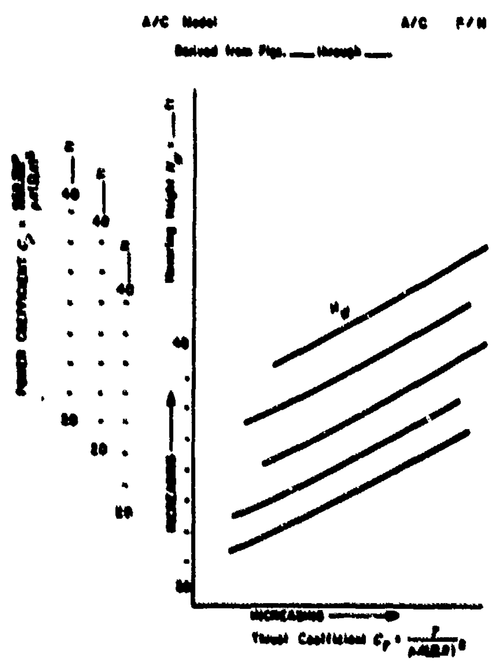


Figure 5-5. Nondimensional Hovering Performance Summary

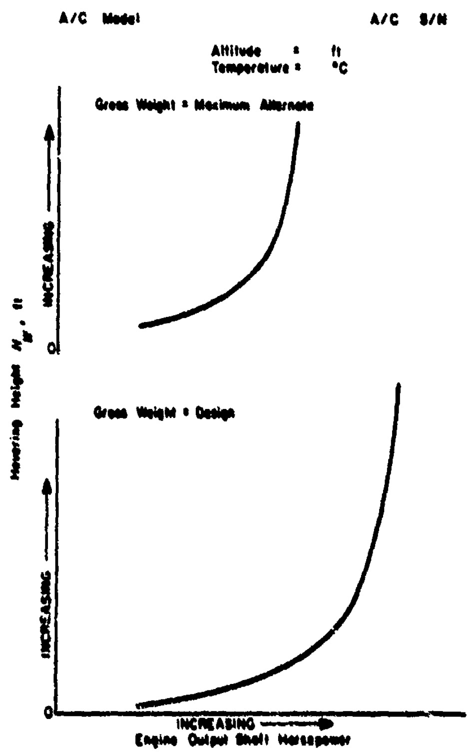


Figure 5-6. Hovering Performance

values. The weights most commonly used are design values or those conditions for which guarantees were made. The gross weight, rotor speed, and altitude conditions selected establish the C_T. This value then is used to enter Fig. 5-5 and obtain the C_P required for each hovering height. The C_P is then solved for *SHP* and plotted to yield Fig. 5-6. A summary data reduction form is presented in Table 5-8. Transmission or power limits should be labeled on the plot to indicate the maximum performance available. These limits are obtained from the power available curves shown in Chapter 14.

TABLE 5-8

SUMMARY HOVERING PERFORMANCE

H_p	T_a	H_D	N_R	ρ	$A(\Omega R)^3$	$A(\Omega R)^2$	C_T	C_P	H_W

The determination of out-of-ground effect hover height is best shown by a plot such as that illustrated in Fig. 5-7. Thrust coefficients are chosen which cover the maximum range of the test data. These values then are used to enter the summary hovering plot (Fig. 5-5) to obtain the C_P and hovering height. The out-of-ground effect line then is drawn through the points where there is no power required increase with additional hover height. The line then is plotted as a function of C_T as shown in Fig. 5-8. From this plot the OGE height may be calculated for any desired operating conditions.

The figure of merit and mean lift coefficient curves may be plotted directly from steps 62 and 64 of the data reduction form, Table 5-4. These test data may show sufficient scatter as to be difficult for both presentation and analysis. An easier and more accurate method is to calculate these param-

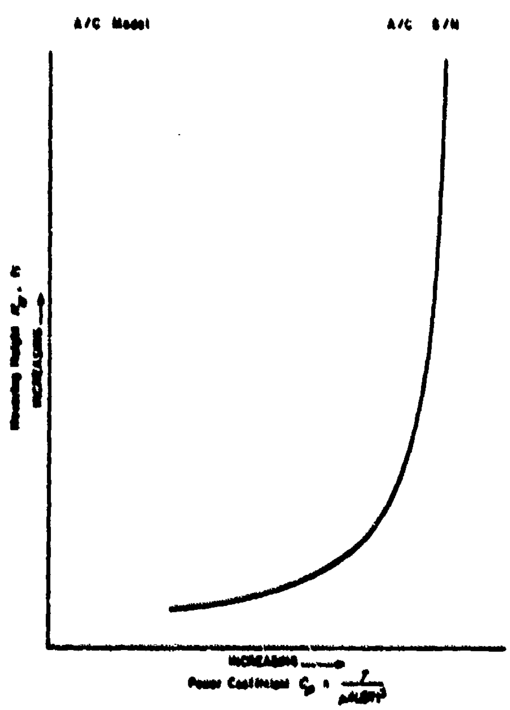


Figure 5-7. Summary Hovering Performance as a Function of Power Coefficient C_P

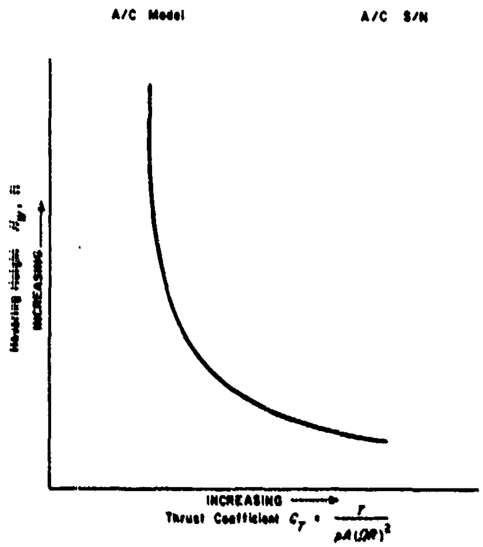


Figure 5-8. Out-of-ground Effect Hovering Height

eters from the smoothed data shown in Fig. 5-5. The calculation form is presented in Table 5-9.

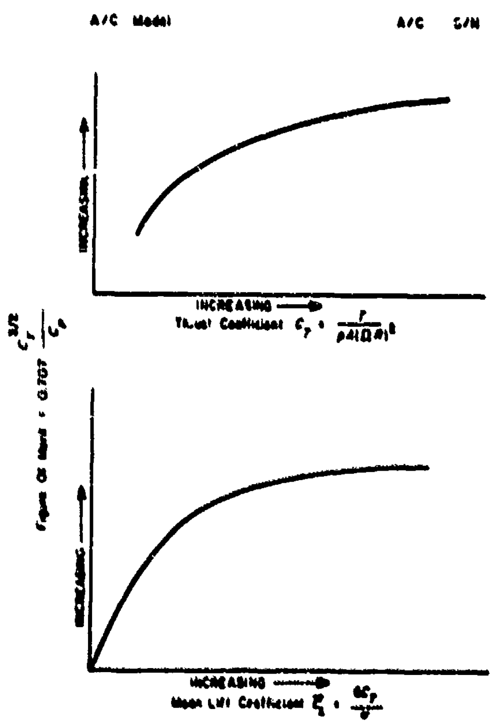


Figure 5-9. Figure of Merit and Mean Lift Coefficient

TABLE 5-9

FIGURE OF MERIT AND MEAN LIFT CALCULATIONS

C_T δ $C_T^{3/2}$ H_W C_P $6C_T/\delta$

$C_T^{3/2}/C_P$ $0.707C_T^{3/2}/C_P$

The resultant values then are presented as shown in Fig. 5-9.

The final summary plot and the most useful information to the operator is shown in Fig. 5-10. Power available is a necessary input to this calculation as shown in Table 5-10.

TABLE 5-10

SUMMARY HOVER CEILING CALCULATION

H_P T_a N_R $A(\Omega R)^2$ $A(\Omega R)^3$

SHP_A H_D C_{P_A} C_T H_W GW

Since this curve is intended to summarize the maximum hovering performance available, the power available must include all the actual installation and operating losses that are prevalent.

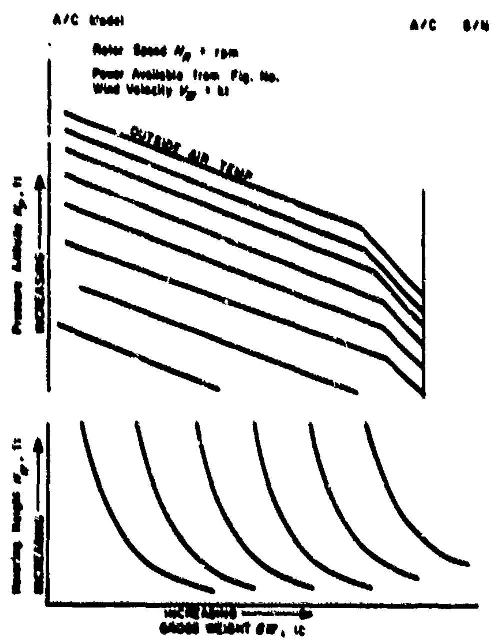


Figure 5-10. Summary Hovering Performance as a Function of Gross Weight GW

TABLE 5-4

DATA REDUCTION FORM FOR HOVERING PERFORMANCE

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1	Point No./Flight No.				
2	Indicated pressure altitude	$H_{p_{IND}}$		ft	From manual or photo panel recording
3	Altimeter instrument correction	$\Delta H_{p_{IC}}$		ft	From altimeter calibration curve
4	Altimeter position error correction	$\Delta H_{p_{PE}}$		ft	From altimeter position error calibration curve
5	Pressure altitude	H_p	$H_p = H_{IND} + \Delta H_{p_{IC}} + \Delta H_{p_{PE}}$	ft	(5) = (2) + (3) + (4)
6	Ambient air pressure	P_a	$P_a = 29.9216 \times \delta$	in. Hg	From tables or calculated at (5) (6) = 29.9216 · (7)
7	Ambient air pressure ratio	δ	$\delta = P/29.9216 = (1 - 6.875586 \times 10^{-6} \times H_p)^{5.25585}$		From tables or calculated panel recording (7) = (6) / 29.9216 = $[1 - 6.875586 \times 10^{-6} \cdot (5)]^{5.255}$
8	Indicated air ambient temperature	$T_{a_{IND}}$		°C	From manual or photo panel recording

TABLE 5-4 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
9	Ambient air temperature instrument correction	$\Delta T_{a_{IC}}$		°C	From instrument calibration curve
10	Ambient air temperature system correction	$\Delta T_{a_{SC}}$		°C	From system calibration curve
11	Ambient air temperature position error correction	$\Delta T_{a_{PE}}$		°C	From position error calibration
12	Ambient air temperature	T_a	$T_a = T_{a_{IND}} + \Delta T_{a_{IC}} + \Delta T_{SC} + \Delta T_{PE}$	°C	(12) = (8) + (9) + (10) + (11)
13	Ambient air temperature ratio	θ_a	$\theta_a = \dfrac{T_a + 273}{T_0 + 273}$		Calculated or taken from tables at T_0 $(13) = \dfrac{(12)}{T_0 + 273}$
14	Air density ratio	σ	$\sigma = \rho/\rho_o = \dfrac{(1-6.875586 \times 10^{-6} H_p)^{5.25585}}{\theta}$		From tables or calculated $(14) = \dfrac{[1 - 6.875586 \times 10^{-6} \cdot (5)]^{5.25585}}{(13)}$
15	Density altitude	H_D	$H_D = \dfrac{10^5}{0.6875586}(1 - \sigma^{0.235})$	ft	From altitude tables or calculated at H_p and T_a $(15) = \dfrac{10^5}{0.6875586}[1 - (14)]^{0.235}$
16	Air density	ρ	$\rho = 0.0023769 \times \sigma$	slug/ft³	From tables or calculated at H_D (16) = 0.0023769 · (14)
17	Indicated fuel specific weight	FS_{IND}		lb/gal	Average of preflight and postflight readings
18	Fuel temperature	T_f		°C	Average of preflight and postflight readings

TABLE 5-4 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
19	Corrected fuel specific weight	FS_C		lb/gal	Corrected to T_f by curve showing variation with temperature
20	Fuel counter reading	FC		ct	From any data source
21	Fuel counter to fuel volume ratio	K_F		ct/gal	From flow meter calibration curve at average power setting during the test
22	Volume of fuel used	FU_{VOL}	$FU_{VOL} = FC/K_F$	gal	(22) = (20) / (21)
23	Weight of fuel used	FU_W	$FU_W = FU_{VOL} \times FS_C$	lb	(23 = (22) / (19)
24	Weight of ballast	W_B		lb	From ballast crew or flight engineer data card
25	Gross weight	GW	$GW = ESGW - FU_W + W_B$	lb	$ESGW$ from maintenance form or engineer data sheet (25) = $ESGW$ − (23) + (24)
26	Indicated restraining thrust	T_{IND}		lb	From load cell reading
27	Restraining thrust correction	ΔT_C		lb	From load cell calibration
28	Restraining thrust	T_C	$T_C = T_{IND} + \Delta T_C$	lb	(28) = (26) + (27)
29	Total thrust	T	$T = T_C + GW + W_{INSTR}$	lb	Where W_{INSTR} is the weight of the load cells and cables (29) = (28) + (25) + W_{INSTR}

TABLE 5-4 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
30	Wheel/skid height	H_W		ft	Measured from cables used by ground crew, photographs or visual theodolite
31	Wind speed	V_W		kt	From weather station data card
32	Wind azimuth	α_W		deg from nose	From weather station data card
33	Indicated ground station ambient air temperature	$T_{aG_{IND}}$		°C	From weather station data card
34	Ambient air temperature instrument correction	$\Delta T_{aG_{IC}}$		°C	From calibration curve
35	Ambient air temperature	T_{aG}	$T_{aG} = T_{aG_{IND}} + \Delta T_{aG_{IC}}$	ft	(35) = (33) + (34)
36	Ambient indicated pressure altitude	$H_{P_{IND}}$		ft	From weather station data card
37	Ambient pressure altitude instrument correction	$\Delta H_{P_{IC}}$		ft	From altimeter calibration curve
38	Ground station pressure altitude	H_{P_G}	$H_{P_G} = H_{P_{IND}} + \Delta H_{P_{IC}}$	ft	(38) = (36) + (37)
39	Standard system indicated ambient air temperature	$T_{a_{IND_S}}$		°C	From standard system recorded by pilot/engineer

TABLE 5-4 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
40	Standard system indicated ambient pressure altitude	$H_{P_{IND_S}}$		ft	From standard system recorded by pilot/engineer
41	True aircraft pressure altitude	H_{P_T}	$H_{P_T} = H_{P_G} + H_W + X$	ft	Where X is the distance from the skid to the altitude sensor (41) = (38) + (30) + X
42	Test system altitude position error due to proximity of ground	$\Delta H_{P_{PE_t}}$	$\Delta H_{P_{PE_t}} = H_{P_T} - H_P$	ft	(42) = (41) − (5)
43	Standard system altitude position error due to proximity of ground	$\Delta H_{P_{PE_S}}$	$\Delta H_{P_{PE_S}} = H_{P_T} - H_{P_{IND_S}}$	ft	(43) = (41) − (40)
44	Test system temperature position error due to proximity of ground	$\Delta T_{a_{PE_t}}$	$\Delta T_{a_{PE_t}} = T_{a_G} - T_a$	°C	(44) = (35) − (12)
45	Standard system temperature position error due to proximity of ground	$\Delta T_{a_{PE_S}}$	$\Delta T_{a_{PE_S}} = T_{a_G} - T_{a_{IND_S}}$	°C	(45) = (35) − (39)
46	Indicated inlet total pressure	$P_{T_{2_{IND}}}$		psi	From manually or photo panel recorded data
47	Inlet total pressure instrument correction	$\Delta P_{T_{2_{IC}}}$		psi	From instrument calibration curves

TABLE 5-4 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
48	Inlet total pressure system correction	$\Delta P_{T_{2_{SC}}}$		psi	From system calibration curves
49	Inlet total pressure	P_{T_2}	$P_{T_2} = P_{T_{2_{IND}}} + \Delta P_{T_{2_{IC}}}$	psi	(49) = (46) + (47) + (48)
			$+ \Delta P_{T_{2_{SC}}}$		
50	Indicated inlet total temperature	$T_{T_{2_{IND}}}$		°C	From manual or photo panel recorded data
51	Inlet total temperature instrument correction	$\Delta T_{T_{2_{IC}}}$		°C	From instrument calibration curves
52	Inlet temperature system error correction	$\Delta T_{T_{2_{SC}}}$		°C	From system error calibration curves
53	Inlet total temperature	T_{T_2}	$T_{T_2} = T_{T_{2_{IND}}} + \Delta T_{T_{2_{IC}}}$	°C	(53) = (50) + (51) + (52)
			$+ \Delta T_{T_{2_{SC}}}$		
54	Rotor speed	N_R		rev/sec	From rotor blips on the oscillograph or tape

TABLE 5-4 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS	
55	Rotor speed	N_R	$N_R = \frac{rev}{sec} \times 60 \frac{sec}{min}$	rpm		$(55) = (54) \times (60)$ $\pi R^2 [2\pi \cdot (55) \cdot R/60]^2$
56	Thrust coefficient constant	K_{c_T}	$\pi R^2 (2\pi N_R R/60)^2$ $= A(\Omega R)^2$	lb/slug/-ft^3	Calculated or from tabulation at N_R R = rotor radius	$(56) = \pi R^2 [(2\pi/60) \cdot (55) \cdot R]^2$
57	Power coefficient constant	K_{c_p}	$K_{c_p} = \frac{550}{\pi R^2 (2\pi N_R R/60)^3}$ $= 550/[A(\Omega R)^3]$	slug/ft^3 / HP	Calculated or from tabulation at N_R	$(57) = \frac{550}{\pi R^2 [2\pi \cdot (55) \cdot R/60]^3}$
58	Test shaft horsepower	SHP_t		HP	From power determination calculations	
59	Thrust coefficient	C_T	$C_T = T/[\rho A(\Omega R)^2] = T/(\rho K_{c_T})$			$(59) = \frac{(29)}{(16) \cdot (56)}$
60	(Thrust coefficient)$^{3/2}$	$C_T^{3/2}$	$C_T^{3/2} = \{T/[\rho A(\Omega R)^2]\}^{3/2}$			$(60) = (59)^{3/2}$
61	Rotor solidity	σ_m	$\sigma_m = b_{c_e}/(\pi R)$		b = number of blades C_e = equivalent blade cord (on thrust basis), ft	
62	Mean lift coefficient	$\bar{C}_L$	$\bar{C}_L = 6C_T/\sigma_m$			$(62) = 6 \cdot (59)/(61)$
63	Power coefficient	C_p	$C_p = 550 \times SHP/[\rho A(\Omega R)^3$ $= K_{c_p} \times SHP_t/\rho$ $= 550\, SHP_t/(K_{c_p} \Omega R)$			$(63) = \frac{(58) \cdot (57)}{(16)}$
64	Figure of merit	FM	$FM = 0.707\, C_T^{3/2}/C_p$			$(64) = 0.707\ (60)/(63)$
65	Advancing blade tip Mach number	M_{TIP}	$M_{TIP} = \Omega R/a$		a = speed of sound (ft/sec) at ambient temperature	$(65) = (2\pi/60) \cdot (55) \cdot R/a$

CHAPTER 6

TRANSLATION PERFORMANCE

6-1 GENERAL

Translation performance, more commonly referred to as sideward and rearward flight performance, generally is confusing when the subject first is encountered. It is simply an extension of the normal concept of hovering. Since it is readily apparent that in operational use the helicopter usually will be hovering in surface wind conditions, information must be provided that will demonstrate any performance changes relative to wind velocity and direction. The hovering in winds condition is simulated by translating the helicopter laterally and rearward at measured velocities and inflow azimuths during calm surface wind conditions. This translation then introduces a relative wind of known magnitude to the helicopter.

The purposes of the test are to determine the power required characteristics at the different airspeeds, to discover any performance limitations, and to evaluate the effect of the relative wind directions on the propulsion system. This information then is used to inform the operator of the performance changes that may be expected relative to the basic zero wind data as surface winds are encountered during operation.

As was discussed during the zero wind hovering tests, the helicopter is sensitive to small relative wind velocities, i.e., translational speeds. The speed range normally tested is that required by the flying quality specifications or that necessary to accomplish the known mission requirements. The exact nature or magnitude of the power requirements cannot be anticipated with any degree of confidence prior to the actual tests. The characteristics are usually nonlinear with numerous discontinuities and/or reversals. In addition, there are usually changes with the flight directions as well as velocities. The speed increments tested should be sufficiently small to show these trends. The test also should be of sufficient scope to provide data for all hovering conditions within the capability or envelope of the helicopter.

During the tests the pilot usually will experience difficulty in stabilizing at the trim speed, and the pilot workload may be expected to be quite high. The fuselage and empennage usually are designed for forward flight and the airflow during other conditions may introduce very destabilizing effects. This instability introduces requirements for control inputs which in turn may affect thrust available and power required. In addition, small changes in height above the ground may cause large changes in thrust which necessitate additional collective control changes. Control or aerodynamic coupling will aggravate the situation and add to the pilot effort. An extremely unstable (with respect to performance) area usually will be encountered at the airspeed for translational lift. This speed may not be the same in all translation directions, and the pilot may not be able to stabilize the aircraft in these areas. At best, the situation becomes one of determining the equivalent static performance values from a dynamic flight condition.

The planning for this test must be accomplished carefully to insure that all the ground elements are functional, organized, and will operate effectively as a team. Communication, transportation, and coordination are the primary areas of neglect. Since the tests usually are accomplished during low wind conditions, it is important that maximum flight productivity be achieved.

The pilot effort is quite high during these tests. The aircraft instabilities usually render it impossible to maintain a fixed control condition for any appreciable time span in this flight regime. The pilot task is appreciated better when one considers that the pilot not only must observe the cockpit instruments but also must maintain a position relative to a pace vehicle. There is also a requirement to monitor constantly the height above the ground while translating the aircraft in a direction other than the line of sight—a formidable task indeed. The data should be recorded when a most representative condition is achieved. This necessarily may not be a stabilized situation. There should be no collective pitch change during the recording time period, and all control motions should be minimized.

The most difficult portion of the data reduction effort is to determine the most appropriate static values from the dynamic data. The variations in performance and pilot parameters must be analyzed to determine the most representative value. This effort is most difficult and tedious. There is generally a great deal of scatter, and one should not be surprised to find that the final result is unsatisfactory from a rigid technical standpoint. This may well be the case when the aircraft cannot be stabilized sufficiently or when the data cannot be recorded adequately and reduced.

6-2 PLANNING

The preliminary planning should be reviewed to evaluate the previously made arrangements in the real time situation. The data requirements for the program are obtained from the test plan and subsequent modifications. These then are evaluated in terms of current knowledge of the aircraft capabilities. The test sequence then should be planned for minimum time and effort requirements with respect to changing gross weights and center of gravity. This is of particular importance during the lateral translations since lateral CG locations may become critical items and will be test parameters. A table of loadings for weight and CG fuselage station should be prepared in advance to reduce any loss of time during the test. The requirements, deviations, and mission accomplishment should be checked for compatibility with the aircraft and test limits. The selected test site must be evaluated and inspected for general suitability.

Requirements must be established for the maintenance, logistic, photographic, and engineering support groups. The maintenance and instrumentation groups should be prepared to support the aircraft at the test site, particularly when it is in a remote area. Provisions should be made for fuel, auxiliary power, and any other items peculiar to the test vehicle. Ballast should be available so that weight and balance changes can be accomplished quickly. The instrumentation personnel should be provided adequate facilities for reloading cameras and oscillographs. When such facilities are not available they should be requisitioned. Sufficient supplies must be provided to support the operation. The photographic support should include motion and still photographic capability. The engineering support should include ground ambient weather environment measuring capability, necessary "on site" data reduction capability, and the personnel necessary to direct technically the pace vehicle as well as the overall ground support effort. When presenting the requirements and the general method of operation to these support groups, every effort should be made to solicit comments that would improve the proposed operation. The personnel involved must be provided the necessary transportation facilities to accomplish their assigned tasks. Arrangements must be made for radio communications, both, "air-to-ground" and "ground-to-ground". Good transportation and communication planning will add a great deal to the productivity and flexibility of the test team.

Flight cards and data recording forms now can be prepared. The weather personnel should record pressure altitude, free air temperature, wind velocity, and wind direction at time intervals of 5 min or less during the tests. This information will establish the "remote" ambient environment. The pace vehicle should record test point numbers, indicated ground speed, aircraft instrumentation numbers, direction of translation, and azimuth direction. When the data cards are completed, they should be coordinated with each group to insure completeness and accuracy. Following this coordination, a briefing should be conducted with all participating individuals attending. The general operation should be discussed with emphasis on timing and interrelated functions. The safety personnel should be briefed and, in turn, should discuss their proposed action in an emergency situation. The photographic coverage should be finalized with respect to location of cameras, type, and frequency of exposures. A map of the area should be available at this briefing and each person or group should be informed as to their relative location at the test site. Assignment of vehicles should be made and communication procedures, call signs, priorities, and assigned radio frequencies should be covered in great detail. Alternate tests should be discussed in the event the primary test cannot be accomplished.

6-3 INSTRUMENTATION

The translational performance generally is considered a static performance test. However, the unstable nature of the flight regime results in a more dynamic than static condition. Average performance values can be obtained from visual instruments for most conditions. There are usually flight areas where the data are extremely transient and the averaging may be very difficult. The necessary parameters are presented in Table 6-1.

TABLE 6-1

VISUAL INSTRUMENTATION FOR TRANSLATIONAL PERFORMANCE

Rotor speed	Pressure altitude
Engine power	Ambient temperature
Skid height	Counter number
Pacer speed (indicated ground speed)	Ballast changes
Direction of translation	Fuel counter

To quantitatively evaluate the magnitude and effect of the transients, it is desirable to use an automatic recording system. This may be either a photo panel, a magnetic tape unit, or an oscillograph. Additional parameters listed in Table 6-2 should be included on this recording system.

TABLE 6-2

AUTOMATIC RECORDING INSTRUMENTATION FOR TRANSLATIONAL PERFORMANCE

Rotor speed	Pedal
Engine power	Collective stick
Longitudinal stick	Lateral stick

The flight control and redundant parameters will aid greatly in the data correlation and analysis.

The instrumentation should include a method for precise time correlation on all the automatic recording devices.

6-4 TEST METHODS

6-4.1 GENERAL

The sideward and rearward flight tests are a simulation of the aircraft condition while hovering with tailwinds and crosswinds. This simulation is somewhat inexact since the tests normally are conducted during low wind conditions while an actual wind environment usually includes gusts. The primary objectives are to determine the engine operating characteristics, inlet performance, variation of power required, flight limits, and stability and control characteristics.

Prior to the test, the aircraft should be in the specified test configuration, at the proper gross weight and with the proper center of gravity location. It is worthwhile for the flight personnel to make a check of the loading, particularly when an adverse CG is being investigated. The test height above the ground should be established at the anticipated normal operational hover height as determined from a mission requirement or pilot opinion.

This height will vary with different aircraft, and it may be necessary to investigate more than one height in the event there is a drastic change in flying qualities, performance, or requirements from the mission profiles. Flying quality changes may be apparent qualitatively while performance variations normally will be evident only after the quantitiative data are analyzed.

The conventional airspeed systems in the helicopter are unusable during these tests. Test systems may be installed above the rotors which will give an indicated airspeed for translation in any direction. Standard differential pressure systems are inoperative at speeds below 10 kt and require a laborious and difficult position error calibration prior to use for this type of test. The most convenient method to determine the translational speed is to use a calibrated pacer vehicle. The pacer vehicle is driven down the runway at predetermined speeds and the helicopter is stabilized relative to the vehicle. When the aircraft is stabilized, the data are recorded for that condition. When sufficient space is available, i.e., a large ramp or taxi area, the flight path should be oriented parallel to the wind which will eliminate errors introduced in compensation for a wind component. For this condition, a sensitive ground station anemometer should be oriented in the direction of translation to obtain the wind velocity to be added to the pace car speed. With this technique, there is no requirement for resolving wind into components and valid data can be obtained in winds to the limit of the pilot's capability to stabilize the helicopter.

A more accurate determination of the airspeed and height can be obtained by using a Fairchild Flight Analyzer to record the flight path and time. With this technique, it is still necessary to use the pace vehicle since the pilot has no other reference with which to hold speed constant. This procedure is less productive than the pacer technique since the camera has no provision for interruption or redirection and one plate must be taken for each point. Thus the limitation of the equipment delays the test progress.

The test should start logically at the neutral center of gravity location (longitudinal and lateral), at a light gross weight, a high rotor speed, and at sea level. There are exceptions, but these are generally the optimum conditions. The tests then should proceed to the limits of the conditions stated previously. This approach provides a maximum safety factor through the build-up concept and a sound engineering basis for any modification of the planned test program. Prior to conducting any tests, the pilot should familiarize himself with the aircraft characteristics and at the same time check out the course for any overlooked or recently placed obstacles or hazards.

6-4.2 SIDEWARD FLIGHT

At the specified hover height, an initial zero speed hover point should be obtained. The aircraft then is translated sideward into the wind to the first aim airspeed and the data recorded. The data recording interval should be approximately 3 to 5 sec. The pace vehicle then is signaled to proceed to the next speed and the procedure is repeated. The helicopter should be maintained at approximately the same height (±2 ft IGE and ±5 ft OGE) above the ground for each point. The allowable height differential can be determined by the power required versus height characteristics. The precise height must be recorded if it is desired to make a correction during the data reduction and analysis procedures. Any questionable points should be repeated, and any unusual characteristics should be noted for future evaluation. The aircraft heading should not be changed during an individual test since this may introduce a performance, stability, and control bias on the parameters. The usual procedure is to translate one direction to the limits of the runway then return by translating in the opposite direction, recording data both ways. At the conclusion of a test, the hover points should be repeated to determine any changes caused by different atmospheric conditions. The aircraft weight should be controlled within ±2% by adding ballast as fuel is consumed. Generally this weight allowance will permit sufficient test time so that the ballasting can be done after a given curve has been completed.

6-4.3 REARWARD FLIGHT

The procedure for the rearward flight is essentially the same as for the sideward tests previously discussed. The hover points should be taken with head and tailwinds. Pilot visibility during this test becomes an increased problem, and the pace vehicle will have to be in front or to the side of the cockpit. Height control becomes more important and more difficult during rearward flight, particularly at nose high pitch attitudes where the tail may approach the ground.

6-5 DATA REDUCTION

The data reduction effort is dependent on the proper correlation of the data from the various recording devices and groups. The weather data should be plotted in time history form, and data points correlated. The pacer airspeeds, point numbers, and counter numbers should be correlated with the flight cards. If camera coverage or Fairchild plates were used, these similarly should be processed and correlated. Pilot comments should be clarified and assigned the proper correlation numbers.

The performance data are treated initially as any other stabilized data. The aircraft is considered to be in a level flight or hover condition. Data reduction forms and perhaps a computer program should have been developed during the planning efforts. Parameters then are read from the appropriate data card, photo film, or oscillograph roll and entered on the data reduction form or computer card. Nondimensional values of the static or average data also should be calculated. The primary objective is to determine the power required as a function of translational velocity and cross flow ratio. The engine characteristics are reduced to the standard corrected engine parameters of power, speed, fuel flow, pressure, ratios, and temperatures. The inlet characteristics are of particular interest for this flight condition. The inlet total pressure and ground station ambient pressure are used to obtain the inlet total pressure ratio. Temperature rises at the inlet may be increased significantly by any large amounts of hot gases that are mixing with the free air in the vicinity of the aircraft. The inlet total temperature and the ground station ambient temperature are used to determine the inlet total temperature rise. Inlet characteristics are discussed in detail in Chapter 13. When possible, the dynamic data should be used to determine the variation in power, engine speed, rotor speed, inlet pressure, and temperature that is occurring as a result of either pilot inputs or external disturbances generated by the aircraft and/or environment. The incremental variations are useful for determining losses that may be encountered during actual hovering operations. A typical data reduction form is presented in Table 6-3*.

6-6 DATA PRESENTATION

The data presentation for the translational performance tests should be in both plot and time history forms. The parameters of primary interest are airspeed, thrust, and power. The results are to be applied to the determination of hovering performance in wind, and the wind or translational direction is thus a necessity. The data from these tests should be correlated and compared to the steady-state hovering and thrust stand test data.

The average power required from Table 6-3 (step 20) is plotted as a function of translation speed (step 30) as shown in Fig. 6-1. The primary power and flight controls are included on the plot to aid in the correlation and analysis. Appropriate limits are included to illustrate the margins remaining. The zero airspeed value should be cross plotted with the hover data from Chapter 5. The level flight data also should be used to establish the curve fairing for the forward translation data.

*The data reduction forms (tables) are located at the end of each chapter.

During the steady-state points there will be a certain amount of variation in the primary variables, depending upon the pilot technique and helicopter stability. The best method to illustrate this is by plotting the differential values (steps 23, 48 and 51, Table 6-3) versus translational speed as shown in Fig. 6-2. These plots also should include any data obtained during actual wind conditions. For these data, the wind gust spread should be noted for analysis purposes. The values may be considerably larger for the transient data than for the steady-state results. The tail rotor performance may be presented as shown in Fig. 6-3.

Plots should be constructed to show the effects of gross weight, rotor speed, altitude, and center of gravity location. There also will be performance differences as a function of ground proximity, particularly at conditions near the performance limits. These may be individual or composite plots.

When sufficient data are available, a nondimensional plot may be constructed with the C_p, C_T and μ parameters. This plot then can be used to calculate performance for any desired conditions of weight, altitude, rotor speed, temperature, or power.

TABLE 6-3

DATA REDUCTION FORM FOR TRANSLATIONAL PERFORMANCE

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1	Point No./Flight No.				
2	Indicated pressure altitude	$H_{P_{IND}}$		ft	From ground station
3	Pressure altitude instrument correction	$\Delta H_{P_{IC}}$		ft	From ground station instrument calibration curves
4	Ground station pressure altitude	H_{P_G}	$H_{P_G} = H_{P_{IND}} + \Delta H_{P_{IC}}$	ft	(4) = (2) + (3)
5	Indicated ambient air temperature	$T_{a_{IND}}$		°C	From ground station
6	Ambient air temperature instrument correction	$\Delta T_{a_{IC}}$		°C	From ground station instrument calibration curves
7	Ambient air temperature	T_a	$T_a = T_{a_{IND}} + \Delta T_{a_{IC}}$	°C	(7) = (5) + (6)
8	Height above the ground	ΔH		ft	
9	Aircraft pressure altitude	H_p	$H_p = H_{P_G} + \Delta H$	ft	(9) = (4) + (8)
10	Average indicated rotor speed	$N_{R_{IND}}$		rpm	From manual or photo panel data
11	Instrument correction for rotor speed	$\Delta N_{R_{IC}}$		rpm	From instrument calibration at (10)
12	Average rotor speed	N_R	$N_R = N_{R_{IND}} + \Delta N_{R_{IC}}$	rpm	(12) = (10) + (11)

TABLE 6-3 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
13	Maximum indicated rotor speed	$N_{R_{IND_{MAX}}}$		rpm	From photo panel or oscillograph
14	Instrument correction for rotor speed	$\Delta N_{R_{IC}}$		rpm	From instrument calibration at (13)
15	Maximum rotor speed	$N_{R_{MAX}}$	$N_{R_{MAX}} = N_{R_{IND_{MAX}}} + \Delta N_{R_{IC}}$	rpm	(15) = (13) + (14)
16	Minimum indicated rotor speed	$N_{R_{IND_{MIN}}}$		rpm	From photo panel or manual data
17	Instrument correction for rotor speed	$\Delta N_{R_{IC}}$		rpm	From instrument calibration at (16)
18	Minimum rotor speed	$N_{R_{MIN}}$	$N_{R_{MIN}} = N_{R_{IND_{MIN}}} + \Delta N_{R_{IC}}$	rpm	(18) = (16) + (17)
19	Rotor speed variation	ΔN_R	$\Delta N_R = N_{R_{MAX}} - N_{R_{MIN}}$	rpm	(19) = (15) – (18)
20	Average power required	SHP_{AVG}		HP	From power required, Chapter 14, Table 14-1, step 43 at (12)
21	Maximum power required	SHP_{MAX}		HP	From power required, Chapter 14, Table 14-1, step 43 at (15)
22	Minimum power required	SHP_{MIN}		HP	From power required, Chapter 14, Table 14-1, step 43 at (18)

TABLE 6-3 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS	
23	Power variation	ΔSHP	$\Delta SHP = SHP_{MAX} - SHP_{MIN}$	HP		(23) = (21) − (22)
24	Indicated ground speed	$V_{G_{IND}}$		kt	From pace vehicle	
25	Instrument correction for ground speed	$\Delta V_{G_{IC}}$		kt	From calibration	
26	Instrument corrected ground speed	$V_{G_{IC}}$	$V_{G_{IC}} = V_{G_{IND}} + \Delta V_{G_{IC}}$	kt		(26) = (24) + (25)
27	Wind velocity	V_W		kt	From ground station	
28	Wind azimuth	α_W		deg from north	From ground station	
28a	Runway azimuth	α_{RW}		deg from north		
29	Wind component	V_{WC}	$V_{WC} = V_W \cos(\alpha_W - \alpha_R)$	kt		(29) = (27) · cos[(28) − (28a)]
30	True airspeed	V_T	$V_T = V_{G_{IC}} + V_{WC}$	kt	Incompressible flow is assumed	(30) = (26) + (29)
31	Fuel used	FU_c		ct	From manual or photo panel data	
32	Fuel counter constant	K_{FC}		gal/ct	From calibration	
33	Volume of fuel used	FU_{VOL}	$FU_{VOL} = FU_c \times K_{FC}$	gal		(33) = (31) · (32)

TABLE 6-3 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
34	Preflight specific fuel weight	FS_1		lb/gal	From instrument preflight
35	Postflight specific fuel weight	FS_2		lb/gal	From instrument postflight
36	Average fuel specific weight	FS_{AVG}	$FS_{AVG} = (FS_1 + FS_2)/2$	lb/gal	(36) = (36 + 34)/2
37	Weight of fuel used	FU_W	$FU_W = FS_{AVG} \times FU_{VOL}$	lb	(37) = (36) · (33)
37a	Gross weight	GW	$GW = ESGW - FU_W$		(37a) = $ESGW$ – (37)
38	Thrust coefficient constant	K_{c_T}	$K_{c_T} = A(\Omega R)^2$ at (12) $= \pi R^2 \left(\frac{2\pi}{60} N_R R\right)^2$	$\frac{\text{lb}}{\text{slug/ft}^3}$	$(38) = \pi R^2 \left[\frac{2\pi}{60} \cdot (12) \cdot R\right]^2$
39	Power coefficient constant	K_{c_p}	$K_{c_p} = 550/[A(\Omega R)^3]$ at (12)	$\frac{\text{slug/ft}^3}{\text{HP}}$	$(39) = \dfrac{550}{\pi R^2 \left(\frac{2\pi}{60} \cdot (12) \cdot R\right)^3}$
40	Density altitude	H_D		ft	At (7) and (9)
41	Air density	ρ	ρ at (40)	slug/ft^3	Calculated or from tables for standard atmosphere
42	Tip speed	ΩR	$\Omega R = \frac{2\pi}{60} N_R R$	ft/sec	$(42) = \frac{2\pi}{60} \cdot (12) \cdot R$
43	Advance ratio	μ	$\mu = V_T/(\Omega R)$		(43) = (30) / (42)

TABLE 6-3 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
44	Thrust coefficient	C_T	$C_T = GW/[\rho A(\Omega R)^2] = GW/(K_{c_T}\rho)$		$(44) = \frac{(37a)}{(41)\cdot(38)}$
45	Power coefficient	C_P	$C_P = 550\, SHP_{AVG}/[\rho A(\Omega R)^3]$ $= SHP \times K_{c_P}/\rho$		$(45) = \frac{(20)\cdot(39)}{(41)}$
46	Maximum collective stick position	$\delta_{s_{c_{MAX}}}$		in.	From photo panel or oscillograph
47	Minimum collective stick position	$\delta_{s_{c_{MIN}}}$		in.	From photo panel or oscillograph
48	Collective stick position variation	$\Delta\delta_{s_c}$	$\Delta\delta_{s_c} = \delta_{s_{c_{MAX}}} - \delta_{s_{c_{MIN}}}$	in.	$(48) = (46) - (47)$
49	Maximum directional pedal position	$\delta_{s_{r_{MAX}}}$		in.	From photo panel or oscillograph
50	Minimum directional pedal position	$\delta_{s_{r_{MIN}}}$		in.	From photo panel or oscillograph
51	Directional pedal position variation	δ_{s_r}	$\delta_{s_r} = \delta_{s_{r_{MAX}}} - \delta_{s_{r_{MIN}}}$	in.	$(51) = (49) - (50)$
52	Average collective stick position	$\bar{\delta}_{s_c}$	$\bar{\delta}_{s_c} = \delta_{s_c}/N$	in.	From photo panel or oscillograph $(52) = \Sigma\delta_{s_{c_i}}/N$ $i = 1, \ldots, 5\; N = 5$
53	Average directional pedal position	$\bar{\delta}_{s_r}$	$\bar{\delta}_{s_r} = \Sigma\delta_{s_r}/N$	in.	From photo panel or oscillograph $(53)\; \Sigma\delta_{s_{r_i}}/N$ $i = 1, \ldots, 5\; N = 5$

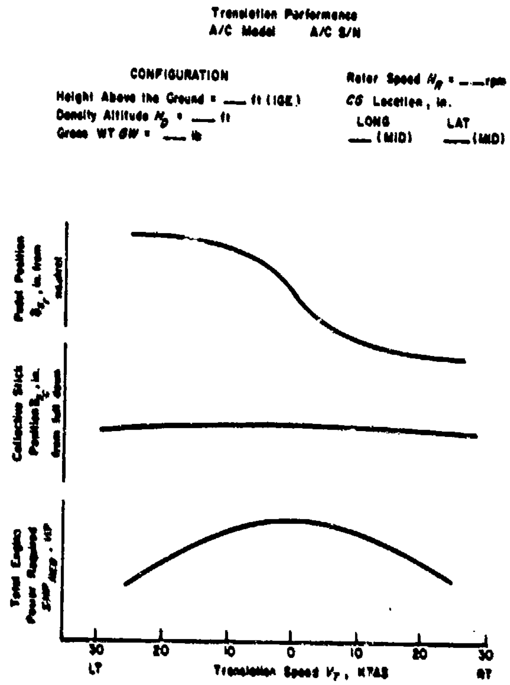


Figure 6-1. Translation Performance

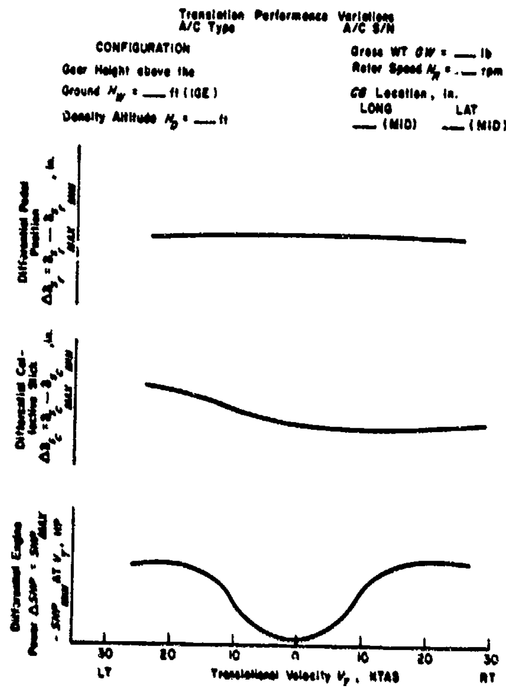


Figure 6-2. Translation Performance Variations

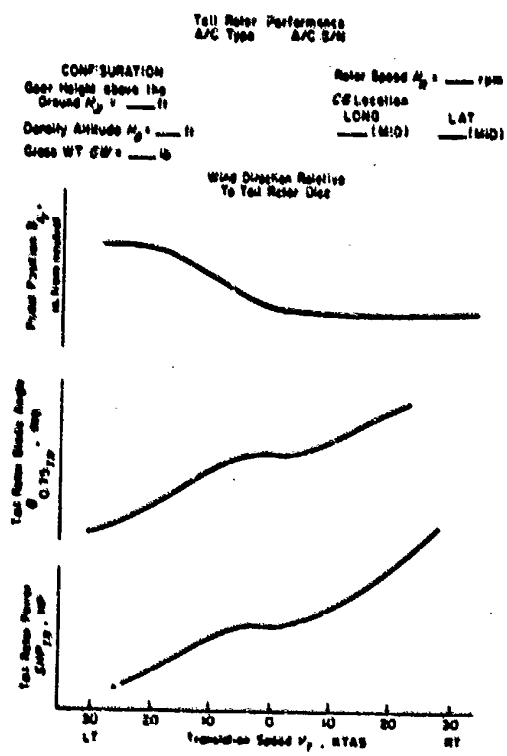


Figure 6-3. Tail Rotor Performance

CHAPTER 7

TAKEOFF PERFORMANCE

7-1 GENERAL

The takeoff profiles possible with a helicopter perhaps most clearly demonstrate the versatility and usefulness of this type vehicle. When sufficient excess thrust is available, the helicopter can, of course, climb-out vertically or at any other flight path angle desired by the pilot. This excess performance margin condition is where most helicopters normally are operated. The situation becomes more interesting and critical when the excess power margin is decreased to the point where some takeoff path other than vertical is necessary. It may surprise some to hear that relatively good takeoff performance still can be obtained when the hovering height is severely limited. This is true even for skid equipped machines when the proper technique is used, and the terrain is reasonably level and obstacle free. Such performance conditions occur when the vehicle is overloaded to the extent that it becomes a STOL aircraft or when ambient atmospheric conditions sufficiently reduce the power available.

The takeoff performance tests are conducted to determine the takeoff distance required to clear a 50-ft obstacle, the optimum takeoff airspeed, and the technique necessary to produce this maximum performance. The tests are conducted as functions of excess thrust, ambient conditions, and various operational procedures. The resulting data can be used to predict performance at various conditions and to recommend particular techniques for given takeoff situations. The discovery of any existing control limits, operational limits, or procedural restrictions generally will result from these tests.

The tests usually are conducted from the maximum gross weight at low altitudes to the maximum allowable for high altitude or power limited conditions. Field elevations should include sea level to the highest elevation available or that within the capability of the test vehicle. Rotor speed usually is limited to the maximum operational value with checks at the minimum allowable or the value for highest rotor efficiency. Since the test is primarily a maximum performance situation, engine power is the maximum available unless limited by some atmospheric or physical parameter.

The test method generally used is to accomplish a series of takeoffs at different incremental airspeeds from minimum to maximum while using a different excess thrust level for each series. The excess thrust is varied by choosing different field elevations, ambient atmospheric conditions, gross weights, or power settings. Ground effect at high elevations cannot be simulated at sea level, thus remote sites are required. Investigations are made to determine the optimum pilot technique for the test vehicle. The influence of any variation in the technique relative to decreased performance is tested when the magnitude appears significant.

The data reduction effort for the takeoff performance is reasonably small when the nondimensional method is used. However, a series of excess thrust coefficients is necessary since single curves cannot be corrected individually to standard conditions. The major efforts required are to correlate the data from the various sources and to read the plates from the Fairchild Flight Analyzer. When this processing is accomplished, the data are in engineering units and the calculations are relatively easy.

7-2 PLANNING

The takeoff tests must be planned very carefully since there is more than one support group involved and the weather criteria are such that flight productivity is of the essence. Prior to actively initiating any flight planning, the test plan should be reviewed carefully to determine what test objectives are to be achieved and what conditions are needed. Most helicopter takeoff performance tests require data at remote sites which imposes additional technical and support problems. The availability of the test site and the necessary support equipment should have been determined during the preliminary planning. The confirmation of previous plans and scheduling may be the only requirement at this point in the program.

There are several detailed problems which are magnified by the operational aspects of the tests. The data recording teams may be widely dispersed in the test area which creates both communication and transportation difficulties. The aircraft must be ballasted periodically for weight control; accordingly, this group must be mobile since it is undesirable to have the aircraft unduly dependent upon ground support.

Before considering the test in detail, the site must be inspected to insure suitability with respect to runway length, camera offset distance, runway surface, obstacles, and proximity of dwellings or personnel. There must be access available for rescue and fire suppression equipment. When possible, the flight pattern should be arranged to obtain the most advantage from any emergency landing areas. The takeoff site must be surveyed and clearly marked. A weather survey should consider any available historical data to determine the predominant wind velocities and patterns. This information can be used to determine the most desirable takeoff direction and advisable test scheduling.

The support groups necessary for the takeoff tests are photographic, maintenance, engineering, fire suppression, and rescue. Each of these groups should be provided with the appropriate test schedule, data cards, flight cards, and operating procedures.

The photographic coverage usually consists of still and motion picture cameras in addition to the Fairchild Flight Analyzer. The photographers normally do not require a flight card and the Fairchild operator usually will have a standard operating procedure for recording the necessary information. This procedure should be reviewed to insure that all of the required engineering data will be recorded. The anticipated limits of the test flight path must be given the Fairchild operator to enable calculation of the proper camera offset distance. The operator should be given the aiming point on the aircraft during tracking. The main rotor or the centerline of the aircraft normally is used as the aim point. When a good reference is not available, one should be painted on the aircraft.

Appropriate preliminary flight cards or instructions should be prepared and discussed with the individual groups. Their comments, in turn, should be incorporated into the final flight card. A briefing then should be given for all of the participating personnel. Particular emphasis should be placed on interrelationships among the groups. Many alternates or deviations from the test plan such as weight changes, aborted takeoffs, technique variations, or camera difficulties may arise which should be discussed and alternate plans established.

The data card for the flight crew should contain the items indicated in Table 7-1.

TABLE 7-1

DATA CARD FOR FLIGHT CREW

Point number	Airspeed (standard)
Plate number	Rotor speed
Free air temperature	Rotor speed bleed
Altitude	Engine speed
Airspeed (boom)	Engine torque
Direction of takeoff	Fuel used
Ballast aboard	Technique
Hover height	Distance (theodilite)

The weather station data card should include the items listed in Table 7-2.

TABLE 7-2

DATA CARD FOR WEATHER STATION

Point number	Wind velocity
Plate number	Wind direction
Time of day	Ambient temperature
Pressure altitude	Wind gust spread

The card should also contain instructions for transmittal of weather data to the theodolite or aircraft and to specify any predetermined wind limits in either direction or velocity. The Fairchild station should record point number and plate number. The theodolite station should record the variables contained in Table 7-3.

TABLE 7-3

DATA CARD FOR THEODOLITE STATION

Point number	Technique
Plate number	Ballast aboard
Boom airspeed	Hover height
Distance	Direction of takeoff

A general map of the theodolite, weather station, and camera stations should be included on the theodolite data card. Spatial orientations of any other test equipment are also pertinent items.

The engineering preparation includes determination of the proper aircraft gross weight and power setting to give the desired thrust and power coefficients.

The necessary in-flight plot forms must be prepared for the flight observers and the theodolite station. The ballast required and the necessary increments to be added should be established when possible.

7-3 INSTRUMENTATION

The takeoff performance tests can be conducted with only visual test instrumentation installed in the aircraft. The flight conditions, though dynamic, can be defined adequately by recording events at critical points such as initial start, lift-off, and during climb-out. Maintaining a uniform pilot technique will insure that the dynamic variations on successive test points will be similar and will be inherently consistent during the complete test. The visual instrumentation required in the aircraft is shown in Table 7-4.

TABLE 7-4

VISUAL INSTRUMENTATION FOR TAKEOFF PERFORMANCE

Airspeed (boom)	Engine speed
Airspeed (standard)	Rotor speed
Altimeter	Ambient air temperature
Fuel used	Engine torque

The instrumentation shown in Table 7-4 is required to conduct, monitor, and validate the test progress. It is desirable to supplement this basic instrumentation with an automatic recording system. This will allow continuous data recording and provide for additional parameters. It will also allow a much higher data density both during steady state as well as during the dynamic portions of the takeoff. The parameters shown in Table 7-5 should be recorded on the automatic system.

TABLE 7-5

AUTOMATIC DATA RECORDING FOR TAKE-OFF PERFORMANCE

Engine torque	Inlet pressure
Engine speed	Longitudinal stick position
Rotor speed	Lateral stick position
CG normal acceleration	Pedal position
Altitude	Collective stick position
Inlet air temperature	Airspeed
Event marker	

The control positions are useful to both correlate performance and to monitor the pilot technique used. Many times these parameters will serve also to resolve discrepancies and variations in the performance results.

In addition to the instrumentation installed on the aircraft, a considerable amount of ground equipment is required to support the test. Fig. 3-2 is a diagram of a typical test area for takeoff performance. The diagram shows the relative location and identity of the ground support equipment.

Camera coverage usually consists of both still and motion picture coverage of each flight. The Fairchild Flight Analyzer described in par. 3-6 is used to record photographically takeoff distance and height above the ground as a function of time.

The visual theodolite also is used to record takeoff distance and height without time correlation. These data are of low accuracy and are used marginally to determine when a height of 50 ft above ground is reached during takeoff. This instrument is described in par. 3-5.

The parameters listed in Table 7-6 should be recorded at the weather station described in par. 3-4.

TABLE 7-6

WEATHER STATION INSTRUMENTATION FOR TAKEOFF PERFORMANCE

Pressure altitude	Wind velocity
Wind direction	Ambient temperature

7-4 TEST METHODS

7-4.1 GENERAL

There are several techniques and variations commonly used to determine the takeoff performance. The maximum performance takeoff normally is used when a vertical climb-out is not possible. Inherent with the general takeoff definition is the requirement to clear a 50-ft obstacle. Thus, the takeoff tests are simulating an overload gross weight and/or a low excess power condition in operational terrain where there is an obstacle. The condition of the terrain before reaching the obstacle will determine to some extent what techniques are usable. For a level runway type area, any technique may be used including a ground run for wheeled vehicles or a sliding procedure for skid gear machines. When there are no large obstacles present but a ground run is impossible (for any reason), an air acceleration technique must be accomplished. The air acceleration technique may be varied with respect to flight path.

The test procedures are generally the same for all techniques used. Prior to initiating each takeoff, all data stations should report a ready status. The aircraft should then be ballasted properly and the hovering height established. It is usually sufficient to estimate the height from the aircraft. A more accurate method is to measure the height with a tape measure. Each takeoff for a given curve is made at the same hover height and the same differential power coefficient.

After the hover check, the aircraft is positioned for the takeoff. The pilot gives a 3- to 5-sec countdown to synchronize the data stations. The theodolite operator notes the aircraft position during countdown and at zero count the takeoff is initiated. The aircraft is accelerated at maximum power available along the desired flight path, rotated, and the climb-out is accomplished. As the helicopter passes through the 50-ft altitude, the theodolite distance is noted and a radio transmission is made to the aircraft that the obstacle height has been reached. All data should have been recorded prior to this time. During the return to the takeoff position, the recorded takeoff airspeed is transmitted to the theodolite crew which in turn responds with the takeoff distance required. Prior to landing, the flight crew informs the ballast personnel of the amount of ballast to be prepared to load, thus minimizing the delay. This procedure is repeated for each takeoff until the necessary data have been accumulated for the given conditions. During the test, radio transmissions should be held to a minimum to avoid interrupting the testing operation or perhaps introducing flight safety factors.

7-4.2 ROLLING TAKEOFF TECHNIQUE

A rolling takeoff technique can be used for helicopters equipped with wheel landing gear.

Prior to each takeoff the ability to hover should be checked to insure that the proper differential power coefficient exists. Following the hover check the aircraft is located in the takeoff position, maximum braking applied, and maximum power added. The proper location of the flight controls prior to takeoff will vary with aircraft and configurations. Generally, forward longitudinal stick is applied to position the thrust vector aft, and collective is used to make the helicopter "light on the wheels". In some instances, the brakes will not overcome the full power thrust and power then must be limited to the braking capability. The countdown then is started and at zero count, the brakes are released and full power and/or thrust is applied. This power condition is maintained during the ground roll acceleration. The standard airspeed system usually is not operative below 20 kt indicated, and will have considerable lag throughout. For low rotation airspeeds, the test system may provide the only speed indication. The capabilities of the standard system should be noted carefully since in the final analysis this may impose a performance limitation. The test speeds should not be terminated or limited by the standard airspeed system since this may be improved subsequently and render all data useful. Rotation must be effected prior to reaching the aim airspeed value to prevent overshooting. Normally, the aim airspeed must be anticipated by approximately 3 to 5 kt. When properly accomplished, the aircraft then will become airborne at the aim airspeed, which is maintained constant during the climb-out over the 50-ft obstacle. The technique is very critical during the acceleration, lift-off, and start of the climb. Caution must be exercised to prevent bleed off of airspeed which will introduce an acceleration factor and may influence significantly the takeoff distance required. When the aim airspeed is missed, the pilot should accept the new value and apply the constant speed technique rather than try to adjust to the original aim speed. The time interval from lift-off to 50-ft is usually very small, and there is not sufficient time to restabilize at a new airspeed. In addition, any large control inputs may disturb the rotor or ground cushion with an attendant loss in lift at a critical point during the takeoff. As the collective is increased to accelerate or to hold airspeed constant, there is usually a tendency for the rotor speed to drop and it is necessary to increase the governor setting (beep) to maintain rpm. Rotor speed bleed significantly increases performance, and a small bleed will be evident in the test data. For some helicopters there may be large and unusual cyclic control requirements during the takeoff. Each maneuver should be accomplished as smoothly as possible with a minimum of adjustments during the climb-out. The excess performance margin must be used very carefully to establish the climb-out path and provide a rate of ascent. Any improper control motions or technique variations at this critical time will reduce greatly the takeoff performance. During the climb-out, the ground effect is decreasing, acceleration factors are not present, and the helicopter is in a stabilized climb.

The takeoff tests are conducted for various airspeeds to determine the value which yields the minimum distance required to clear the obstacle. The difficulty in flying a given condition is usually directly proportional to the resulting performance. By this it is implied that good performance, short distance, and large excess power conditions are not so difficult as those where the performance is marginal. The various speeds and flight paths to be investigated are shown in Fig. 7-1. Although at the very high speeds the aircraft will climb rapidly after becoming airborne, a longer runway distance is required to reach the lift-off airspeed. The pilot task for this takeoff condition should be minimal. As the airspeed is decreased, the technique becomes

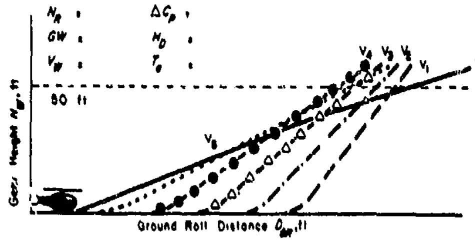


Figure 7-1. Rolling Takeoff Performance

increasingly critical and the total distance to clear the obstacle becomes less. Below some minimum airspeed, the total distance required will increase rapidly and a condition quickly will be reached where a climb-out cannot be accomplished at lower airspeeds. In this area, the helicopter initially will climb after the rotation and then, as the influence of ground effect decreases, the climb-out will terminate in level flight or an ensuing descent may develop. Extreme caution should be used when entering these areas since the aircraft is in a full power condition and little can be done in the event a high rate of descent condition develops. Increased collective at this point merely will decrease the rotor speed and may induce a stall.

7-4.3 LEVEL ACCELERATION FROM HOVER TECHNIQUE

For helicopters not equipped with a rolling takeoff capability, a similar profile may be accomplished by an air acceleration close to the ground. The acceleration should be as close to the ground as feasible in order to utilize fully the maximum ground effect. The aircraft attitude, flying qualities, and configuration are factors in determining the minimum acceleration height. A hover check should be accomplished to establish the power and the excess thrust available. The helicopter then is stabilized at the desired hovering height. A countdown is accomplished and the acceleration is initiated by applying forward stick and increased collective. For conditions where maximum power was not required at the initial hover height, an excess of power is available for the acceleration. In addition, the forward velocity usually puts the aircraft in a lower power required area and at some speed translational lift is reached. At this point the helicopter has a strong tendency to climb and it may be difficult to maintain the low acceleration height. The attitude may be extremely nose down and uncomfortable during the acceleration at high excess power conditions. This attitude, of course, tilts the rotor thrust vector forward and increases the acceleration. As was the case with the rolling technique, the airspeed lag and the time differential required for rotation must be anticipated. This time differential will vary with airspeed and excess thrust available. For the higher excess thrust conditions, the acceleration is higher, the lag is greater, and the rotation may require larger stick inputs. The characteristics for each condition must be evaluated to determine the proper airspeed differential to accomplish this. Following rotation, the climb-out is similar to that previously discussed for the rolling technique. The flight profiles resulting from this air acceleration technique are shown in Fig. 7-2. The minimum airspeed takeoff is at the point where translational lift occurs. Rotation below this speed will invariably result in the aircraft climbing slightly then descending and striking the ground. For all the low excess power points, extreme caution must be used to avoid large or rapid control inputs which can influence the ground cushion and adversely influence the takeoff performance. A successful takeoff can be accomplished, terrain permitting, when the available hovering height is zero. The helicopter can be slid along the ground until translation lift is reached and then a climb-out can be made. Needless to say, there is some excess thrust level where even this technique will not permit a successful takeoff.

Since the ground effect is a significant factor in the takeoff performance, it often is necessary to conduct tests at some acceleration height other than the minimum. This normally is done for helicopters that often carry large external sling loads below the

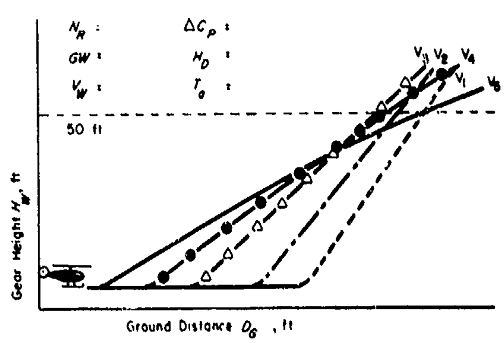


Figure 7-2. Level Acceleration From a Hover Technique

helicopter. Each helicopter should be evaluated to determine the sizes and shapes of the operational loads within the mission spectrum. From this, a determination can be made as to the necessary gear height to be tested, the length of the sling, and the nature of the load to be carried. The takeoff airspeeds are relatively low and the flat plate drag of the load is not significant at the low speeds attained during takeoff. The weight of the load and its distance from the aircraft CG location may introduce significant control moments during the takeoff. These moments may detract significantly from the flying qualities of the aircraft and cause control requirements that decrease the performance. The technique at an elevated hover height is essentially the same as that described for the minimum height acceleration. Care must be used to prevent a loss of altitude which may allow the sling load to contact the ground.

7-4.4 SIMULTANEOUS CLIMB AND HOVER TECHNIQUE

A third technique that may be used is a simultaneous climb and accelerate technique. This technique is not useful where the available hover height is less than approximately 10 ft. Typical flight profiles for this technique are shown in Fig. 7-3. This technique is perhaps the most difficult for the pilot to accomplish precisely and consistently. It is very difficult to maintain a straight line flight path throughout the climb-out. The takeoff is initiated from a low hover height or a "light on skids" condition. The pilot fixes his eye on

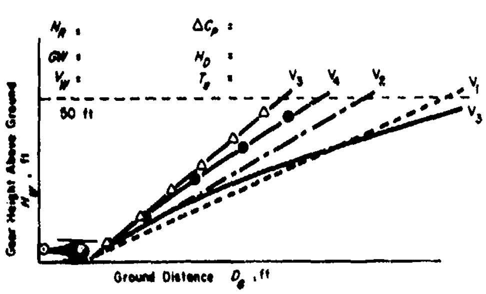


Figure 7-3. Simultaneous Climb and Accelerate Technique

some target object on the horizon to establish the flight path. A countdown is given, then longitudinal and collective controls are applied to initiate the takeoff. Collective is applied as rapidly as possible to maintain flight along the desired flight path. The collective application must not be so rapid as to cause a significant rotor speed decrease. The flight path angle and acceleration rate must be such that translational lift is reached prior to reaching the maximum hover height limit.

At high speeds, the flight path is very shallow with translational lift being attained long before the maximum hover height is reached and there is little likelihood that a descent may be encountered. Translational lift introduces a significant performance increase at which the helicopter will tend to climb vertically from the flight path. Forward stick will be required to maintain the flight profile.

At lower speeds the climb angle becomes steeper and the distance required to make the takeoff over the obstacle will decrease. The maximum performance is obtained where translational lift is reached at the height and time where power required is equal to the power available. The increased performance at translational lift then will allow the climb-out or an acceleration to be continued. At any lower speed the aircraft will climb to a height where excess thrust is not available to continue the climb-out or to accelerate to a condition where a climb-out can be accomplished. There is always a danger that the aircraft may descend and strike the ground during this type of maneuver.

7-4.5 ROTOR SPEED BLEED TECHNIQUE

A variation that may be used with any of the previously discussed techniques is that of rotor speed bleed. This technique allows the kinetic energy of the rotor to be used during critical portions of the takeoff. Although this method is normally very effective, the results will vary with aircraft depending upon the rotor inertia and aerodynamic characteristics.

If negligible power loss to the rotor as a function of rotor speed and nominal lift loss with increased blade angle of attack (increased collective) are assumed then the total rotor energy is available to augment the takeoff. The acceleration to translational lift requires the greatest portion of the takeoff distance at low airspeed, low excess thrust conditions. It is in this area that the rotor rpm bleed can be used most effectively. To provide the most reserve kinetic energy and the greatest performance benefit, the rotor speed should be at the maximum allowable at the start of the takeoff. Generally, rotor speed should be bled, while applying maximum beep, at a rate that will provide a minimum rotor speed value as translational lift is reached. On some aircraft and configurations, bleeding to the minimum rotor speeds can result in large rotor lift losses. Prior to testing with this technique, various rates of bleed and minimum rotor speed values should be investigated to determine the optimum performance and safest procedure.

As translational lift is reached, the collective may be lowered slightly to allow the rotor speed to increase as much as possible. At this point, the rotation may be accomplished and the climb-out started. The collective can now be increased and bleed again used to augment the engine power available during the climb-out. The bleed rate should be such that minimum rotor speed occurs after reaching the 50-ft obstacle. Again, it may be necessary to determine the optimum bleed rate and minimum rotor speed values. The rate at which the rotor rpm bleeds off and the rate at which it can be regained will strongly influence the technique.

While the rotor speed bleed technique will result in significant performance gains with most helicopters, this is not the case for all machines nor perhaps for all conditions on a given machine. An example of the former would be a helicopter with low rotor inertia where the energy gained from using the kinetic energy is less than the loss of lift and lower rotor speeds. The collective control position at the start of a takeoff in the latter case may not indicate an impending performance change at the gross weight and/or altitude. At some point the increased collective required to bleed rotor speed can introduce stall with hazardous results. The safety aspects of this performance technique cannot be over-emphasized since the performance changes are insidious and may not be readily apparent to the pilot.

7-4.6 DATA RECORDING

The data recording procedures for the flight crew, the weather personnel, and the theodolite group now will be discussed in detail. During each of the hover height checks, the flight engineer should record the gear height, atmosphere, and power required parameters. The theodolite station also should note the gear height. During the countdown the fuel counter reading should be recorded in the cockpit and the weather station should record pressure altitude, wind velocity, and direction. As the acceleration is being accomplished, the maximum power and rotor speed values reached should be recorded. At rotation, a data event marker should be used to note the point on any automatic data recording equipment. The pilot should note the rotation speed for recording. Any rotor speed bleed used during the rotation should be noted. The maximum power and rotor speed used during the climb-out should be recorded. As the takeoff is being accomplished, the weather conditions should be monitored to record any changes from the initial readings. At the obstacle height the theodolite should signal the aircraft, and the pilot should note airspeed. On the assumption that there is a clearly positive rate of climb, the Fairchild camera can now swing through and prepare for the next plate. When there is any indication that the aircraft may descend through the obstacle height, the camera should continue tracking.

During the return to the takeoff area, the aircraft should transmit the airspeed and the theodolite station should reply with the takeoff distance required. The airspeed versus distance values should be plotted both in the aircraft and at the theodolite station. This plot then is used by the flight crew to determine the next aim airspeed. The flight engineer should calculate the fuel used and establish the ballast needed for the next takeoff. This information then should be transmitted to the ballast crew to minimize the ground time necessary for the ballast operation. The theodolite and camera stations should call when ready for the next takeoff. After the ballasting is completed, the next condition can be accomplished using the previously discussed methods.

7-5 DATA REDUCTION

The data reduction procedures for the takeoff performance utilize manually recorded data from the aircraft and ground stations, photo panel data, oscillograph data, and plates from the Fairchild camera. Prior to any data reduction effort, the data cards must be reviewed for clarity, completeness, and correlation. The Fairchild plate number is the single most important correlating parameter. This is followed in importance by point number, time of day, and ballast values. The manually recorded data are entered on the data reduction forms or punched on cards for insertion into an automatic data processing (ADP) system. The flight crew data cards should be compared with the theodolite data to insure that no erroneous entries were recorded. The wind velocity, direction, and ambient temperature should be plotted as a function of time of day. The photo panel film should be developed and scanned for correlation with the flight card data. As a rule, a single average value for each parameter will suffice for the data reduction.

Since the whole takeoff maneuver is a dynamic situation, certain stipulations and rules must be applied to obtain consistent, meaningful data with an appropriate degree of accuracy. The power used in the data reduction is the maximum obtained during the takeoff or that power which occurred at some reference point such as during the acceleration, rotation, climb-out, or at the obstacle clearance point. The airspeeds of primary interest are at rotation and at the obstacle height. The test gross weight can be defined arbitrarily as that at the start of the takeoff, at the obstacle height, or an average of the weight during the takeoff maneuver. Any of these approaches will provide equal validity provided consistency is observed. The photo panel data should be used only for backup in the event some item was not hand recorded or when a high density time history is necessary. This procedure will reduce considerably the time interval between the end of a flight and the start of data reduction, and will reduce the total data reduction effort required. This also will simplify the remote site operation where film processing equipment may be limited or nonexistent. Since the photo panel data are not used normally, it may be desirable to return the film to the permanent installation where it can be processed in a more timely and economical manner.

The oscillograph data must be processed and correlated prior to any data reading. Event markers on the roll should be identified and correlated with the flight cards, and the counter numbers should be equated to the Fairchild plate numbers. The data should be reviewed for any unusual aircraft motions or control requirements, and the control margins should be determined. The control positions of interest are read and recorded in the appropriate data reduction forms. A time history of the control positions, airspeed, and aircraft motions can be constructed to illustrate and evaluate the pilot technique being used.

The Fairchild camera plates must be developed and printed as shown in Fig. 7-19. These initial prints then are evaluated to determine the need for enlargements. Proper offset camera distances will provide sufficient readability and accuracy without enlargements although both can be enhanced by using enlargements of the contact print. Prior to reading the plate, a graphical scale is constructed. The runway distance represented by the two targets is calculated as shown in pars. 3-5 and 3-6. A Gerber (variable) scale is used to divide equally the distance between targets which is then used to construct the scale. On the Fairchild print, the vertical distance is in the same proportion as the horizontal distance and a graphical scale thus can be constructed for the vertical height. A typical scale incorporating both vertical and horizontal di:tances is illustrated in Fig. 7-4.

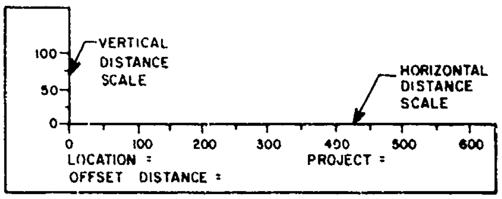


Figure 7-4. Graphical Distance Scale for Fairchild Plates

This scale is used to locate and mark the frame where the obstacle height is reached. Normally, five frames on either side of the obstacle height frame and the takeoff start frame also are marked at this time. For a mathematical solution, the distance between the targets on the plate in inches and the runway distance in feet also can be used to establish a K-factor in units of feet/inch. The distances on the plate then can be scaled and multiplied by the K-factor to yield hori ,ital and vertical distances for any particular frame. The type of solution desired will depend upon the data reduction procedures

...off area is not level and the distance data must be corrected for runway gradient. A typical situation is shown in Fig. 7-5.

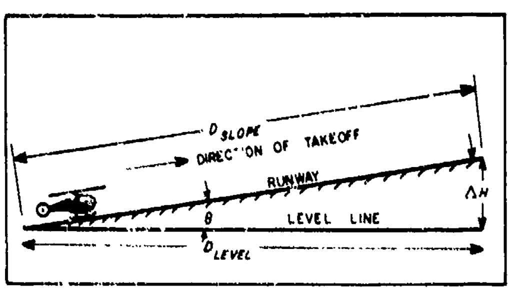


Figure 7-5. Runway Takeoff Gradient

For a ground roll takeoff, the distance correction based on energy and geometric considerations is given by

$$D_{LEVEL} = \frac{D_{SLOPE}}{1 + 2g\, D_{SLOPE} \sin \theta / V_{TO}^2} \quad (7\text{-}1)$$

where

D_{SLOPE} = distance along runway to takeoff, ft
D_{LEVEL} = horizontal projection of D_{SLOPE}, ft
θ = runway gradient, deg
g = acceleration due to gravity, ft/sec^2
V_{TO} = takeoff speed, ft/sec

For most helicopter takeoff tests the correction is numerically small since the gradient is not large, the ground distance is relatively short, and the speed is low. However, it may be a significant percentage of the total takeoff distance or ground roll distance. The gradient easily is obtained from the Fairchild plate since the camera is maintained in a level position during the test and the fiducial marks represent a level line. A line is drawn parallel to the fiducial marks with the origin at the takeoff start point. Another line then is drawn through the path transcribed by the gear. The angle between these two lines represents the runway gradient. The most accurate method to determine the effect of the runway gradient is by experimentally conducting a high and a low speed takeoff in both directions under similar performance and environmental conditions. This is most effective in validating theoretical correction factors.

Reading the Fairchild plate consists of reading the elapsed time, and the horizontal and vertical distances for the previously marked frames. This is accomplished best by using a short data reduction sheet for each plate. This form combines the necessary data reduction and the plotting on the same form. When reading the Fairchild plates, it must be remembered that the internal camera timer is progressing continuously and does not give a zero time directly. However, it is known from which frame the takeoff was initiated, and the time at this point is treated as a tare. Subtracting the tare from the recorded frame times yields the elapsed time from the start of takeoff. The start time is read and recorded. The time recorded by the Fairchild camera is in seconds, tenths, and hundredths of a second. The time must be accumulated manually across the plate since each time the seconds counter passes zero, 10 sec must be added to the subsequent time value recorded for each frame. The time for the previously marked frames is recorded. Vertical and horizontal distances are now read directly from the print by use of the graphical scale illustrated in Fig. 7-4. The horizontal distance must be read for the same aircraft reference point in each frame. The camera operator tracked the aim point of the aircraft and this appears in each frame, usually near the center. The time, height, and distance values are entered first on the data reduction form and then plotted on the form as illustrated in Fig. 7-6.

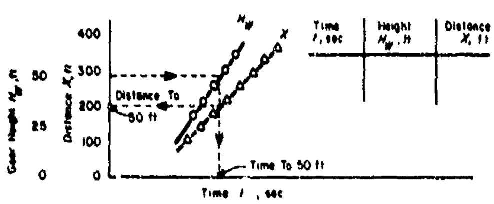


Figure 7-6. Takeoff Profile

The wind velocity and direction relative to the helicopter are obtained from the weather station data and entered on the form. A headwind velocity component V_{HW} is now calculated.

$$V_{HW} = V_W \times \cos(\alpha_{RW} - \alpha_W) \qquad (7\text{-}2)$$

where

V_{HW} = velocity of headwind component, kt

V_W = veloctiy of wind, kt

α_{RW} = runway azimuth, deg from north

α_W = wind azimuth, deg from north

Lines are now faired through the plotted points as shown in Fig. 7-6, and slopes are taken to give the vertical and horizontal speed, i.e., dh/dt and dx/dt. The value of the horizontal speed, now is added to the headwind component to obtain the true airspeed along the direction of takeoff. True airspeed along the flight path is the resultant of the horizontal, wind, and vertical speeds as illustrated in Fig. 7-7.

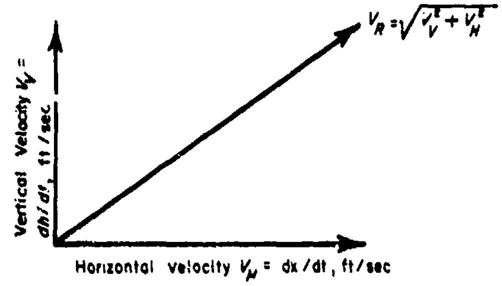


Figure 7-7. Velocity Relationships During Takeoff

The resultant of course, can, be solved mathematically and is done best in this manner when automatic data processing is being used. When a manual data reduction method is being employed, a graphical method is easier, more expedient, and introduces little sacrifice in accuracy. This graphical method is accomplished by utilizing a device such as illustrated in Fig. 7-8.

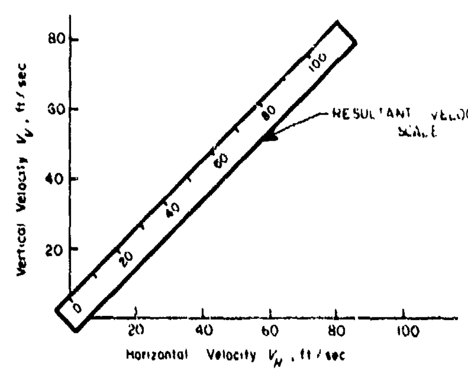


Figure 7-8. Device for Graphical Determination of True Airspeed

The true airspeed and distance are then entered on the general data reduction form shown in Table 7-7*.

The individual takeoff curves purposely are conducted at widely varying weight, power, and atmospheric conditions. This is necessary to produce data at the desired excess power ΔC_p coefficients i.e.,

$$\Delta C_P = C_{P_A} - C_{P_{REQ\ to\ H}} \tag{7-3}$$

These curves usually cannot be compared directly. A quasi-nondimensional method has been used in the data reduction form, and this method will now be discussed in considerable detail.

The data from steps 39 and 50, are plotted as shown in Fig. 7-9.

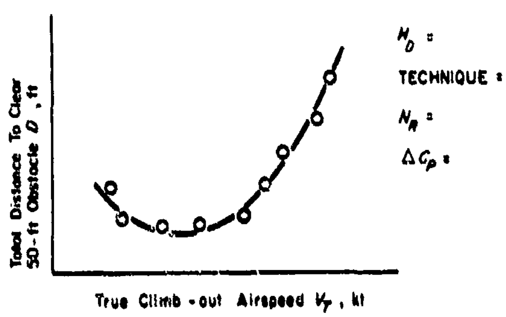


Figure 7-9. Takeoff Performance

There will be a similar group of data points for each takeoff test that was conducted, and a curve is faired through the points illustrated. These faired curves then are entered at equally spaced airspeed values to obtain the most representative distance for the particular excess power coefficient. A three-dimensional carpet plot now is constructed with distance, airspeed, and ΔC_P values derived from the individual plots. A plot similar to that shown in Fig. 7-10 is the result.

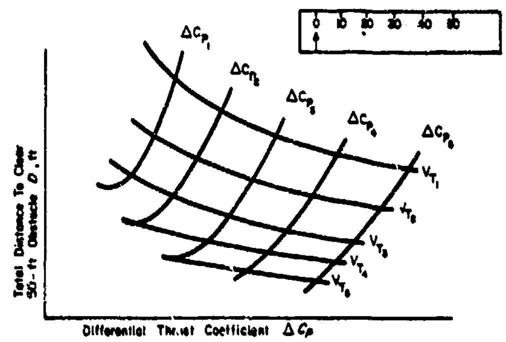


Figure 7-10. Nondimensional Takeoff Performance

A sliding scale such as shown in Fig. 7-10 is used to locate the curves in the proper relative positions. The reference is placed on the ΔC_P value, then the airspeed and distance values are plotted. The lines for each ΔC_P then are faired as illustrated. The constant airspeed lines now can be drawn through the appropriate points on each ΔC_P curve. The initial fairings may need adjusting to accomplish the latter operation. When the curves are drawn, the fairings then are transferred back to the initial plot of data points for comparison. The final result should be a smooth carpet plot with fairings that also agree with the curve faired through the test points. From this plot, with the power available and the hovering power required data known, it is possible to calculate the takeoff performance for any gross weight, atmospheric, and power conditions.

A most difficult portion of the curve to define is the bottom or minimum distance required area. Since this is the maximum performance, it must be determined precisely. The back side points are usually a minimum number and the curve may be very steep in this area, both of which complicate definition of the curve. A useful technique that may help to define the minimum distance required

*The data reduction forms (tables) are located at the end of each chapter.

is to plot the minimum distances and airspeeds as shown in Fig. 7-11.

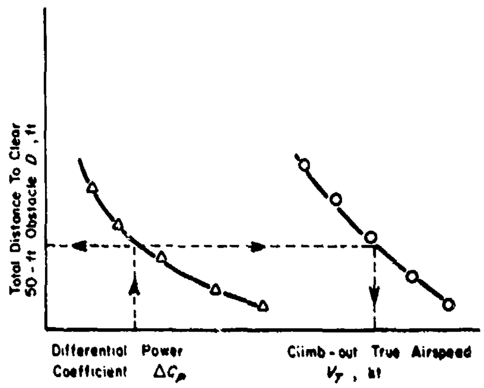


Figure 7-11. Airspeed and Distance for Maximum Takeoff Performance

The data for Fig. 7-11 may be obtained from either the test plots or the carpet plot. The values should result in a smooth curve that can then be used in finalizing the other original data and carpet plot fairings. Given a ΔC_p value, the plots may be used to calculate quickly the maximum performance which is available for any condition as shown in Fig. 7-15.

The previous methods are used to develop a carpet plot for each technique used during the tests. Data from various techniques normally require independent plots.

7-6 DATA PRESENTATION

The discussion presented in Chapter 5 should be consulted for general guidelines. A typical presentation for the individual takeoff curves is shown in Fig. 7-12. The fairing derived from the carpet plot is shown in addition to the curve through the test data. Generally, these lines will coincide. For a more graphic illustration, they are shown separated in this case.

The carpet plot is often strange to many people and often is misunderstood. Every effort must be made to present a clear concise and complete plot. An example is illustrated in Fig. 7-17.

Fig. 7-13 is a plot of the distance required to accelerate to the true climb-out airspeed (step 42, Table 7-7). This distance includes ground roll if applicable.

The air distance required to clear a 50-ft obstacle is plotted in Fig. 7-14. Ground roll, if any, is not included.

Fig. 7-16 illustrates the effect of increased pressure altitude on the distance required to clear a 50-ft obstacle.

Fig. 7-18 illustrates typical control positions during takeoff.

TABLE 7-7

DATA REDUCTION FORM FOR TAKEOFF PERFORMANCE

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1	Flight No./Point No.				
2	Indicated pressure altitude	$H_{P_{IND}}$		ft	From weather station
3	Altimeter instrument correction	$\Delta H_{P_{IC}}$		ft	From instrument calibration
4	Pressure altitude	H_P	$H_P = H_{P_{IND}} + \Delta H_{P_{IC}}$	ft	(4) = (2) + (3)
5	Time of day	t		hr	From ground station
6	Indicated ambient air temperature	$T_{a_{IND}}$		°C	From weather station
7	Instrument correction for ambient temperature	$\Delta T_{a_{IC}}$		°C	From weather station instrument calibration
8	Ambient air temperature	T_a	$T_a = T_{a_{IND}} + \Delta T_{a_{IC}}$	°C	(8) = (6) + (7)
9	Indicated rotor speed	$N_{R_{IND}}$		rpm	From manual or photo panel recording
10	Instrument correction for rotor speed	$\Delta N_{R_{IC}}$		rpm	From instrument calibration
11	Rotor speed	N_R	$N_R = N_{R_{IND}} + \Delta N_{R_{IC}}$	rpm	(11) = (9) + (10)

TABLE 7-7 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
12	Density altitude	H_D		ft	Calculated or from tables at (8) and (4)
13	Air density	ρ		slug/ft^3	Calculated or from tables at (12)
14	Air density ratio	σ	$\sigma = \rho/\rho_o$		Calculated or from tables at (12) (14) = (13) /0.0023769
15	Weight of ballast	W_B		lb	
16	Engine start gross weight	$ESGW$	$ESGW = ESGW_o + W_B$	lb	(16) = $ESGW_o + W_B$
17	Fuel counter difference	ΔFC	$\Delta FC = FC_2 - FC_1$	ct	From photo panel or manually recorded data
18	Fuel counter constant	K_{FC}		gal/ct	From instrument calibration
19	Preflight fuel specific weight	FS_1		lb/gal	From instrument preflight
20	Postflight fuel specific weight	FS_2		lb/gal	From instrument postflight
21	Average fuel specific weight	FS_{AVG}	$FS_{AVG} = (FS_1 + FS_2)/2$	lb/gal	(21) = [(19) + (20)] /2
22	Volume of fuel used	FU_{VOL}	$FU_{VOL} = \Delta FC \times K_{FC}$	gal	(22) = (17)·(18)
23	Weight of fuel used	FU_W	$FU_W = FU_{VOL} \times FS_{AVG}$	lb	(23) = (22)·(21)

TABLE 7-7 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
24	Gross weight	GW	$GW = ESGW - FU_W$	lb	(24) = (16) – (23)
25	Test power	SHP_t		HP	From power determination, Chapter 14, Table 14-1
26	Thrust coefficient constant	K_{c_T}	$K_{c_T} = A(\Omega R)^2$		Calculated or from tables
27	Power coefficient constant	K_{c_P}	$K_{c_P} = 550/[A(\Omega R)^3]$		Calculated or from tables
28	Thrust coefficient	C_T	$C_T = GW/(K_{c_T}\rho)$		(28) = (24) / [(26)·(13)]
29	Power coefficient	C_P	$C_P = (SHP) K_{c_P}/\rho$		(29) = (25)·(27) / (13)
30	Power coefficient required	$C_{P_{REQ}}$			At (28) and reference hover height (hover carpet plot)
31	Differential power coefficient	ΔC_P	$\Delta C_P = C_P - C_{P_{REQ}}$		(31) = (29) – (30)
32	Runway azimuth	α_{RW}		deg from north	
33	Wind azimuth	α_W		deg from north	
34	Wind velocity	V_W		kt	From weather station
35	Wind shear correction at 50 ft above ground level	$V_{WS_{50}}$		kt	
36	Head wind velocity	V_{HW}	$V_{HW} = V_W \cos(\alpha_{RW} - \alpha_W)$	kt	(36) = (34) cos [(32) – (33)]

TABLE 7-7 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
37	Ground distance required to clear 50-ft obstacle	$D_{G_{50}}$		ft	From Fairchild plate data, Fig. 7-19
38	Time (elapsed) to clear 50-ft obstacle	t_{50}		sec	
39	Ground distance required to clear 50-ft obstacle with zero wind	$D_{G_{50}}$	$D_{G_{50}} = D_G + V_{HW} t_{50}$		(39) = (37) + (36) (38)
40	Elapsed time to rotation	t_R		sec	
41	Ground distance to rotation	D_{G_R}		ft	From Fairchild plate data, Fig. 7-19
42	Acceleration distance	D_{ACC}	$D_{ACC} = D_{G_R} + V_{HW} t_R$	ft	(42) = (41) + (36) (40)
43	Indicated standard airspeed	$V_{S_{IND}}$		kt	From photo panel or manual recorded data
44	Instrument airspeed correction for standard system	$\Delta V_{S_{IC}}$		kt	From instrument calibration
45	Instrument corrected standard airspeed	$V_{S_{IC}}$	$V_{S_{IC}} = V_{S_{IND}} + \Delta V_{S_{IC}}$	kt	(45) = (43) + (44)
46	Horizontal velocity	V_H	$V_H = dx/dt + V_{HW}$	ft/sec	From Fairchild data slope $(46) = \left(\frac{dx}{dt}\right) + (36)$ at 50 ft from Fig. 7-6

TABLE 7-7 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
47	Vertical velocity	V_V	$V_V = dh/dt$	ft/sec	From Fairchild data slope at 50 ft from Fig. 7-6
48	True climb-out airspeed	V_T	$V_T = V_R = \sqrt{V_H^2 + V_V^2}$	ft/sec	Calculated or derived graphically from Fig. 7-8 $(48) = \sqrt{(46)^2 + (47)^2}$
49	Calibrated airspeed	V_{CAL}	$V_{CAL} = V_T\sqrt{\sigma}$	ft/sec	$(49) = (48)\sqrt{(14)}$
50	True climb-out airspeed	V_T	$V_T = V_T$(ft/sec) x 0.592	kt	$(50) = (48) \cdot 0.592$
51	Standard system airspeed position error	$\Delta V_{S_{PE}}$	$\Delta V_{S_{PE}} = V_{CAL} - V_{S_{IC}}$	kt	$(51) = (49) - (45)$

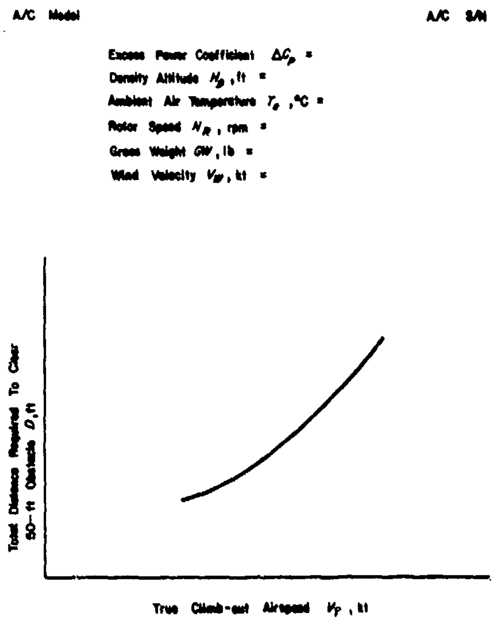


Figure 7-12. Takeoff Distance Required

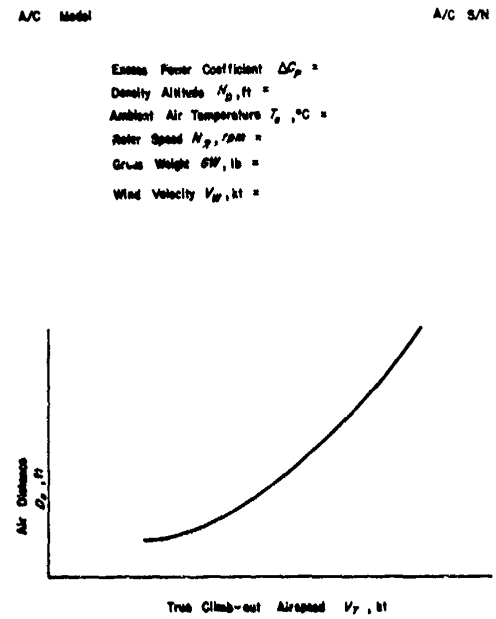


Figure 7-14. Air Distance Required

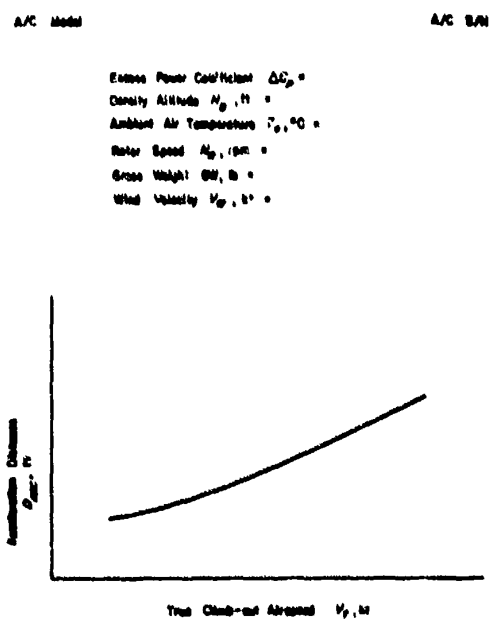


Figure 7-13. Acceleration Distance Required

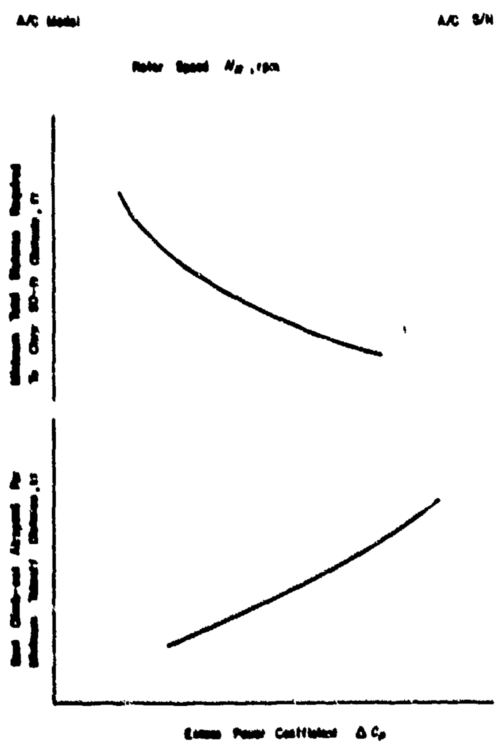


Figure 7-15. Maximum Takeoff Performance

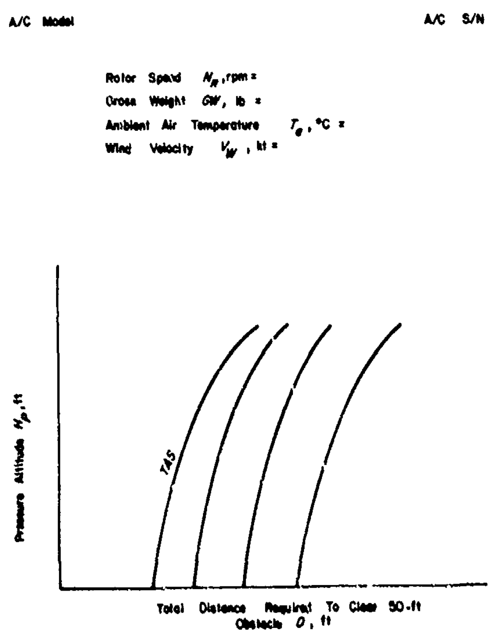


Figure 7-16. Takeoff Distance Required To Clear a 50-ft Obstacle

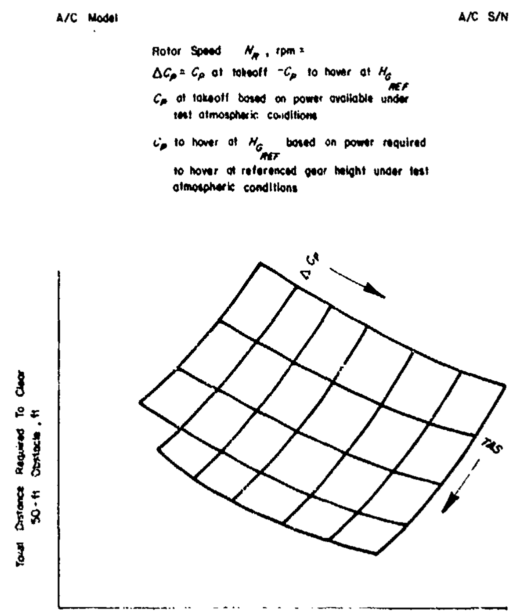


Figure 7-17. Takeoff Performance Variation With Excess Power Coefficient

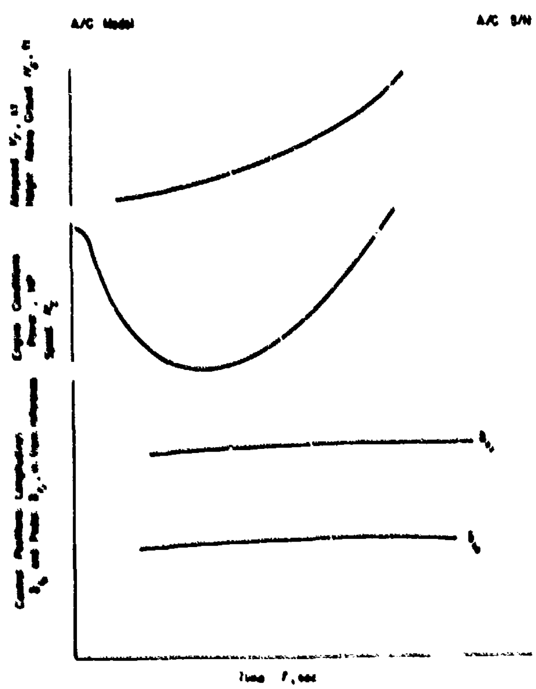


Figure 7-18. Takeoff Characteristics

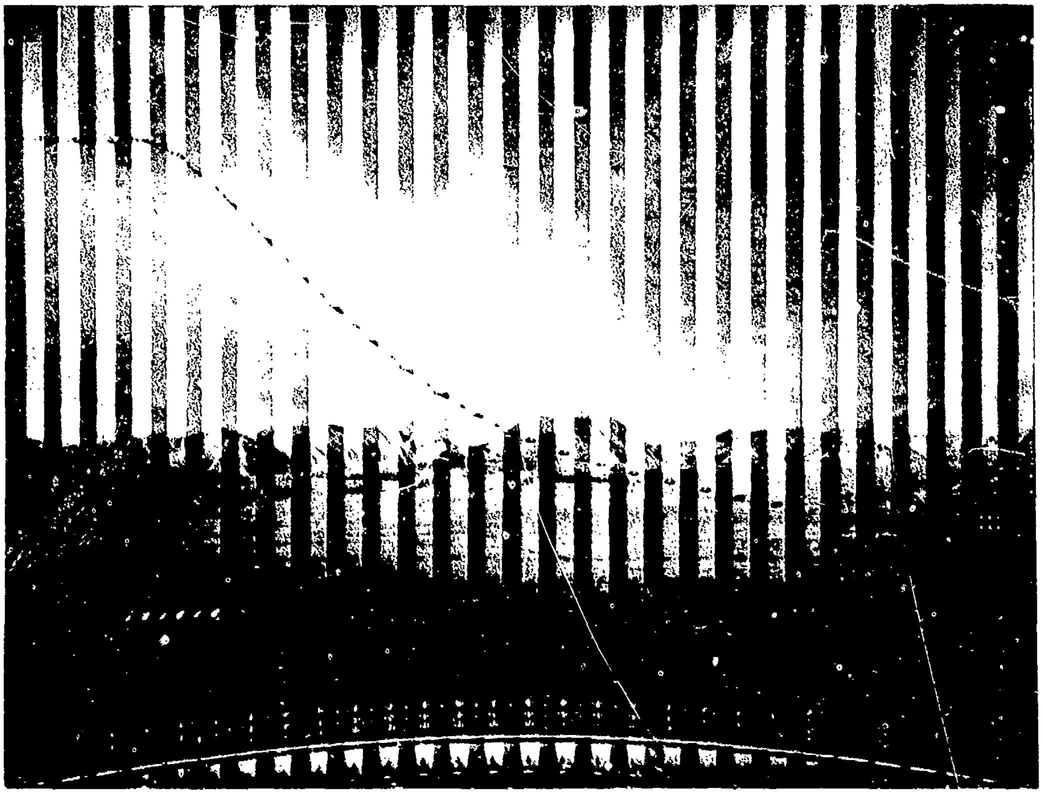


Figure 7-19. Fairchild Camera Data

CHAPTER 8

CLIMB PERFORMANCE

8-1 GENERAL

Climb performance is an important aspect of the total performance characteristics since the helicopter very often operates in restricted areas where many obstacles are present. Following a maximum performance takeoff, a sufficient excess performance margin is imperative to continue the climb after any initial takeoff obstacles have been cleared. Generally, under conditions of limited excess thrust, the rate of climb is not excessive and precise schedules and techniques must be used. Maneuvering under these conditions may reduce the performance available and constitute an operational limit.

The climb tests are conducted to determine the rate of climb available as a function of power, gross weight, rotor speed, airspeed, and altitude. Special tests are conducted to determine the various correction factors applicable to the variables mentioned. The general technique is an important consideration in obtaining maximum climb performance. Power management, power available, inlet performance, and engine operating characteristics normally are evaluated during the course of the tests.

Climbs should be conducted at the maximum and minimum gross weights to establish the rotor performance for the widest range and most adverse conditions. The airspeed and rotor speed investigations should be of sufficient magnitude to cover the general operating range including the cruise climb condition. Power variations should include the range from maximum available to the minimum for which a climb is possible.

The planning for the climb tests is relatively simple except for the tests to obtain correction factors. During these tests the ground personnel responsible for the ballast changes must be especially careful to load and document the proper amount. The tests should be conducted from the lowest takeoff altitude available, preferably at sea level.

Test techniques are relatively straightforward with the exception of the vertical climbs. The continuous climbs, sawtooth climbs, and correction factor climbs are primarily at steady-state conditions except for altitude. The speed and power schedules are critical and must be closely followed and rigidly controlled. The vertical climb is difficult to accomplish precisely, and a certain amount of practice is advisable before recording the final data. Wind conditions are also critical since the climb is initiated from the ground or a hover and the rotor is sensitive to small inflow velocities. In addition, surface winds give little or no indication of the existence of shear or gradients with increased altitude.

8-2 PLANNING

The planning for the climb performance tests generally differs for vertical climb, sawtooth climb, correction factor climb, and continuous climb tests. For vertical climb tests, a ground crew is necessary and a Fairchild Flight Analyzer or motion picture camera generally is used to record position data.

Sawtooth climbs should be conducted at the same average gross weight as the continuous climbs. The engine start gross weight should include allowances to compensate for the fuel used while climbing to the test altitude. The climb airspeed schedule should be calculated from the level flight performance summaries at the airspeed for minimum power required. A minimum of three points should be flown above and below this speed value. In lieu of test data, the calculated best climb speed may be obtained from the handbook or from the manufacturer. The upper and lower rotor speed limits should be determined and the range divided into six increments. A climb then should be planned at each of these speeds. Airspeed is held constant during the rotor speed tests. Flight cards are prepared which list the altitudes, airspeeds, and rotor speeds to be investigated. A briefing of the crew members is conducted to explain the flight card and test procedures; followed by a briefing of the maintenance crew concerning the configuration and ballast requirements, and the instrumentation group concerning instrumentation requirements.

Gross weight must be controlled carefully when conducting the climbs to determine the gross weight K_W and power K_P correction factors. For the K_W climbs, the helicopter is tested first at maximum gross weight, and then ballast is removed incrementally on each succeeding climb. The fuel used on each climb should be estimated and compensated for in the ballast and gross weight calculations for the following climb. The number of personnel should be adequate to quickly accomplish the ballast changes. For the K_P climbs, gross weight should be maintained constant by adding ballast to compensate for fuel used on the previous climb. To facilitate the ballast operation, the amount required for each climb should have been estimated previously and readily available for loading.

Flight cards are prepared which specify the gross weight, rotor speed, power, altitude, and airspeed schedules. The flight and ground personnel should be briefed. The ballasting procedure should be covered in detail with emphasis on speed and safety of operation.

The continuous climbs are conducted to the service ceiling and are normally the last climb tests to be accomplished. The aircraft ballast calculations should result in an engine start gross weight which compensates for the fuel used during engine start, takeoff, and travel to the test area to arrive at the test gross weight. The climb schedule must be available and in a form that is suitable for the pilot. Oxygen equipment should be available when the anticipated maximum altitude is greater than 12,000 ft. Flight cards are made which specify gross weight, power, rotor speed, and airspeed schedules. This card is explained during a briefing of the flight crew with emphasis on power and airspeed schedules during the climbs. After the briefing, the maintenance and instrumentation requirements are provided these groups.

8-3 INSTRUMENTATION

The climb conditions are relatively static and the time average value of the primary parameters can be hand recorded from visual instruments. The required visual instrumentation is listed in Table 8-1.

TABLE 8-1

VISUAL CLIMB INSTRUMENTATION

Altitude	Engine torque
Airspeed (boom)	Ambient air temperature
Rotor speed	Fuel used
Engine speed	Fuel flow
Airspeed (standard)	Elapsed time

Although this instrumentation is a minimum, a considerable manual recording workload will exist during a high performance climb. A photo panel should be installed with the additional parameters shown in Table 8-2.

TABLE 8-2

PHOTO PANEL INSTRUMENTATION FOR CLIMB PERFORMANCE

Rotor speed	Airspeed (standard)
Engine speed	Airspeed (boom)
Fuel flow	Fuel temperature
Engine torque	Altitude (boom)
Time	Altitude (standard)
Ambient air temperature	Inlet air temperature
Inlet air pressure	

A photo panel installation will reduce the number of hand recorded parameters to those necessary for in-flight monitoring requirements. A permanent record is obtained which can be read after the flight and used for data reduction purposes. In addition, more data points are obtained to define better the changes in performance with altitude.

An oscillograph is normally unnecessary during climb tests since sufficiently accurate data can be obtained from the visual and photo panels. However, when an oscillograph is installed, the parameters shown in Table 8-3 are recorded.

TABLE 8-3

OSCILLOGRAPH INSTRUMENTATION FOR CLIMB PERFORMANCE

Engine speed	Event marker
Engine torque	Airspeed
Rotor speed	Longitudinal stick position
Altitude	Lateral stick position
Pedal position	Collective stick position
Fuel flow	

8-4 TEST METHODS

8-4.1 VERTICAL CLIMB TEST METHOD

An absolute minimum of these flights should be conducted since a significant portion of the flight is within the height-velocity (or Deadman's) curve. The vertical climb tests normally are conducted from ground level to a maximum height of 1000 ft above the ground. As power is applied, the induced flow into the rotor causes a rapid reduction in power required with a resultant vertical acceleration. The acceleration will be greater with greater excess power margins and a longer climb will be necessary before the peak value is reached. This will determine the point at which the climb can be terminated. The airspeed system is, of course, inoperative and the pilot has only ground reference and aircraft attitude to aid in maintaining a vertical flight path. The value of the ground reference rapidly diminishes with increasing height.

The primary purpose of these tests is to determine the vertical acceleration and climb capability at the various power and collective pitch settings. The climbs are conducted best by maintaining full power while varying the gross weight. This relieves the pilot from the difficult task of attempting to maintain a precise power schedule in addition to his other tasks. The first test should be conducted at the maximum planned gross weight. This will be the lowest rate of climb condition and will allow the pilot more time to maintain schedules while building proficiency for the more difficult light weight, high rate of climb tests. If available, a sensitive attitude indicator may provide a reliable reference for the pilot. The hovering attitude is noted prior to the climb although the climb attitude may be somewhat different from the hovering attitude as a result of increased thrust and drag considerations. The changes should be small and a satisfactory climb should result.

When the climb is initiated with the aircraft on the ground, large initial control inputs may be necessary to counter the effects of increasing power during the lift-off. These inputs may limit the rate at which power can be applied and cause a climb path other than vertical. Starting the climb from a low hover height normally reduces the control requirements and allows the pilot to establish the initial baseline point. From the hover, power is applied as rapidly as possible, and a vertical climb is maintained until a constant rate of climb or the predetermined altitude is reached. A countdown is given at the start of the climb to the recording ground crews for operational and data correlation purposes. Event markers should be used by the pilot during the climb to denote any unusual characteristics or occurrences. At the termination of the climb, a hover point should be obtained which is used later in the hover performance and thrust to weight calculations.

From the hover, power is reduced slowly and a stabilized vertical descent is established. This is maintained until the aircraft is close to the ground and then power is applied to establish a hover at the initial start height, where the data are recorded. Prior to testing any descent conditions, the aircraft reaction to power application should be investigated. The height and excess power required to slow the aircraft to a hover also should be noted carefully. Aircraft with lifting surfaces (wings, fairings, horizontal stabilizers, etc.) are particularly susceptible to unusual control moments with high vertical rates of descent. A build-up technique should be used to insure that a sufficient performance margin is present at all times to contend with any unforeseen contingencies.

The procedure is repeated for various decreasing weight increments to include the minimum gross weight. The rate of climb will increase as the minimum gross weight is approached, and the flight characteristics may change significantly. The pilot must use extreme care to employ the same technique throughout the tests. Variations will be reflected in the data, and an appropriate correction procedure may not be available. At best, the correction procedures probably will be difficult to accomplish and have an unknown accuracy.

As a general rule the performance is most critical where the performance margin is the smallest—i.e., heavy weight, low power, high altitude. Invariably, control inputs detract from the lift available with magnitude, method, and rate of application being important factors. This, of course, becomes particularly important when control is obtained by direct thrust modulation or direction. After the climb is initiated, large corrections to the flight path should be avoided. A small deviation from the vertical can be tolerated or corrected, whereas, a discontinuity in the climb rate caused by thrust loss, resulting from control inputs, usually will render the data invalid. As the performance margin increases, the control inputs become a smaller percentage of the total excess thrust and should have a smaller influence on the climb rate. This is a fortunate situation since the control requirements are usually higher during these conditions.

Transitioning from a climb to forward level flight requires control inputs that may cause thrust losses and a descent. The control requirements to transition from a hover should be the same, and there should be no variation when full power is being used. When excess power is available, it can be used to compensate for the control requirements and minimize or eliminate any altitude loss. The transition performance should be determined for the excess performance conditions at which the climbs are conducted. The tests should include pitch and thrust control variations to determine the optimum technique.

A thrust loss causes a decrease in the climb speed or an increase in descent sink rate. In a descent, the aircraft enters ground effect where recirculation ingestion and ground turbulence may become factors. Should this occur too near the ground, insufficient performance margin may be available to safely arrest the rate of descent prior to ground contact.

The vertical climb and descent are highly transient, poorly defined flight regimes and an automatic recording system is necessary to obtain the maximum useful data. The data should be recorded during the hover to determine the power required at the in-ground effect, thrust to weight ratio of unity condition. The automatic system should be activated when the pilot begins his countdown and remain on until the termination of the climb. Event markers should be inserted at the start of the climb.

The hover point at the top of the climb is recorded to provide power required at the out-of-ground effect thrust equals weight condition. A time history of the descent is recorded with event markers at different altitudes during the stabilized descent. It is very important to identify the point where action is taken to stop the descent and enter the hover. This point also should be announced to the ground recording personnel for later correlation of their information with the flight data.

8-4.2 SAWTOOTH CLIMB TEST METHOD

The purposes of the sawtooth climb performance tests are to verify the calculated best climb speed schedule, determine the loss that results from a nonoptimum airspeed, and to establish the variation in climb performance as a function of rotor speed and sideslip angle. The flight card specifies the sideslip angles, density altitude, rotor speed, and airspeed ranges. The first variable to be investigated is airspeed, starting with the calculated best climb speed. The minimum test altitude increment is 2000 ft. After takeoff and during the climb the aircraft is stabilized at the desired conditions 1000 ft below the aim test altitude. This will allow usable data to be taken through the maximum altitude increment. The climb then is continued to an altitude 1000 ft above the aim test altitude. Generally, data are taken during the descent as discussed in Chapters 10 and 11. Airspeed then is varied by the specified increment and the procedure is repeated until the necessary range of conditions is tested. The size of the airspeed increments will vary with the individual speed capabilities, although 10 kt is a common value. Regardless of the speed range, a minimum of eight points usually is needed to define the curve adequately. The tests should be accomplished at the power setting for the data that will be corrected subsequently using the results of these tests and a zero sideslip angle, and a constant rotor speed. During the test an in-flight determination is made of the best climb speed as shown in Fig. 8-1.

By use of the previously determined best climb speed, zero sideslip angle, and maximum power or the power setting for continuous climbs, a series of tests is conducted at various rotor speeds. The altitude increment and the stabilization technique are the same as in the variable airspeed tests. The first test usually is at the maximum allowable rotor

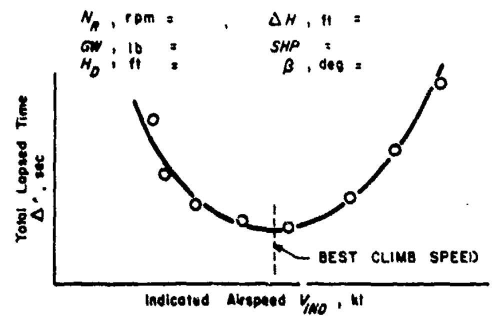


Figure 8-1. Rate of Climb Variation With Airspeed

speed. Each succeeding test is conducted at some incrementally lower rotor speed. The magnitude of the increment will vary with the rotor speed range. A common increment is 10 rpm, with a minimum of six points needed to define the curve adequately. At the low rotor speed and high collective setting, incipient retreating blade stall may be encountered. This usually is characterized by an apparent decrease in stability, lower control response, and an increase in airframe vibration level. Descent performance as a function of rotor speed variation is obtained at the end of each climb. The technique for these is discussed in Chapters 10 and 11.

Most helicopters exhibit a change in power required as a function of sideslip angle and airspeed. These characteristics will result in a change in the climb performance. These climbs are made at the best climb speed, normal operating rotor speed, and maximum power, or at the values which will be used during continuous climbs. The procedure with respect to altitude increments and operational considerations is the same as for the previously discussed sawtooth climb tests. The first climb condition is zero sideslip and best climb speed. On each succeeding climb, the sideslip angle is varied as specified by the flight card. The angle should be increased to the envelope limit or until a limit becomes evident. Both right and left sideslip angles shoud be investigated. A minimum of three points will be required to define the curve. The performance change with sideslip should not be the same for all airspeeds, and the test should be repeated at an airspeed above and below the initial best climb speed.

The sawtooth climb data can be recorded manually for data reduction. During high rates of climb, the time increments become smaller which increases the manual data recording density and, therefore, an automatic system is more desirable. The engine and inlet performance variables are highly transient, and only a limited amount of time averaged or instantaneous data can be recorded manually. This type of data has limited use and when accurate information of this type is desired, an automatic system must be used. The manual recording technique is similar for all the sawtooth climb tests discussed. Data are recorded at the start of the altitude increment and at a minimum of 500-ft intervals during the climb. Smaller altitude increments should be used when possible. When an automatic system is used, it should be activated prior to reaching the test altitude with event markers used to identify the various altitudes during the climb. Time and altitude are the most critical parameters during these tests and must be measured with the greatest possible accuracy. The total elapsed time for the test altitude increment is used to monitor the test results and insure completeness and validity. An in-flight plot similar to Fig. 8-1 is constructed during the airspeed variation tests.

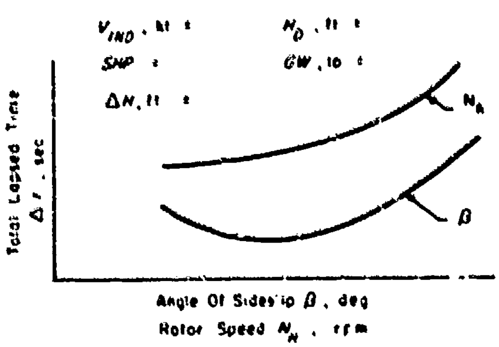


Figure 8-2. Climb Performance Variation With Rotor Speed and Angle of Sideslip

On this plot, the smallest lapsed time represents the highest rate of climb condition. The best climb speed as indicated by this plot then is used in the remaining sawtooth climb tests. In most cases, the best climb speed shown on the plot will be a faired line value, and it is good practice to conduct one climb at this precise value to insure the validity of the fairing. In a similar manner, the other tests can be monitored by plotting the lapsed time versus the variable being investigated as illustrated in Fig. 8-2.

8-4.3 WEIGHT AND POWER CORRECTION FACTORS CLIMB TEST METHOD

The gross weight K_W-factor and power correction K_P-factor (K-factors) climbs are conducted to determine the rate of climb correction coefficients for deviations in gross weight and power from the aim values. From the flight test technique viewpoint, they are simply a series of climbs at various gross weights and power settings.

The K_W-factor test usually is conducted with power maintained at the maximum while incrementally varying the gross weight on each successive climb. The rotor speed should be the same as the future continuous climbs, and the altitude increment used should be approximately 50% of the estimated service ceiling altitude or a minimum of 2000 ft. The first test usually is at the maximum gross weight, low rate of climb condition. This allows the pilot to start at an easier condition and increase proficiency as the test progresses. The aircraft is stabilized in level flight at the scheduled climb speed. For elevations above sea level, the climb should be started as near the ground as practical unless the ambient density altitude is below sea level. In this case the aircraft should be in a stabilized climb as it passes through the sea level density altitude. From the stabilized level flight condition at the scheduled climb speed, the collective is increased rapidly to apply full power while maintaining the climb speed schedule. This technique will reduce the acceleration factors involved during the climb entry and allow the aircraft to become stabilized within the minimum altitude and/or time. During the stabilized climb, all schedules are maintained as closely as possible. Descent performance should be conducted at the completion of the climb as discussed in Chapters 10 and 11. Upon return to the ground, the predetermined ballast increment is off-loaded as rapidly as possible. When the rotors are not stopped, the pilot should exercise caution during this ballast operation to insure that ground safety is not jeopardized and that the established ground procedure is observed. The amount of ballast off-loaded should be counted carefully and recorded both on the ground and flight cards. The weight increment will vary with aircraft although six increments should be considered a minimum. The climb procedure then is repeated at these different weights until the specified weight range is covered.

The K_p factor climbs are conducted with a different power setting on each climb while maintaining a constant weight. The power that is used will vary from the maximum available to some minimum that will produce a reasonable rate of climb (500 ft/min) at the start of the climb. A light gross weight is used since this will permit the greatest variation in power. The rotor speed and airspeed schedules should be the same as the K_μ climbs, and the altitude increment also should be similar. The climb entry technique and the stabilized portion of the climb are the same as previously discussed. After each climb the aircraft is returned to the ground and ballast is added to compensate for the fuel used during the test.

Safety must be maintained during the ballasting operation and care exercised to insure proper ballast addition and documentation. On each succeeding climb, the power setting is changed a predetermined increment until the power range has been encompassed. Descent performance should be determined as discussed in Chapters 10 and 11.

These climbs are relatively steady-state with respect to the airframe although the inlet and engine conditions may be expected to change rapidly with variations in altitude, rate of climb, attitude, and climb angle. To obtain the maximum data, both a manual and an automatic recording capability are required. The automatically recorded data supplement the hand recorded aircraft parameters and provide a time history of the transient inlet and engine performance parameters. The automatic system should record data continuously during the whole of each climb. Event markers are inserted at altitude increments to aid in correlation with the hand recorded data. In some cases the automatic recording capability may be limited or the climb may be too long for a continuous recording. In this case, data are recorded only for different altitude increments, for example, within ±200 ft of each even 1000 ft altitude. Upon landing, the fuel used is recorded, the ballast requirement is provided the ground crew, and the ballast is loaded and recorded on the flight card. The ambient conditions are checked to insure that the pre-selected weight and power increments are in fact attainable and will produce the desired data.

The documentation of each test must be complete to prevent loss of data and to insure adequate correlation. The time and altitude parameters are of the most importance. These should be recorded as often as possible during each climb. The power, rotor speed, and airspeed are monitored and any deviations from the schedule are noted.

8-4.4 CONTINUOUS CLIMB TEST METHOD

The purposes of the continuous climb tests are to determine the variation in climb performance with altitude and to establish the service ceiling. These usually are conducted at design and maximum gross weight. Additional weights and configurations may be specified in the test plan. The airspeed, rotor speed, and power schedule should be studied carefully prior to the test since any deviation will influence the performance and require a correction. Most helicopters have a speed value. In lieu of test data, the calculated best climb speed may be obtained from the handbook or from the manufacturer. The upper and lower rotor speed limits should be determined and the range divided into six increments. A climb then should be planned at each of these speeds. Airspeed is held constant during the rotor speed tests. Flight cards are prepared which list the altitudes, airspeeds, and rotor speeds to be investigated. A briefing of the crew members is conducted to explain the flight card and test procedures; followed by a briefing of the maintenance crew concerning the configuration and ballast requirements, and the instrumentation group concerning instrumentation requirements.

Gross weight must be controlled carefully when conducting the climbs to determine the gross weight K_W and power K_P correction factors. For the K_W climbs, the helicopter is tested first at maximum gross weight, and then ballast is removed incrementally on each succeeding climb. The fuel used on each climb should be estimated and compensated for in the ballast and gross weight calculations for the following climb. The number of personnel should be adequate to quickly accomplish the ballast changes. For the K_P climbs, gross weight should be maintained constant by adding ballast to compensate for fuel used on the previous climb. To facilitate the ballast operation, the amount required for each climb should have been estimated previously and readily available for loading.

Flight cards are prepared which specify the gross weight, rotor speed, power, altitude, and airspeed schedules. The flight and ground personnel should be briefed. The ballasting procedure should be covered in detail with emphasis on speed and safety of operation.

The continuous climbs are conducted to the service ceiling and are normally the last climb tests to be accomplished. The aircraft ballast calculations should result in an engine start gross weight which compensates for the fuel used during engine start, takeoff, and travel to the test area to arrive at the test gross weight. The climb schedule must be available and in a form that is suitable for the pilot. Oxygen equipment should be available when the anticipated maximum altitude is greater than 12,000 ft. Flight cards are made which specify gross weight, power, rotor speed, and airspeed schedules. This card is explained during a briefing of the flight crew with emphasis on power and airspeed schedules during the climbs. After the briefing, the maintenance and instrumentation requirements are provided these groups.

8-3 INSTRUMENTATION

The climb conditions are relatively static and the time average value of the primary parameters can be hand recorded from visual instruments. The required visual instrumentation is listed in Table 8-1.

TABLE 8-1

VISUAL CLIMB INSTRUMENTATION

Altitude	Engine torque
Airspeed (boom)	Ambient air temperature
Rotor speed	Fuel used
Engine speed	Fuel flow
Airspeed (standard)	Elapsed time

Although this instrumentation is a minimum, a considerable manual recording workload will exist during a high performance climb. A photo panel should be installed with the additional parameters shown in Table 8-2.

TABLE 8-2

PHOTO PANEL INSTRUMENTATION FOR CLIMB PERFORMANCE

Rotor speed	Airspeed (standard)
Engine speed	Airspeed (boom)
Fuel flow	Fuel temperature
Engine torque	Altitude (boom)
Time	Altitude (standard)
Ambient air temperature	Inlet air temperature
Inlet air pressure	

A photo panel installation will reduce the number of hand recorded parameters to those necessary for in-flight monitoring requireules during the tests will reduce greatly the magnitude of the corrections and increase the data accuracy and validity. At the moment there is no suitable nondimensional method for reducing climb data, and the standard practice is to correct the test values directly to standard day conditions. The nature of the data is such that a great many points are required for each climb and, since each point must be corrected, the workload becomes formidable. Although similar, the data reduction forms for the various types of climbs are presented separately for clarity. The use of a computer subroutine for processing climb performance data will reduce significantly the manual data reduction task for continuous climbs. Additional gains in data reduction efficiency can be realized when the subroutine is used in conjuction with automatic, computer compatible data recording devices such as airborne magnetic tape units or telemetry systems.

8-5.2 VERTICAL CLIMBS

The data reduction form for vertical climbs is presented in Table 8-4*.

8-5.3 SAWTOOTH CLIMBS

The data reduction procedures for sawtooth climbs are presented in Table 8-5*. These climbs are conducted to enable derivation of rate of climb correction factors for airspeed and rotor speed deviations. A discussion of the applications of the data reduction procedures follows.

For the airspeed variation tests, the time and altitude from Table 8-4, steps 5 and 43, are plotted as shown in Fig. 8-3. The altitude increment was 2000 ft, and the minimum number of points manually recorded was four. Additional points should be obtained from the automatic (photo panel) recording system to define the curve better. A plot such as Fig. 8-3 will be constructed for each of the climbs at a different airspeed. Fig. 8-3 is entered at the aim test altitude, and the slope (indicated rate of climb) is taken. This procedure is repeated for each of the curves. These indicated values are corrected for nonstandard temperature effects on the altimeter. The corrected rate of climb then is plotted as a function of the calibrated airspeed as shown in Fig. 8-4. The curve shown in Fig. 8-4 can be used to derive an airspeed correction curve in the following manner. Fig. 8-4 is entered at chosen airspeed increments, and the slope of the curve is determined to obtain $\Delta(R/C)/\Delta(A/S)$. This derivative then is plotted as a function of speed as shown in Fig. 8-5. Any variation in airspeed that may have occurred during the test may be corrected by entering Fig. 8-5 and obtaining a correction factor. This then is multiplied by the speed variation to obtain the $\Delta(R/C)$ value.

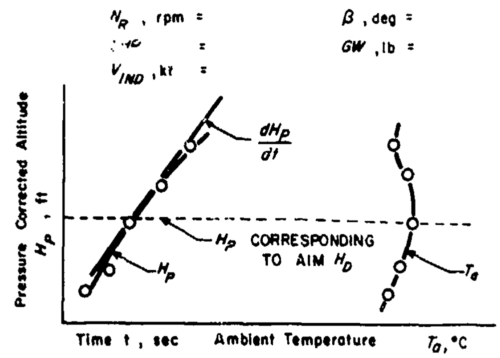


Figure 8-3. Sawtooth Climb Performance

The rotor speed variation test data are processed in much the same way as the airspeed variation data with the exception that airspeed is now a constant. A typical plot is shown in Fig. 8-6. The curve shown in Fig. 8-6 then can be differentiated to produce rotor speed correction factors as illustrated in Fig. 8-7.

The rotor speed correction factors now may be applied to the airspeed curve to correct for any small rotor speed variations.

This process of correcting and refairing the curves may be repeated several times to reduce the scatter and increase the accuracy

*The data reduction forms (tables) are located at the end of each chapter.

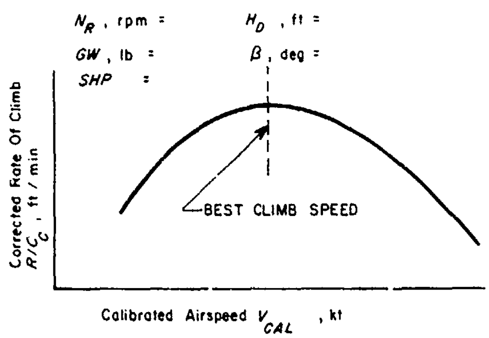


Figure 8-4. Sawtooth Corrected Climb Performance

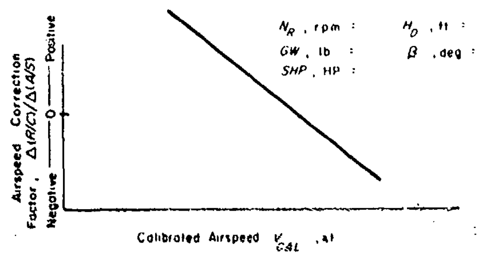


Figure 8-5. Rate of Climb Correction Factor for Airspeed

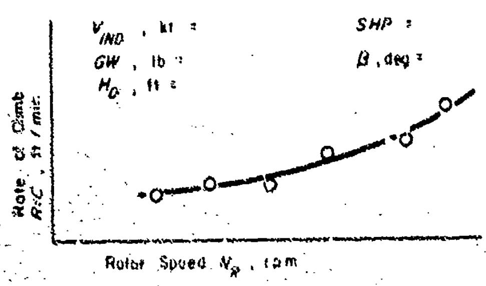


Figure 8-6. Climb Performance Variation With Rotor Speed

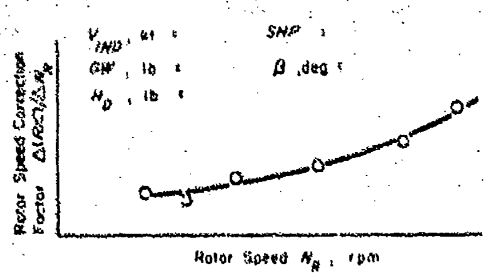


Figure 8-7. Rate of Climb Correction Factor for Rotor Speed Variation

of the data. These corrections usually are accomplished on a separate summary reduction form as distinct from the general reduction form previously presented.

8-5.4 CORRECTION FACTOR CLIMBS

The basic data reduction form for the K-factor climbs is contained in Table 8-5*.

The time and corrected altitude from the correction factor climbs are plotted as shown in Fig. 8-8.

Plotting the additional parameters of airspeed, gross weight, power, and rotor speed will permit using the faired line values rather than working with the individual points. The slope of the curve is determined at 500 ft intervals to establish the rate of climb versus altitude curve, illustrated in Fig. 8-9.

The other parameters are read at each altitude increment and transcribed to a data reduction form for use in subsequent data procedures. Plots similar to Figs. 8-8 and 8-9 should be prepared for each of the climbs conducted. From Fig. 8-9, the rate of climb versus gross weight at a set of flight conditions is read at a minimum of two altitudes, usually 5,000 and 10,000 ft. Additional altitudes are advisable when unusual characteristics are encountered. This rate of climb is then corrected for altimeter error and plotted versus gross weight as shown in Fig. 8-10. Curves then are faired through the points and the slope is read at the gross weight at which the power variation tests were conducted. This provides an initial weight correction factor $\Delta(R/C)/\Delta GW$.

The initial data reduction for the power variation tests is similar to the weight variation tests through the procedure required to obtain Figs. 8-8 and 8-9. The data must be read at the same altitudes as those used for the K_W climbs. Since gross weight was not

*The data reduction forms (tables) are located at the end of each chapter.

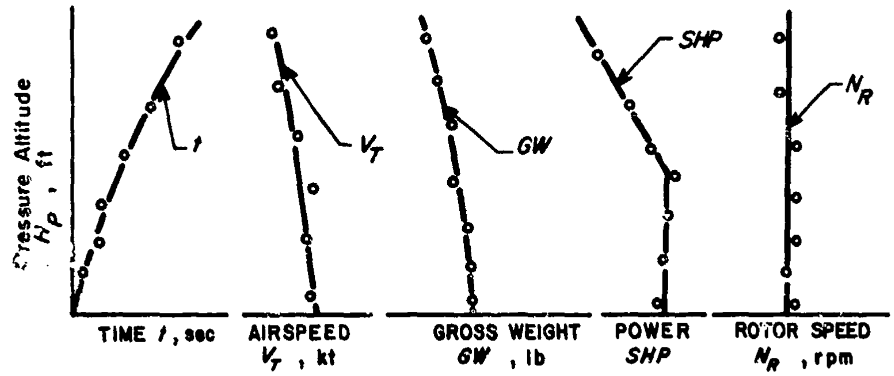


Figure 8-8. Climb Performance

precisely constant for the climbs, the previously determined weight correction factor should be used to make a rate of climb correction. The change in rate of climb is

$$\Delta(R/C)_W = \Delta(R/C)/GW \times \Delta GW \qquad (8\text{-}1)$$

$$(R/C)_C = R/C + \Delta(R/C)_W \qquad (8\text{-}2)$$

This corrected rate of climb is then plotted as illustrated in Fig. 8-11. The curve then is entered at the power for which the weight variation tests were conducted to obtain a power correction factor $\Delta(R/C)/\Delta SHP$. This factor now is used to correct the K_W climbs for any power variations that may have existed. The points are then replotted, the curves retaired, and the slope of the R/C versus GW is again read. This refined gross weight correction is now applied to the tapeline rate of climb to obtain more accurate data for plotting in Figs. 8-10 and 8-11. The process should be repeated alternately to the two sets of data until the data scatter and the succeeding changes in rate of climb are minimized. Further refinements and more accurate data can be obtained by applying the previously established rotor speed and airspeed correction factors.

8-5.5 CONTINUOUS CLIMBS

The continuous climb data reduction procedures are much more complex and time consuming than the previously discussed climbs. This is caused by the requirement to incorporate standard day power available and fuel consumption corrections, calculate additional parameters, and by the sheer volume of points associated with a climb to high altitude. Since various contractual guarantees—

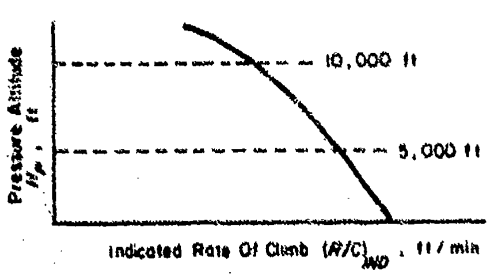


Figure 8-9. Climb Performance Variation With Altitude

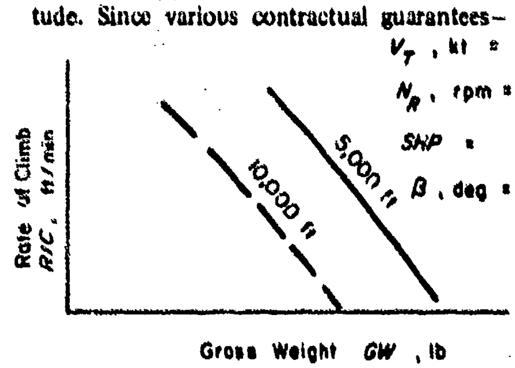


Figure 8-10. Climb Performance Variation With Gross Weight

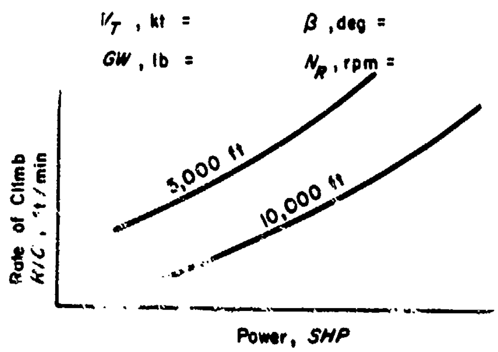


Figure 8-11. Climb Performance Variation With Power

such as rate of climb, time to climb, and service ceiling–may be associated with the test results, the importance of these tests is evident. A data reduction form for the continuous climb test is presented in Table 8-6*.

*The data reduction forms (tables) are located at the end of each chapter.

A computer subroutine may be used to reduce the amount of manual data reduction effort required.

As can be seen, there are a multitude of corrections that can be made to the test rate of climb. Close adherence to the planned conditions should eliminate some of the corrections such as those for rotor speed and airspeed. The power and weight corrections are inevitable unless the test is conducted on a standard day. The magnitude of these corrections can be minimized by close weight control and proper power management for the ambient conditions.

8-6 DATA PRESENTATION

Sample final plot formats of the reduced climb data are presented in Figs. 8-12 through 8-14.

TABLE 8-4

DATA REDUCTION FORM FOR VERTICAL CLIMBS

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1	Point No./Flight No.				
2	Indicated pressure altitude	$H_{P_{IND}}$		ft	From photo panel or manual recording
3	Pressure altitude instrument correction	$\Delta H_{P_{IC}}$		ft	From instrument calibration
4	Pressure altitude position error	$\Delta H_{P_{PE}}$		ft	From altimeter position error calibration
5	Pressure altitude	H_P	$H_P = H_{P_{IND}} + \Delta H_{P_{IC}} + \Delta H_{P_{PE}}$	ft	(5) = (2) + (3) + (4)
6	Indicated ambient air temperature	$T_{a_{IND}}$		°C	From photo panel or manual recording
7	Ambient air temperature instrument correction	$\Delta T_{a_{IC}}$		°C	From instrument calibration
8	Ambient air temperature system correction	$\Delta T_{a_{SC}}$		°C	From total system correction
9	Ambient air temperature	T_a	$T_a = T_{a_{IND}} + \Delta T_{a_{IC}} + \Delta T_{a_{SC}}$	°C	(9) = (6) + (7) + (8)
10	Indicated rotor speed	$N_{R_{IND}}$		rpm	From photo panel or manual recording
11	Instrument correction for rotor speed	$\Delta N_{R_{IC}}$		rpm	From instrument calibration

TABLE 8-4 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
12	Rotor speed	N_R	$N_R = N_{R_{IND}} + \Delta N_{R_{IC}}$	rpm	(12) = (10) + (11)
13	Density altitude	H_D		ft	Calculated or from tables at (5) and (9)
14	Air density	ρ		slug/ft^3	Calculated or from tables at (13)
15	Air density ratio	σ	$\sigma = \rho/\rho_0$		Calculated or from tables at (13) (15) = (14) /0.0023769
16	Square root density ratio	$\sqrt{\sigma}$	$\sqrt{\sigma} = \sqrt{\rho/\rho_0}$		Calculated or from tables at (13) (16) = $\sqrt{(15)}$
17	Ambient air pressure	P_a		in. Hg	Calculated or from tables at (5)
18	Ambient air pressure ratio	δ_a	$\delta_a = P_a/P_0$		Calculated or from tables at (5) (18) = (17)/29.92
19	Ambient air temperature ratio	θ_a	$\theta_a = (T_a + 273.15)/288.15$		Calculated or from tables at (5) (19) = [(9) + 273.15] /288.15
20	Air temperature ratio	$\sqrt{\theta}$			Calculated or from tables at (5) (20) = $\sqrt{(19)}$
21	Standard temperature	T_S		°C	Calculated or from tables at (5)

TABLE 8-4 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
22	Engine start gross weight	$ESGW$		lb	From maintenance preflight
23	Preflight fuel specific weight	FS_1		lb/gal	from instrument or maintenance preflight
24	Preflight fuel temperature	T_f		°C	From instrument preflight
25	Test fuel temperature	T_{f_t}		°C	From photo panel or manual recording
26	Postflight fuel specific weight	FS_2		lb/gal	From instrument postflight
27	Average fuel specific weight	FS_{AVG}	$FS_{AVG} = (FS_1 + FS_2)/2$	lb/gal	(27) = [(23) + (26)] /2
28	Fuel counter difference	ΔFC	$\Delta FC = FC_2 - FC$	ct	From photo panel or manual recording
29	Fuel counter constant	K_{FC}		gal/ct	From fuel flow calibration
30	Volume of fuel used	FU_{VOL}	$FU_{VOL} = \Delta FC \times K_{FC}$	gal	(30) = (28)·(29)
31	Weight of fuel used	FU_W	$FU_W = FU_{VOL} \times FS_{AVG}$	lb	(31) = (30)·(27)
32	Weight of ballast	W_B		lb	From flight card or ballast crew
33	Test gross weight	GW_t	$GW_t = ESGW + W_B - FU_W$	lb	(33) = (22) + (32) – (31)

TABLE 8-4 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
34	Altimeter temperature correction factor	K_{HP}	$K_{HP} = \frac{T_a + 273}{T_S + 273}$		$(34) = \frac{(9) + 273}{(21) + 273}$
35	Wheel/skid height	H_W		ft	From measurements or movie camera data
36	Height above the ground	ΔH_W		ft	From Fairchild or movie camera data at (37)
37	Time from takeoff	Δt		sec	From cameras or timing device
38	Thrust coefficient constant	K_{c_T}	$K_{c_T} = A(\Omega R)^2$	lb-slug/ft³	$(38) = \pi R^2 [\pi \cdot (12) \cdot R/30]^2$
39	Power coefficient constant	K_{c_P}	$K_{c_P} = 500/[A(\Omega R)^3]$	slug/ft³-HP	$(39) = 550/[\pi R^2 \ \pi \cdot (12) \cdot R/30]$
40	Thrust coefficient	C_T	$C_T = GW_t/(\rho K_{c_T})$		$(40) = 33/[(14) \cdot (38)]$
41	Power required	SHP_{REQ}		HP	From power required calculations table
42	Power available	SHP_a		HP	From power available calculations table

TABLE 8-4 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
43	Power coefficient available	C_{P_A}	$C_{P_A} = SHP_A \times K_{c_p}/\rho$		(43) = (42)·(39) / (14)
44	Thrust coefficient available	C_{T_A}			From hover C_p vs C_T curve at (42) and (35)
45	Thrust available	T_A	$T_A = C_{T_A} \times \rho K_{c_T}$	lb	(45) = (44)·(14)·(38)
46	Takeoff thrust to test gross weight ratio	T/W	$T/W = T/GW_t$		(46) = (45 / (33)
47	Hover power required	SHP_H		HP	From power calculations table
48	Test thrust coefficient	C_{T_t}	$C_{T_t} = GW_t/(\rho K_{c_T})$		(48) = (33) / [(14)·(38)]
49	Corrected thrust coefficient	C_{T_C}	$C_{T_C} = C_{T_t}/\sigma$		(49) = (48) / (15)
50	Indicated rate of climb	dh/dt		ft/min	Slope of curve (37) versus (36)
51	Tapeline rate of climb	R/C	$R/C = dh/dt \times K_{H_p}$	ft/min	(51) = (50)·(34)

TABLE 8-5

DATA REDUCTION FORM FOR SAWTOOTH CLIMBS

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1-34					See Table 8-4
35	Boom indicated airspeed	$V_{B_{IND}}$		kt	From photo panel or manual recording
36	Boom airspeed instrument correction	$\Delta V_{B_{IC}}$		kt	From instrument calibration
37	Boom airspeed position error correction	$\Delta V_{B_{PE}}$		kt	From position error calibration
38	Calibrated boom airspeed	$V_{B_{CAL}}$	$V_{B_{CAL}} = V_{B_{IND}} + \Delta V_{B_{IC}} + \Delta V_{B_{PE}}$	kt	(38) = (35) + (36) + (37)
39	Indicated standard system airspeed	$V_{S_{IND}}$		kt	From photo panel or manual recording
40	Airspeed correction for standard system	ΔV_S		kt	From airspeed system calibration
41	Standard system calibrated airspeed	$V_{S_{CAL}}$	$V_{S_{CAL}} = V_{S_{IND}} + \Delta V_S$	kt	Incompressible flow is assumed (41) = (39) + (40)
42	True airspeed	V_T	$V_T = V_{B_{CAL}} / \sqrt{\sigma}$	kt	(42) = (38) / (16)
43	Time	t		sec	From photo panel or manual recording
44	Indicated rate of climb	dh/dt		ft/min	Slope of curve (43) preceding versus (5) in Table 8-4

TABLE 8-5 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
45	Tapeline rate of climb	R/C	$R/C = dh/dt \times K_{H_2}$	ft/min	(45) = (44)·(34)
45a	Climb correction for rotor speed deviation	$\frac{\Delta(R/C)}{\Delta N_R}$		$\frac{\text{ft/min}}{\text{rpm}}$	Slope of R/C vs N_R from sawtooth climbs
46	Standard (aim) rotor speed	N_{R_S}		rpm	
47	Rotor speed deviation	ΔN_R	$\Delta N_R = N_{R_S} - N_R$	rpm	(47) = (46) – (12)
48	Rate of climb correction for rotor speed deviation	$\Delta(R/C)_{\Delta NR}$	$\Delta(R/C)_{\Delta NR} = \frac{\Delta(R/C)}{\Delta N_R} \times \Delta N_R$	ft/min	(48) = (45a)·(47)
49	Scheduled airspeed	V_{SCHED}		kt	From calculated airspeed schedule
50	Climb correction for airspeed deviation	$\frac{\Delta(R/C)}{\Delta V}$		$\frac{\text{ft/min}}{\text{kt}}$	Slope of R/C versus V_T from sawtooth climbs
51	Airspeed deviation	ΔV	$\Delta V = V_{SCHED} - V_T$	kt	(51) = (49) – (42)
52	Rate of climb correction for airspeed deviation	$\Delta(R/C)_{\Delta V}$	$\Delta(R/C)_{\Delta V} = \frac{\Delta(R/C)}{\Delta V} \times \Delta V$	ft/min	(52) = (50)·(51)

TABLE 8-5 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
53	Climb correction for power deviation	$\Delta(R/C)/\Delta SHP$		(ft/min)/HP	Slope of *R/C* versus *SHP* from correction factor climbs
54	Test shaft horsepower	SHP_t		HP	From power calculation tables
55	Power available	SHP_A		HP	From power available calculation table
56	Excess power	ΔSHP	$\Delta SHP = SHP_A - SHP_t$	HP	(56) = (55) – (54)
57	Standard day power available	SHP_S		HP	From standard day power available table
58	Standard (aim) gross weight	GW_S		lb	
59	Gross weight deviation	ΔGW	$\Delta GW = GW_S - GW_t$	lb	(59) = (58) – (33)
60	Gross weight correction	$\Delta(R/C)/\Delta GW$		$\frac{ft/min}{lb}$	Slope of *R/C* versus gross weight curve from correction factor climbs
61	Rate of climb correction for gross weight deviation	$\Delta(R/C)_{\Delta W}$	$\Delta(R/C)_{\Delta W} = \frac{\Delta(R/C)}{\Delta GW} \times \Delta GW$	ft/min	(61) = (60)·(59)

TABLE 8-5 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
62	Power deviation (std day)	ΔSHP_S	$\Delta SHP_S = SHP_S - SHP_t$	HP	(62) = (57) − (54)
63	Power correction	$\Delta(R/C)/\Delta SHP$		(ft/min)/HP	Slope of R/C versus SHP curves from correction factor climbs
64	Rate of climb correction for power deviation	$\Delta(R/C)_{\Delta P}$	$\Delta(R/C)_{\Delta P} = \frac{\Delta(R/C)}{\Delta SHP} \times \Delta SHP_S$	ft/min	(64) = (63)·(62)
65	Corrected rate of climb	$(R/C)_C$	$(R/C)_C = R/C + \Delta(R/C)_{\Delta V} + \Delta(R/C)_{\Delta N_R} + \Delta(R/C)_{\Delta P} + \Delta(R/C)_{\Delta W}$	ft/min	(65) = (45) + (52) + (48) + (64) + (61)
66	Power correction factor	K_P	$K_P = \frac{\Delta(R/C) \times GW_t}{\Delta SHP \times 33000}$		From faired line value curves of SHP vs R/C (66) = $\frac{(63)\cdot(33)}{33{,}000}$
67	Gross weight correction factor	K_W	$K_W = \frac{\Delta(R/C)}{(\Delta SHP) \times 33000\left(\frac{1}{GW_2} - \frac{1}{GW_1}\right)}$		From faired lines from curves of GW vs R/C (67) = $\frac{(63)}{33{,}000\left(\frac{1}{GW_2} - \frac{1}{GW_1}\right)}$

TABLE 8-6

DATA REDUCTION FORM FOR CONTINUOUS CLIMBS

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1-34					See Table 8-4
35-45					See Table 8-5
46	Standard (aim) rotor speed	N_{R_S}		rpm	From photo panel or manual recording
47	Rotor speed deviation	ΔN_R	$\Delta N_R = N_{R_S} - N_{R_T}$	rpm	(47) = (46) − (12)
48	Rotor speed correction	$\Delta(R/C)/\Delta N_R$		$\frac{ft/min}{rpm}$	From slope of R/C vs N_R curve from sawtooth climbs
49	Rate of climb correction for rotor speed deviation	$\Delta(R/C)_{\Delta N_R}$	$\Delta(R/C)_{\Delta N_R} = \frac{\Delta(R/C)}{\Delta N_R} + \Delta N_R$	ft/min	(49) = (48)·(47)
50	Scheduled airspeed	V_{SCHED}		kt	From calculated airspeed schedule
51	Airspeed deviation	ΔV	$\Delta V = V_{SCHED} - V_T$	kt	(51) = (50) − (42)
52	Airspeed correction	$\Delta(R/C)/\Delta V$		$\frac{ft/min}{kt}$	Slope of R/C vs A/S from sawtooth climbs
53	Rate of climb correction for airspeed deviation	$\Delta(R/C)_{\Delta V}$	$\Delta(R/C)_{\Delta V} = \frac{\Delta(R/C)}{\Delta V} \times \Delta V$	ft/min	(53) = (52)·(51)

TABLE 8-6 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
54	Corrected test rate of climb	$(R/C)_C$	$(R/C)_C = R/C + \Delta(R/C)_{\Delta N_R} + \Delta(R/C)_{\Delta V}$		(54) = (45) + (49) + (53)
55	Power correction factor	K_P			From Table 8-5
56	Gross weight correction factor	K_W			From Table 8-5
57	Standard day fuel flow	W_{f_S}		lb/hr	From engine performance calculations
58	Standard day power available	SHP_S		HP	From engine performance calculations
59	Initial altitude	H_{P_1}		ft	Initial value (starting at sea level) from plot of step (54) preceding vs (5), Table 8-4
60	Second altitude	H_{P_2}		ft	Second altitude (at 1000-ft increments) from plot of step (54) preceding vs (5), Table 8-4
61	Altitude increment	ΔH	$\Delta H = H_{P_2} - H_{P_1}$	ft	(61) = (60) – (59)
62	Initial rate of climb	$(R/C)_1$		ft/min	At (59)
63	Second rate of climb	$(R/C)_2$		ft/min	At (60)

TABLE 8-6 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
64	Average rate of climb from H_{p_1} to H_{p_2}	$\overline{(R/C)}_{\Delta H}$	$\overline{(R/C)}_{\Delta H} = [(R/C)_1 + (R/C_2)]/2$	ft/min	(64) = [(63) + (62)] /2
65	Incremental time	Δt	$\Delta t = \Delta H\ \overline{(R/C)}_{\Delta H}$	min	(65) = (61) / (64)
66	Initial fuel flow	W_{f_1}		lb/hr	From plot of W_{f_S} vs altitude at (59) for SHP_S
67	Second fuel flow	W_{f_2}		lb/hr	From plot of W_{f_S} vs altitude at (60) for SHP_S
68	Average fuel flow	$W_{f_{AVG}}$	$W_{f_{AVG}} = (W_{f_1} + W_{f_2})/2$	lb/hr	(38) = [(66) + (67)] /2
69	incremental fuel used	ΔW_{f_S}	$\Delta W_{f_{AVG}} \times \Delta t/60$	lb	(69) = (68)·(65) /60
70	Standard climb start gross weight	GW_{S_o}		lb	From military specifications, model specification, or guarantee
71	Standard gross weight	GW_S	$GW_S = GW_{S_o} - \Delta W_{f_S}$	lb	Plotted versus altitude at median of altitude$_1$ and altitude$_2$ (71) = (70) − (69)

TABLE 8-6 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
72	Gross weight deviation	ΔGW	$\Delta GW = GW_S - GW_t$	lb	(72) = (71) − (33)
73	Rate of climb correction for gross weight deviation	$\Delta(R/C)_{\Delta W}$	$\Delta(R/C)_{\Delta W} = K_W\, SHP_S \cdot 33000 \left(\frac{1}{GW_S} - \frac{1}{GW_t}\right)$	ft/min	$(73) = 33000 \cdot (56) \cdot (58) \left[\frac{1}{(71)} - \frac{1}{(33)}\right]$
74	Test power	SHP_t		HP	From engine performance calculations
75	Power deviation	ΔSHP	$\Delta SHP = SHP_S - SHP_t$	HP	(75) = (58) − (74)
76	Rate of climb correction for power deviation	$\Delta(R/C)_{\Delta P}$	$\Delta(R/C)_{\Delta P} = K_P \frac{\Delta SHP}{GW_t}\, 33000$	ft/min	(76) = 33000 (55) · (75) / (33)

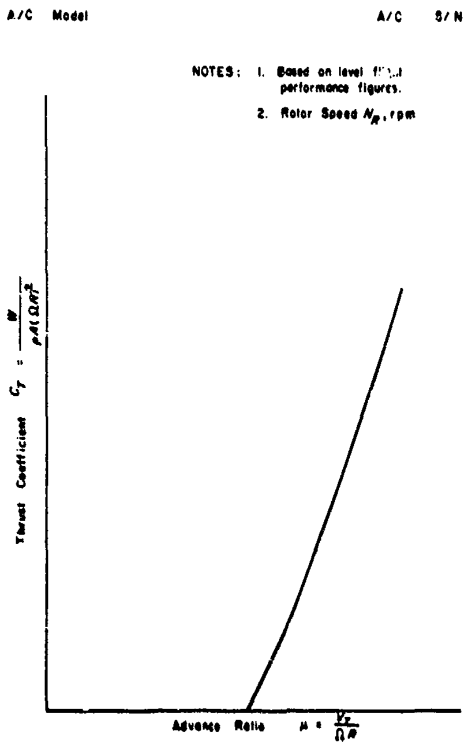


Figure 8-12. Nondimensional Minimum Power Required Performance

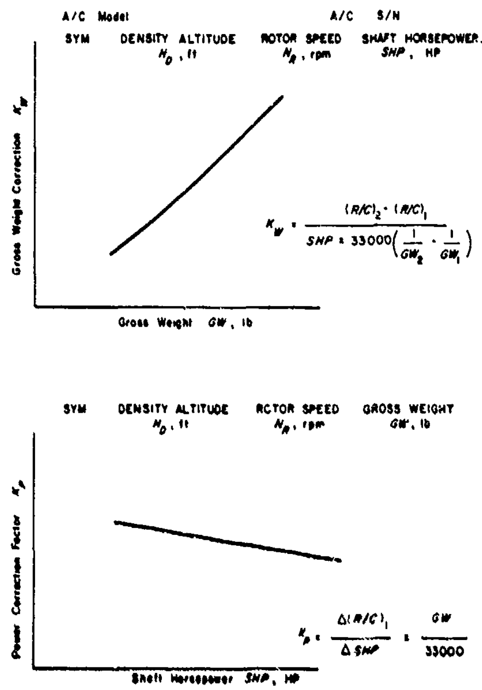


Figure 8-13. Gross Weight and Power Correction Factors

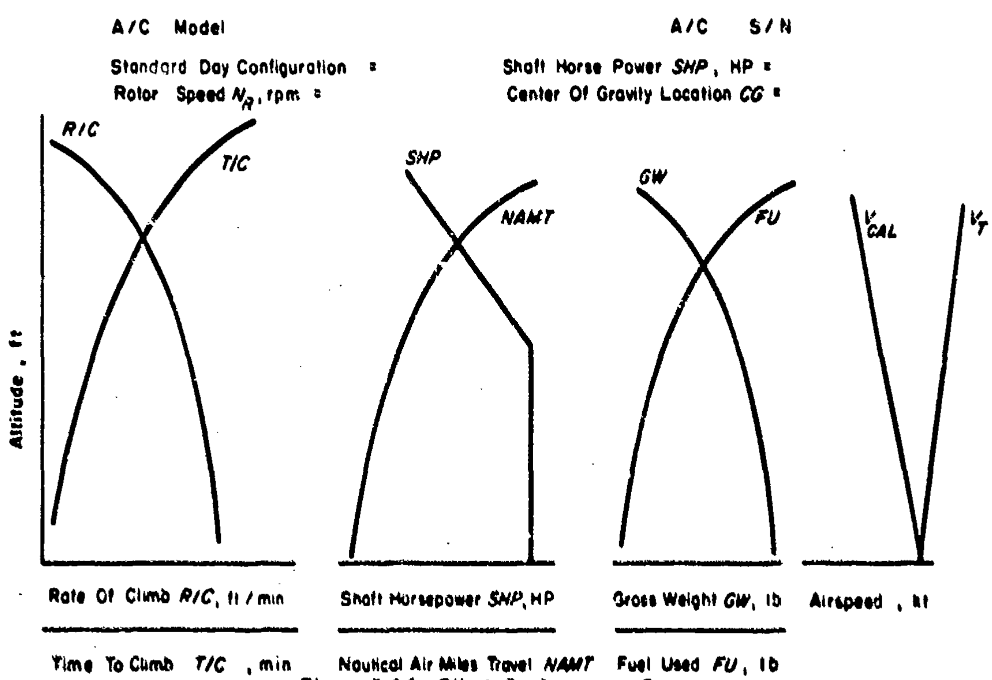


Figure 8-14. Climb Performance Summary

CHAPTER 9

LEVEL FLIGHT PERFORMANCE

9-1 GENERAL

The greatest flight test effort appropriately is devoted to the level flight regime where the helicopter is operated a majority of the time. The level flight data should include all the parameters necessary to define completely the engine and aircraft characteristics throughout the level flight envelope. These parameters may be generalized into two groups: power required and power available. Power required is defined as that power necessary to overcome all of the operating system losses and all the component drag losses at a given flight condition. This power usually is influenced by the aerodynamic configuration, center of gravity location, flight regime, atmospheric conditions, and rotor speed. Aircraft attitudes, blade stall, and compressibility effects also may influence the power required. Power available is determined primarily by the engine operating characteristics in the various flight conditions. It is influenced by factors that change the engine or inlet operating conditions; i.e., attitude, airspeed, rotor speed, and atmospheric conditions. Transient conditions may occur which also affect the power available. This is of particular significance when the effect is detrimental and the maneuver is critical.

Data from the speed power polars are used to determine the power required as a function of center of gravity location, gross weight, airspeed, altitude, rotor speed, and various aerodynamic configurations. The changes in power required may be large or small for each of these conditions. Sufficient tests should be conducted to establish the nature and magnitude of the variations throughout the flight envelope. Special test techniques and conditions may be needed to evaluate adequately blade stall and compressibility effects. The engine parameters recorded during these tests are used to evaluate engine performance and determine the standard day power available during level flight.

The level flight test techniques are perhaps the simplest of those employed during a test program. Since the condition is static, the pilot must use the necessary time and care to stabilize precisely. Pilot comments concerning the suitability of the test vehicle should include its stability, flight control harmony, cockpit presentation, comfort, and operation of the engine controls. These comments are necessary for obtaining an overall rating of the aircraft and can be useful also during the reduction and analysis of the test data. The data obtained during these tests are used to calculate the various summary performance capabilities such as maximum airspeed, range performance, endurance, blade stall limits, specification conformance, guaranteed compliance, and mission suitability.

Data reduction by the nondimensional method is relatively straightforward unless compressibility and blade stall effects are encountered. Pilot comments, vibration, and stress information are used to recognize significant blade stall areas within the data and to account for the phenomenon during data processing. Compressibility is more difficult to detect from in-flight cues and usually is not apparent until the data reduction has been accomplished. It then may be necessary to reprocess the data while utilizing some special data reduction techniques.

9-2 PLANNING

The planning for the level flight performance tests is primarily an engineering effort. Each of the speed power tests is similar in nature and easy to arrange in sequence with respect to gross weight, center of gravity location, altitude, and rotor speed. From the test plan, a complete listing of all the tests should be made in the order they will be conducted. This order should consider the relative efforts required by the maintenance, weight and balance, and data reduction support groups. Consideration also should be given to any safety aspects that might require a build-up procedure. This information is used by the support groups to requisition the necessary materials and schedule the workload. This procedure will insure a timely response to aircraft support requirements, and prompt data reduction and analysis after a test flight.

To minimize the number of data corrections, the in-flight technique must be rigidly controlled and adequately monitored. By assuming a constant C_P, C_T, μ, and data reduction method, a W/ρ graph must be prepared for each speed power polar. The standard test altitude and gross weight should be determined from the test schedule. Then C_T, W/ρ, and altitude are calculated.

Since the standard thrust coefficient C_{T_S} is

$$C_{T_S} = \frac{W_S}{\rho_S A(\Omega R)^2} \tag{9-1}$$

then the standard density ρ_S is

$$\rho_S = \frac{W_S}{C_{T_S} A(\Omega R)^2} \tag{9-2}$$

The variation in weight during the flight is estimated by the weight of fuel required for one hour of operation at normal rated power. The total fuel used is then halved and the differential is added to the helicopter test gross weight to obtain the takeoff gross weight. Engine start, warmup, and takeoff fuel requirements are included, and when a long climb is necessary, an additional allowance should be made for fuel used during the climb. In a similar manner, the gross weight at the end of the test is determined by subtracting the weight differential from the test gross weight. This gross weight range is divided into weight increments at which Eq. 9-2 is solved to yield the proper ρ values for a constant C_T. Altitude tables are entered at these values to obtain the corresponding density altitudes. The points obtained in this manner then are plotted as shown in Fig. 9-1. Since the plot will be used during flight, it must be compatible with the cockpit environment. Since the in-flight gross weight determination requires some calculations, a direct reading instrument such as a fuel used counter should be used. A scale then is constructed which relates the fuel counter reading to weight of fuel used. This scale is based on an average fuel specific weight determined from previous preflight and postflight readings. In the absence of this data, a value of 6.5 lb per gal may be used with confidence. The scale is constructed on a separate piece of paper and attached to the W/ρ chart with paper clips. This method of attaching the scale will allow the zero fuel used index to be moved to account for the usual small variations in engine start gross weight. It also permits the use of the same W/ρ chart for different start gross weights while maintaining the same constant C_T value. This is done by shifting the zero fuel counter index to the new engine start weight GW_1.

Since density altitude is not an indicated parameter in the cockpit, the density altitude obtained from Fig. 9-1 must be converted to the proper pressure altitude at the existing

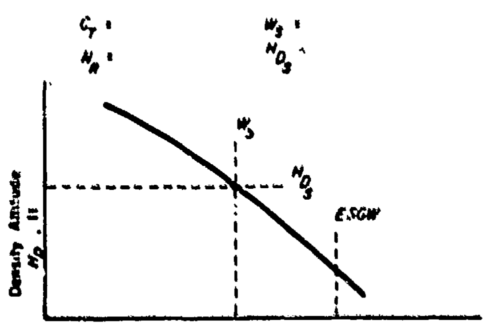


Figure 9-1. Density Altitude and Gross Weight Variation for Constant C_T

ambient temperature conditions. This is accomplished by employing an ambient temperature versus pressure and density altitude chart, such as the one illustrated in Fig. 9-2. Care should be taken that the proper charts are available in the cockpit for any particular flight.

Prior to the test flight, data recording forms must be constructed which have provisions for all the necessary aircraft, atmospheric, and propulsion parameters. A typical data recording card is shown in Table 9-1.

The final portion of the planning process is to brief all the participating and/or responsible personnel. The maintenance personnel are given the fuel loading, aircraft configuration, center of gravity location, internal seating, and any other aircraft requirements. The instrumentation group is provided with the parameter requirements and any postflight processing needs.

The flight profile, schedule, and test conditions are described in the briefing of the flight crew and any chase or pace aircraft support crews. Copies of the flight card are distributed to each active test participant. The takeoff time, location, and flight duration information are covered in the briefing. All support groups such as telemetry, space positioning, or photography are present and briefed as required. It is desirable to have representatives

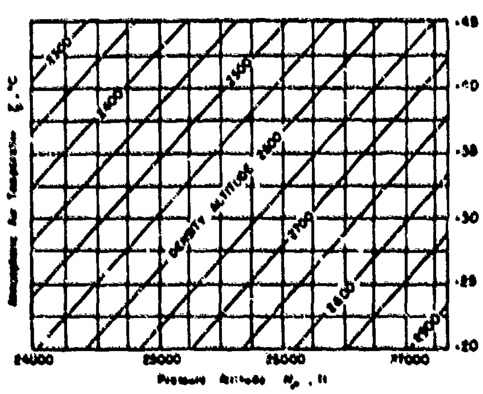


Figure 9-2. Altitude Chart

TABLE 9-1

LEVEL FLIGHT DATA RECORDING CARD

Photo counter number	Engine speed
Oscillograph recording number	Engine torque
	Fuel flow
Altitude	Boom airspeed
Ambient air temperature	Standard system airspeed
Rotor speed	Pitch attitude
Fuel counter	Collective stick position
Fuel temperature	Longitudinal stick position
Pedal position	

from each group present at the common briefing. This will familiarize all members with their duties and their relation to the overall test team. Many operational problems will arise at these meetings, and solutions can be effected which will prevent an interruption of the test or failure to accomplish the objective. In the event this type of meeting is not feasible, each group should be briefed individually.

9-3 INSTRUMENTATION

The level flight performance test parameters are stable, time averaged values with respect to the atmospheric and flight conditions. Other parameters such as engine, inlet, rotor, and stress conditions are transient or cyclic in nature. The minimum instrumentation should measure the averaged power required, and atmospheric and flight conditions. These data can be recorded manually from a visual instrument panel in the cockpit. The required parameters are listed in Table 9-2.

TABLE 9-2

VISUAL INSTRUMENTATION FOR LEVEL FLIGHT PERFORMANCE

Pressure altitude	Engine speed
Ambient air temperature	Engine torque
Fuel used	Engine temperature
Fuel flow	Engine inlet temperature
Rotor speed	Engine inlet pressure
Airspeed (standard and test system)	

Manual recording of these parameters will allow the data reduction to be accomplished from the flight card rather than reading the photo panel or oscillograph. This will reduce the data processing workload and provide more timely results.

A photo panel may be used which greatly reduces the flight personnel workload and provides a higher data density capability. Although the data reduction workload will be increased when data are recorded in this manner, the inlet conditions and other dynamic parameters are available as a function of time. Engine and flight control parameters should be added to those listed in Table 9-2 for photo panel recording.

An oscillograph or magnetic tape system is most desirable for recording dynamic parameters. With either of these, the data reduction effort is much less than that required to read and plot high speed camera photographic (photo panel) data. From these continuous recordings, any desired data density can be read from the traces. the instrumentation usually placed on the oscillograph is shown in Table 9-3.

TABLE 9-3

OSCILLOGRAPH INSTRUMENTATION FOR LEVEL FLIGHT PERFORMANCE

Rotor speed	Engine torque
Engine speed	Throttle position
Engine inlet temperature	Longitudinal stick position
Engine inlet pressure	Lateral stick position
Fuel flow	Pedal position
Event marker	Collective stick position

Additional parameters peculiar to the test vehicle or requirements, of course, should be added to these basic lists.

9-4 TEST METHODS

9-4.1 GENERAL

The speed power tests are conducted to determine the total power required and the engine charateristics as a function of airspeed during stabilized level flight conditions. Some special techniques are necessary to evaluate vibration, compressibility, and blade stall effects. To define properly the power required variations with airspeed, the aircraft must be stabilized precisely with no acceleration about any axis. Since the linear accelerations are usually subtle, they are the most significant. The pilot normally can recognize and correct for angular accelerations. The pilot must insure that all of the many parameters are stable, which largely accounts for the relatively long period of time required to stabilize the airframe or power plant. Failure to stabilize the aircraft properly will cause excessive data scatter and may result in erroneous data.

9-4.2 LEVEL FLIGHT PERFORMANCE

The normal procedure for conducting the speed power tests is for the pilot to have an aim rather than an exact airspeed for each point. Power is added slowly until the aircraft is near the aim speed. The power then is fixed while the pilot adjusts rotor speed, altitude, and airspeed. The desired parameters are considered stabilized when the pilot and flight crew verify that the indicators are not fluctuating. When a steady-state condition is obtained, the data should be recorded while the flight condition is monitored. The stabilized condition is then maintained for at least 2 min to allow the propulsion system parameters to become stable. This time does not cause a delay since the manual data recording procedure normally will require this period of time to record all the necessary parameters. Of course, automatic data recording systems are much more rapid, although the propulsion system and pilot requirements are unchanged. As a result, the flight time required per point is generally independent of the data recording system being used. While monitoring the flight condition, the pilot can determine the correct airspeed for the power setting. This method is preferred rather than attempting to obtain an exact airspeed. Many small power adjustments may be necessary to achieve an exact airspeed with a corresponding increase in time required to stabilize at each condition.

Various in-flight techniques are used for recording the data. Each of these has merit, although only one method will be discussed in detail. As soon as the power is fixed, fuel flow timing should be initiated and all power parameters are recorded as rapidly as possible. Meanwhile, the pilot monitors airspeed and altitude to simplify recording the data. Following the power measurement, atmospheric conditions and propulsion system operating conditions are of utmost importance. When all parameters have been recorded, the most critical readings such as power parameters should be checked to insure they were constant during the data gathering period. The best airspeed value for the power condition is determined by the pilot.

For the constant C_T method, the pilot must be given a new altitude for each successive point. This procedure is accomplished as follows:

1. Determine the current gross weight from the fuel totalizer reading and the engine start gross weight.

2. Estimate the fuel consumption that will occur while the pilot stabilizes at the next test condition.

3. From steps 1 and 2, determine the approximate gross weight for the test point.

4. Enter the W/ρ chart (Fig. 9-1) to obtain the proper density altitude for the test gross weight.

5. From the test ambient air temperature (instrument corrected) and the density altitude from step 4, enter the altitude chart (Fig. 9-2) to obtain the proper pressure altitude.

6. Apply the instrument correction to the pressure altitude from step 5 and give the proper indicated altitude value to the pilot. Since the test requires being at the proper test altitude for each point, it is important that these procedures be accomplished with a minimum of delay. The flight test engineer should have the necessary items properly arranged with a procedure established that quickly will allow this altitude determination. The time required to stabilize also can be reduced by the pilot anticipating the next aim airspeed and knowing that an altitude increase of approximately the same magnitude as the previous points will be necessary.

The speed increment between points will vary with different aircraft and possibly for any given speed power test. The speed range available is the most important item when determining the airspeed increment. The minimum number of points per test is eight, regardless of the envelope magnitude. The curve must be defined adequately in all areas, particularly at minimum power, and maximum and minimum airspeed. In addition, any unusual flight characteristics discovered during the course of the test should be investigated thoroughly. Failure to accomplish this may require repeating the test at a later date. No attempts should be made to obtain data in the airspeed range from 0 to 20 kt. These data are of minimum importance and are not needed to define the level flight performance. In addition, the airspeed system is usually inoperative in this area, and the pilot has no means to establish accurately a steady-state condition. Stability and control characteristics usually add to the difficulty in stabilizing in this area.

The speed power tests are conducted for all operating conditions which should reveal any blade stall areas within the envelope. Blade stall generally is characterized by increased vibration, control feedback forces, control power deterioration, increased stress loads, and power required. Some degree of blade stall is nearly always present, and many times the best range performance is obtained while operating in an incipient stall area. This type of stall usually is not evident to the pilot and does not change significantly the performance characteristics. When sufficient blade stall data are not obtained during the normal speed power tests, specific additional blade stall tests may be required. These are accomplished by varying the rotor speed and airspeed at different altitudes. At a given altitude and rotor speed, the airspeed is increased until the pilot considers that the first signs of blade stall are present. The data and pilot comments are recorded at this point, and then airspeed is increased slowly until the next stall condition is reached. Generally these stall areas are well defined as mild, moderate, or severe. The procedure is repeated until the limiting stall factor is encountered. Caution must be used to avoid increasing airspeed at a rate that introduces acceleration effects.

9-4.3 BLADE STALL

Since vibration loads and dynamic stresses increase with blade stall, recording these parameters will aid greatly in determining the stall areas and envelope limits. For aircraft with unboosted control systems, the feedback forces - in addition to the vibration - will allow the pilot to define accurately the degree of stall present. A highly boosted control system without control force feedback may preclude the use of pilot opinion to determine the blade stall characteristics. This is particularly the case for incipient and mild stall where vibration increases are small. It is then necessary to instrument certain portions of the rotor and dynamic components to record operating conditions and stress levels. This method has been used with moderate success since the accuracy and sensitivity leaves something to be desired. For some types of aircraft, this instrumentation may be installed as standard equipment with the stress level presented on an indicator mounted on the pilot panel to serve as an airspeed or envelope limiting indicator. Monitoring stress in this manner increases flight safety and allows a longer life to be realized for the dynamic components.

Control power deterioration may occur when severe blade stall conditions are reached. Pilot comments, in addition to stress and stability data, are required to evaluate the results of this type of stall condition. The aircraft motion resulting from a control power deterioration will vary with aircraft configuration and type as well as the operating conditions. The maximum airspeed usually will be limited by power, vibration, or stress considerations before such a control power deterioration condition is reached.

Advancing blade compressibility effects generally are not apparent qualitatively to the pilot. Analysis of the stress and power required data generally will show the existence and magnitude of this phenomenon. As previously discussed in the speed power tests, normal tests will reveal the compressibility areas. In the event it is necessary specifically to investigate compressibility, the most common method is to vary incrementally rotor speed at the maximum airspeed and at the coldest ambient temperature available. This will show the effects of the rotor tip speed on power required at the highest Mach number. Although not clearly defined by current literature or test data, the retreating blade also may be experiencing Mach number effects caused by early drag divergence at the very high blade angles. In this instance, the blade stall/Mach number relationship is not known and test or correction methods are not established.

9-4.4 MAXIMUM AIRSPEED

The maximum level flight speed is an important performance characteristic for any aircraft and must be determined accurately. This value normally is calculated from the data obtained during the speed power tests. When the maximum airspeed capability is a guarantee point or when mission suitability is in question, actual demonstration points should be obtained rather than rely on calculations. For this type of test, extreme care should be taken to insure the proper gross weight, center of gravity location, altitude, rotor speed, and power setting during data recording. The takeoff gross weight must include sufficient fuel to takeoff, climb, and stabilize at the test altitude. All other conditions must be met precisely as specified in the performance guarantee documentation. The results of this test then are used to validate the previously calculated maximum airspeed.

9-4.5 RANGE

Range missions are conducted primarily to validate the level flight power required and fuel flow data. The altitude, airspeed, and range are calculated and the range mission then is conducted to determine the accuracy of the prediction. This initially should be accomplished in atmospheric conditions closely approximating those that existed during the speed power tests. Since the helicopter normally is operated in other types of environments, it is worthwhile to repeat some of the range missions in a more representative operational atmosphere. The sensitive test instrumentation is used, and the pilot attempts to maintain precise airspeed and altitude schedules. The resulting data will show the effects of turbulence. The final and perhaps most important test is to determine the range performance that actually will be realized by the operational pilot. This is best accomplished by conducting the same tests with standard instruments at a pilot effort level representative of operational pilots. The performance realized during this test will include all losses caused by variations in pilot technique as well as nonoptimum atmospheric conditions and instrument errors.

9-5 DATA REDUCTION

9-5.1 GENERAL

The steady-state level flight performance data are the most suitable for automatic processing of all the different types of data obtained during a performance program. This is because there is both a large amount of data and the opportunity for similar processing of each speed power polar. The initial data reduction effort is to determine the test power required, fuel flow, atmospheric conditions, and airspeed conditions. Total power required is the sum of the drag and system losses. Fuel flow, in conjunction with airspeed and atmospheric conditions, determines the range performance. These basic test data are also used for blade stall, vibration, maximum airspeed, endurance, and many other calculations. These items will be discussed in detail.

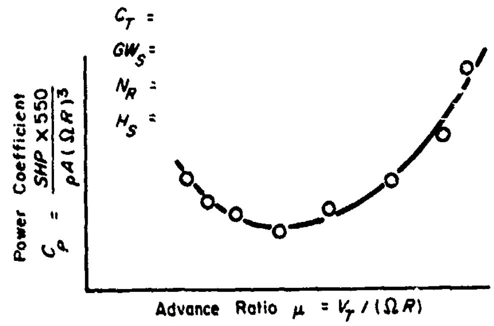


Figure 9-3. Nondimensional Level Flight Performance

9-5.2 LEVEL FLIGHT PERFORMANCE

The method presented here is the nondimensional C_P, C_T, and μ method. This method is valid only when there are no significant Mach number or blade stall effects. Although existing theories predict the occurrence of these effects, the accuracy of the predictions is somewhat questionable. The best approach is to assume initially that the effects are small. Experience has shown that large effects will cause an excessive amount of scatter in the data and special methods will then be required. A level flight performance data reduction form is shown in Table 9-4.*

The basic reduction method contained in Table 9-4 treats each point as a separate entity within each flight. The power and weight are corrected to constant altitude, temperature, and rotor speed values for each speed power polar. These speed powers are plotted collectively in a three-dimensional plot (carpet plot) that describes the total level flight performance map. The data initially are plotted in nondimensional advance ratio, thrust coefficient, and power coefficient form as shown in Fig. 9-3.

A best fit curve then is faired through the points. This procedure is accomplished for each speed power polar.

*The data reduction forms (tables) are located at the end of each chapter.

The individual plots illustrated in Fig. 9-3 are entered at even μ values and the C_P value is recorded. The plot shown in Fig. 9-11 is now entered at the appropriate C_T, and the C_P values are plotted as illustrated. As previously stated, each flight has been corrected to a unique, constant C_T value. The C_P scale is staggered to allow sufficient space between the constant μ lines, which are faired after all the data have been plotted. A good planning effort should produce data which encompass the maximum C_T range available. This plot constitutes a total power required performance map and can be used to calculate the various types of performance–as will be illustrated.

The airspeed for best climb performance will be at the minimum power required condition which is also where greatest excess power is available. These conditions are determined from the level flight performance data. The nondimensional speed powers are plotted as shown in Fig. 9-4.

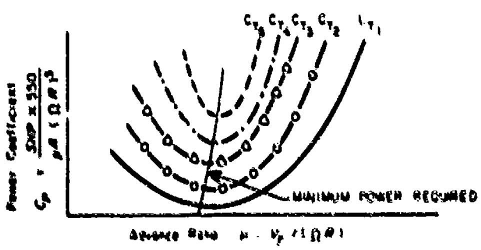


Figure 9-4. Nondimensional Speed-Power

For each curve, the minimum power required point is determined, and a curve then is faired through these points. The line determines the variation of minimum power required with the airspeed, altitude, gross weight, and rotor speed. Since power is of no further concern, the minimum power curve in Fig. 9-4 then is replotted as shown in Fig. 9-5 and used to calculate climb schedules for the desired conditions. Note that nonstandard temperatures can be used to calculate performance degradation for hot day conditions. It is also important to realize that these data are valid only for forward speed climbs and should not be used for the vertical case.

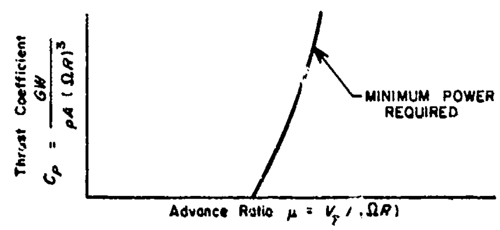


Figure 9-5. Nondimensional Airspeed for Minimum Power Required

Gross weight, altitude, temperature, and rotor speed values are used to calculate C_T and then determine μ from the faired line. From the μ value and the airspeed calibration data, an indicated airspeed can be determined which will provide the best climb performance. The data reduction forms for this summary calculation are presented in Tables 9-5 and 9-6*.

Best endurance is obtained at minimum fuel flow, which occurs at minimum power required and is thus the same conditions as for best climb performance. In this case, power is of importance and the data from Fig. 9-4 is plotted as shown in Fig. 9-6.

These figures are then used to obtain values for calculating the endurance airspeed and power required. Inlet correction factors for the given airspeeds are obtained from Fig. 13-2. From these and the atmospheric pressure and temperature, engine correction factors δ_{T_1} and $\sqrt{\theta_{T_1}}$ then are calculated. Corrected power $SHP/(\delta_{T_1}\sqrt{\theta_{T_1}})$ then is determined and used to enter the engine characteristic curve, Fig. 14-19, and obtain the corrected fuel flow $W_f/(\delta_{T_1}\sqrt{\theta_{T_1}})$. This then is used to calculate fuel flow at the specified conditions, and, in conjunction with the fuel available, an endurance is established. A data reduction form for calculating the endurance is shown in Table 9-6.*

9-5.3 BLADE STALL

During the course of the level flight performance tests, the blade stall points should

*The data reduction forms (tables) are located at the end of each chapter.

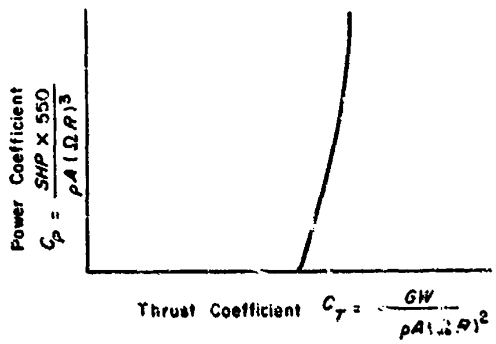


Figure 9-6. Nondimensional Power Required at Best Endurance Airspeed

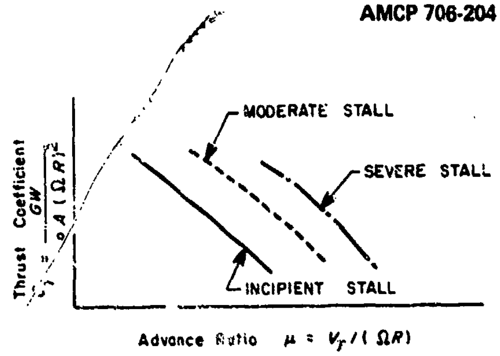


Figure 9-7. Nondimensional Blade Stall

have been identified qualitatively by pilot comment. Retreating blade pitch angle, stress level, and vibration data are quantitative methods that are used often to determine when blade stall conditions exist. For the points identified as blade stall, the nondimensional data are plotted as shown in Fig. 9-7.

A curve is fitted through the test points and the line then is transferred to the nondimensional carpet plot as shown in Fig. 9-11. The line then represents an airspeed limit based on the allowable level of blade stall. Vibration and stress considerations can be treated in much the same way to determine the maximum speed limits.

9-5.4 MAXIMUM AIRSPEED

The maximum airspeed capability is a most important characteristic of any air vehicle. This is calculated from the nondimensional level flight power required data shown in Fig. 9-11 and the power available information shown in Fig. 14-19. The calculations should be made for maximum allowable power, normal rated power, maximum continuous power, and cruise power. In most cases some of these will be identical but on other occasions different definitions may be used. The significant ones usually are defined in the model specification, test plan, or guarantees documentation. Gross weight, altitude, and rotor speed are used to calculate the thrust coefficient. The specified power then determines the power coefficient that is available. Entering Fig. 9-11 at the C_T and C_p available values will provide the resultant μ, from which the true airspeed is calculated. The μ value rarely will be an even value, and a graphical interpolation of the Fig. 9-11 will be required. This can be accomplished most easily by using a Gerber (variable) Scale to construct additional μ lines. A summary calculation form is presented in Table 9-7*.

The resulting airspeed envelope may exceed the desired limits from a blade stall, stress, or vibration viewpoint. Envelopes for these limitations may be calculated in a similar manner by selecting the μ for the appropriate parameter boundary shown in Fig. 9-11. The data also should be checked to insure that stability and control factors will not prevent attaining the calculated performance.

9-5.5 RANGE

The range performance is calculated from the nondimensional power required in Fig. 9-11, the inlet performance shown in Table 13-1, and the fuel flow characteristics from Fig. 14-19. Specific range is defined as the nautical air miles that can be traveled per pound of fuel used. The common practice is to include some conservatism by using 0.99 $NAMPP_{MAX}$ on the high speed side of the curve. Range usually is required for a number of conditions, and it is best to calculate a summary performance map. Pertinent variables are rotor speed, gross weight, altitude,

*The data reduction forms (table) are located at the end of each chapter.

center of gravity location, aerodynamic configuration, and airspeed. It is most convenient to calculate the performance for each of the variables in terms of altitude, gross weight, and airspeed. Rotor speeds are generally the normal operating speed, the maximum speed, and the minimum speed. Aerodynamic configurations include stores, gear, doors open, and other significant changes for which a power required change was noted. Changes in power required with center of gravity also will necessitate additional range calculations. A typical calculation procedure is shown in Table 9-8*.

As previously noted, the calculations are made for a sufficient range of conditions to allow an entire map to be constructed. The specific range values are plotted as shown in Fig. 9-8, for each flight.

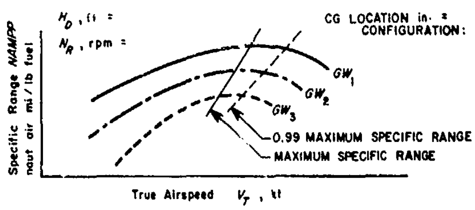


Figure 9-8. Specific Range Summary

Lines are faired through the data points. Values are read and multiplied by 0.99. The resultant then is plotted on the individual gross weight curves and a line faired through the points. This establishes the recommended cruise speed. When this line is replotted as shown in Fig. 9-12, specific range for any gross weight may be obtained.

The illustration shown in Fig. 9-8 is for gross weight; however, a similar treatment can be used for rotor speed, altitude, center of gravity locations, and various configurations. In some instances, it may be necessary to obtain a range for specified limiting factors such as airspeed, power, or blade stall. The method is the same with the exceptions that the limiting value is that shown in Fig. 9-11. For power, C_P will be limited; for airspeed, μ becomes a limit; and with blade stall, vibration may impose either a power available or a μ limit.

9-6 DATA PRESENTATION

Certain of the plots used during the reduction process should be shown in the result of test reports. Also some of the original data should be shown so that other calculations can be made by interested readers. Typical plots that normally are presented are illustrated in Figs. 9-9 through 9-14. Plots such as Fig. 9-9 can be made for each speed power polar. It also may be necessary to modify the title of Fig. 9-13 for the appropriate data being shown such as blade stall, power available, stress, or compressibility.

*The data reduction forms (tables) are located at the end of each Chapter.

TABLE 9-4

DATA REDUCTION FORM FOR LEVEL FLIGHT PERFORMANCE

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1	Point No./Flight No.				
2	Indicated boom pressure altitude	$H_{P_{IND}}$		ft	From photo panel or cockpit recorded data
3	Boom altitude instrument correction	$\Delta H_{P_{IC_B}}$		ft	From instrument calibration curves
4	Boom altitude position error	$\Delta H_{P_{PE_B}}$		ft	From airspeed calibration curves
5	Boom pressure altitude	H_{P_B}	$H_{P_B} = H_{P_{IND_B}} + \Delta H_{P_{IC_B}} + \Delta H_{P_{PE_B}}$	ft	(5) = (2) + (3) + (4)
6	Indicated boom airspeed	V_{IND_B}		kt	From photopanel or cockpit recorded data
7	Boom airspeed instrument correction	ΔV_{IC_B}		kt	From instrument calibration curves
8	Boom airspeed position error	ΔV_{PE_B}		kt	From airspeed calibration curves
9	Calibrated boom airspeed	V_{CAL_B}	$V_{CAL_B} = V_{IND_B} + \Delta V_{IC_B} + V_{PE_B}$	kt	(9) = (6) + (7) + (8)
10	Indicated ambient air temperature	$T_{a_{IND}}$		°C	From photo panel or cockpit recorded data
11	Instrument correction for ambient air temperature	$\Delta T_{a_{IC}}$		°C	From instrument calibration curves

TABLE 9-4 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
12	Instrument corrected ambient air temperature	$T_{a_{IC}}$	$T_{a_{IC}} = T_{a_{IND}} + \Delta T_{a_{IC}}$	°C	$(12) = (10) + (11)$
13	Speed of sound	a		kt	Calculated or from tables at (12)
14	Density altitude	H_D		ft	Calculated or from tables at (5) and (12)
15	Square root of air density ratio	$\sqrt{\sigma}$			Calculated or from tables at (14)
16	True airspeed	V_T	$V_T = V_{CAL}/\sqrt{\sigma}$		Incompressible flow is assumed $(16) = (9) / (15)$
17	Mach number	M	$M = V_T/a$		$(17) = (16) / (13)$
18	Ambient air temperature	T_a	$T_a = \dfrac{T_{a_{IC}} + 273}{1 + K\dfrac{M^2}{5}} - 273$	°C	K is the recovery constant for probe $(18) = \dfrac{(12) + 273}{1 + K\dfrac{(17)^2}{5}} - 273$
19	Indicated rotor speed	$N_{R_{IND}}$		rpm	From photo panel or cockpit recorded data
20	Rotor speed instrument correction	$\Delta N_{R_{IC}}$		rpm	From instrument calibration curves
21	Rotor speed	N_R	$N_R = N_{R_{IND}} + \Delta N_R$	rpm	$(21) = (19) + (20)$

TABLE 9-4 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
22	Preflight fuel specific weight	FS_1		lb/gal	From instrumentation preflight
23	Postflight fuel specific weight	FS_2		lb/gal	From instrumentation postflight
24	Average fuel specific weight	FS_{AVG}	$FS_{AVG} = (FS_1 + FS_2)/2$	lb/gal	(24) = [(22) + (23)]/2
25	Fuel counter difference	ΔFC	$\Delta FC = FC_2 - FC_1$	ct	From photo panel or cockpit recording
26	Fuel counter constant	K_{FC}		ct/gal	From instrumentation calibration curves average value used
27	Volume of fuel used	FU_{VOL}	$FU_{VOL} = \Delta FC \times K_{FC}$	gal	(27) = (25)·(26)
28	Weight of fuel used	FU_W	$FU_W = FU_{VOL} \times FS$	lb	(28) = (27)·(24)
29	Gross weight	GW	$GW = ESGW - FU_W$	lb	$ESGW$ from preflight form (29) = $ESGW$ – (28)
30	Blade rotational tip speed	ΩR	$\Omega R = \frac{2\pi}{60} \times N_R \times R$	ft/sec	(30) = $\frac{2\pi}{60}$·(21)·R
31	Advance ratio	μ	$\mu = 1.69V_T/(\Omega R)$		(31) = 1.69·(16) / (30)
32	Advancing blade tip speed	V_{TIP}	$V_{TIP} = \frac{\Omega R}{1.69} + V_T$	kt	(32) = (30) / 1.69 + (16)

TABLE 9-4 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
33	Advancing blade Mach number	M_B	$M_B = V_{TIP}/a$		(33) = (32) / (13)
34	Maximum retreating blade angle	θ_B		deg	From oscillograph recording
35	Azimuth for maximum blade angle	ψ		deg	From oscillograph recording
36	Thrust coefficient constant	K_{c_T}	$K_{c_T} = A(\Omega R)^2$		Calculated at (30) or from tables (36) = A $(30)^2$
37	Power coefficient constant	K_{c_p}	$K_{c_p} = 550/[A(\Omega R)^3]$		Calculated at (30) or from tables (37) = 550 / [A $(30)^3$]
38	Air density	ρ		slug/ft^3	Calculated or from tables at (14)
39	Thrust coefficient	C_T	$C_T = GW/(\rho K_{c_T})$		(39) = (29) / [(36)·(38)]
40	Test power required	SHP_t		HP	From engine performance Table 14-1
41	Power coefficient	C_P	$C_P = SHP_t \times K_{c_p}/\rho$		(41) = (40)·(37) / (38)
42	Aim thrust coefficient	C_{T_S}			At N_{R_S}, H_{D_S}, T_S
43	Thrust coefficient correction	ΔC_T	$\Delta C_T = C_{T_S} - C_T$		(43) = (42) − (39)

TABLE 9-4 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
44	Power coefficient correction factor	$\Delta C_p/\Delta C_T$			Slope from carpet plot at (39)
45	Power coefficient correction	ΔC_p	$\Delta C_p = \Delta C_p/\Delta C_T \times C_T$		(45) = (44)·(43)
46	Standard power coefficient	C_{P_S}	$C_{P_S} = C_P + \Delta C_P$		(46) = (41) + (45)
47	Standard gross weight	GW_S	$GW_S = C_{T_S} \times K_{C_T} \times \rho_S$	lb	ρ_S at H_{D_S} (47) = (42)·(36) ρ_S
48	Standard day power	SHP_S	$SHP_S = SHP_t \times \rho_S/\rho$	HP	(48) = (40) ρ_s / (38)
49	Fuel used	FU		ct	From photo panel or cockpit recorded data
50	Elapsed fuel flow time	Δt		sec	From photo panel or cockpit recording
51	Test fuel flow	W_{f_t}	$W_{f_t} = FU/\Delta t$	ct/sec	(51) = (49) / (50)
52	Fuel counter constant	K_{FC}		gal/ct	At (51) from fuel flow calibration curves
53	Fuel temperature	T_f		°C	From photo panel or cockpit recording
54	Fuel specific weight	FS_3		lb/gal	At (53) from method shown in Chapter 14, Eq. 14-11
55	Volumetric fuel flow	$W_{f_{VOL}}$	$W_{f_{VOL}} = W_{f_t} \times K_{FC}$	gal/sec	(55) = (51)·(52)

TABLE 9-4 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
56	Fuel flow	W_f	$W_f = W_{f_{VOL}} \times FS_3$	lb/sec	(56) = (55)·(54)
57	Inlet total pressure	P_{T_2}		psi	From inlet pressure recording
58	Inlet air pressure ratio	δ_{T_2}	$\delta_{T_2} = P_{T_2} / P_0$		(58) = (57) / 14.7
59	Inlet total temperature	T_{T_2}		°C	From average inlet temperature recording
60	Inlet temperature ratio	θ_{T_2}	$\theta_{T_2} = \dfrac{T_{T_2} + 273}{T_0}$		$(60) = \dfrac{(59) + 273}{288}$
61	Standard pressure	P_S		psi	Pressure at (42)
62	Standard pressure ratio	δ_S	$\delta_S = P_S/14.7$		Calculated or from tables at (61)
63	Standard temperature	T_S		°C	Temperature at (42)
64	Standard temperature ratio	θ_S	$\theta_S = \dfrac{T_S + 273}{T_0}$		$(64) = \dfrac{(63) + 273}{288}$
65	Referred fuel flow	$W_{f_{T_R}}$	$W_{f_{T_R}} = W_f / (\delta_{T_2} \sqrt{\theta_{T_2}})$	lb/hr	(65) = (56) / [(58) · $\sqrt{(60)}$]
66	Standard day fuel flow	W_{f_S}	$W_{f_S} = W_f / (\delta \sqrt{\theta}) \times \delta_S \sqrt{\theta_S}$	lb/hr	$W_f / (\delta\sqrt{\theta})$ — from referred fuel flow curve at (48)
67	Specific range	$NAMPP$	$NAMPP = V_T / W_f$	Nautical air miles per pound of fuel	(67) = (16) / [(56) 3600]

TABLE 9-5

DATA REDUCTION FORM FOR BEST CLIMB SPEED CALCULATION

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1	Point No./Flight No.				
2	Gross weight	GW		lb	Selected from guarantee or mission considerations
3	Rotor speed	N_R		rpm	Selected from power or specified conditions
4	Pressure altitude	H_p		ft	Selected from guarantee or mission considerations
5	Ambient air temperature	T_a		°C	
6	Density altitude	H_D		ft	Calculated or from tables at (4) and (5)
7	Rotor tip speed	V_{TIP}	$V_{TIP} = \Omega R$	ft/sec	Calculated at (3)
8	Thrust coefficient constant	K_{c_T}	$K_{c_T} = A\,(\Omega R)^2$	lb/slug-ft^3	Calculated at (3)
9	Air density	ρ		slug/ft^3	Calculated or from tables at (4)
10	Thrust coefficient	C_T'	$C_T = GW / (K_{c_T}\rho)$		(9) = (2) / [(8)·(7)]
11	Advance ratio	μ	$\mu = V_T / (\Omega R)$		From Fig. 9-5

TABLE 9-5 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
12	True airspeed	V_T	$V_T = \mu \times \Omega R / 1.69$	kt	(12) = (11)·(7) / 1.69
13	Square root of air density ratio	$\sqrt{\sigma}$			Calculated or from tables at (6)
14	Calibrated airspeed	V_{CAL}	$V_{CAL} = V_T \times \sqrt{\sigma}$	kt	Incompressible flow assumed (14) = (12)·(13)
15	Airspeed position error	ΔV_{PE}		kt	From airspeed calibration curve at (14)
16	Instrument corrected airspeed	V_{IC}	$V_{IC} = V_{CAL} + \Delta V_{PE}$	kt	(16) = (14) + (15)
17	Airspeed instrument correction	ΔV_{IC}		kt	From instrument calibration curves at (16)
18	Indicated airspeed	V_{IND}	$V_{IND} = V_{IC} + \Delta V_{IC}$	kt	(18) = (16) + (17)

TABLE 9-6

DATA REDUCTION FORM FOR ENDURANCE CALCULATIONS

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1-17					See Table 9-5
18	Power coefficient constant	K_{c_p}	$K_{c_p} = 550 / [A(\Omega R)^3]$	slug/ft^3—HP	Calculated at selected value of N_R
19	Power coefficient required	C_p	$C_p = SHP \times K_{c_p} / \rho$		From Fig. 9-6 at (9) [(19) / (18)] · (9)
20	Power required	SHP	$SHP = (C_p / K_{c_p}) \rho$	HP	(20) = [(19) / (18)] · (9)
21	Ambient air pressure	P_a		in. Hg	Calculated or from Tables at (4)
22	Ambient air pressure ratio	δ_a	$\delta_a = P_a / 29.92$		(22) = (21) / 29.92
23	Square root of ambient air temperature ratio	$\sqrt{\theta}$	$\sqrt{\theta} = \sqrt{(T_a + 273) / 288.15}$		Calculated or from Tables at (5) (23) = $\sqrt{[(5) + 273] / 288.15}$
24	Referred power required	SHP_{REF}	$SHP_{REF} = SHP / (\delta_a \sqrt{\theta})$	HP	(24) = (20) / [(22)·(23)]
25	Referred fuel flow	$W_{f_{REF}}$	$W_{f_{REF}} = W_f / (\delta_a \sqrt{\theta})$	lb/hr	From engine characteristics curve, Fig. 14-19 at (24)
26	Fuel flow required	W_f	$W_f = W_{f_{REF}} \times \delta_a \sqrt{\theta}$	lb/hr	(26) = (25)·(22)·23

TABLE 9-6 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
27	Fuel available	*WF*		lb	From fuel tank capacity and calculated fuel allowances
28	Endurance	*E*	$E = WF/W_f$	hr	(28) = (27) / (26)
29	Distance traveled	*NAMT*	$NAMT = V_T \times E$	n mi	(29) = (12)·(28)

TABLE 9-7

DATA REDUCTION FORM FOR CALCULATION OF MAXIMUM AIRSPEED

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1	Pressure altitude	H_p		ft	Selected or from specified conditions
2	Ambient air temperature	T_a		°C	Selected or from specified conditions
3	Density altitude	H_D		ft	Calculated or from tables at (1) and (2)
4	Air density	ρ			Calculated or from tables at (3)
5	Rotor speed	N_R		rpm	Selected or from specified conditions
6	Power coefficient constant	K_{c_P}	$K_{c_P} = 550 / [A(\Omega R)^3]$	slug/ft^3-HP	Calculated at (5)
7	Thrust coefficient constant	K_{c_T}	$K_{c_T} = A(\Omega R)^2$	lb/slug-ft^3	Calculated at (5)
8	Advancing blade tip speed	V_{TIP}	$V_{TIP} = \Omega R$	ft/sec	Calculated at (5)
9	Gross weight	GW		lb	Selected or from specified conditions
10	Power available	SHP_A		HP	From power available calculations Table 14-1

TABLE 9-7 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
11	Thrust coefficient	C_T	$C_T = GW / (\rho K_{c_T})$		(11) = (9) / [(4)·(7)]
12	Power coefficient	C_P	$C_P = K_{c_P} SHP_A / \rho$		(12) = (6)·(10) / (4)
13	Advance ratio	μ	$\mu = 1.69 V_T / (\Omega R)$		Obtained from level flight power required, Fig. 9-11, at (11) and (12)
14	True airspeed	V_T	$V_T = \mu \times 0.59\ \Omega R$	kt	(14) = (13)·0.59·(8)

TABLE 9-8

DATA REDUCTION FORM FOR SUMMARY RANGE PERFORMANCE

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1	Pressure altitude	H_p		ft	Selected or from specified conditions
2	Ambient air temperature	T_a		°C	Selected or from specified conditions
3	Density altitude	H_D		ft	Calculated or from tables at (1) and (2)
4	Air density	ρ		slug/ft^3	Calculated or from tables at (3)
5	Ambient air pressure	P_a		lb/in.^{2}	Selected or from specified conditions
6	Ambient air pressure ratio	δ_a	$\delta_a = P_a/P_o$		P_o = 14.7 psi (6) = (5) / 14.7
7	Gross weight	GW		lb	Selected or from specified conditions
8	Rotor speed	N_R		rpm	Selected or from specified conditions
9	Advancing blade tip speed	V_{TIP}	$V_{TIP} = \Omega R$	ft/sec	Calculated at (8)

TABLE 9-8 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
10	Thrust coefficient constant	K_{c_T}	$K_{c_T} = A(\Omega R)^2$	lb/slug-ft^3	Calculated at (8)
11	Power coefficient constant	K_{c_P}	$K_{c_P} = \dfrac{550}{A(\Omega R)^3}$	slug/ft^3-HP	Calculated at (8)
12	True airspeed	V_T		kt	At selected intervals from minimum power required to maximum speed
13	Advance ratio	μ	$\mu = [V_T/(\Omega R)] \times 1.69$		(13) = [(12) / (9)] x 1.69
14	Rotor thrust coefficient	C_T	$C_T = GW/(K_{c_T}\rho)$		(14) = (7) / [(10)·(4)]
15	Power coefficient required	C_P	$C_P = SHP \times K_{c_P}/\rho$		From Fig. 9-11 at (13) and (14)
16	Power required	SHP_{REQ}	$SHP_{REQ} = C_P\rho/K_{c_P}$	HP	(16) = (15)·(4) / (11)
17	Square root of ambient air temperature ratio	$\sqrt{\theta}$	$\sqrt{\theta} = \sqrt{(T_a + 273) / 288}$		(17) = $\sqrt{[(2) + 273] / 288}$
18	Referred power required	SHP_{REF}	$SHP_{REF} = SHP_{REQ}/(\delta_a\sqrt{\theta})$	HP	(18) = (16) / [(6)·(17)]

TABLE 9-8 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
19	Referred fuel flow	$W_{f_{REF}}$	$W_{f_{REF}} = W_f / (\delta_a \sqrt{\theta})$	lb/hr	From engine characteristics curve Fig. 14-19, at (18)
20	Fuel flow required	$W_{f_{REQ}}$	$W_{f_{REQ}} = W_{f_{REF}} \times \delta_a \sqrt{\theta}$	lb/hr	(20) = (19)·(6)·(17)
21	Specific range	$NAMPP$	$NAMPP = V_T / W_f'$	air miles per pound of fuel	(21) = (12) / (20)
22	Fuel available	WF		lb	From calculated fuel allowances and fuel tank capacity
23	Range	R	$R = NAMPP \times WF$	n mi	(23) = (21)·(22)

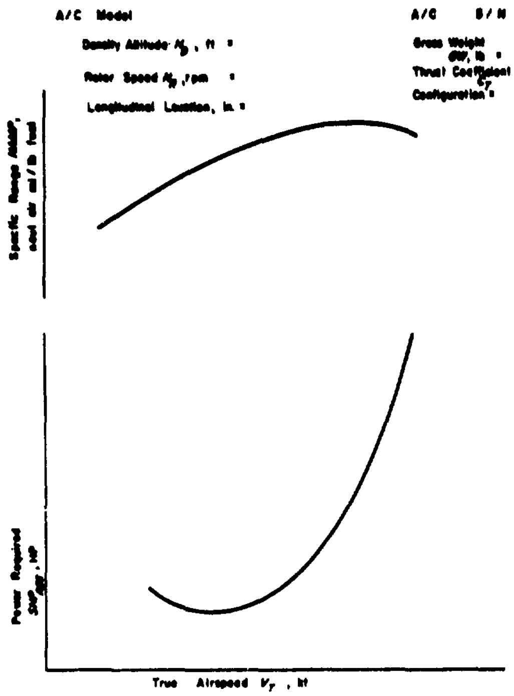


Figure 9-9. Level Flight Power Required

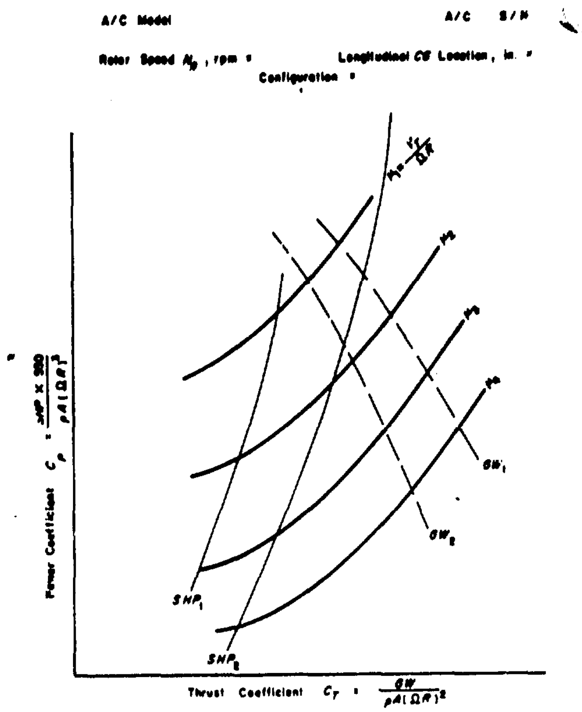


Figure 9-11. Nondimensional Level Flight Power Required

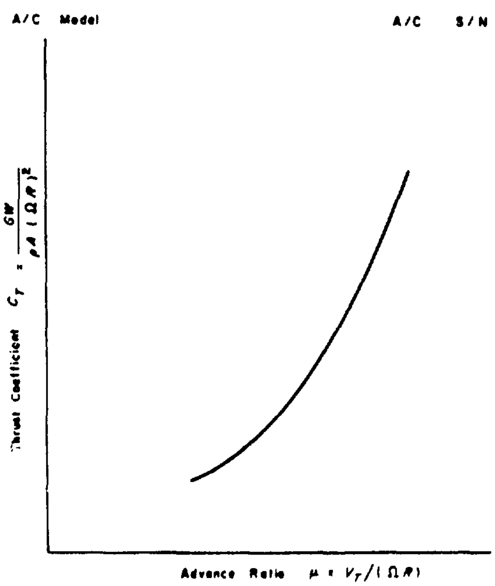


Figure 9-10. Advance Ratio for Minimum Power Required

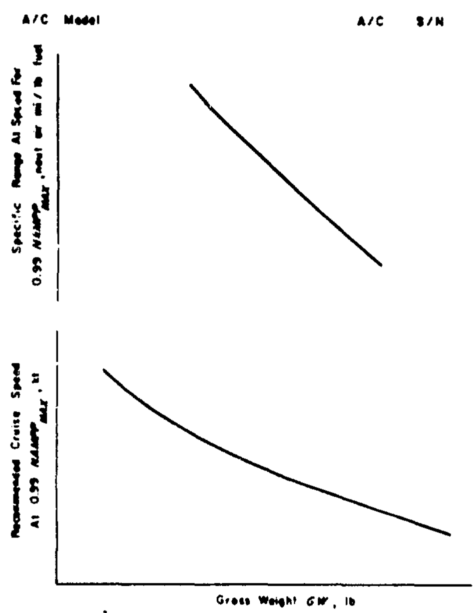


Figure 9-12. Level Flight Range Summary

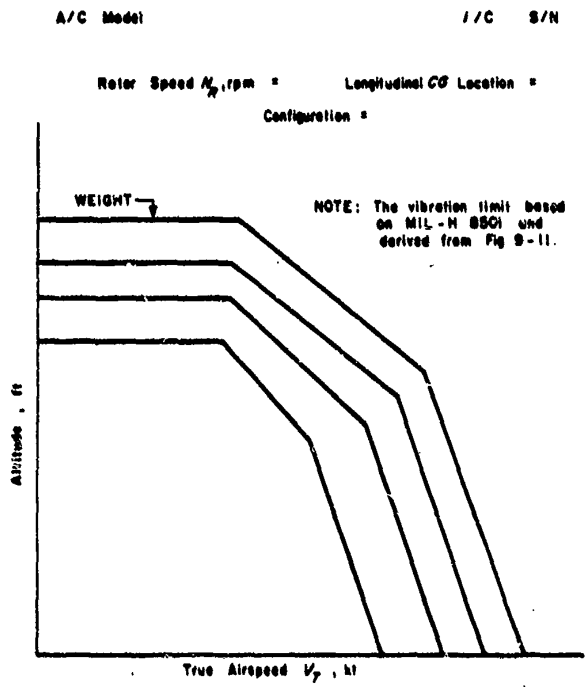


Figure 9-13. Maximum Airspeed Limit for Vibration

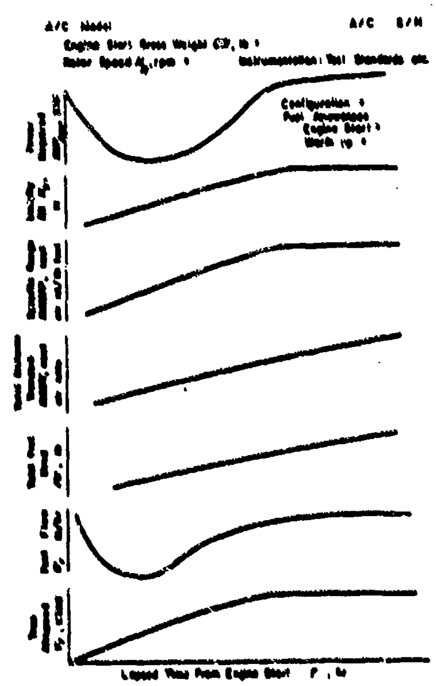


Figure 9-14. Range Mission Performance

CHAPTER 10

AUTOROTATIONAL DESCENT AND LANDING PERFORMANCE

10-1 GENERAL

The capability—after suffering a complete power failure—to maneuver to an unprepared, restricted area, and finally make a controlled safe landing is a feature unique to the helicopter. The steady-state autorotation condition itself is as natural to the helicopter as is a power condition. However, hazardous conditions may be encountered during the dynamic maneuvers between the power loss and landing. The rapidity of the power loss coupled with the change in aerodynamic conditions may introduce moments resulting in large and rapid changes in attitude, rates, and accelerations. Airspeed, power, gross weight, altitude, rotor speed, and flight control positions usually will be significant influence factors.

The pilot reaction time is the delay time required to realistically simulate pilot recognition of the actual power loss condition. This value must be correlated with the capability of the operational pilot to be sure that the test results are representative of operational conditions. The test value usually is defined in various specifications or requirements but may be inaccurate due to a general lack of information in this area and an omission of pertinent requirements.

Pilot factors are the most maligned items. The aircraft performance capability can be no better than the pilot's ability to cope with the aircraft entry reactions. Configuration variations will be expected to exhibit different reactions and motions when power is lost. The autorotational landing maneuver usually requires a high degree of pilot effort and skill when conditions are other than optimum. Adverse landing site conditions, with respect to terrain and wind, contribute the greatest hazards.

The autorotation tests are conducted to determine the flight characteristics, aircraft performance, and optimum handling techniques during entry, steady state descent, and landing performance. The results of these tests are used to define the maximum airspeed envelope, the necessary pilot technique, and the available recognition time during the entry. The steady-state performance tests define the rate of descent in terms of airspeed, rotor speed, gross weight, and altitude. From these data the optimum airspeed and rotor speed are determined for minimum rate of descent and maximum glide distance. The autorotation landing tests define the minimum height and airspeed relationships which permit a safe landing after an engine failure occurs. Included in these latter data are the horizontal distance required, the expected touchdown speed, the slide or roll distance after touchdown, and the optimum technique to be utilized.

The landing tests usually are conducted at several gross weights including the maximum weight. Entry airspeeds tested are from hover to the maximum at all test configurations and conditions. These tests should consider the effects of center of gravity location and the operational range of rotor speeds available. The steady-state descent and landing tests in particular should reflect gross weight and altitude considerations. The landing tests must be conducted at high field elevations since ground proximity effects may vary with altitude. Also, atmospheric wind velocity and direction variations should be included to determine the magnitude of any performance deterioration or stability changes that may result.

The planning for entry and steady-state descent tests need not be elaborate since only the maintenance, instrumentation, and flight personnel are involved. However, every effort should be made to have a chase aircraft available during the entry tests. This aircraft is valuable for observing the physical condition of the test aircraft and commenting on the behavior during critical test conditions. During landing tests, the increased ground support personnel and remote site operation greatly complicate the planning. Under these conditions, good preliminary planning as well as close attention to detail and logistics is necessary to insure a successful, safe, and productive operation.

The test techniques used during the entry and landing tests may be extremely critical. Both of these generally are considered high risk tests where a build-up technique should be used. The buildup for the entry test is with respect to altitude, gross weight, and airspeed. The critical portions of the landing tests are usually a determination of the proper flare airspeed, flare technique, and touchdown procedure. In all these tests the instrumentation and photographic coverage should be the maximum available. Since this is a high risk test, crash and rescue facilities must be available and safety procedures should be closely heeded.

The data reduction effort is complicated by the large number of data recording sources such as the aircraft, weather station, theodolite, Fairchild camera, and photographic stations. Data correlation and recording procedures should be specified during planning and immediately checked after the test as the initial part of the data reduction. Generally, the data will include manually recorded data, automatic data recording, movie camera coverage, and the Fairchild plates. After the initial data processing is completed the reduction can then be accomplished by either manual or automated procedures.

10-2 PLANNING

The test plans should be used to define the test requirements in terms of gross weights, altitudes, and configurations. The autorotational performance tests, in reality, are composed of several individual parts. These parts are entry to autorotation, steady-state descent, and the height-velocity diagram. The entry to autorotation tests are conducted best in conjunction with the level flight tests or during the sawtooth climb and descent tests. In a similar manner, the steady-state descent performance can be done while returning from the climbs for weight and power correction factors or the climb to service ceiling tests. The height-velocity diagram tests must be conducted independently of other tests.

The single most important factor in the planning for the height-velocity test is the choice of landing site. For wheeled machines, the surface must be sufficiently hard to prevent penetration by the wheels and be relatively smooth with respect to bumps, holes, or obstructions. Even small, gentle bumps or dips may cause the rotor blades to flap downward, particularly when the rpm is low after full collective has been applied. For skid gear, the ground surface should be softer with a low friction coefficient such as a sod surface. The site must be large enough to allow for wind drift during the descent, overshoot during the approach, and, when necessary, a power recovery or go around. The area should be free of obstacles that may influence the aircraft performance or introduce any additional pilot workload or apprehension factors. These sites are difficult to find at best and almost impossible to find at remote high altitude sites. The area must allow a sufficient offset distance for the Fairchild camera.

The requirements now can be established for the logistic and support groups. These groups are maintenance, instrumentation, photographic, weather, engineering, crash, and rescue. The maintenance group must perform the ballast operations and the normal maintenance operations. An adequate amount of ballast and the necessary transportation and communication equipment must be requisitioned. Routine maintenance planning should be covered by the usual maintenance procedures. Additional requirements caused by weight and balance or test procedures must be established and provided to the maintenance personnel. The instrumentation personnel should be informed of the type of data to be recorded and the volume of data to be processed. The timeliness of the instrumentation service and data processing are also pertinent items. Aircraft turn around and preflight times should be established. The photographic support for the height-velocity tests includes movie coverage as well as the Fairchild Flight Analyzer. Plans must include means to transport the camera equipment and provide communications with the aircraft and other test groups. In some cases the Fairchild operator may require help to manipulate the heavy equipment. Arrangements should be made to have the camera site surveyed or checked in the case where an existing site is to be used. The operator should be informed of the number of plates needed per day and the total number anticipated for the entire test. This will allow the operator to order adequate supplies and to preload the plates prior to each test. The weather group should provide the equipment and operators for measurement of ground wind velocity and direction, pressure altitude, and ambient air temperature. When possible, equipment should be obtained to record the wind conditions at other heights above ground level. The engineering support will consist of a theodilite crew and sufficient personnel to manage the ground operation. These groups must all be provided with suitable communication and transportation equipment. Radio frequencies and procedures must be clearly established.

Preliminary flight and data cards are written and coordinated with the individual support groups. Their procedures and comments are incorporated in the final flight and data cards. A briefing is then held with all the support groups present to discuss the operation and establish the respective duties. Vehicle assignments should be finalized and the communication procedures explained. Each group should review their operational procedures to determine if they will influence the duties of the other groups or the test as a whole. When there is an influence, the appropriate group should be informed. The emphasis in this meeting should be on operational procedures rather than the technical details that were presumably resolved during the group briefings.

10-3 INSTRUMENTATION

The autorotation entry tests require visual instruments for monitoring and recording critical flight parameters during the build-up procedure. The necessary visual instruments are listed in Table 10-1.

TABLE 10-1

VISUAL INSTRUMENTATION FOR ENTRY TESTS

Altitude	Fuel used
Ambient air temperature	Collective stick position
Rotor speed	Normal acceleration
Airspeed	

The entry is a dynamic maneuver with stabilized parameters before the power reduction, followed by variable indications from the visual instruments. In view of these considerations, an automatic data recording system is desirable during the dynamic maneuver. Some of the parameters that should be recorded automatically are shown in Table 10-2.

TABLE 10-2

OSCILLOGRAPH INSTRUMENTATION FOR ENTRY TESTS

Altitude	Throttle position
Airspeed	Longitudinal stick position
Rotor speed (linear)	Rotor speed (blip)
Pitch attitude	Lateral stick position
Normal acceleration	Pedal position
Roll attitude	Engine speed
Yaw attitude	Collective stick position
Engine torque	Longitudinal acceleration

The steady-state descent performance is a stabilized condition and can be recorded adequately from the visual instrumentation shown in Table 10-1. High data density as well as correlation parameters are available if desired from an automatic data recording system for the parameters listed in Table 10-2.

The landing tests include stabilized entry conditions followed by a transient descent and a dynamic landing maneuver. The instrumentation shown in both Tables 10-1 and 10-2 are required in the aircraft. The ground instrumentation should consist of a Fairchild camera, a thermometer, an altimeter, an anemometer (wtih directional sensing capability), a vertical theodolite, and a horizontal theodolite. Some means must be provided to accomplish positive data correlation and identification from the various sources. This correlation should be simultaneous for all parameters if possible.

10-4 TEST METHODS

10-4.1 ENTRY TESTS

The autorotation entry tests are conducted to determine the initial aircraft reaction to a sudden power loss. This power loss is simulated by rapidly reducing the throttle from a specified flight condition. These throttle chops usually are conducted from stabilized level flight and climb conditions although, if desired, they may be conducted from any other powered flight situation. The primary goals of these tests are to determine the rotor behavior characteristics, the aircraft stability and control characteristics, and the necessary pilot actions. The elapsed time from power loss to specific aircraft attitudes, rates, and pilot input is a primary consideration. The tests should be conducted at an altitude at least 2000 ft above the terrain in order to allow time and space for any necessary recovery action. This minimum altitude will vary with different aircraft, depending upon the information available concerning the anticipated behavior. In some cases it may be prudent to test at bailout altitude (5000 ft). Air-to-air camera coverage can provide invaluable information relative to aircraft behavior. The aircraft should be stabilized at the specified flight condition, the countdown should be given, and the power rapidly reduced to flight idle. One pilot technique used is to hold all controls fixed and let the aircraft characteristics determine the aircraft motion. At some point in time the attitude, rate, or some other flight parameter will require the pilot to initiate recovery action. This then will define the critical parameter and the maximum allowable time delay. Controlling the critical parameters as they are defined while varying the remaining parameters will allow the aircraft performance to be defined in terms of all the flight controls. This technique may introduce unpredictable reactions as unknown areas are entered and a build-up procedure should be employed.

Another, perhaps simpler and safer, method is to maintain the attitude from throttle chop to stabilized autorotation. The control positions then are used to determine when a limit has been reached. For example, when full aft stick is required, the maximum tolerable pitching moment has been encountered. Other controls are treated is a similar manner. In some cases the test vehicle will be capable of exceeding a contractual specification which would then be used as the limiting criteria. The undesirable feature about this latter technique is that dynamics and couplings are not included in the measured control requirements. The control margin necessary to overcome these is difficult to establish other than through testing.

The altitude loss during the entry should be noted since this will have a significant bearing on the height-velocity diagram. Following the altitude work, it may be desirable to repeat the test at a low ground height using camera or theodolite coverage to establish more accurately this altitude loss.

The entry tests are conducted for a range of gross weights, altitudes, center of gravity locations, and rotor speeds as a function of entry airspeed. It is best to start the test at some intermediate speed and then proceed toward the high airspeed and hover conditions. Speed increments of 10 or 20 kt are usual, depending upon the speed range to be covered. The sequence of the other variables may be adjusted according to the characteristics of the test helicopter. The usual rule of the safest condition first, with a build-up to the most critical, is the recommended procedure. The data recording system should include an automatic capability since the maneuver is dynamic. The steady-state conditions should be hand recorded prior to the countdown. At the start of the countdown, the automatic data system should be activated. During the entry, the visual indicators should be monitored and data recorded. The time should be recorded when the recovery is initiated and at the same time an identifying marker (event or blanking) is inserted into the automatic data system. This will allow an exact determination of the conditions at this critical point in time. Pilot comments should be recorded and analyzed during the progress of the test to insure that the planned buildup is both safe and productive. The test procedure then is repeated for all required conditions.

10-4.2 SAWTOOTH DESCENT TESTS

The sawtooth descents are conducted to determine the airspeed for minimum rate of descent, best glide distance, and to determine the autorotational performance as a function of both airspeed and rotor speed. Gross weight, altitude, center of gravity location, and aircraft configuration may be other significant variables. The aim density altitude is chosen and then the pressure altitude increment for the test usually is established at ±1000 ft from this desired height. The increment must not be so large as to introduce any altitude effects that properly should be investigated during another separate test. A small increment makes stabilization, time, and altitude measurements very critical since only a small error or change can influence the rate of descent values significantly. The airspeed range will vary with different aircraft, and the test should use airspeed values from well on the backside to well on the front side of the power required curve. This will encompass the airspeed for minimum rate of descent which is near the minimum power required airspeed, and the speed for best glide distance which is at a slightly higher airspeed. Airspeed increments of 10 kt are most suitable.

The rotor speed used during the airspeed sweep should be that recommended in the operator's manual. When unavailable from this source, the rotor speed should be set at the maximum or that most suitable from a flying quality or vibration consideration. The initial test airspeed should be an intermediate speed prior to proceeding to the more critical extremes. The aircraft should be stabilized in autorotation at an altitude above the test altitude increment. This stabilized condition then should be maintained as the aircraft descends through the test altitude increment. All flight control motions should be minimized during this recording period. This procedure then is repeated for each of the specified test conditions.

At the airspeed extremes, the rate of descent may be very high and allowance should be made to provide sufficient altitudes above the ground to accomplish the recovery. At these conditions the flying qualities may change significantly and have an influence on the maximum performance values that can be obtained from the aircraft In some cases, these changes have been sufficiently large as to introduce flight safety considerations. Engine response time and rotor characteristics at high collective angles must be considered when determining the minimum recovery altitude.

The data can be recorded manually during this test since the aircraft is in a stabilized flight condition. When the aircraft is stabilized and reaches the specified altitude, time is started; and altitude, rotor speed, gross weight, ambient air temperature, and airspeed are recorded. These parameters should be recorded at a minimum of 500-ft altitude intervals during the descent. Usually the pilot maintains the proper airspeed and rotor speed, and only deviations from the schedule need be noted. Following each descent, the elapsed time is plotted versus the airspeed as shown in Fig. 10-1.

This in-flight plot then will yield the indicated airspeed for the minimum rate of descent conditions.

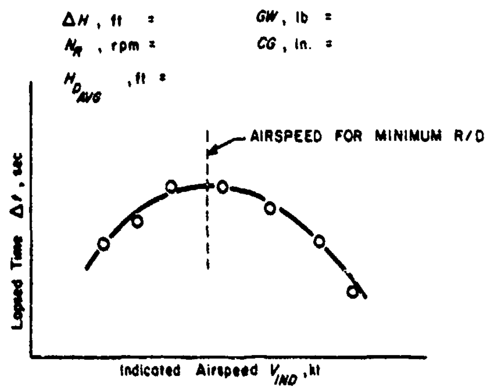


Figure 10-1. Autorotational Descent Performance With Airspeed Variation

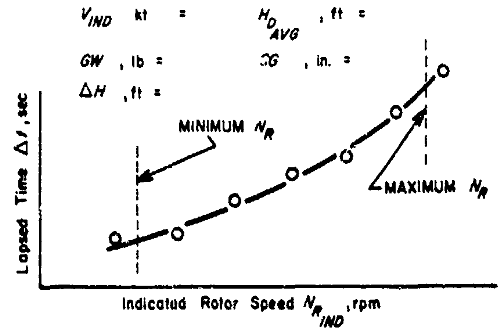


Figure 10-2. Autorotational Descent Performance Variation With Rotor Speed

The change in rate of descent and rotor characteristics is determined by conducting tests at various rotor speeds while maintaining constant airspeed. The maximum rotor speed range available should be divided in such a manner that a minimum of six test values are obtained. The altitude increment should be the same as the airspeed variation tests, and the in-flight curve shown in Fig. 10-1 should be used to determine the test airspeed. The test procedure is the same as that previously discussed with the exception that rotor speed rather than airspeed is varied for each descent. At the low rotor speeds, the maximum collective available may prevent reaching the minimum rotor speed. The data can be hand recorded from visual instrumentation with the airspeed, time, altitude, and rotor speed being the critical parameters. A sample of a normal in-flight plot is shown in Fig. 10-2.

Plotting the data during the test will reveal any erratic points that should be repeated and will provide information concerning any undesirable characteristics that progressively may be developing.

10-4.3 STEADY-STATE DESCENT TESTS

The steady-state descent tests usually are conducted while descending from high altitude level flight tests or from the service ceiling climbs. The objective of the test is to determine the variation in rate of descent with altitude, collective stick position, and gross weight. The tests usually are conducted at the airspeed for minimum rate of descent and at the operational rotor speed.

The aircraft is stabilized at the specified flight conditions which then are maintained throughout the descent. As the altitude decreases, the collective must be lowered to maintain a constant rotor speed. At light weight conditions the full down collective stop may be encountered. In this case, the collective then should be maintained in that position and the test continued provided the rotor speed does not exceed any limits. These steady-state data may be hand recorded from visual instrumentation. The altitude and time should be recorded at minimum altitude increments of 1000 ft, and any deviations from the rotor speed or airspeed schedule should be noted. The recorded altitude and the accompanying elapsed times, both incremental and total, are the critical parameters.

10-4.4 HEIGHT-VELOCITY DIAGRAM TEST

The height-velocity diagram test is accomplished to determine the height and airspeed limits from which a safe autorotational landing following a sudden, complete power failure cannot be accomplished. The engine failure is simulated by a sudden engine power reduction (throttle chop) by the pilot. The complete maneuver includes all the factors previously discussed in this chapter, in addition to the flare and landing.

Prior to operating near the ground, an estimated height required should be determined at altitude. The aircraft initially is stabilized at some predetermined altitude, rotor speed, and airspeed. The entry characteristics previously have been determined so the initial aircraft behavior and necessary pilot reaction can be anticipated. The first entry speed should be at some value slightly below the airspeed for minimum rate of descent. Following the throttle chop, the controls should be held fixed for the time increment contractually specified or the maximum allowable as determined from the autorotational entry tests. At the end of this delay time there usually has been a significant rotor speed decay and the collective stick is lowered. An optimum longitudinal control technique then is used to attain some pre-selected airspeed such as 50 kt. A cyclic flare then is executed, and the subsequent rotor speed increase is noted. This procedure then is repeated at the same entry speed while the flare is executed at different airspeeds. At low speeds the flare may not provide a sufficient buildup of rpm while at high airspeeds the flare may produce an excessive rotor speed increase. The desired flare airspeed is the minimum airspeed that will produce the maximum allowable rotor speed increase during the cyclic flare. Extreme care must be used to hold the flare technique constant so that the rotor speed increase will be representative of the flare airspeed. An excessive flare airspeed will not decrease significantly the minimum height required but should increase the margin of safety. Typical effects of variation of flare airspeed on the rotor speed and height loss are shown in Fig. 10-3. When the airspeed is above the optimum flare airspeed, collective must be added during the flare to prevent rotor over-speed. Thus, the same rotor speed increment is available to accomplish the landing. The collective application and deceleration should provide a

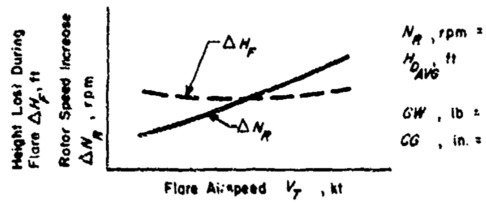


Figure 10-3. Rotor Speed and Altitude Variation With Flare Airspeed

slower touchdown speed although the higher entry speed may negate these factors.

When the flare airspeed has been determined for a given gross weight, entry rotor speed, and altitude, the complete airspeed range is tested incrementally to determine the optimum manner of achieving this value. At low airspeeds (airspeeds below the flare speed), the aircraft must be dived to increase speed to the flare value as rapidly as possible while still maintaining a reasonable pitch attitude. A reasonable pitch attitude will be determined by the flight characteristics, pilot, and aircraft configuration. For speeds above the flare value, a flare can be accomplished immediately after the time delay interval.

After the baseline data are obtained at the maximum operational rotor speed, the minimum rotor speed condition should be investigated. For this condition, the rotor speed at entry is lower and the collective stick position is higher. These factors may increase the rotor speed decay rate to such a point that dangerously low transient minimum speeds result, and the delay time previously used is no longer usable. The procedure then must be repeated to determine the maximum delay time and the limiting factors. There also may be a significant deterioration of flying qualities at the very low rotor speeds. When a lower minimum rotor speed is used, the allowable rotor speed increase is greater and the required flare airspeed may be higher.

Gross weight and altitude effects should be investigated in a similar manner. Increased gross weight usually will require a higher flare airspeed.

Center of gravity location is also important since sufficient control power must be available to accomplish the entry, gain flare airspeed, flare the aircraft in the desired manner, and then to level the aircraft from the flare prior to the touchdown. An aft CG location may limit the control available to pitch the nose down to gain flare airspeed and to level the aircraft after the cyclic flare. A forward CG location may limit aft pitching moment and limit the flare attitude that can be obtained. Lateral CG locations usually are not investigated, and any effects from this should be significant only in the case of very unusual lateral control characteristics. The vertical CG location tests normally are omitted since unusual loadings should be avoided during the test to prevent tip-over tendencies during touchdown and deceleration.

An evaluation of all factors now has been accomplished with the exception of the ground proximity effects and the actual touchdown procedure. Unfortunately, these must be accomplished near the ground. An extensive, systematic practice should be established to enable the pilot to determine the minimum flare height above the ground, the cyclic control necessary to accomplish the flare, and to level the aircraft prior to touchdown. Another factor to be determined is the rate of descent. The flare attitude and the physical dimensions of the aircraft can be used to calculate the minimum flare height (from the pilot's seat). To achieve a realistic height, this calculated value must be increased to compensate for the vertical distance used during the time increment necessary for the flare, leveling the aircraft, and the landing. The minimum touchdown speed requirements also will influence the technique and the height required.

At a given condition, the aircraft has a certain amount of kinetic energy that must be absorbed by the rotor and the landing gear in order to bring the aircraft to a stop. The vertical and horizontal touchdown velocities are fixed by the structural limits of the landing gear. Thus, the remainder of the energy must be dissipated through the rotor during the flare and landing.

An extremely high forward touchdown speed is undesirable primarily from a terrain consideration. Most engine failures occur in areas other than over runways and, as a result, wheels are of little practical value. Most wheels can endure less horizontal force than can skid gear. High forward touchdown speeds introduce high drag forces on the skids, and the aircraft may have a tendency to nose over. This usually results in an aft cyclic input which tilts the rotor aft with increased downward flapping. The tail area may now be endangered by the subsequent reduced fuselage to blade clearance. Of course, for all machines the greater the horizontal distance covered in unprepared terrain, the greater is the possibility of encountering an undesirable object. The forward speed can be reduced by flaring the helicopter to a more severe attitude, higher above the ground, and holding the flare for a longer period of time. (This will usually require that collective pitch be applied during the flare to prevent a rotor overspeed.) As a result, the forward speed will be reduced during the flare. Extreme caution must be used with this technique to prevent the tail from making ground contact during the extreme, prolonged flare; and, since the thrust vector is inclined from vertical with this technique, the dissipation of vertical speed may be insufficient to prevent a hard landing with full application of pitch. The amount of collective used should be noted carefully to insure a sufficient margin to reduce the vertical speed to within gear limits. The collective margin may become critical when the helicopter is leveled inadvertently at an excessive height above the ground. High vertical touchdown speeds also introduce large downward acceleration forces on the blades which can cause considerable downward flapping. This, in addition to the low rotor speed, may cause blade-to-fuselage interference.

After sufficient practice has been accomplished to establish the flare and touchdown techniques, the actual height-velocity test can be conducted. An estimated curve can be calculated by summing the distances required for entry, to gain flare speed, flare, and touchdown. The first airspeed tested should be an intermediate speed, and the initial vertical height above the ground should be greater than the value calculated from the previous tests. The maneuver and landing should be accomplished and the excess height margin estimated. The margin will be apparent from the distance to the ground after the flare is completed and the aircraft leveled in a landing position. This height differential then can be compared to the estimated margin which will validate the original calculation and estimate. In the event of a large discrepancy, a reassessment of the previous work is advisable. This should isolate the cause of the discrepancy. Photographic data can be very useful in showing any changes in technique or performance from that obtained during the individual tests. Pilot comments should be evaluated carefully to determine any changes as a result of the maneuver or ground proximity. The subject is treated in general here since the parameters are so numerous, the effects so subtle, and so little understood as to render detailed discussion most difficult.

The test then is repeated at some lower starting height. For some initial vertical height there will be no altitude increment remaining prior to touchdown and the minimum safe conditions have been achieved. The procedure then is repeated through the airspeed range. The vertical height can be reduced further by using a lower flare airspeed or a shorter time in the flare attitude. Decreasing the flare airspeed while maintaining the same flare technique will result in higher vertical touchdown speeds. Reducing the flare duration will increase both vertical and horizontal touchdown speeds. Flaring to a lower pitch attitude will increase significantly the forward touchdown speed.

The desired data recording system for the height-velocity tests include an automatic high density system as well as a visual manual recording capability. During the initial determination of flare airspeed the entry airspeed, altitude, gross weight, and rotor speed should be recorded from visual instrumentation. The pilot should give a 5-sec countdown at which

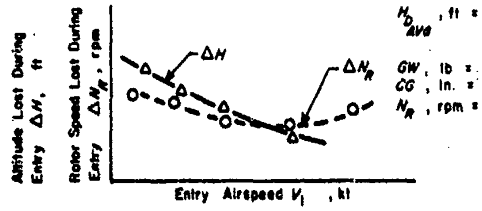


Figure 10-4. Altitude and Rotor Speed Variation With Entry Airspeed

time the automatic data system is activated. When the flare is initiated the altitude, airspeed, and rotor speed should again be recorded. An identifying notation (event marker) also should be inserted into the automatic system. A differential altitude and rotor speed should be determined for each test condition. These should be plotted in the form illustrated in Fig. 10-4.

The rotor speed at entry of the flare N_{R_1} and the speed at the completion of the flare N_{R_2} are used to determine the rotor speed increase, $\Delta N_R = N_{R_2} - N_{R_1}$. These data then are plotted as shown in Fig. 10-5.

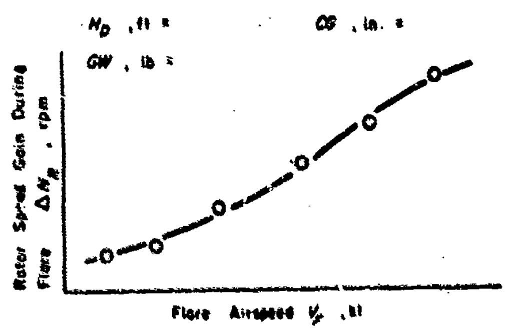


Figure 10-5. Rotor Performance During Flare

The indicated airspeed at the end of the flare should be noted and can be plotted as a function of flare airspeed or ΔN_R, see Fig. 10-6.

The altitude increment required for entry, to attain flare airspeed, and to flare are assumed and plotted as shown in Fig. 10-7. Although this curve is not a true height-

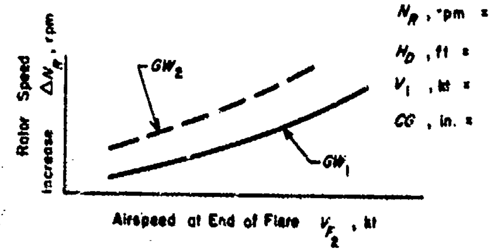


Figure 10-6. Flare Performance

velocity diagram, the values are reasonably close to the final result and give an indication of the height above the ground that will be required. The curve shown in Fig. 10-7 is compared with any previously obtained data. The initial test is then an attempt to incorporate all of the previously tested segments into one continuous maneuver. In the interest of safety, the tests should be started at some height above the estimated values as shown in Fig. 10-7.

The curve obtained to this point (Fig. 10-7) omits takeoff corridor. A separate program should be conducted to determine this area. The techniques in this area are different and in general the test is more hazardous. The timing and pilot technique are more critical near the ground, and there is less time or opportunity for corrections. The zero airspeed height is determined by accomplishing throttle chops at different hovering heights. The technique is deceptively simple. The usual pilot delay after the throttle chop is pilot reaction time with no additional increment. Following pilot recognition of the engine failure, the collective pitch increase is timed to achieve maximum pitch at the time of ground contact. The hovering height is increased until the landing impact loads approach the gear limits. Extreme care must be used to insure that the full up collective stick position will be reached by the helicopter when very near the ground. The rate of application should be a continuous motion to arrest the rate of descent. After the baseline data have been obtained, the performance should be evaluated with different types of collective applications. Significant improvements in performance may be possible by a rapid input just prior to contact or, possibly, by slowing the initial rotor decay by lowering the collective immediately after the power loss. These techniques should be used with extreme caution since pilot actions are very critical. The results also must be judged carefully with respect to operational pilot capabilities. Longitudinal control should be monitored carefully and controlled to prevent blade-to-fuselage contact at touchdown. The resultant hover point is denoted as point A on the curves shown in Fig. 10-8.

The line $A'B$, Fig. 10-8, is determined initially from a level flight condition. The helicopter is stabilized at some airspeed and at a height above the ground no greater than the hovering point A. As in the hovering case, following the power loss, the aircraft attitude

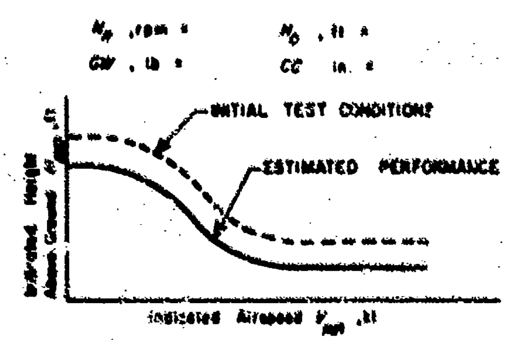


Figure 10-7. Recommended Minimum Height Required for a Safe Landing Following an Engine Failure

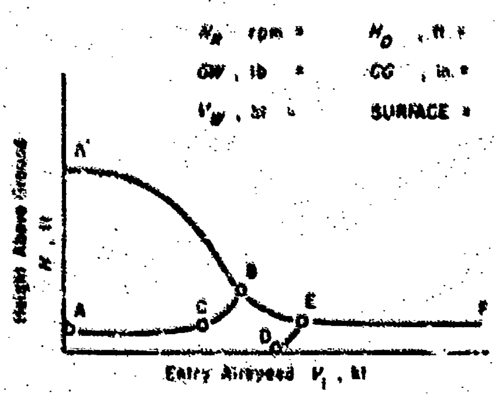


Figure 10-8. Recommended Altitude for Safe Landing Following an Engine Failure

is held level and the collective pitch is increased to slow the rate of descent. The helicopter will drop rapidly and at this low height there is neither time nor altitude to flare to any extent. As a result, there is deceleration only from the aerodynamic drag, and the forward speed at touchdown will be essentially the same as the entry airspeed. The same caution notes for the hover test condition apply to this test.

The entry height above the ground is then increased incrementally until a limiting factor is reached. The limiting factor will vary with aircraft, gear configuration, program requirements, and gear instrumentation. The ultimate test limiting factor is the structural strength of the landing gear which will require extreme care and a careful buildup to determine. Direct readout, real time instrumentation is most advisable. Less strenuous criteria such as pilot comfort and touchdown speed limits usually are used which do not approach any gear limits.

The entry airspeed then is increased and the procedure is repeated until the flare airspeed, denoted by point *C*, Fig. 10-8, is reached. Since the height usually increases somewhat near the flare airspeed, a slight flare may be possible to reduce forward speed and build the rotor speed. Airspeed increments of 10 kt usually are satisfactory values. Although the altitude increment will vary with different test vehicles, pilot proficiency, and availability of previous test data, increases measured in inches may be significant in this area.

A level flight boundary now has been established. Since the primary purpose of this test is to define a safe takeoff path, the helicopter may be in a climbing or accelerating condition. The power and collective pitch settings will be higher with larger resultant rotor speed decay rates, altitude losses, and attitude changes. Due to the acceleration, the airspeed loss at power failure will be smaller which should result in higher forward touchdown speeds. To conduct proper build-up tests, the first point should be at a lower height than the previously established level flight curve and at a slightly greater power. Power then is increased incrementally on succeeding points until full power is reached. If possible, the height then is increased further to a maximum as limited by some factor. The size of the power increments will depend on the capabilities of the test vehicle and the flight personnel. Following this determination, the airspeed then is increased and the procedure is repeated to encompass the speed range. Although flare airspeed has been attained, the region—bounded by the line *DEF*, Fig. 10-8, and the ground—may exist where insufficient time or height is available to flare.

Before attempting the tests to establish the boundaries of the region, the reader should be aware of conditions which may be encountered. The cyclic delay is usually more critical in these tests than the collective delay. As the entry airspeed is increased, the rotor speed decay normally increases, often accompanied by rolling and pitching tendencies. Pitch up and roll left motions caused by blade stall are common. The excessive cyclic control delay may result in insufficient height to flare and reduce speed. Tail rotor clearance is a primary concern in this area. A series of abbreviated flares may be better than one continuous flare to reduce the airspeed. The entry tests will provide information concerning aircraft motions and the proper rate of progress to the more critical high speeds. Speed increments of 10 kt are usually sufficient for these tests. At each airspeed, the height above the ground is lowered until a critical value is reached. This technique does not apply, for example, to a wheeled helicopter over a runway during a high speed accelerating climb-out since the landing can be accomplished easily from a very low height above the ground if control delays are nominal. For these conditions, this portion of the curve, in fact, does not exist. One can imagine the results if the prepared runway were replaced with an area unsuitable for high speed, run-on landings.

The data recording equipment required for the height-velocity tests includes airborne as well as ground installations. Since there is a requirement for certain parameters to be read in real time, visual indicators or readouts are mandatory. These consist of visual indicators in the aircraft and a theodolite on the ground. The aircraft may be influenced strongly by surface wind conditions, accordingly, weather information must be available at all times. The maneuvers are dynamic, and it is necessary to have an automatic recording system to record the rapid transient conditions. This can be accomplished with either a photo panel or an oscillograph. The ground weather station should record wind velocity, wind direction, altitude, and ambient temperature at 5-min intervals during the test, with additional readings desirable at each test point. The visual theodolite reading is recorded on each test point and entered on the data card. The Fairchild camera is used to record each test point from countdown to the end of the landing portion. Motion picture coverage should be used to portray visually any unusual aircraft motions during the maneuver and to give the pilot an outside view of the technique with respect to heights, attitudes, and aircraft motions.

10-5 DATA REDUCTION

There are relatively few calculations involved, and the data normally are not corrected to standard conditions. Regardless of the data reduction procedure to be used, the first portion of the effort is to correlate and define the data recorded at the different locations—i.e., weather station, aircraft, theodolite, ballast station, and Fairchild camera.

The data reduction for the autorotation entry tests uses data from the cockpit, the photo panel, and the oscillograph. This data reduction form is presented in Table 10-3*.

The data are generally difficult to read because of the highly transient conditions. Careful flight techniques will minimize the transients and greatly reduce the reduction effort and improve the quality of the data. Establishing the recovery time is difficult when the controls are not held fixed, and in some cases an estimation may have to suffice. For the rates and accelerations, a maximum value may occur before the recovery is initiated. In this case, the maximum should be used rather than the value at recovery or at any time interval. The control position data will provide a check on the technique as well as indicate any imminent limitations.

The data reduction required to determine flare generally is accomplished more easily since the conditions are considerably less transient. The data reduction form for the flare airspeed tests is shown in Table 10-4*.

The data reduction form for hover, low speed, and low altitude landings is shown in Table 10-5*. The data for the hover points must be obtained from a tilt head Fairchild Flight Analyzer or a motion picture camera.

10-6 DATA PRESENTATION

The data are presented graphically as shown in Figs. 10-9 through 10-34. These plots do not show all the possible variations in each parameter, and it may be necessary to make more than one of each type shown.

*The data reduction forms (tables) are located at the end of chapter.

TABLE 10-3

DATA REDUCTION FORM FOR AUTOROTATION ENTRIES

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1	Point No./Flight No.				
2	Indicated entry pressure altitude	$H_{P_{IND_1}}$		ft	From photo panel
3	Altimeter instrument correction	ΔH_P		ft	From instrument calibration
4	Entry pressure altitude	H_{P_1}	$H_{P_1} = H_{P_{IND_1}} + \Delta H_P$	ft	(4) = (2) + (3)
5	Indicated ambient air temperature	$T_{a_{IND}}$		°C	From photo panel
6	Instrument correction for ambient air temperature	ΔT_a		°C	From instrument calibration
7	Entry ambient air temperature	T_{a_1}	$T_{a_1} = T_{a_{IND}} + \Delta T_a$	°C	(7) = (5) + (6)
8	Density altitude	H_D		ft	From tables at (7) and (4) or calculated
9	Air density ratio	σ			Calculated or from tables at (8)
10	Indicated entry air speed	V_{IND_1}		kt	From photo panel

TABLE 10-3 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
11	Instrument correction for entry airspeed	ΔV_{IC}		kt	From instrument calibration
12	Airspeed correction for position error	ΔV_{PE}		kt	From position error calibration
13	Calibrated entry airspeed	V_{CAL_1}	$V_{CAL_1} = V_{IND_1} + \Delta V_{IC} + \Delta V_{PE}$	kt	(13) = (10) + (11) + (12)
14	True entry airspeed	V_{1_T}	$V_{1_T} = V_{CAL_1} / \sqrt{\sigma}$	kt	Incompressible flow is assumed (14) = (13) / $\sqrt{(9)}$
15	Collective delay time	Δt_{δ_c}		sec	From oscillograph time from throttle off to collective movement
16	Longitudinal delay time	Δt_{δ_e}		sec	From oscillograph time from throttle chop to longitudinal stick movement
17	Lateral delay time	Δt_{δ_a}		sec	From oscillograph time from throttle chop to lateral stick movement
18	Pedal delay time	Δt_{δ_r}		sec	From oscillograph time from throttle chop to pedal movement
19	Indicated pressure altitude at recovery	$H_{P_{IND_2}}$		ft	From photo panel

TABLE 10-3 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
20	Altimeter correction	$\Delta H_{P_{IC}}$		ft	From instrument calibration
21	Recovery altitude	H_{P_2}	$H_{P_2} = H_{P_{IND_2}} + \Delta H_{P_{IC}}$	ft	(21) = (19) + (20)
22	Altitude loss during entry	ΔH	$\Delta H = H_{P_1} - H_{P_2}$	ft	(22) = (4) − (21)
23	Rotor speed at recovery	N_{R_2}		rpm	From photo panel
24	Rotor speed at entry	N_{R_1}		rpm	From photo panel
25	Rotor speed loss during entry	ΔN_R	$\Delta N_R = N_{R_1} - N_{R_2}$	rpm	(25) = (24) − (23)
26	Rotor decay rate	$\Delta N_R/\Delta t$	$\Delta N_R/\Delta t = \Delta N_R/\Delta t_{\delta_c}$	rpm/sec	(26) = (25) / (15)
27	Minimum rotor speed during entry and recovery	N_{R_3}		rpm	From oscillograph at minimum value recorded
28	Indicated recovery airspeed	V_{IND_2}		kt	From photo panel
29	Instrument correction for airspeed	ΔV_{IC}		kt	From instrument calibration
30	Airspeed position error	ΔV_{PE}		kt	From position error calibration

TABLE 10-3 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
31	Airspeed at recovery	V_2	$V_2 = V_{IND_2} + \Delta V_{IC} + \Delta V_{PE}$	kt	(31) = (28) + (29) + (30)
32	Airspeed loss during entry	ΔV	$\Delta V = V_1 - V_2$	kt	(32) = (13) − (31)
33	Pitch altitude at entry	θ_1		deg	From oscillograph
34	Pitch attitude at recovery	θ_2		deg	From oscillograph at recovery
35	Pitch attitude change during entry	$\Delta\theta$	$\Delta\theta = \theta_2 - \theta_1$	deg	(35) = (34) − (33)
36	Pitch rate	$\dot{\theta}$		deg/sec	From oscillograph
37	Pitch acceleration (maximum)	$\ddot{\theta}$		deg/sec^2	From oscillograph maximum during entry
38	Roll attitude	ϕ		deg	From oscillograph
39	Roll rate	$\dot{\phi}$		deg/sec	From oscillograph
40	Roll acceleration (maximum)	$\ddot{\phi}$		deg/sec^2	From oscillograph maximum during entry
41	Yaw attitude	ψ		deg	From oscillograph

TABLE 10-3 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
42	Yaw rate	$\dot{\psi}$		deg/sec	From oscillograph
43	Yaw acceleration (maximum)	$\ddot{\psi}$		deg/sec^2	From oscillograph maximum during entry
44	Longitudinal stick position at entry	$\delta_{s_{e_1}}$		in.	From oscillograph
45	Longitudinal stick position at end of recovery	$\delta_{s_{e_2}}$		in.	From oscillograph
46	Longitudinal stick required for recovery	$\Delta\delta_{s_e}$	$\Delta\delta_{s_e} = \delta_{s_{e_2}}\ \delta_{s_{e_1}}$	in.	(46) = (45) – (44)
47	Lateral stick position at entry	$\delta_{s_{a_1}}$		in.	From oscillograph
48	Lateral stick position at end of recovery	$\delta_{s_{a_2}}$		in.	From oscillograph
49	Lateral stick required for recovery	$\Delta\delta_{s_a}$	$\Delta\delta_{s_a} = \delta_{s_{a_2}} - \delta_{s_{a_1}}$	in.	(49) = (48) – (47)
50	Pedal position at entry	$\delta_{s_{r_1}}$		in.	From oscillograph
51	Pedal position at end of recovery	$\delta_{s_{r_2}}$		in.	From oscillograph

TABLE 10-3 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
52	Pedal required during recovery	$\Delta\delta_{s_r}$	$\Delta\delta_{s_r} = \delta_{s_{r_2}} - \delta_{s_{r_1}}$	in.	(52) = (51) – (50)
53	Collective stick position at entry	$\delta_{s_{c_1}}$		in.	From oscillograph
54	Collective stick position at end of recovery	$\delta_{s_{c_2}}$		in.	From oscillograph
55	Collective used during recovery	$\Delta\delta_{s_c}$	$\Delta\delta_{s_c} = \delta_{s_{c_2}} - \delta_{s_{c_1}}$	in.	(55) = (54) – (53)
56	Time to lower collective	Δt		sec	Time from initial movement to minimum value
57	Rate of lowering collective	$\Delta\delta_{s_c} / \Delta t$		in./sec	(57) = (55) / (56)

TABLE 10-4

DETERMINATION OF FLARE AIRSPEED

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1	Point No./Flight No.				
2	Indicated pressure altitude at flare entry	H_{IND_i}		ft	From photo panel or cockpit recording
3	Altimeter instrument correction at flare entry	$\Delta H_{p_{IC_i}}$		ft	From instrument calibration curves at (2)
4	Pressure altitude at flare entry	H_{p_i}	$H_{p_i} = H_{p_{IND_i}} + \Delta H_{p_{IC_i}}$	ft	(4) = (2) + (3)
5	Indicated ambient air temperature at flare entry	$T_{a_{IND_i}}$		°C	From photo panel
6	Ambient air temperature correction at flare entry	ΔT_{a_i}		°C	From instrument calibration curves at (5)
7	Ambient air temperature at flare entry	T_{a_i}	$T_{a_i} = T_{a_{IND_i}} + \Delta T_{a_i}$	°C	(7) = (5) + (6)
8	Density altitude at flare entry	H_{D_i}		ft	Calculated or from tables at (7) and (4)
9	Air density ratio at flare entry	σ_i			Calculated or from tables at (8)
10	Indicated airspeed at flare entry	V_{IND_i}		kt	From photo panel or cockpit recording

TABLE 10-4 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
11	Airspeed indicator correction at flare entry	ΔV_{IC_t}		kt	From instrument calibration curves at (10)
12	Airspeed system position error correction at flare entry	ΔV_{PE_t}		kt	From airpseed position error calibration curve at (10)
13	Calibrated airspeed at flare entry	V_{CAL_t}	$V_{CAL_t} = V_{IND_t} + \Delta V_{IC_t} + \Delta V_{PE_t}$	kt	(13) = (10) + (11) + (12)
14	True airspeed at flare entry	V_{T_t}	$V_{T_t} = V_{CAL_t} \sqrt{\sigma_t}$	kt	(14) = (13) / $\sqrt{(9)}$
15	Indicated rotor speed at flare entry	$N_{R_{IND_t}}$		rpm	From photo panel or cockpit recording
16	Instrument correction for rotor speed at flare entry	$\Delta N_{R_{IC_t}}$		rpm	From instrument calibration curves at (15)
17	Rotor speed at flare entry	N_{R_t}	$N_R = N_{P_{IND_t}} + \Delta N_{R_{IC_t}}$	rpm	(17) = (15) + (16)
18	Pitch attitude at flare entry	θ_t		deg	From oscillograph
19	Longitudinal stick position at flare entry	$\delta_{s_{e_t}}$		in. from full aft	From oscillograph
20	Normal acceleration at flare entry	A_{z_t}		g's	From oscillograph
21	Longitudinal acceleration at flare entry	A_{x_t}		g's	From oscillograph

TABLE 10-4 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
22	Engine start gross weight	$ESGW$		lb	From preflight instrument sheet
23	Weight of fuel used	FU_W		lb	From fuel used determination, Table 9-4
24	Gross weight	GW	$GW = ESGW - FU_W$	lb	(24) = (22) − (23)
25	Indicated pressure altitude at flare termination	$H_{P_{IND_3}}$		ft	From photo panel or cockpit recorded data
26	Altimeter instrument correction	$\Delta H_{P_{IC_3}}$		ft	From instrument calibration curves at (25)
27	Pressure altitude at flare termination	H_{P_3}	$H_{P_3} = H_{P_{IND_3}} + \Delta H_{P_{IC_3}}$	ft	(27) = (25) + (26)
28	Height required to flare	ΔH	$\Delta H = H_{P_i} - H_{P_3}$	ft	(28) = (4) − (27)
29	Indicated ambient air temperature at flare termination	$T_{a_{IND_3}}$		°C	From photo panel or cockpit recorded data
30	Ambient air temperature instrument correction at flare termination	$\Delta T_{a_{IC_3}}$		°C	From instrument calibration curves
31	Ambient air temperature at termination of flare	T_{a_3}	$T_{a_3} = T_{a_{IND_3}} + \Delta T_{a_{IC_3}}$	°C	(31) = (29) + (30)

TABLE 10-4 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
32	Density altitude at termination of flare	H_{D_3}		ft	Calculated or from tables at (27) and (31)
33	Indicated airspeed at flare termination	V_{IND_3}		kt	From photo panel or cockpit recorded data
34	Airspeed instrument correction at flare termination	ΔV_{IC_3}		kt	From instrument calibration curves at (33)
35	Airspeed system position error at flare termination	ΔV_{PE_3}		kt	From airspeed system calibration curve at (33)
36	Calibrated airspeed at termination of flare	V_{CAL_3}	$V_{CAL_2} = V_{IND_2} + \Delta V_{IC_3} + \Delta V_{PE_3}$	kt	(36) = (33) + (34) + (35)
37	Air density ratio at flare termination	σ_3			Calculated or from tables at (32)
38	True airspeed at termination of flare	V_{T_3}	$V_{T_3} = V_{C_3} / \sqrt{\sigma_3}$	rpm	(38) = (36) / $\sqrt{(37)}$
39	Airspeed loss during the flare	ΔV	$\Delta V = V_{T_1} - V_{T_3}$	kt	(39) = (14) − (38)
40	Indicated rotor speed at termination of flare	$N_{R_{IND_3}}$		rpm	From oscillograph or photo panel data
41	Instrument correction for rotor speed	$\Delta N_{R_{IC_3}}$		rpm	From instrument calibration curves at (40)

TABLE 10-4 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
42	Rotor speed at termination of flare	N_{R_3}	$N_{R_3} = N_{R_{IND_3}} + \Delta N_{R_{IC_3}}$	rpm	(42) = (40) + (41)
43	Rotor speed increase during flare	ΔH_R	$\Delta N_R = N_{R_3} - N_{R_1}$	rpm	(43) = (42) – (17)
44	Normal acceleration at termination of flare	A_{z_3}		g's	From oscillograph or photo panel data
45	Normal acceleration increase during the flare	ΔA_z	$\Delta A_z = A_{z_3} - A_{z_1}$	g's	(45) = (44) – (20)
46	Longitudinal stick position at termination of flare	$\delta_{s_{e_3}}$		in. from full aft	From oscillograph
47	Longitudinal stick used during flare	$\Delta\delta_{s_e}$	$\Delta\delta_{s_e} = \delta_{s_{e_3}} - \delta_{s_{e_1}}$	in.	(47) = (46) – (19)

TABLE 10-5

DATA REDUCTION FORM FOR LOW ALTITUDE HOVER AND FORWARD SPEED LANDINGS

STEP NO	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1	Point No./Flight No.				
2	Indicated pressure altitude at power failure	$H_{P_{IND}}$		ft	From photo or cockpit recorded data
3	Altimeter instrument correction at power failure	$\Delta H_{P_{IC}}$		ft	From instrument calibration curves
4	Pressure altitude at power failure	H_P	$H_P = H_{P_{IND}} + \Delta H_{P_{IC}}$	ft	(4) = (2) + (3)
5	Gear height above the ground at power failure	H_G		ft	From camera or theodolite
6	Indicated ambient air temperature at power failure	$T_{a_{IND}}$		°C	From photo panel or cockpit recorded data
7	Temperature indicator correction at power failure	$\Delta T_{a_{IC}}$		°C	From instrument calibration curves
8	Ambient temperature at power failure	T_a	$T_a = T_{a_{IND}} + \Delta T_{a_{IC}}$	°C	(8) = (6) + (7)
9	Density altitude at power failure	H_D		°C	Calculated or from tables at (4) and (8)
10	Air density at power failure	ρ		slug/ft^3	Calculated or from tables at (9)

TABLE 10-5 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
11	Square root of air density ratio	$\sqrt{\sigma}$	$\sqrt{\sigma} = \sqrt{\rho/0.0023769}$		Calculated or from tables at (9) $(11) = \sqrt{(10)/0.0023769}$
12	Indicated airspeed at power failure	V_{IND}		kt	From photo panel or cockpit recorded data
13	Airspeed instrument correction at power failure	ΔV_{IC}		kt	From instrument calibration curves at (12)
14	Airspeed position error at power failure	ΔV_{PE}		kt	From airspeed system calibration curves at (12)
15	Calibrated airspeed at power failure	V_{CAL_1}	$V_{CAL_1} = V_{IND} + \Delta V_{IC} + \Delta V_{PE}$	kt	(15) = (12) + (13) + (14)
16	True airspeed at power failure	V_{T_1}	$V_{T_1} = V_{CAL_1}/\sqrt{\sigma}$	kt	(16) = (15) / (11)
17	Surface wind at touchdown	V_W		kt	From ground station
18	Wind direction degrees from nose	θ_W		deg	From ground station
19	Headwind velocity component	V_{HW}	$V_{HW} = V_W \cos\theta_W$	kt	(19) = (17) cos (18)
20	True ground speed	V_{T_G}	$V_{T_G} = V_{T_1} - V_{HW}$	kt	(20) = (16) − (19)
21	Indicated ground speed	V_{IND_G}		ft/sec	From Fairchild Flight Analyzer

TABLE 10-5 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS	
22	True ground speed	V_{T_G}	$V_{T_G} = V_{IND_G} \times 0.59$	kt	Fairchild data converted to knots	(22) = (21)·0.592
23	True airspeed	V_T	$V_T = V_{T_G} + V_{HW}$	kt	Fairchild Analyzer corrected for head-wind component	(23) = (22) + (19)
24	Indicated rotor speed at power failure	$N_{R_{IND}}$		rpm	From photo panel or cockpit recorded data	
25	Rotor speed instrument correction at power loss	$\Delta N_{R_{IC}}$		rpm	From instrumentation calibration curves	
26	Rotor speed at power failure	N_{R_1}	$N_{R_1} = N_{R_{IND}} + \Delta N_{R_{IC}}$	rpm		(26) = (24) + (25)
27	Collective stick position at power loss	$\delta_{s_{c_1}}$		in. from full down	From oscillograph or cockpit recorded data	
28	Indicated airspeed at ground contact	V_{IND_2}		kt	From A/C system photo panel or cockpit recording	
29	Airspeed indicator correction	ΔV_{IC}		kt	From instrumentation calibration curves	
30	Airspeed position error correction	ΔV_{PE}		kt	From airspeed system calibration curves	
31	Calibration airspeed at ground contact	V_{CAL_2}	$V_{CAL_2} = V_{IND_2} + \Delta V_{IC} + \Delta V_{PE}$	kt		(31) = (28) + (29) + (30)

TABLE 10-5 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
32	True airspeed at ground contact	V_{T_2}	$V_{T_2} = V_{CAL_2} / \sqrt{\sigma}$		(32) = (31) / (11)
33	Airspeed lost during landing	ΔV	$\Delta V = V_{T_1} - V_{T_2}$	kt	(33) = (16) − (32)
34	Ground speed at contact	V_{G_T}		kt	From Fairchild Flight Analyzer
35	Ground distance traveled after contact	S		ft	From Fairchild Flight Analyzer
36	Indicated rotor speed at ground contact	$N_{R_{IND}}$		rpm	From oscillograph or cockpit recording
37	Rotor speed instrument correction at ground contact	$\Delta N_{R_{IC}}$		rpm	From instrument calibration curves at (36)
38	Rotor speed at ground contact	N_{R_2}	$N_{R_2} = N_{R_{IND}}' + \Delta N_{R_{IC}}$	rpm	(38) = (36) + (37)
39	Rotor speed loss during landing	ΔN_R	$\Delta N_R = N_{R_1} - N_{R_2}$	rpm	(39) = (26) − (38)
40	Collective stick position at ground contact	$\delta_{s_{c_2}}$		in. from full down	
41	Collective used during landing	$\Delta\delta_{s_c}$	$\Delta\delta_{s_c} = \delta_{s_{c_2}} - \delta_{s_{c_1}}$	in. from full down	(41) = (40) − (27)

TABLE 10-5 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
42	Normal acceleration at ground contact	A_{z_2}		g's	From oscillograph or cockpit recorded data
43	Sink rate at ground contact	$(R/D)_2$		ft/sec	From Fairchild Flight Analyzer

A/C Model A/C S/N

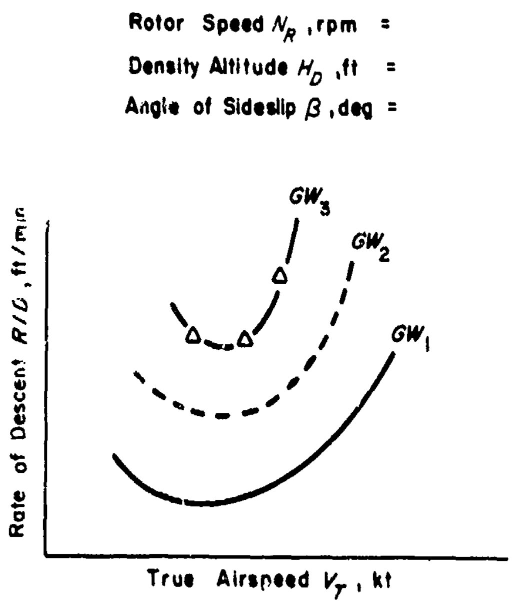


Figure 10-9. Autorotation Performance Variation With Airspeed and Gross Weight

A/C Model A/C S/N

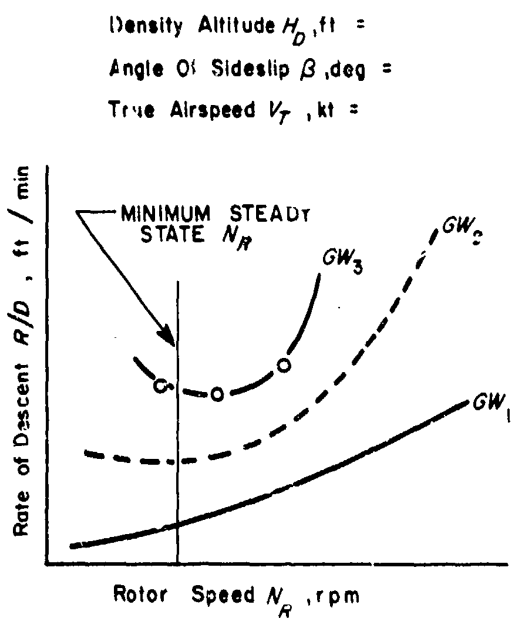


Figure 10-11. Autoroation Performance Variation With Rotor Speed and Gross Weight

A/C Model A/C S/N

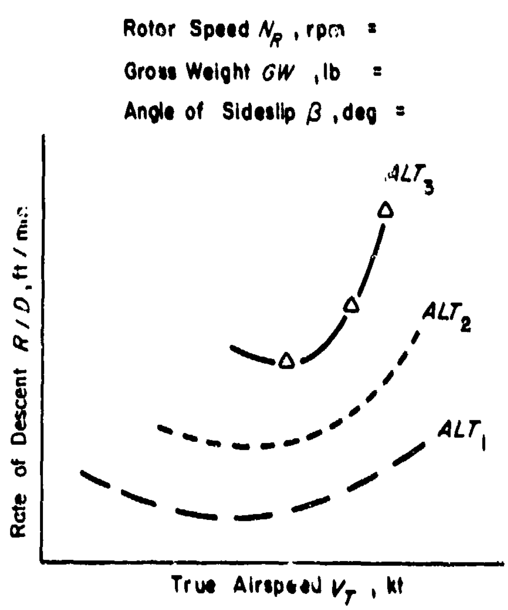


Figure 10-10. Autorotation Performance Variation With Airspeed and Altitude

A/C Model A/C S/N

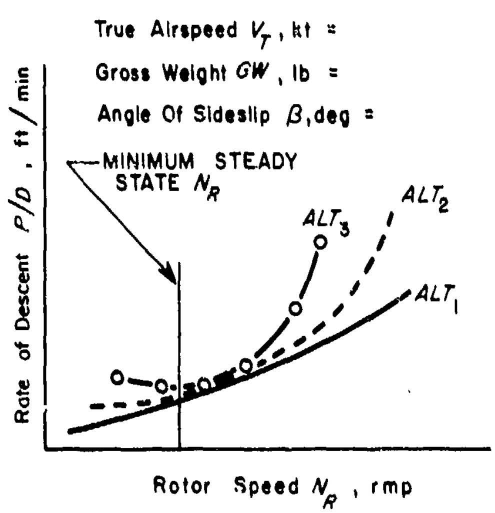


Figure 10-12. Autorotation Performanc Variation With Rotor Speed and Altitude

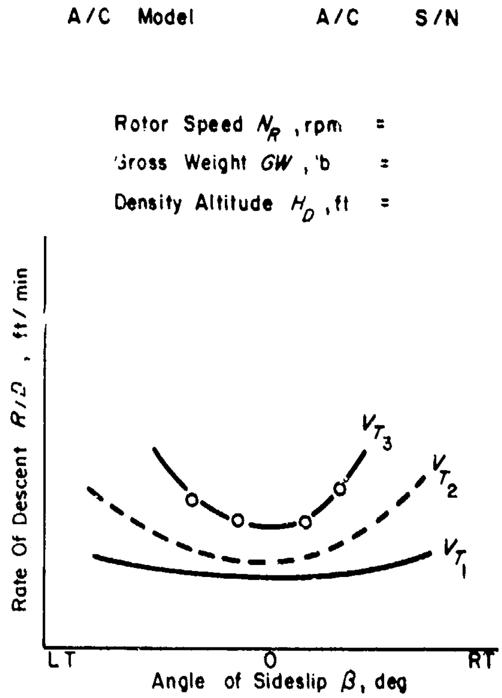


Figure 10-13. Autorotation Performance Variation With Sideslip Angle and Airspeed

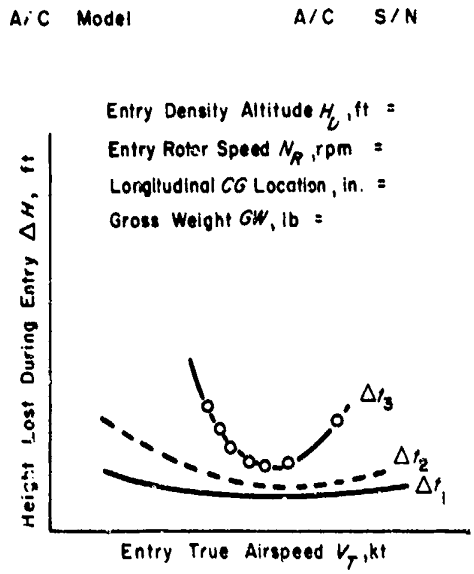


Figure 10-15. Height Required for Autorotation Entry

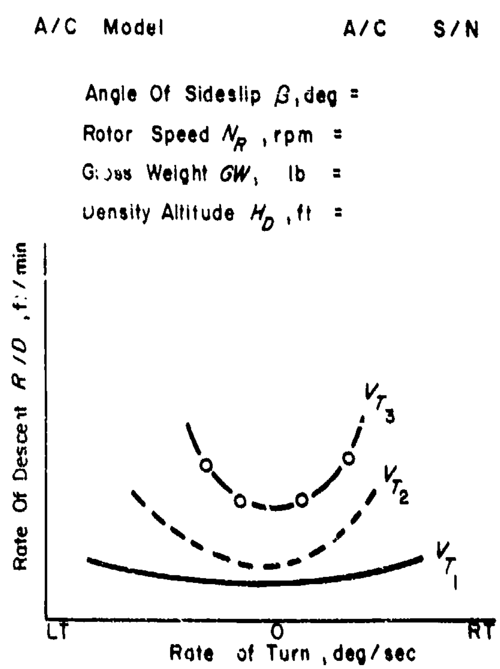


Figure 10-14. Autorotation Performance Variation With Turn Rate and Airspeed

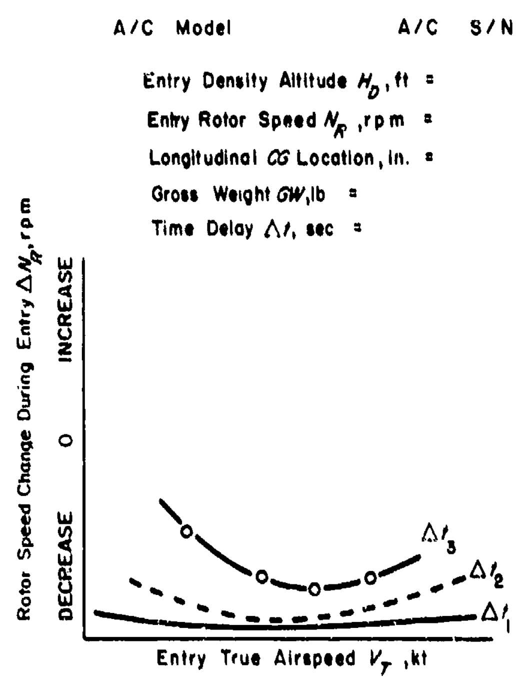


Figure 10-16. Rotor Speed Change During Autorotation Entry

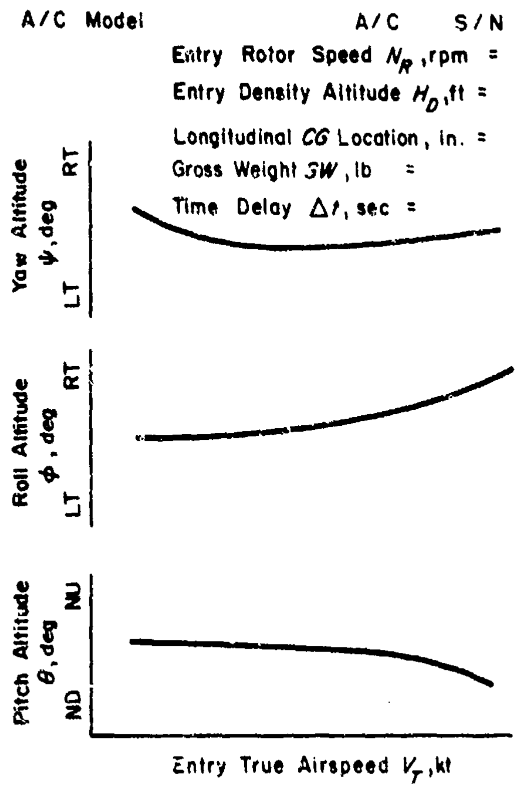


Figure 10-17. Attitude Change During Autorotation Entry

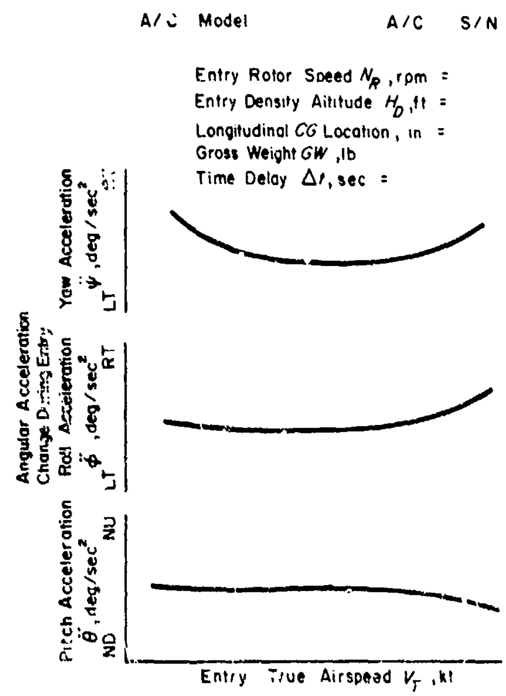


Figure 10-19. Angular Acceleration Changes During Autorotation Entry

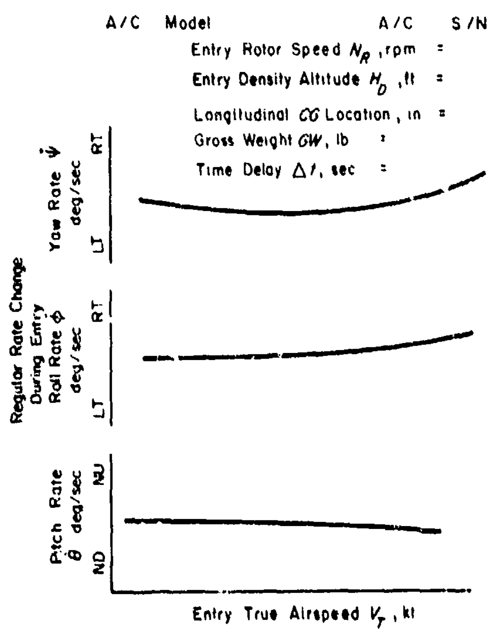


Figure 10-18. Angular Rate Changes During Autorotation Entry

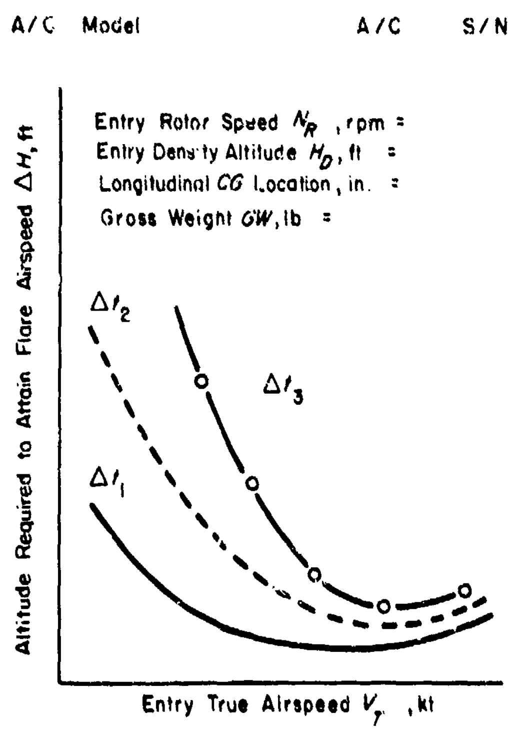


Figure 10-20. Height Required To Attain Flare Airspeed

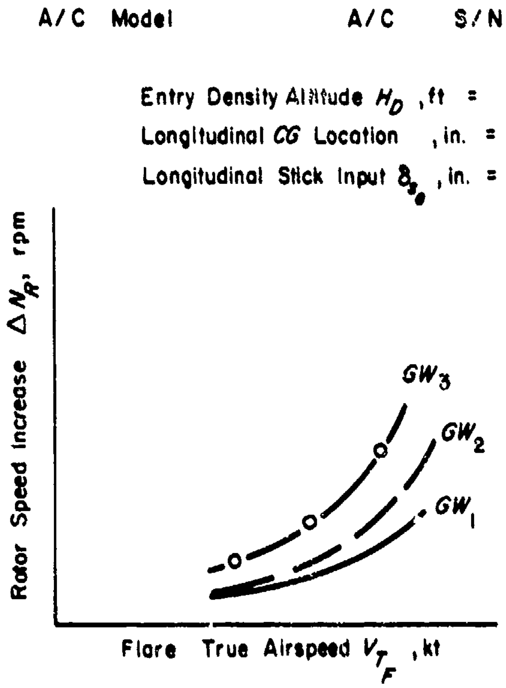


Figure 10-21. Rotor Speed Increase During Flare

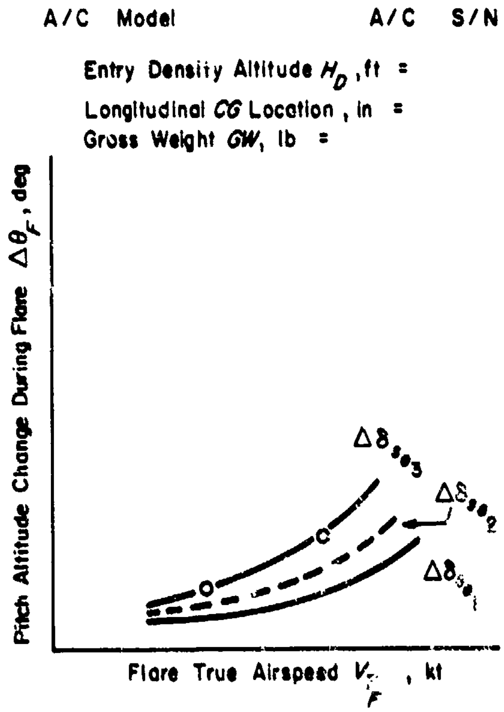


Figure 10-23. Pitch Attitude Change During Flare

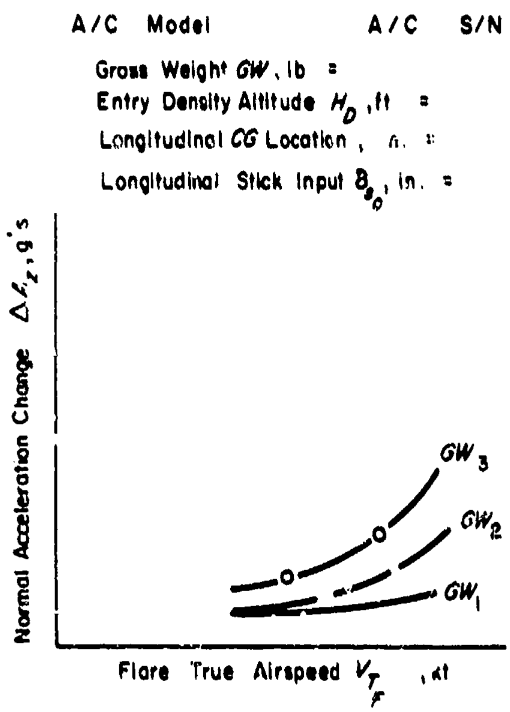


Figure 10-22. Normal Acceleration Change During Flare

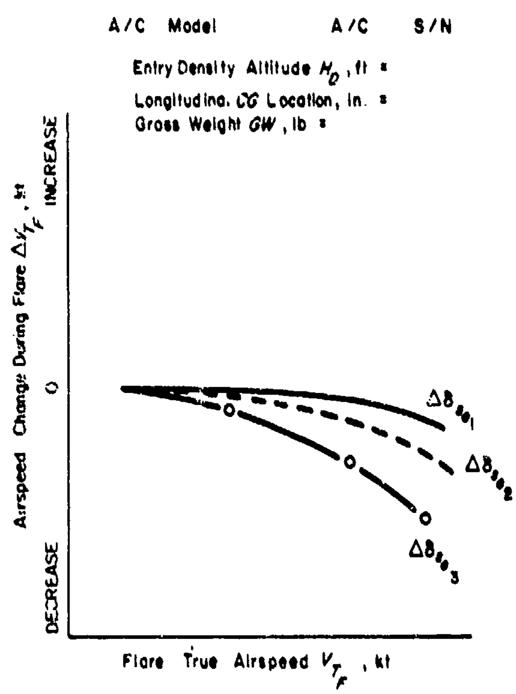


Figure 10-24. Airspeed Change During Flare

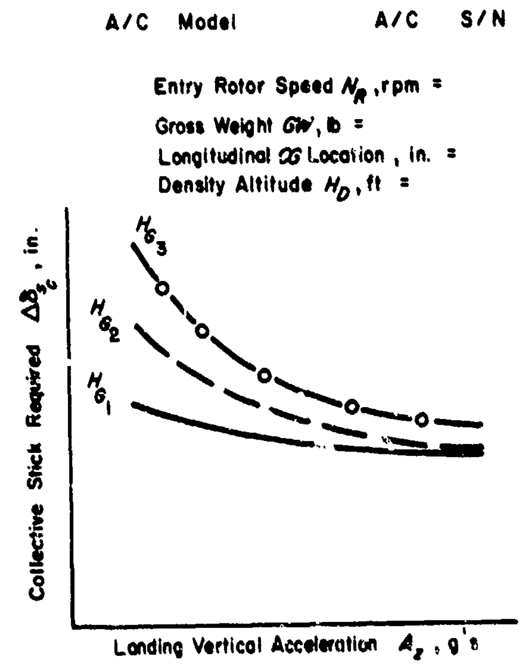


Figure 10-25. Collective Stick Required During Hover Landing at Various Entry Gear Heights

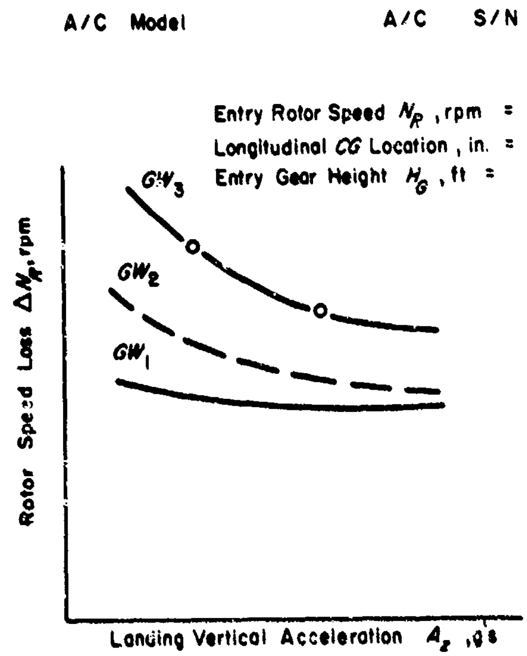


Figure 10-27. Rotor Speed Required During Hover Landing

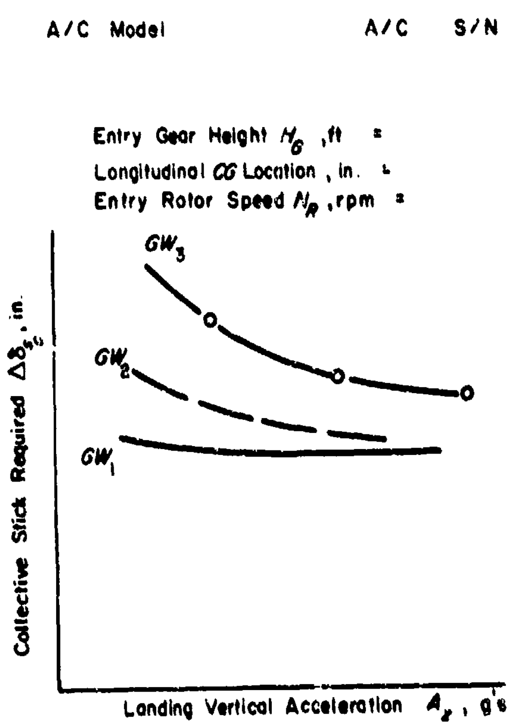


Figure 10-26. Collective Stick Required During Hover Landing at Various Gross Weights

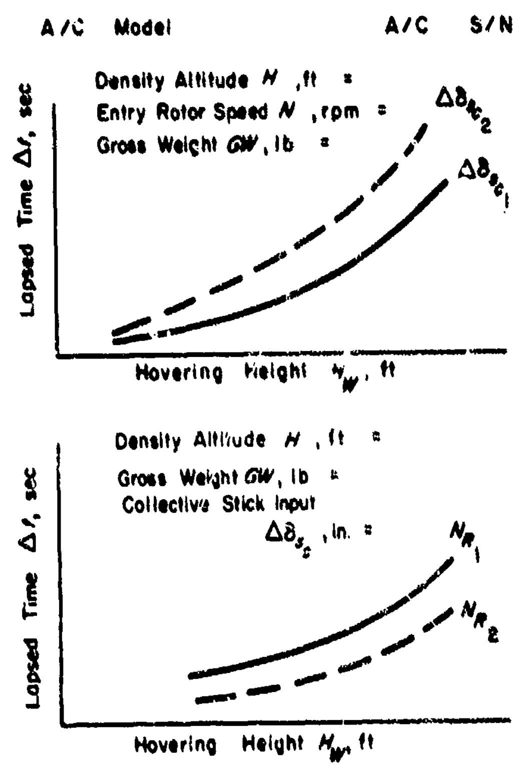


Figure 10-28. Lapsed Time During Hover Landing

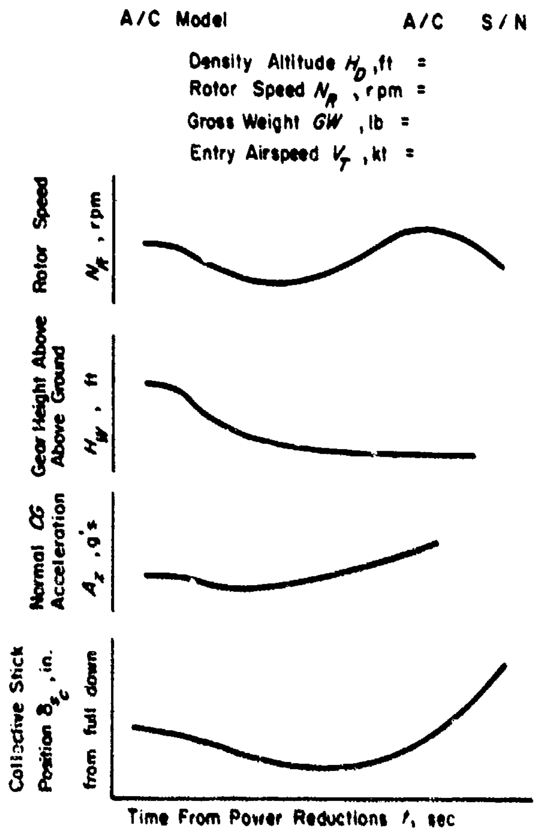


Figure 10-29. Hover Landing Characteristics

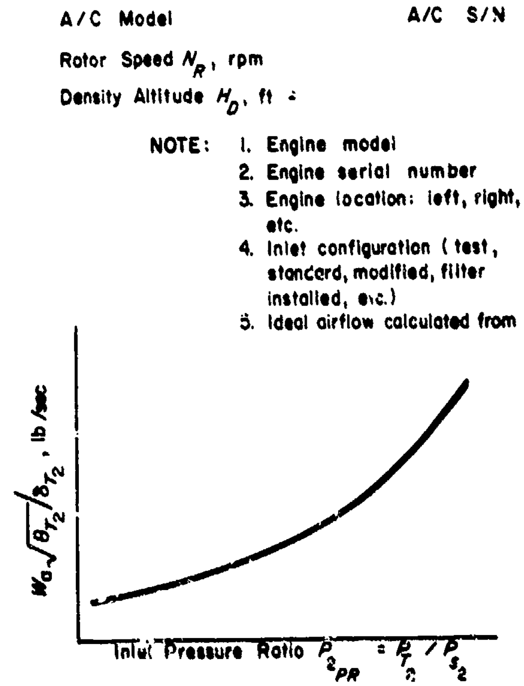


Figure 10-31. Low Speed, Low Height Landing Acceleration

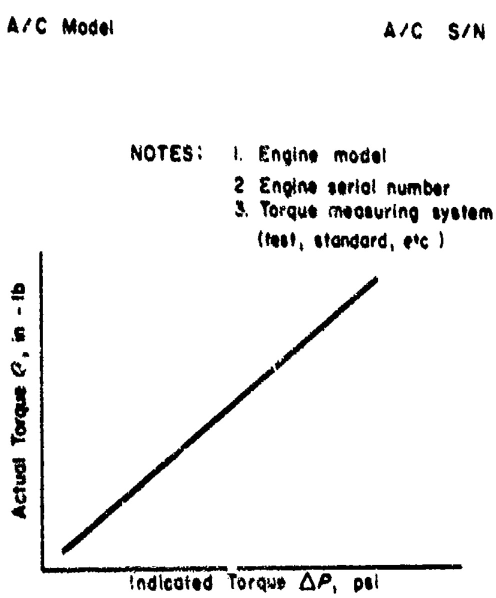


Figure 10-30. Low Speed, Low Height Landing Distance

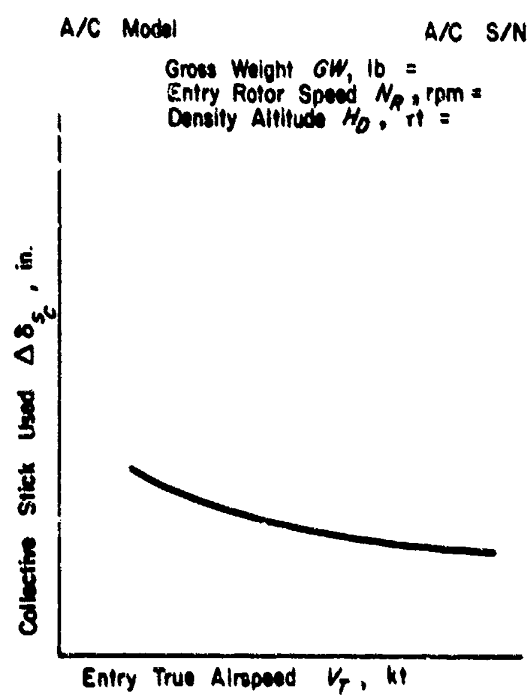


Figure 10-32. Collective Stick Required During Low Speed, Low Height Landing

A/C Model A/C S/N

Entry Rotor Speed N_R, rpm =
Gross Weight GW, lb =
Density Altitude H_D, ft =

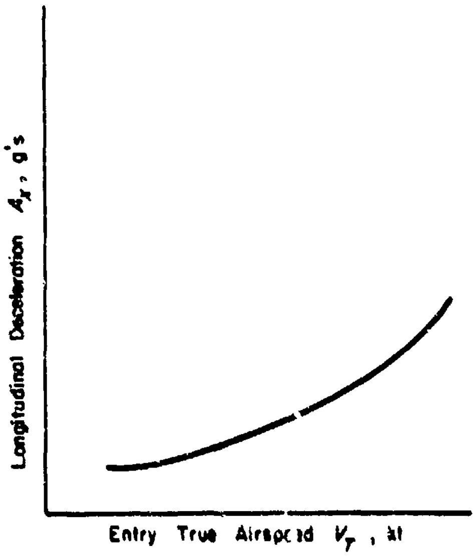


Figure 10-33. Longitudinal Landing Decelerations During Low Speed, Low Height Landing

A/C Model A/C S/N

Wind Velocity V_W, kt =
Density Altitude H_D, ft =
Gross Weight GW, lb =
Minimum % Gear Load On Touchdown =
Rotor Speed N_R, rpm =

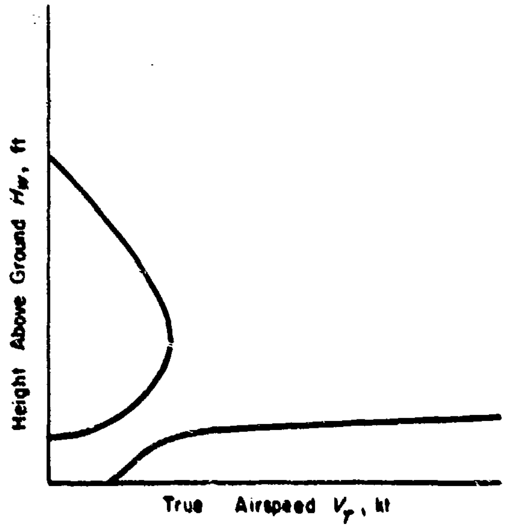


Figure 10-34. Minimum Height Required To Make a Safe Autorotational Landing

CHAPTER 11

POWERED DESCENT AND LANDING PERFORMANCE

11-1 GENERAL

Descent performance in a helicopter most often is associated with an autorotation condition. It is interesting to note, however, that most descents actually involve some power input to the rotor. During a normal descent, this power level is the minimum required to control the glide angle and the rate of descent during the approach. For single engine helicopters, there is no general requirement to conduct performance tests in this partial power flight regime. For multiengined helicopters, loss of power on one engine may necessitate a powered descent depending upon the remaining power available from the other engine and the power required to sustain level flight. Thus, it is important to know the performance characteristics for the envelope range of gross weight, altitude, and power available.

The maximum performance powered landing capability is of importance to all helicopters. For large excess power margins, the helicopter can descend easily to a hover at any desired height and then a landing can be made from this point. This should be the normal operating condition. As the excess performance margin decreases, the hover capability is less and the descent rate is greater. For some gross weight, altitude, and power conditions, a vertical descent and landing is impossible due to the rotor and aircraft structural limitations. Thus, it is possible for the aircraft to takeoff at a low altitude site and ferry a large cargo to a site where a hovering capability does not exist, offload, and then depart at a greatly reduced gross weight. During these types of landings, full power is applied during the approach, and the aircraft becomes committed to a landing while relatively high above the ground. The proper technique must be used to manage airspeed and rotor speed carefully in order to prevent an excessive rate of descent from developing. Though demanding, this maneuver can be accomplished safely when the proper technique is used and sufficient rotor kinetic energy is available. The information obtained during these tests is needed to establish the emergency procedures in the operator's manual for engine-out landing of multiengine aircraft. The data are also valuable for evaluating the aborted takeoff capability and techniques.

The powered descent tests are conducted to determine the steady-state rate of descent as a function of the difference between descent power available and the level flight power required. The rotor and aircraft dynamic response following power loss on one engine is also of importance, and techniques must be developed to cope with the particular behavior of the vehicle. These tests will provide the data needed to calculate the range and time available for the pilot to select the most suitable landing area. The landing tests provide the data necessary to determine the horizontal distance required to land in a confined area, the engine power and rotor energy required to arrest rate of descent, and the minimum approach airspeed as a function of performance margin. During the approach and landing, the engine response, dynamic thrust characteristics, and flight control limitations are significant factors. The techniques established should be suitable for the different types of landings and conditions, and should be compatible with the anticipated mission requirements within the aircraft capabilities.

The steady-state descent performance tests are similar in nature to the level flight performance tests, while the landing tests are most similar to the takeoff and autorotational landing tests with marginal test conditions and techniques for most cases. The scope of the tests should include the maximum range of speed, gross weight, and altitude conditions. Rotor speed and center of gravity should encompass the most critical conditions within the operational limits. In the event high altitude landing sites are unavailable, the performance margin conditions should be simulated at low altitude through control of power or weight parameters. Maximum flight safety during the tests requires a buildup of experience based on related performance values as they are obtained.

The planning for the descent performance tests should be coordinated closely with the level flight test planning since the two easily can be conducted concurrently. The logistics and equipment requirements are identical to the takeoff and landing or autorotational landing performance tests. Consideration must be given to the technical requirements such as engineering support and data reduction, particularly when operating at remote sites. In the test sequence, the descent tests should follow the level flight performance tests, while the landing tests are dependent upon data from the hover and vertical climb results.

The necessary instrumentation for these tests includes both airborne and ground equipment. While the descent performance is essentially static, the landing flight conditions are primarily dynamic and require automatic data recording systems to evaluate properly the aircraft response. Photographic equipment also should be used to record the flight path and aircraft behavior.

11-2 PLANNING

The descent and landing tests should be conducted in conjunction with the level flight and takeoff performance tests. The planning procedure discussed during the sawtooth climbs is adequate for the powered descent tests, while the landing test requirements are similar to those for the takeoff tests. When these tests are properly integrated, conflicts and flight requirements to obtain the additional data will be minimized.

11-3 INSTRUMENTATION

The instrumentation discussed in the level flight and takeoff performance is suitable for the descent tests with the exception of that needed to record the landing information. Linear longitudinal and vertical accelerometers should be installed at the center of gravity to record landing loads and decelerations. Appropriate landing gear instrumentation also should be installed to record forces. These data are recorded on the oscillograph and displayed visually in the cockpit to enable the flight crew to monitor the test progress and evaluate any variations occurring in the critical landing parameters. For the landings, real time data presentation on a ground station should be used when possible.

11-4 TEST METHODS

11-4.1 GENERAL

The partial power entry tests are conducted to evaluate a normal power reduction for all configurations and for an engine failure on multiengine helicopters. The specific objectives of these power reduction tests are to discover any safety of flight implications with respect to rotor, power train, or aircraft characteristics.

For normal power reductions, the most significant test variables are airspeed, gross weight, and altitude. The airspeed range investigated should be from a hover to the maximum permissible at the limit gross weight and altitude. Since these conditions are the most critical end points, an incremental build-up technique should be used. With this approach, the first test then will be a sea level altitude and a light gross weight condition. The magnitude of the aircraft reaction usually follows the trend of the power required curve and will thus be smallest at the minimum power condition. From the initial test condition, the airspeed is varied incrementally to both the higher and lower airspeed values.

The performance and reaction normally will vary with the rate and magnitude of the power reduction. These parameters should be varied both individually and simultaneously to detect any adverse characteristics and their magnitude. This procedure is continued until the most critical power reduction condition is established. The initial tests are conducted best at the least critical rotor speed, which is usually the maximum and then proceed to the more critical conditions. The information gained during these tests is evaluated for inclusion in the emergency procedures of the operator's manual. The stabilized data at the specified entry conditions are recorded prior to the power reduction. The necessary in-flight data to be recorded are shown in Table 11-1.

TABLE 11-1

MANUALLY RECORDED DATA FOR PARTIAL POWER ENTRY TESTS

Airspeed	Ambient air temperature
Rotor speed	Airspeed (recovery)
Altitude	Rotor speed at recovery
Engine power	Altitude at recovery
Gross weight	Engine power at recovery

A countdown should be used to start the automatic recording system which then records continuously during the maneuver. The power reduction is accomplished and the aircraft is stabilized in a steady-state partial power descent. The control requirements and aircraft characteristics should be monitored closely during the build-up tests for any progressive changes that might indicate hazardous trends. It should be expected that the most critical condition will be a rapid, total power failure.

The detailed test methods presented in par. 10-4, will suffice for the in-flight monitoring and analysis of results. The test procedure is repeated for increased gross weights and altitudes. The higher collective settings at these conditions usually will result in more pronounced aircraft reactions and greater changes in rotor speed and altitude.

The test procedure used to evaluate an inadvertent power failure on a multiengine helicopter is basically the same as that previously discussed for a normal entry. The primary difference is that the power reduction is on a single engine and, for that engine, is both sudden and complete. As might be expected, the response will vary with the flight condition and the aircraft characteristics. The pilot reaction time needed to recognize the failure and to initiate corrective action must be established. The delay time will depend on the particular aircraft response and should be less critical for a single engine failure on a multiengine aircraft than a complete power failure on a single engine helicopter. A build-up procedure should be used to evaluate the rate of power reduction and recognition time. The data recording and in-flight procedures are similar to those previously discussed for the single engine helicopter.

11-4.2 POWERED DESCENT PERFORMANCE

The difference between descent performance tests and level flight tests is that insufficient power is available to sustain level flight and a rate of descent develops. Two flight test techniques commonly are used to obtain the data. In both cases, the aircraft is in a stable descent. The test is conducted at density altitude and weight conditions which will yield the desired thrust coefficients. To preclude introducing any density altitude effects, the test should be conducted through an altitude increment of 1000 ft. The aircraft should be stabilized at such a height that the data can be recorded from 500 ft above the aim altitude to 500 ft below it. A minimum of five points should be obtained during each descent. The flight and data recording procedures are similar to those described in detail for the sawtooth climbs, par 8-5.3. Several gross weights, altitudes, and power settings should be tested. This should include the maximum gross weight, full power condition, which is normally the most critical. Unless precluded by some aircraft peculiarity, both engines should remain operative during the tests and power should be adjusted to simulate the desired excess performance condition. Since the test is steady-state, manually recorded data will suffice although an automatic data recording device will provide higher density, greater accuracy, and instantaneous values.

The first technique is to stabilize the aircraft in level flight at the best climb speed. This constitutes the first data point and establishes the maximum power that is to be used during this test. On succeeding points, power is held constant while speed is varied incrementally from the initial value. Data are recorded during each of these stabilized descents. Airspeed increments of 10 kt are generally suitable and a minimum of six points is needed to define the curve adequately. The high speed extreme is not as critical nor as important as the low speed region. The low speed data are indicative of the performance to be expected during the maximum performance landing tests. These data are much easier and safer to obtain at altitude than near the ground and will aid greatly in planning and conducting the landing tests. As the speed is decreased, the rate of descent may build rapidly and power settling may occur. Care should be taken to allow sufficient altitude for a power recovery. Stability and control characteristics should be evaluated prior to conducting any tests in this area.

During the descent, a change in power available may result from altitude or ambient temperature changes. The pilot should not attempt to correct for this since a correction will be made during the data reduction. The lapsed time for the altitude increment is recorded and is a relative measure of the rate of descent. These data should be used to construct an in-flight plot as shown in Fig. 11-1. A curve faired through the points will aid in monitoring the test and will indicate questionable points or trends that need further investigation.

The second technique is to stabilize at the best climb speed and record the trim conditions. Descents then are obtained by incrementally reducing power while maintaining airspeed constant. The power increments should be selected so that six points will be obtained and the minimum power point should be a full autorotation. The data recording procedures are similar to the first technique, and an in-flight plot similar to Fig. 11-2 is constructed.

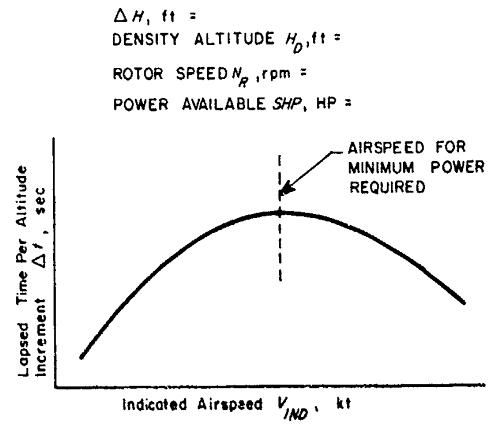


Figure 11-1. Rate of Descent Variation With Airspeed

A third test often is conducted to evaluate rotor performance changes with rotor speed. This may be accomplished in two ways. The aircraft is trimmed at best climb speed as in the previous tests and rotor speed varied incrementally. One technique is to maintain power constant and measure the rate of descent. Another is to maintain altitude and measure the power required. In-flight plots similar to Fig. 11-1 can be made of appropriate variables to monitor test progress and validity.

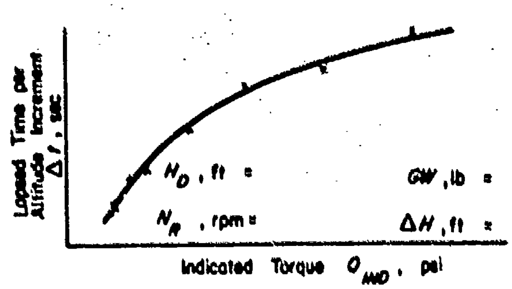


Figure 11-2. Rate of Descent Variation With Power

11-4.3 POWERED LANDING PERFORMANCE

11-4.3.1 VERTICAL LANDING

The vertical landing tests often are conducted concurrently with the vertical climb and takeoff performance tests as discussed in par. 8-4. These normally do not cover an extensive altitude range and may not reflect the capability over the performance envelope. The best coverage is obtained by making landings at the same conditions used for the takeoff tests. Previous descent information should be evaluated prior to entering any critical landing conditions.

11-4.3.2 FORWARD SPEED LANDINGS

This technique is used when an insufficient performance margin is available for a vertical landing and sufficient space exists for a forward speed approach and landing. The low speed descent data previously obtained as a function of power deficiencies will be an indication of the descent rates that can be expected, while hover and vertical landing data may be used to estimate the rotor capability and ground proximity effects. The horizontal distance required to land over a 50-ft obstacle will be dictated by the approach angle. For reduced excess performance margins, shallow approach angles usually will result in low rates of descent and high forward touchdown speeds. The terrain must be suitable for these landings. Increasing the approach angle will reduce horizontal distance, approach velocity, and touchdown speed, but will increase the rate of descent. When the approach airspeed is less than the minimum autorotation flare airspeed, the aircraft may be in an unsafe portion of the height-velocity diagram for a relatively extended period of time. The exposure time is generally at a maximum during a vertical descent. The basic tests generally are conducted at the maximum rotor speed with the number of additional explorations at lower values dependent upon the nature and magnitude of the changes discovered during the test progress. Most airspeed systems are unreliable at the low speeds and there may be large position errors. Altitude or rate of descent systems are also inaccurate, and the unusual flow conditions can cause significant lag errors.

The flight risk can be reduced by first conducting the tests for conditions that are less critical—i.e., large excess performance margins—to determine the proper technique, the nature of the ground effects, and effectiveness of collective application. The performance margin then is decreased to evaluate qualitatively the aircraft characteristics in a condition near those that will be encountered during the tests. When the preparatory tests have been completed, the actual tests are initiated. This build-up technique easily is accomplished during the takeoff tests since they are conducted in a similar manner.

For the first test, the height above the ground should be approximately 300 ft, at a light gross weight, a low density altitude, maximum rotor speed, and an approach speed slightly above minimum autorotational flare airspeed. Rate of descent should be approximately 300 ft/min, or that which can be arrested easily. From this trim condition, the airspeed is decreased incrementally on each successive landing while power is maintained at the original value. This value is less than the power available that provides a margin of safety should an excessive sink rate develop or another contingency arise. The rate of descent will increase on each successive landing when using this technique. The speed increments for these points will be dictated by the previous performance results. The increased steady state rate of descent will steepen the flight path, decrease the horizontal distance, and require more arresting energy. This will necessitate increased collective pitch just prior to the landing. The flight profiles obtained during the test will appear as shown in Fig. 11-3.

The aircraft is stabilized prior to entering the camera range to allow time to establish accurately the baseline entry conditions. At entry, speed is decreased to the aim value and

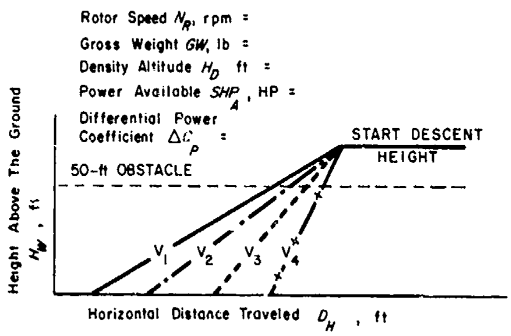


Figure 11-3. Maximum Performance Landing Profiles

held constant during the descent. The entry airspeed is decreased on subsequent tests until some limiting factor is reached. The most common restriction will be gear limits although collective stops or stability and control characteristics may be limiting factors. The real time instrumentation is monitored closely during the buildup to the critical points. Gross weight control during successive runs is important since the effects on performance may not be linear and a small change in weight can result in a significant change in performance.

In the tests where full power is needed, the aircraft will be committed to a landing relatively high above the ground and the rotor kinetic energy and ground effect must be sufficient to arrest the rate of descent. Various techniques of timing and rate of collective application should be evaluated to determine the most effective way to utilize the kinetic energy stored in the rotor. If the collective is applied too soon, the descent rate will be slowed at some height above the ground where insufficient energy may remain to cushion the touchdown due to the low rotor speed. A rapid application of collective may cause an accelerated stall on some portion of the rotor disc, with attendant losses in lift at a critical time. This technique is difficult and very critical. The collective available should be monitored carefully to insure that an unexpected limit is not encountered at an inopportune time.

The absolute minimum descent entry speed will result in a touchdown velocity equal to the gear limit. This, of course, leaves no safety margin and should not be considered as a practical limit. The value for a practical limit must be determined on an individual basis. Surprisingly, this limit is relatively easy for the test crew to establish and, fortunately, is generally much less than the landing gear limits.

When the minimum entry speed for one differential power coefficient condition has been established, sufficient information should be available to decrease the magnitude of the build-up procedures; bearing in mind, of course, that the first condition was least critical and characteristics may change significantly with changes in gross weight, altitude, and rotor speed. The test procedure is repeated for each of the specified conditions. The in-flight technique, with respect to flight parameters, and the weight and environmental control requirements are similar to the take-off performance tests.

11-5 DATA REDUCTION

The powered descent data are recorded manually from visual instruments and may be recorded automatically by airborne equipment. The data are for stabilized conditions, and thus time-averaged values will suffice for each point. The reduction is essentially the same as that used for the sawtooth climb data. A typical data reduction form is shown in Table 11-2*.

The data reduction for powered landing performance must treat the transient conditions and pilot input much the same as for the autorotational landing tests. A typical form is illustrated in Table 11-3*.

11-6 DATA PRESENTATION

The data presentation for powered descent performance is usually time-averaged values with little need for time history type of graphs. Some typical presentations are shown in Figs. 11-4 through 11-12.

*The data reduction forms (tables) are located at the end of each chapter.

TABLE 11-2

DATA REDUCTION FORM FOR POWERED DESCENT

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1	Flight No./Point No.				
1a	Indicated pressure altitude	$H_{p_{IND}}$		ft	From cockpit recording or photo panel data
2	Altimeter instrument correction	$\Delta H_{p_{IC}}$		ft	From instrument calibration curves at (1)
3	Altimeter position error correction	$\Delta H_{p_{PE}}$		ft	From altimeter position error calibration at (2)
4	Pressure altitude	H_p	$H_p = H_{p_{IND}} + \Delta H_{p_{IC}} + \Delta H_{p_{PE}}$	ft	(4) = (1a) + (2) + (3)
5	Indicated ambient air temperature	$T_{a_{IND}}$		°C	From cockpit or photo panel data
6	Ambient temperature instrument error	$\Delta T_{a_{IC}}$		°C	From instrument calibration curves at (5)
7	Ambient temperature position error system correction	$\Delta T_{a_{PE}}$		°C	From instrumentation system calibration at (5)
8	Ambient air temperature	T_a	$T_a = T_{a_{IND}} + \Delta T_{a_{IC}} + \Delta T_{a_{PE}}$	°C	(8) = (5) + (6) + (7)
9	Density altitude	H_D		ft	Calculated or from tables at (4) and (8)
10	Air density	ρ		slug/ft^3	Calculated or from tables at (9)

TABLE 11-2 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
11	Air density ratio	σ	$\sigma = \rho/0.0023769$		Calculated or from tables at (9)
12	Indicated airspeed	V_{IND}		kt	From cockpit recording or photo panel data
13	Airspeed instrument correction	ΔV_{IC}		kt	From instrument calibration curves at (12)
14	Airspeed position error correction	ΔV_{PE}		kt	From instrument position error calibration
15	Calibrated airspeed	V_{CAL}	$V_{CAL} = V_{IND} + \Delta V_{IC} + \Delta V_{PE}$	kt	(15) = (12) + (13) + (14)
16	True airspeed	V_T	$V_T = V_{CAL} / \sqrt{\sigma}$	kt	$(16) = (15) / \sqrt{(11)}$
17	Engine start gross weight	$ESGW$		lb	From preflight data sheet
18	Engine start fuel specific weight	FSW_1		lb/gal	From preflight data sheet
19	Engine stop fuel specific weight	FSW_2		lb/gal	From postflight data sheet
20	Average fuel specific weight	FSW_{AVG}	$FSW_{AVG} = (FSW_1 + FSW_2)/2$	lb/gal	
21	Fuel counter difference	ΔFC	$\Delta FC = FC_2 - FC_1$	ct	From cockpit recording or photo panel data

TABLE 11-2 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
22	Fuel constant	K_{FC}		gal/ct	From average of fuel flow calibration curve
23	Volume of fuel used	FU_{VOL}	$FU_{VOL} = \Delta FC \times K_{FC}$	gal	(23) = (21)·(22)
24	Weight of fuel used	FU_W	$FU_W = FU_{VOL} \times FSW_{AVG}$	lb	(24) = (23)·(20)
25	Gross weight	GW	$GW = ESGW - FU_W$	lb	(25) = (17) – (24)
26	Indicated rotor speed	$N_{R_{IND}}$		rpm	From cockpit recording or photo panel data
27	Instrument correction for rotor speed	$\Delta N_{R_{IC}}$		rpm	From instrument calibration curves
28	Rotor speed	N_R	$N_R = N_{R_{IND}} + \Delta N_{R_{IC}}$	rpm	(28) = (26) + (27)
29	Thrust coefficient constant	K_{c_T}	$K_{c_T} = A(\Omega R)^2$	lb/slug-ft³	Calculated at (28)
30	Power coefficient constant	K_{c_P}	$K_{c_P} = 550/[A(\Omega R)^3]$	slug/ft³-HP	Calculated at (28)
31	Advancing blade tip speed	V_{TIP}	$V_{TIP} = \Omega R$	ft/sec	Calculated at (28)
32	Thrust coefficient	C_T	$C_T = GW/(\rho K_{c_T})$		(32) = (25) / [(10)·(29)]
33	Advance ratio	μ	$\mu = [V_T/(\Omega R)] \times 1.69$		(33) = [(16) / (31)]·1.69

TABLE 11-2 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
34	Level flight power coefficient required	$C_{P_{REQ}}$			From Fig. 9-11 at (32) and (33)
35	Available power	SHP_A		HP	From Table 14-1
36	Available power coefficient	C_{P_A}	$C_{P_A} = SHP_A \times K_{c_p} / \rho$		(36) = [(35)·(30)] / (10)
37	Power coefficient deficiency	ΔC_p	$\Delta C_p = C_{P_A} - C_{P_{REQ}}$		(37) = (36) – (34)
38	Power deficiency	ΔSHP	$\Delta SHP = \Delta C_p \rho / K_{c_p}$	HP	(38) = [(37)·(10)] / (30)
39	Time	t		sec	From cockpit recording or photo panel data
40	Indicated rate of descent	dh/dt		ft/min	From slope of H_p, V_T, t curve at aim altitude
41	Altimeter temperature correction factor	K_{H_p}	$K_{H_p} = \dfrac{T_a + 273}{T_S + 273}$		T_S and T_a at aim altitude
42	Tapeline rate of descent	$(R/D)_T$	$(R/D)_T = dh/dt \times K_{H_p}$	ft/min	(42) = (40)·(41)
43	Flight path angle	γ	$\gamma = \text{Arcsine} [(R/D)_T / V_T] \times 0.00986$	deg	From tables (43) = Arcsine [(42) / (16)] · 0.00986
44	Horizontal distance traveled	$NAMT$	$NAMT = [(H_{p_1} - H_{p_2}) / \tan\gamma] \times 1.646 \times 10^{-4}$	naut air mi	$\tan\gamma$ from tables

TABLE 11-3

DATA REDUCTION FORM FOR POWERED LANDING PERFORMANCE

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1-38					See Table 11-2
39	Time	t		sec	From Fairchild Camera Data, Table 7-7
40	Gear height above the ground	H_G		ft	From Fairchild Camera Data, Table 7-7
41	Horizontal distance traveled from 50-ft H_G to touchdown	X_{50}		ft	From Fairchild Camera Data, Table 7-7
42	Touchdown rate of descent	R/D		ft/sec	From slope of H_G vs t curve at touchdown
43	Touchdown forward ground speed	V_G		ft/sec	From slope of X vs t curve at touchdown
44	Ground roll distance after touchdown	X_G		ft	From Fairchild Camera Data, Table 7-7
45	Collective stick position at H_G = 50 ft	$\delta_{s_{c_1}}$		in. from full down	From oscillograph or cockpit recording
46	Collective stick position at touchdown	$\delta_{s_{c_2}}$		in. from full down	From oscillograph or cockpit recording

TABLE 11-3 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
47	Collective stick used during landing	$\Delta\delta_{s_c}$	$\Delta\delta_{s_c} = \delta_{s_{c_2}} - \delta_{s_{c_1}}$	in.	(47) = (46) − (45)
48	Rotor speed at H_G = 50 ft	N_{R_1}		rpm	From oscillograph or cockpit recording
49	Rotor speed at touchdown	N_{R_2}		rpm	From oscillograph or cockpit recording
50	Rotor speed used during landing	ΔN_R	$\Delta N_R = N_{R_2} - N_{R_1}$	rpm	(50) = (49) − (48)
51	Gear load at touchdown	F_G		lb	From landing gear instrumentation on oscillograph or cockpit recording
52	Normal acceleration at touchdown	A_z		g's	From oscillograph or cockpit recording
53	Horizontal deceleration	A_x		g's	Maximum from touchdown to stop, from oscillograph or cockpit recording

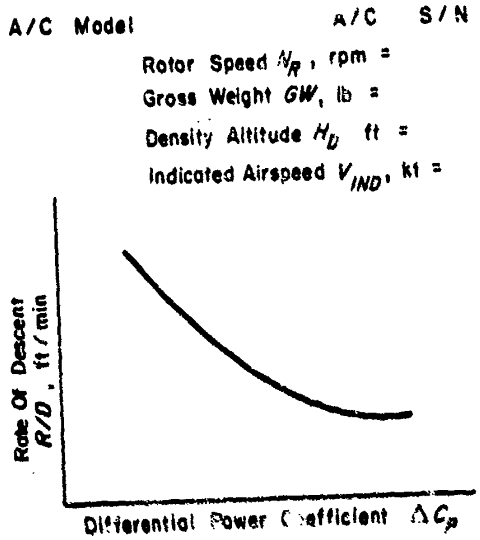


Figure 11-4. Variation in Powered Descent Performance With Airspeed

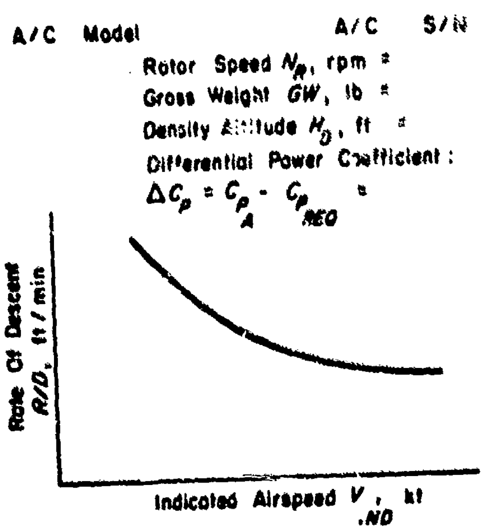


Figure 11-5. Variation in Powered Descent Performance With Differential Power Coefficient

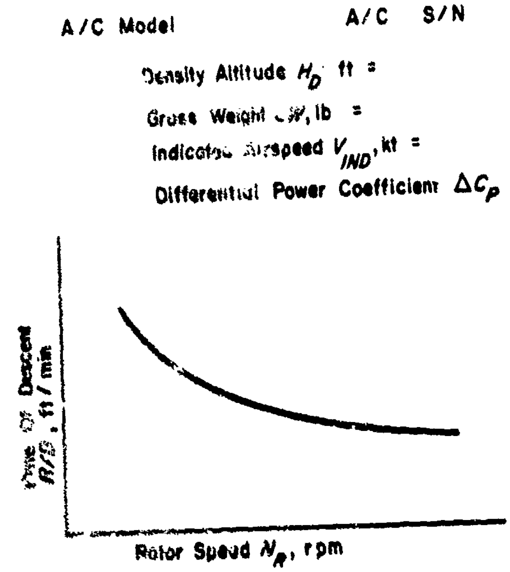


Figure 11-6. Powered Descent Performance Change With Rotor Speed

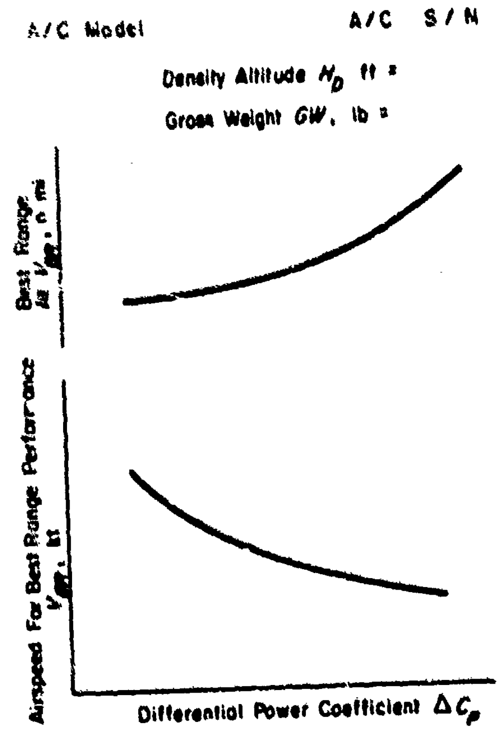


Figure 11-7. Range Performance in Powered Descent

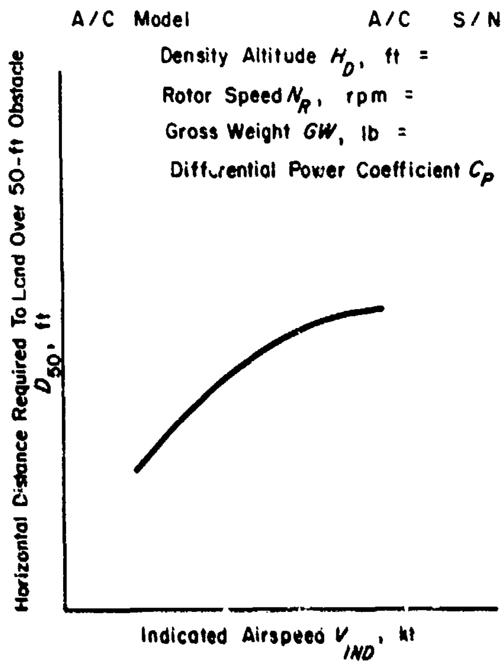


Figure 11-8. Distance Required To Land Over a 50-ft Obstacle

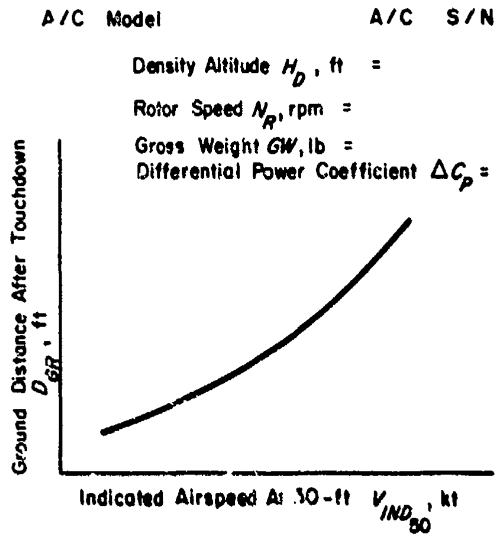


Figure 11-10. Ground Distance Required After Touchdown

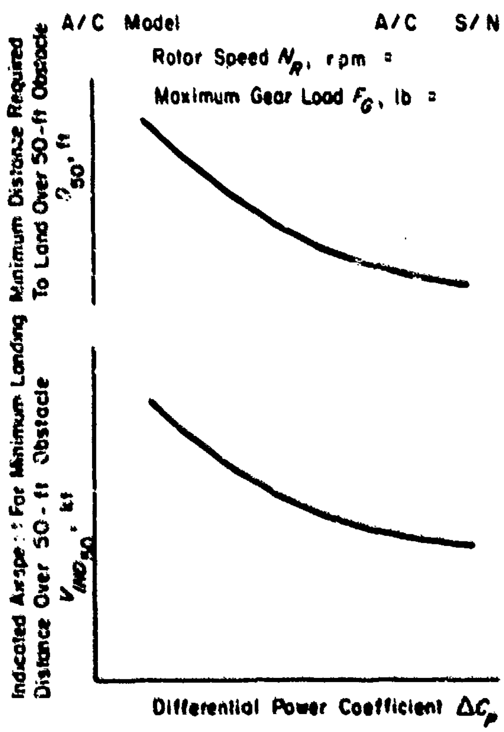


Figure 11-9. Powered Descent Landing Performance Variation With Differential Power Coefficient

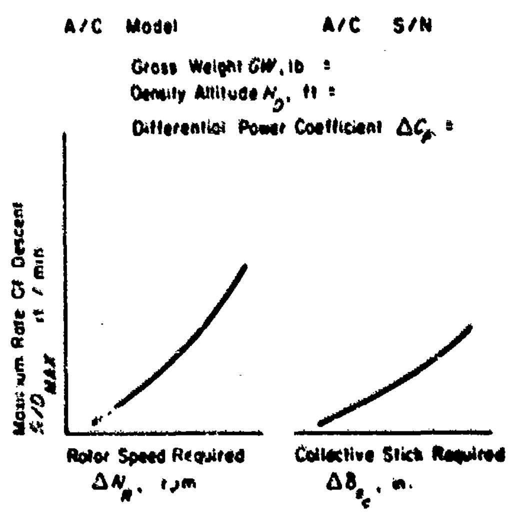


Figure 11-11. Rotor Characteristics During Maximum Performance Landing

A/C Model   A/C   S/N

Gross Weight GW, lb =

Density Altitude H_D, ft =

Rotor Speed N_R, rpm =

Differential Power Coefficient ΔC_p =

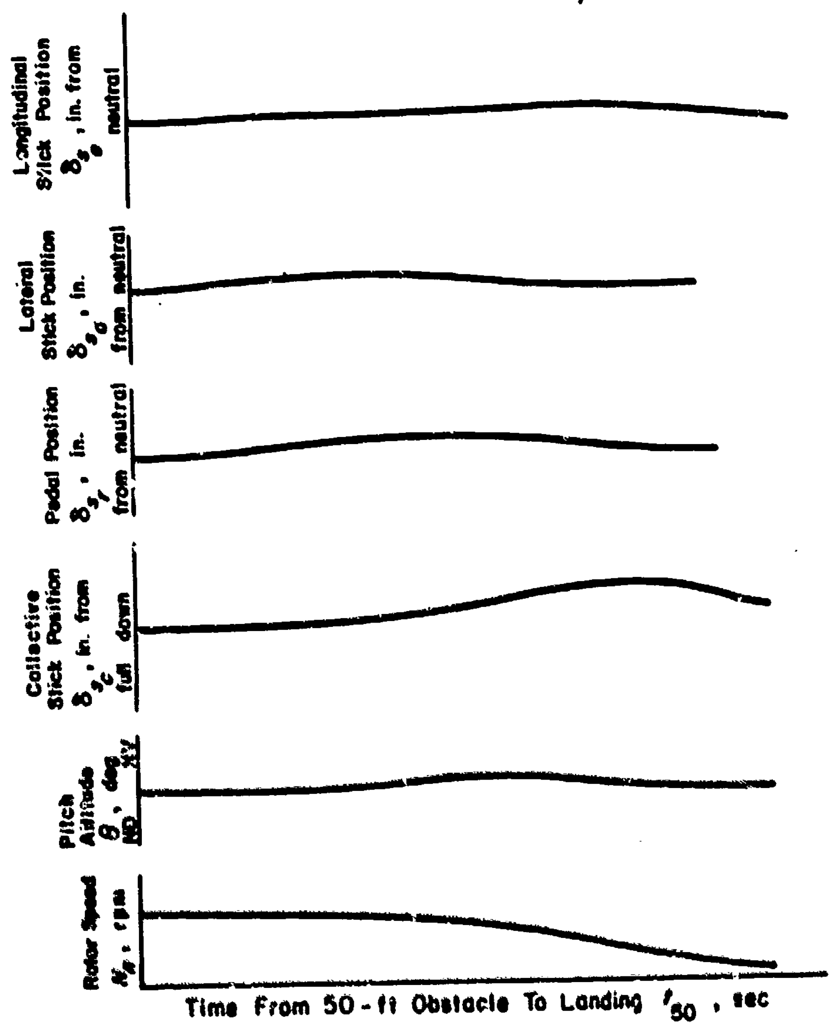


Figure 11-12. Control Positions During Maximum Performance Landing

CHAPTER 12

AIRSPEED, ALTITUDE, AND TEMPERATURE SYSTEM CALIBRATIONS

12-1 GENERAL

The altitude, airspeed, and nature of the air through which the vehicle is passing must be known in order to define adequately the performance characteristics of an aircraft. The data required for determination of these items are obtained from airspeed, altitude, and ambient air temperature systems. The accuracy of this determination becomes largely dependent upon the accuracy with which these conditions are sensed and the capability of the recording equipment. The indicators are part of a sensing system, and thus the accuracy of the system is also of concern. These systems are subjected to the aircraft flight environment which may cause their performance to vary accordingly. Therefore, a static ground calibration may not correct for errors introduced by flight conditions. The primary flight calibration difficulties are caused by the general inconsistency of air, the transient nature of the airflow, and the magnitude of the values being recorded. Local airflow around an aircraft cannot be predicted with any degree of accuracy or certainty and, as a result, most production systems are developed experimentally. Because of the complex nature of the airflow around a helicopter and the wide range of flight regimes available, a compromise is usually necessary relative to the magnitude of the errors that can be tolerated for the different conditions. The end result usually will be a system optimized for the level flight cruise condition. This can vary with mission requirements, in that a crane helicopter may be optimized for hover while an attack helicopter may be optimized for high speed.

For most test systems, the atmospheric pressure and temperature sensors are mounted on a boom extending forward of the aircraft. This is an attempt to locate them in an area of undisturbed flow that more truly represents the ambient air conditions. When possible, a swiveling head that automatically aligns with the flow also is used to eliminate total pressure errors. Although the static pressure errors of such a system may be quite large, the less turbulent air reduces the fluctuation of the indicated values.

The purpose of the calibrations is to determine the pressure and temperature variations caused by the passage of the aircraft through the air. The system errors are determined by comparing known and indicated values of airspeed, altitude, and temperature. All installed systems including the test and standard systems are calibrated. The calibrations then are used to correct the indicated test values and standardize the performance data.

The scope of the calibrations must be sufficient to provide the information needed to correct data throughout the flight test envelope. The conditions should include flight regime, altitude, gross weight, rotor speed, center of gravity location, and ground proximity. The accuracy of the calibrations must be compatible with the data requirements as specified by the objectives and test plans.

The flight calibrations can be accomplished by utilizing pacer, trailing bomb, tower flyby, and ground speed course methods. Some of the inherent advantages and disadvantages of these methods must be weighed in terms of accuracy, time, and equipment available. The method used must provide suitable data while being compatible with the test vehicle.

The planning effort that is required will be determined largely by the test method to be used. The trailing bomb and pacer calibrations are relatively easy to plan since personnel requirements are limited to flight crews. The tower flyby and ground speed courses require the use of ground personnel and equipment. In these cases, careful and complete planning is required to produce an efficient, productive test operation. Planning and scheduling must be accomplished in a manner conducive to obtaining data with the necessary accuracy within the time frame and testing sequence of the program.

12-2 PLANNING

As in all cases involving support groups, an extra planning effort is needed to produce an effective and safe test operation. The test plan usually will specify the calibration method to be used. This should be evaluated in light of the existing equipment and situation.

The aircraft pace method requires the utilization of pacer aircraft and crew. The pacer performance must be compatible with the anticipated speed range for the test vehicle. For high speed helicopters or perhaps for other considerations, it may be desirable to use different pacers for various segments of the speed envelope. The calibrations of the pacer aircraft must be checked carefully to be sure they are current and acceptable. The indicators in the pacer should be of sufficient sensitivity to provide the desired accuracy. The communications equipment for the aircraft must be compatible and the frequency to be used should be established and coordinated prior to the flights. To lose data from a flight because of communication problems is most disturbing. The local area and traffic pattern should be studied to determine the most suitable time and method of operation. Flight cards are needed for both the test and pace aircraft, and a briefing should be held for the flight personnel. A typical flight card is shown in Table 12-1. Items should be added as required for individual tests or aircraft peculiarities.

TABLE 12-1

FLIGHT DATA CARD FOR PACER CALIBRATION METHOD

Aim altitude	Ambient air temperature
Aim airspeed	Test area
Engine start gross weight	Aircraft serial number
Rotor speed	Configuration
Radio frequency	Center of gravity location
Test altitude	Fuel used
Test airspeed	

The trailing bomb method can be accomplished during the performance tests or conducted as individual tests. In the former case, the performance tests must be reviewed to determine which tests can be combined best to produce the necessary data in all the flight regimes. Previous data should be examined to determine the bomb capabilities relative to limitations and stability characteristics. Provision must be made for attaching the trailing bomb to the aircraft while on the ground or letting it out during flight. The calibrations for the bomb must be current and accurate. Contrary to popular opinion, when the test is conducted solely for calibration purposes, more formal planning is needed. Airspeed increments must be selected and other parameters—such as altitude, rotor speed, gross weight, and center of gravity locations—must be compatible with the test plan requirements. When it is an integrated program, these are dictated by the performance requirements. For this test method a flight card such as shown in Table 12-1 should be modified to include the bomb parameters.

The tower flyby method requires an established low-level test course and a properly equipped recording tower. The ground support equipment and personnel must be available when scheduled. The course schedule should be checked for conflicting tests by other aircraft and for general flight activity in the area. The recording equipment in the tower must be inspected and the effect of drift from the course marker should be determined. Calibrations of the recording equipment should be validated, and accurate. The test plan usually specifies the gross weight, altitude, rotor speed, center of gravity location, configuration, and airspeed range. A flight card such as shown in Table 12-1 should be developed and discussed during the briefing for both tower and aircraft personnel. Communication signals should be established and a procedure developed so that data can be recorded simultaneously in the test aircraft and the flyby tower.

The planning for the ground speed course method varies somewhat with the different types of ground support used. The two primary variations in methods are that the flight crews record all of the data or that a Fairchild camera be used to record height, distance, and time. In the first case, available courses should be examined for suitability and, if necessary, an appropriate course should be established and measured. Test airspeed values must be selected and wind tolerances established. For the Fairchild camera variation, the camera station must be established, surveyed, and measured. Special care must be used to insure that the course markers are clearly visible from altitude for climb and descent calibrations. Tests also should be planned to determine the minimum recovery altitude during descent and the best method to establish the climb in a minimum time and distance. Correlation and communication procedures should be established between the camera and the test aircraft. Provisions should be made to establish a weather station to record wind direction, wind velocity, ambient altitude, and air temperature.

12-3 INSTRUMENTATION

The airspeed calibration data are taken under stable conditions and can be recorded manually from visual instrumentation. Sensitive instruments are necessary to provide accurate and reliable data. The required visual instruments are listed in Table 12-2.

TABLE 12-2

VISUAL INSTRUMENTATION FOR AIRSPEED CALIBRATION

Boom airspeed	Rotor speed
Standard system airspeed	Test ambient temperature
Boom altitude	Standard ambient temperature
Fuel counter	Aircraft pitch attitude

Although not required, a photo panel incorporating the parameters shown in Table 12-2, plus engine parameters, may be used to provide a permanent, high density data record and to resolve any discrepancies found within the manually recorded data. A photo panel also will reduce the in-flight workload during climb and descent, and is needed when dynamic performance information will be obtained.

Most installations are susceptible to volume mismatches for the total and static parts of the system. This usually is caused by the difference in the numbers of instruments involved and the plumbing needed. These volume differences will cause lag errors in the airspeed which can lead to erroneous data unless the errors are recognized. Instantaneous pressures applied to the static and total pressure inlets will cause a needle fluctuation should the volumes be unequal. The volumes then can be changed as appropriate by adding either canisters or tubing to one of the systems. Altimeter lag results from unequal pressure losses caused by air motion through different lengths of tubing. Thus, an increase in volume achieved by adding tubing can improve both the airspeed and altimeter systems. References should be consulted for details on lag errors and system balancing.

12-4 TEST METHODS

12-4.1 GENERAL

The flight technique varies with the calibration method used. In all methods, the aircraft must be in a stabilized condition in order to determine the position error. Nonstabilized conditions can introduce lag error of a magnitude which, of course, will be dependent on the system characteristics and the pressure rate of change. The airspeed and altitude are of primary concern to the pilot. However, gross weight, rotor speed, and altitude may change the position error characteristics. Configuration changes such as external stores or windows and doors open also may change the pressure field around the static port and influence the position error. During the calibration, the aircraft should be trimmed at zero sideslip unless otherwise specified. Sideslip angles can cause total pressure errors through losses in the pitot sensor. If required, the temperature probe calibration should be accomplished at the same time the airspeed and altitude systems are calibrated. A map or chart of the test area should be in the cockpit for reference purposes.

12-4.2 TRAILING BOMB

The trailing bomb method requires the minimum ground and flight support, and provides a capability of calibrating for all flight regimes and altitudes. With proper planning, the calibration data can be obtained during the performance testing. The level flight, sawtooth climbs, and descent performance procedures provide the proper stabilized conditions for a sufficient period of time to insure a valid calibration. When the test is conducted concurrently with the performance tests, it is necessary only to record the parameters from the trailing bomb since complete aircraft parameters will be recorded as performance items. For the case where the test is solely a calibration effort, the values indicated by the instruments listed in Table 12-2 should be recorded. The lines to the bomb should be measured and the relative distance from the bomb to the aircraft sensors should be recorded for inclusion in the data reduction procedures. The distance from the bomb to the aircraft will vary as speed is changed and the bomb trails in different positions. An exact altimeter calibration requires that this vertical distance be determined. Photographs taken from a chase aircraft can be used to obtain these data. Generally, the bomb is not incorporated into an automatic data gathering system and the values are recorded manually. A typical airspeed calibration bomb with associated tubing and indicators is shown in Fig. 12-1.

The bomb must be launched carefully to preclude damage from ground contact or rotor wash. A common technique is to attach the bomb with the aircraft on the ground and then extend out the lines as the helicopter slowly ascends vertically. When launching the bomb in flight, consideration must be given to the additional aerodynamics due to velocity.

Level flight tests should be conducted prior to climb or descent. On the initial flight the bomb should be observed carefully to be sure that it is not influenced by the rotor wash or in close proximity to the aircraft. At low speed, the bomb is below the aircraft and the fuselage tends to shield it from the rotor wash. The bomb usually tends to trail further aft with increased airspeed. The vertical distance to the rotor decreases and the fuselage protection may be removed. Since the rotor wash also travels aft as the speed increases, the bomb stability and data validity may be affected.

The various relationships between the bomb and aircraft with respect to flight regimes are illustrated in Fig. 12-2.

Thus, the initial test condition should be at a low or intermediate speed. Most bombs become unstable at some high speed which is evidenced by porpoising and perhaps longi-

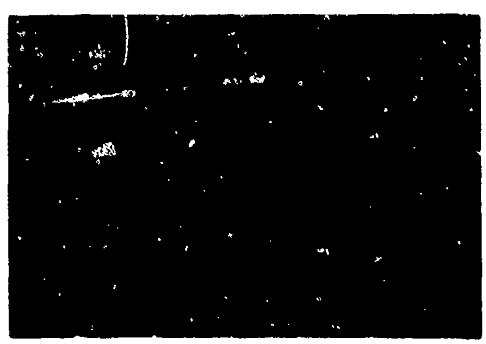


Figure 12-1. Airspeed Calibration Bomb

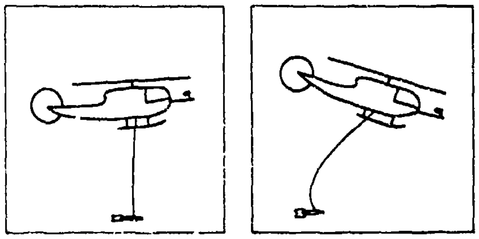


Figure 12-2. Airspeed Calibration Bomb in Flight

tudinal motion. This will cause fluctuations in the indicated values and also will introduce a safety hazard if the aerodynamic loads from the bomb cause the restraint to fail, releasing the bomb. Most bombs are very sensitive to turbulence and only a slight amount will render the data unusable. Since the bomb may be in a different flight condition during climb or descent, the rates should be built up while noting the relative position and motion of the bomb.

12-4.3 AIRCRAFT PACE

The aircraft pace method is perhaps the quickest and easiest method to accomplish. The method can be used for all airspeeds or altitudes and in all flight regimes. The test aircraft is accelerated slowly to the aim airspeed and altitude condition. Power, rotor speed, and collective are then adjusted to stop the acceleration and allow the aircraft to stabilize. A more rapid acceleration to the aim speed will require larger adjustments and more time for the stabilization to be accomplished. An extremely slow acceleration will curtail unduly the test productivity. Usually the pilot of the test aircraft stabilizes at the test conditions while the actual pacing is done by the pace vehicle. This technique reduces the workload for the pilot of the test vehicle and allows him to devote more attention inside the cockpit. However, this is a matter of personal preference to be resolved by the pilot.

When the test aircraft is stabilized, an appropriate signal is given to the pacer. As a rule, both aircraft are stabilizing simultaneously to place the pace aircraft in position shortly after the signal is given. At this time, the pacer will indicate a stabilized condition, and the data from both aircraft are recorded. Unless a change occurs in the test conditions during the recording interval, the test point can be terminated at the end of the interval. This procedure then is repeated for the necessary airspeed ranges and increments. The test sequence may begin at high speed with subsequent tests at reduced airspeed values, or the initial point may be at low speed with subsequent test at increased airspeeds as a matter of personal preference. In the interest of productivity, the less critical test aircraft parameters can be recorded while the pace aircraft is completing the stabilization. When the pacer signals to read data, previously noted values should be checked for changes. For data correlation purposes, pacer data also should be recorded on the test aircraft data card.

The aircraft pacer method must entail a careful stabilization and pace technique. The pacer should be as close as practical so that small relative motions can be recognized. However, downwash and close proximity effects can cause pressure field changes and introduce flight hazards. The relative positions of the test and pace vehicles are at the discretion of the pilots, although this should be established prior to flight. The predominant tendency during this test is for the pilot to hurry the stabilizing process. The signals exchanged between the aircraft must be understood mutually to insure that the proper data are recorded at the right moment.

12-4.4 TOWER FLYBY

The test aircraft pilot task for the tower flyby method is very similar to the pacer method. The primary differences are the ground proximity and, occasionally, the limited space for the approach. This calibration method usually is limited to the level flight regime. The course generally consists of a marked path for the aircraft to follow and a tower for recording the height above the ground. A typical installation is illustrated in Fig. 12-3.

The pilot must allow sufficient distance to stabilize the aircraft prior to reaching the tower. The distance needed will change with speed and aircraft characteristics. The accurate judgment of the distance required will increase significantly the productivity, particularly during the low speed tests. Any obstacles that may influence a low level flight operation should be noted before any tests and the flight pattern altered accordingly. The test parameters are monitored during the approach to the tower. As the aircraft passes the tower, it should be directly above the designated ground station. Deviations from the course centerline are recorded since these will influence the accuracy of the calibration. Data should be read in the aircraft on signal from the tower while theodolite readings are being made in the tower. For data correlation purposes, counter numbers and other identifying items should be recorded both in the tower and in the test aircraft. Accuracy is dependent upon instantaneous, simultaneous readings at both locations. Wind effects during the run should be monitored and the drift recorded.

The tower flyby courses often are located near high density traffic areas where more than one aircraft may be using the course at the same time. This can divert pilot attention from the test and demands that the aircraft/tower communications be precise and responsive. When flying close to the ground at low speeds where the aircraft is on the back side of the power required curve, large control or power changes which detract from lift should be avoided to preclude undesirable loss of altitude or increase in the rate of descent.

12-4.5 GROUND SPEED COURSE

Although the ground speed course method usually is employed to obtain only a level flight calibration, a Fairchild camera or Askania facilities may be used to obtain climb and descent calibrations. The ground speed course generally is flown in close ground proximity, therefore, obstacles are of constant concern. The course, with its markings, should be known and checked prior to starting the tests. Markings and obstacles may be changed without the knowledge of the test personnel, and a check should be made prior to each individual operation. A ground speed course is shown in Fig. 12-4. The acceptable turbulence level should be such that airspeed will vary less than 1 kt from the test value. A crosswind will cause a drift and, generally, a component of more than 3 kt will invalidate the data. When manual timing is being used, the pilot should fly the course in a manner which will facilitate this effort. The height and distance from the timing markers should be a minimum to allow more accurate timing. For a given course length, the timing function

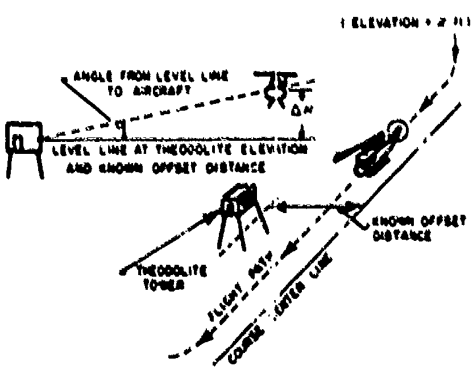


Figure 12-3. Tower Flyby Course

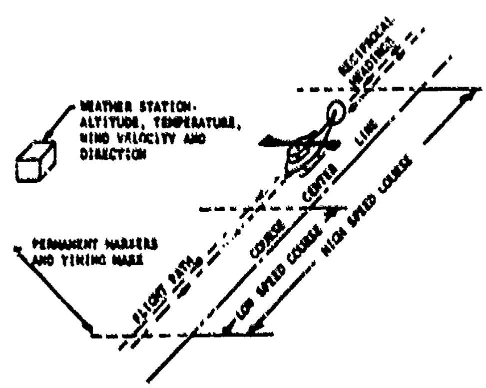


Figure 12-4. Ground Speed Course

becomes more critical as airspeed increases. Several methods may be used to record elapsed time, with electronic means being the most accurate and least common. When the timing is accomplished manually from within the aircraft, extreme care must be taken that the method remains consistent from point to point. A most effective way to accomplish this is to maintain the same body position and use the same line of sight. A structural member of the aricraft provides a good sighting device and, in the absence of a suitable member, a grease pencil mark on the canopy will serve admirably. Any change in course length should include a common airspeed at both distances. An in-flight plot of time versus indicated airspeed readily will show any timing error that may have been made. By comparing the times for reciprocal headings at a given speed, the effects of wind may be determined. The same procedure is used for all speeds tested.

The aircraft is stabilized at the aim conditions during the approach to the course and held in this condition during the run. Each airspeed should be flown on reciprocal headings to compensate for wind effects. In most cases, it will be necessary to change course lengths at the high and low speed extremes. When this change is made, a given speed should be flown on each course to provide continuity and comparative data for the course lengths. When a Fairchild camera is being used, its distance from the reference line is a measured value which is used in the data reduction; deviations from the centerline will introduce errors. The deviation or cross-wind drift tolerance limits can be determined from the calibration accuracy requirements. Pilot correction for drift will introduce side-slip and may change the static pressure field surrounding the aircraft.

For a Fairchild camera installation (see Fig. 3-2), the usable camera coverage is usually very limited (approximately 2000 ft field width). The climb and descent tests therefore will involve stabilizing the aircraft in a more difficult flight regime in a much shorter time—which greatly complicates the pilot tasks. The extent of the camera coverage should be determined carefully prior to starting the tests. Extremely good course markings are required. These must be clearly visible from both near the ground and from altitude. The climb should be initiated close to the ground and just prior to entering the camera range. The climbing flight condition is maintained until beyond the camera coverage. During descent, the aircraft can be stabilized above camera coverage. However, the flight path must be judged carefully to obtain the maximim coverage. Extreme care must be used to allow sufficient recovery time and altitude during the high descent rate tests. For autorotational and low power descent tests, the engine response time and aircraft lag time are critical factors. The pilot should insist on a thorough engineering effort to obtain information defining the recovery capabilities of the test aircraft (rotor and engine). The pilot also should conduct a sufficient number of tests to be intimately familiar with the stability and control characteristics during maximum power recovery maneuvers.

12-5 DATA REDUCTION

Typical data reduction forms are presented in Tables 12-3 through 12-6* for the various calibration methods. Reduction steps common to all methods are presented only in Table 12-3*. Table 12-5* refers to processing of Fairchild camera data which was also discussed extensively in par. 7-5.

The data from the trailing bomb calibration method can be reduced by using Table 12-3. The bomb is considered in the same light as the pacer vehicle and substituted into the appropriate steps. An additional factor that can be included is the vertical distance from the bomb to the aircraft sensors.

12-6 DATA PRESENTATION

The calibration data are presented as shown in Figs. 12-5 through 12-9.

*The data reduction forms (tables) are located at the end of each chapter.

TABLE 12-3

DATA REDUCTION FORM FOR PACER AIRSPEED CALIBRATION METHOD

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1	Indicated test system altitude	$H_{P_{IND_t}}$		ft	From manual recording or photo panel data
2	Instrument correction for test system altitude	$\Delta H_{P_{IC_t}}$		ft	From instrument calibration curves at (1)
3	Instrument corrected test system altitude	$H_{P_{IC_t}}$	$H_{P_{IC_t}} = H_{P_{IND_t}} + \Delta H_{P_{IC_t}}$	ft	(3) = (1) + (2)
4	Indicated standard system altitude	$H_{P_{IND_S}}$		ft	From manual recording or photo panel data
5	Instrument correction for standard system altitude	$\Delta H_{P_{IC_S}}$		ft	From instrument calibration curves at (4)
6	Instrument corrected standard system altitude	$H_{P_{IC_S}}$	$H_{P_{IC_S}} = H_{P_{IND_S}} + \Delta H_{P_{IC_S}}$	ft	(6) = (4) + (5)
7	Indicated test system ambient air temperature	$T_{a_{IND_t}}$		°C	From manual recording or photo panel data
8	Instrument correction for test system ambient air temperature	$\Delta T_{a_{IC_t}}$		°C	From instrument calibration curves at (7)
9	Instrument corrected test system ambient air temperature	$T_{a_{IC_t}}$	$T_{a_{IC_t}} = T_{a_{IND_t}} + \Delta T_{a_{IC_t}}$	°C	(9) = (7) + (8)

TABLE 12-3 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
10	Standard system indicated ambient air temperature	$T_{a_{IND_S}}$		°C	From manual recording or photo panel data
11	Instrument correction for standard system ambient air temperature	$\Delta T_{a_{IND_{IC_S}}}$		°C	From instrument calibration curves at (10)
12	Instrument corrected standard system ambient air temperature	$T_{a_{IC_S}}$	$T_{a_{IC_S}} = T_{a_{IND_S}} + \Delta T_{a_{IND_{IC_S}}}$	°C	(12) = (10) + (11)
13	Indicated test system airspeed	V_{IND_t}		kt	From manual recording or photo panel data
14	Instrument correction for test airspeed system	ΔV_{IC_t}		kt	From instrument calibration curves at (13)
15	Instrument corrected test system airspeed	V_{IC_t}	$V_{IC_t} = V_{IND_t} + \Delta V_{IC_t}$	kt	(15) = (13) + (14)
16	Standard system indicated airspeed	V_{IND_S}		kt	From manual recording or photo panel data
17	Instrument correction for standard system airspeed	ΔV_{IC_S}		kt	From instrument calibration curves at (16)
18	Instrument corrected standard system airspeed	V_{IC_S}	$V_{IC_S} = V_{IND_S} + \Delta V_{IC_S}$	kt	(18) = (16) + (17)

TABLE 12-3 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
19	Indicated rotor speed	$N_{R_{IND}}$		rpm	From manual recording or photo panel data
20	Instrument correction for rotor speed	$\Delta N_{R_{IC}}$		rpm	From instrument calibration curves at (19)
21	Rotor speed	N_R	$N_R = N_{R_{IND}} + \Delta N_{R_{IC}}$	rpm	(21) = (19) + (20)
22	Advancing blade tip speed	V_{TIP}	$V_{TIP} = \Omega R$	ft/sec	From tables or calculated at (21)
23	Engine start gross weight	*ESGW*		lb	From A/C preflight records
24	Engine start fuel specific weight	FS_1		lb/gal	From A/C preflight records
25	Engine stop fuel specific weight	FS_2		lb/gal	From postflight records
26	Average fuel specific weight	FS_{AVG}	$FS_{AVG} = (FS_1 + FS_2)/2$	lb/gal	(26) = [(24) + (25)] /2
27	Fuel used in counts	FU_c		ct	From manual recording or photo panel data
28	Fuel counter constant	K_{FC}		gal/ct	From fuel flow instrument calibration curves
29	Volume of fuel used	FU_{VOL}	$FU_{VOL} = FU_c \times K_{FC}$	gal	(29) = (27)·(28)

TABLE 12-3 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
30	Weight of fuel used	FU_W	$FU_W = FU_{VOL} + FS_{AVG}$	lb	(30) = (29)·(26)
31	Gross weight	GW	$GW = ESGW - FU_W$	lb	(31) = (23) − (30)
32	Pacer indicated pressure altitude	$H_{P_{IND_P}}$		ft	From pacer recorded data
33	Instrument correction for indicated pacer pressure altitude	$\Delta H_{P_{IC_P}}$		ft	From pacer instrument calibration curves at (32)
34	Position error correction for pacer altitude	$\Delta H_{P_{PE_P}}$		ft	From pacer altimeter calibration curve at (32)
35	Pacer pressure altitude	H_{P_P}	$H_{P_P} = H_{P_{IND_P}} + \Delta H_{P_{IC_P}} + \Delta H_{P_{PE_P}}$	ft	(35) = (32) + (33) + (34)
36	Test altitude system position error	ΔH_{PE_t}	$\Delta H_{P_{PE_t}} = H_{P_P} - H_{P_{IC_t}}$	ft	(36) = (35) − (3)
37	Standard altitude system position error	$\Delta H_{P_{PE_S}}$	$\Delta H_{P_{PE_S}} = H_{P_P} - H_{P_{IC_S}}$	ft	(37) = (35) − (6)
38	Pacer indicated ambient air temperature	$T_{a_{IND_P}}$		°C	From pacer recorded data
39	Pacer instrument correction ambient temperature	$\Delta T_{a_{IC_P}}$		°C	From pacer instrument calibration curves

TABLE 12-3 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
40	Pacer instrument corrected ambient temperature	$T_{a_{IC_p}}$	$T_{a_{IC_p}} = T_{a_{IND_p}} + \Delta T_{a_{IC_p}}$	°C	(40) = (38) + (39)
41	Pacer temperature recovery factor	$K_{T_{a_p}}$			From pacer temperature calibration curve
42	Ambient air temperature	T_a	$T_a = T_{a_{IC_p}} \cdot K_{T_{a_p}}$	°C	(42) = (40) · (41)
43	Pacer indicated airspeed	V_{IND_p}		kt	From pacer recorded data
44	Pacer airspeed instrument correction	ΔV_{IC_p}		kt	From pacer instrument calibration curves at (43)
45	Pacer airspeed system position error	ΔV_{PE_p}		kt	From pacer airspeed calibration curves at (44)
46	Pacer calibrated airspeed	V_{CAL_p}	$V_{CAL_p} = V_{IND_p} + \Delta V_{IC_p} + \Delta V_{PE_p}$	kt	(46) = (43) + (44) + (45)
47	Test system airspeed position error	ΔV_{PE_t}	$\Delta V_{PE_t} = V_{CAL_p} - V_{IC_t}$	kt	(47) = (46) – (15)
48	Standard system airspeed position error	ΔV_{PE_S}	$\Delta V_{PE_S} = V_{CAL_p} - V_{IC_S}$	kt	(48) = (46) – (18)
49	Density altitude	H_D		ft	From tables or calculated at (42) and (35)

TABLE 12-3 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
50	Square root of density ratio	$\sqrt{\sigma}$			From tables or calculated at (49)
51	True airspeed	V_T	$V_T = V_{CAL_p} / \sqrt{\sigma}$	kt	(51) = (46) / (50)
52	Speed of sound	a		ft/sec	Calculated or from tables at (42)
53	Mach number	M	$M = (V_T/a) \times 1.69$		(53) = [(51) / (52)] · 1.69
54	Test system ambient temperature probe recovery factor	$K_{T_{a_t}}$	$K_{T_{a_t}} = \left[\frac{T_a + 273}{T_{a_{IC_t}} + 273} - 1\right] \frac{5}{M^2}$		$(54) = \left[\frac{(42) + 273}{(9) + 273} - 1\right] \frac{5}{(53)^2}$
55	Angle of sideslip	β		deg	From oscillograph or cockpit recording
56	Angle of attack	α		deg	From oscillograph or cockpit recording
57	Test indicated rate of climb or descent	$(R/C)_t$		ft/min	From cockpit reading or slope of H vs T curve
58	Pacer rate of climb or descent	$(R/C)_p$		ft/min	Calibrated data from pacer recording

TABLE 12-3 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
59	Correction for rate of climb or descent indication system	$\Delta(R/C)$	$\Delta(R/C) = (R/C)_p - (R/C)_t$	ft/min	(59) = (58) − (57)
60	Altimeter position error correction	ΔH_{PE}	$\Delta H_{PE} = H_{P_p} - H_{P_{IC_S}}$	ft	(60) = (35) − (6)

TABLE 12-4

DATA REDUCTION FORM FOR GROUND SPEED COURSE CALIBRATION METHOD

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1-31					From Table 12-3
32	Indicated ground station pressure altitude	$H_{P_{IND_G}}$		ft	Recorded at ground station
33	Instrument correction for ground station altitude	$\Delta H_{P_{IC_G}}$		ft	From calibration curves for ground instrument
34	Ground station pressure altitude	H_{P_G}	$H_{P_G} = H_{P_{IND_G}} + \Delta H_{P_{IC_G}}$	ft	(34) = (32) + (33)
35	Indicated ground station ambient air temperature	$T_{a_{IND_G}}$		°C	Recorded at ground station
36	Instrument correction for ground station ambient air temperature	$\Delta T_{a_{IC_G}}$		°C	From instrument calibration curves for ground station temperature
37	Ground station ambient air temperature	T_{a_G}	$T_{a_G} = T_{a_{IND_G}} + \Delta T_{a_{IC_G}}$	°C	(37) = (35) + (36)
38	Speed course length	X		ft	From measurement of course
39	Height above the ground	H_G		ft	From visual theodolite or estimated
40	Ground station altitude corrected to A/C altitude	H_P	$H_P = H_{P_G} + H_G$	ft	(40) = (34) + (39)

TABLE 12-4 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
41	Test altitude system position error	$\Delta H_{P_{PE_t}}$	$\Delta H_{P_{PE_t}} = H_P - H_{P_{IC_t}}$	ft	(41) = (40) – (3)
42	Standard altitude system position error	$\Delta H_{P_{PE_S}}$	$\Delta H_{P_{PE_S}} = H_P - H_{P_{IC_S}}$	ft	(42) = (40) – (6)
43	Elapsed time	t		sec	From manual recording
44	Density altitude	H_D		ft	From tables or calculated at (40) and (37)
45	Square root of air density ratio	$\sqrt{\sigma}$			From tables or calculated at (44)
46	Ground speed	dx/dt	$dx/dt = X/t$	ft/sec	(46) = (38) / (43)
47	True ground airspeed	V_{T_G}	$V_{T_G} = \dfrac{V_{T_{OUTBOUND}} + V_{T_{INBOUND}}}{2}$	ft/sec	Average of reciprocal headings to compensate for wind
48	Ground calibrated airspeed	V_{CAL_G}	$V_{CAL_G} = V_{T_G} \times \sqrt{\sigma})/1.69$	kt	(48) = [(47)·(45)] / 1.69
49	Test airspeed system position error	ΔV_{PE_t}	$\Delta V_{PE_t} = V_{CAL_G} - V_{IC_t}$	kt	(49) = (48) – (15)

TABLE 12-4 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
50	Standard airspeed system position error	ΔV_{PE_S}	$\Delta V_{PE_S} = V_{CAL_G} - V_{IC_S}$	kt	(50) = (48) – (18)
51	Speed of sound	a		ft/sec	From tables or calculated at (37)
52	Mach number	M	$M = V_{T_G}/a$		(52) = (47) / (51)
53	Test system ambient temperature probe recovery factor	$K_{T_{a_t}}$	$K_{T_{a_t}} = \left(\dfrac{T_{a_G} + 273}{T_{a_{IC_t}} + 273} - 1\right)\dfrac{5}{M^2}$		$(53) = \left[\dfrac{(37) + 273}{(9) + 273} - 1\right] \cdot \dfrac{5}{(52)^2}$

TABLE 12-5

DATA REDUCTION FORM FOR FAIRCHILD CAMERA AIRSPEED CALIBRATION METHOD

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1-31					From Table 12-3
32-38					From Table 12-4
39	Aircraft height above the ground	H_W		ft	From Fig. 7-6
40	Ground level altitude corrected to A/C height	H_p	$H_p = H_{p_G} + H_W$	ft	(40) = (34) + (39)
41	Test altitude system position error	$\Delta H_{p_{PE_t}}$	$\Delta H_{p_{PE_t}} = H_p - H_{p_{IC_t}}$	ft	(41) = (40) − (3)
42	Standard altitude system position error	$\Delta H_{p_{PE_S}}$	$\Delta H_{p_{PE_S}} = H_p - H_{p_{IC_S}}$	ft	(42) = (40) − (6)
43	Distance traveled	X		ft	From Fig. 7-6
44	Elapsed time	t		sec	From Fig. 7-6
45	Ground speed	dx/dt	$dx/dt = X/t$	ft/sec	(45) = (43) / (44)
46	True ground speed	V_{T_G}	$V_{T_G} = \dfrac{\frac{dx}{dt}_{OUTBOUND} + \frac{dx}{dt}_{INBOUND}}{2}$	ft/sec	Average of reciprocal headings

TABLE 12-5 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
47	Density altitude	H_D		ft	From tables or calculated at (37) and (40)
48	Square root of air density ration	$\sqrt{\sigma}$			From tables or calculated at (47)
49	Calibrated airspeed	V_{CAL}	$V_{CAL} = V_{T_G} \times \sqrt{\sigma})/1.69$	kt	(49) = [(46)·(48)] / 1.69
50	Test airspeed system position error	ΔV_{PE_t}	$\Delta V_{PE_t} = V_{CAL} - V_{IC_t}$	kt	(50) = (49) − (15)
51	Standard airspeed system position error	ΔV_{PE_S}	$\Delta V_{PE_S} = V_{CAL} - V_{IC_S}$	kt	(51) = (49) − (18)
52	Speed of sound	a			From tables or calculated at (37)
53	Mach number	M	$M = V_{T_G}/a$		(53) = (46) / (52)
54	Test system ambient temperature probe recovery factor	$K_{T_{a_t}}$	$K_{T_{a_t}} = \left[\dfrac{T_{a_G} + 273}{T_{a_{IC_t}} + 273} - 1\right] \dfrac{5}{M^2}$		$(54) = \left[\dfrac{(37) + 273}{(9) + 273} - 1\right] \cdot \dfrac{5}{(53)^2}$
55	Angle of sideslip	β		deg	From oscillograph or cockpit recording

TABLE 12-6 (Continued)

STEP NO.	DEFINITON	SYMBOL	EQUATION	UNITS	REMARKS
56	Angle of attack	α		deg	From oscillograph or cockpit recording
57	Test indicated rate of climb or descent	$(R/C)_t$		ft/min	From oscillograph or cockpit recording
58	Rate of climb or descent	R/C		ft/min	From slope of H vs T data obtained from the Fairchild Camera
59	Correction for rate of climb indication system	$\Delta(R/C)_{PE}$	$\Delta(R/C)_{PE} = R/C - (R/C)_t$	ft/min	(59) = (58) – (57)

TABLE 12-6

DATA REDUCTION FORM FOR TOWER FLYBY AIRSPEED CALIBRATION METHOD

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1-31					From Table 12-3
32	Tower indicated pressure altitude	$H_{P_{IND_{TO}}}$		ft	From tower recording
33	Tower altitude instrument correction	$\Delta H_{P_{IC_{TO}}}$		ft	From calibration curves for tower instrument
34	Corrected tower pressure altitude	$H_{P_{TO}}$	$H_{P_{TO}} = H_{P_{IND_{TO}}} + \Delta H_{P_{IC_{TO}}}$	ft	(34) = (32) + (33)
35	Horizontal distance from tower to flight path	X			From tower data recording
36	Tower height above the ground	H_{TO}		ft	From measurement
37	Theodolite angle	θ		deg	From tower data recording
38	Angle function	$\tan\theta$			From tables
39	Aircraft height above the tower	ΔH	$\Delta H = X \tan\theta$	ft	(39) = (35)·(38)
40	Tower altitude corrected to aircraft	$H_{P_{TO_C}}$	$H_{P_{TO_C}} = H_{P_{TO}} + \Delta H$	ft	(40) = (34) + (39)
41	Standard altitude system position error	$\Delta H_{P_{PE_S}}$	$\Delta H_{P_{PE_S}} = H_{P_{TO_C}} - H_{P_{IC_S}}$	ft	(41) = (40) − (6)

TABLE 12-6 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
42	Test altitude system position error	$\Delta H_{P_{PE_t}}$	$\Delta H_{P_{PE_t}} = H_{P_{TO_C}} - H_{P_{IC_t}}$	ft	(42) = (40) – (3)
43	Corrected atmospheric pressure	P_{a_C}		in. Hg	From tables or calculated at (40)
44	Test aircraft system instrument corrected atmospheric pressure	$P_{a_{IC_t}}$		in. Hg	From tables or calculated at (3)
45	Standard system aircraft instrument corrected atmospheric pressure	$P_{a_{IC_S}}$		in. Hg	From tables or calculated at (6)
46	Test airspeed system static pressure error	ΔP_{a_t}	$\Delta P_{a_t} = P_{a_C} - P_{a_{IC_t}}$	in. Hg	(46) = (43) – (44)
47	Standard airspeed system static pressure error	ΔP_{a_S}	$\Delta P_{a_S} = P_{a_C} - P_{a_{IC_S}}$	in. Hg	(47) = (43) – (45)
48	Speed of sound	a	$a = 38.967\sqrt{T_a\,(^\circ K)}$	kt	a_{SL} = 661.48 kt at sea level standard $a = 38.967\sqrt{T_a + 273}$
49	Standard system airspeed position error	ΔV_{PE_S}	$\Delta V_{PE_S} = \frac{\Delta P_{a_S} \times a^2}{1.4 \times 29.92 \times V_{IC_S}} \times \left[\frac{2.5}{1+0.2\left(\frac{V_{IC_S}}{a}\right)^2}\right]$		$(50) = \left[\frac{(47)\cdot(48)^2}{1.4 \times 29.92 \times (17)}\right] \times \left\{\frac{2.5}{1+0.2\left[\frac{(17)}{(48)}\right]^2}\right\}$

TABLE 12-6 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
50	Angle of sideslip	β		deg	From oscillograph or cockpit recording
51	Angle of attack	α		deg	From oscillograph or cockpit recording

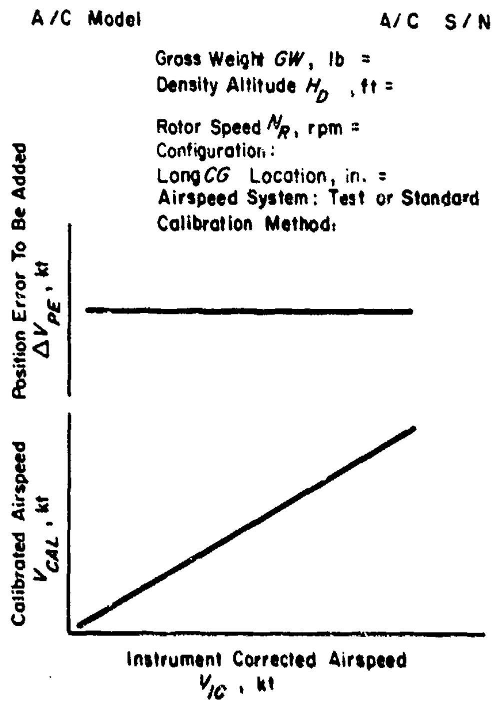


Figure 12-5. Airspeed Calibration

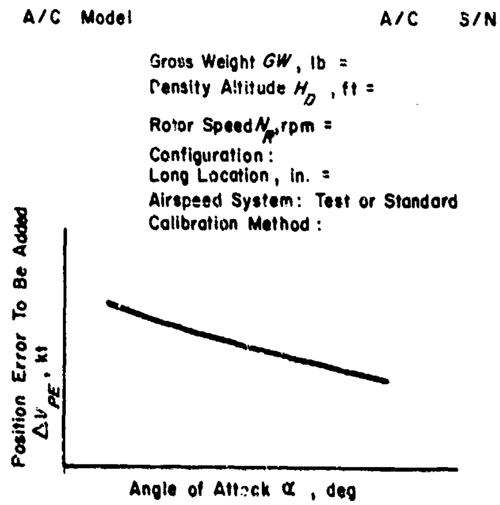


Figure 12-7. Position Error Correction for Angle of Attack

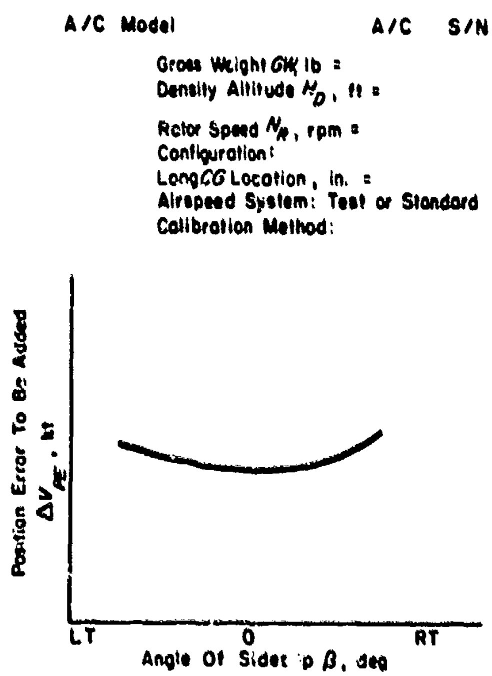


Figure 12-6. Position Error Correction for Angle of Sideslip

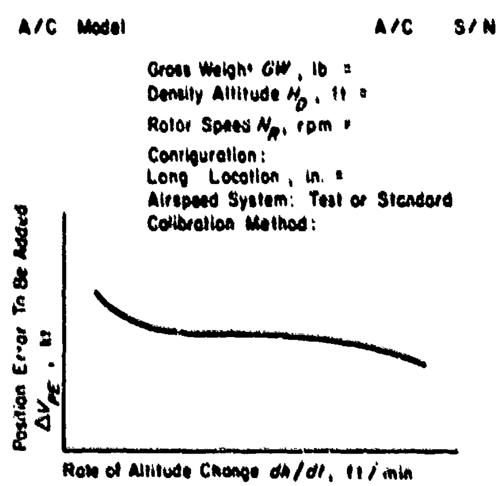


Figure 12-8. Position Error Correction for Rate of Climb or Descent

A/C Model A/C S/N

Gross Weight GW, lb =
Density Altitude H_D, ft =
Rotor Speed N_R, rpm =
Configuration:
Long CG Location, in. =
Altimeter System: Test or Standard
Calibration Method:

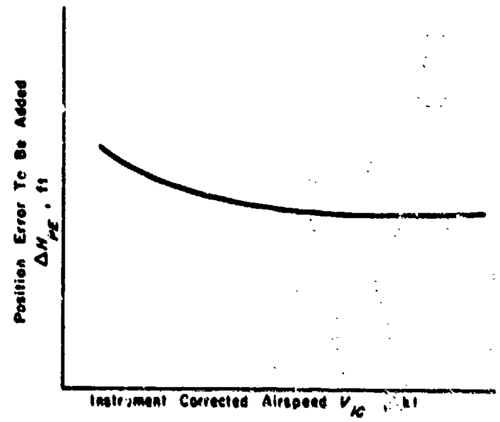


Figure 12-9. Altimeter Calibration

CHAPTER 13

INLET PERFORMANCE

13-1 GENERAL

The term inlet performance in this chapter pertains to only engine inlets. The inlet generally is considered the entire ducting, shaping, guiding, or holding apparatus between the compressor face and the free airstream. The design of this apparatus may vary from simple to complex and will have a bearing on the inlet performance. The most important aspect of inlet design with respect to the airframe and power plant is the effect of inlet performance on power available which is turn can directly influence or compromise the total vehicle capability. Often this is not understood clearly or appreciated. The engine usually is designed with general consideration for ideal flow but without intent for any particular inlet since the engine manufacturer's knowledge of future applications of the power plant is limited. Thus, one can expect only that inlet losses will be present and will detract from the contractor specified engine performance. A second item is that inlets are very difficult to design for other than optimum free stream conditions. The actual airflow in a helicopter environment varies greatly through the flight regime from hover to high speed and including climb or descent. Also, the engine location usually is dictated by other than inlet considerations which tend to aggravate the situation. The measurement of these inlet conditions is most difficult. The flow usually is disturbed most under the rotor, and the losses cannot be considered in the classical sense i.e., flow separation and ingestion of boundary layer flow. The magnitude of the values involved is normally small which renders measurement and accuracy critical. Although experience has shown that the inlet performance need not be measured precisely in most cases, some instances have occurred where significant errors were introduced by inadequate inlet considerations. The inlet performance usually becomes more critical with increased rotor thrust which introduces higher downwash velocities, greater turbulence, and more recirculation. The VTOL machines will accent these phenomena as well as add others, including hot gas re-ingestion. Although not specifically discussed here, engine cooling and exhaust ducts may be treated with the same procedures as those outlined for inlets.

The objective of the inlet tests is to determine the effect that the inlet performance has on the propulsion system characteristics, which is turn determines the aircraft capability. The inlet performance normally is defined in terms of the pressure and temperature conditions of the air delivered to the engine. These are compared to the design criteria and, if necessary, inlet or installation modifications and improvements are made to provide the inlet a more suitable environment.

The tests should be of sufficient scope to define adequately the pressure and temperature conditions across the face of the compressor throughout the flight regime. The number and location of the pickups or probes should be compatible with the engine manufacturers design points. Test data at these points then can be input to the manufacturer's computer program and used to predict installed performance and suggest areas of improvement. The data also should allow computation of distortion factors in the event they cause engine oscillations that in turn cause performance variations.

The tests should be planned sufficiently in advance that engine performance data can be gathered during the engine calibration before it is installed in the airframe. Usually, the collection of these data during the flight test is impractical although it is needed to evaluate the installation losses. Formidable flight test instrumentation requisition and installation problems also may require long lead times. Additional time must be allowed for development and airworthiness qualification of the inlet hardware. To avoid duplication of flight tests, the inlet data usually are gathered during the performance testing. Thus, the inlet requirements must be integrated carefully with the other data requirements.

The data reduction can be accomplished easily with manual methods when the number of probes are limited and when time averaged data will suffice. For more complex installations or when it is desired to determine profiles and distortion characteristics, the mass of data becomes unwieldy and automatic data processing is more appropriate. This method will allow high density data processing as well as a statistical analysis of the results.

13-2 PLANNING

The schedule for engine calibration and aircraft delivery should be obtained as early as possible. During the engine calibration, which is usually conducted by the engine contractor, the inlet characteristics should be determined. The instrumentation to accomplish this is usually a part of the engine test cell. When possible, the calibration also should be accomplished with the aircraft inlet installed, including the probes that will be used in the flight tests. In the event this cannot be done, the test cell instrumentation should be positioned in a manner similar to that anticipated for the test equipment. This will assist greatly in data correlation and provide information for comparison purposes.

The flight test instrumentation requirements should be planned on the basis of the particular aircraft and the data necessary to achieve the test objectives. The number of parameters must be minimized due to the impact on the total instrumentation package. During the static ground tests, it may be necessary to use special ground recording equipment. This must be arranged with the flight test group and the personnel responsible for the vertical and horizontal thrust stands. Necessary restraining devices must be designed and constructed for the static tests. These special devices normally require a long lead time and there may be financial implications. Compatible restraining points for the aircraft and the ground facility also may be a problem. When other than established points such as cargo hooks are used, aircraft structural problems should be anticipated. Appropriate action must be taken to insure that the aircraft will not be endangered.

During the flight test portions of the program, the inlet data requirements should be coordinated and integrated into the total data recording effort. Arrangements must be made to include the preflight checks necessary to check any instrumentation peculiar to the inlet data. Planning must be accomplished to insure that the recording process for the other tests will be compatible with the requirements for the inlet performance.

13-3 INSTRUMENTATION

The instrumentation consists of pressure and temperature sensors, and an appropriate recording system. The pressure sensors must record both static and total pressures. All sensors must have a sensitivity and response time compatible with the type of data required. One overriding consideration is the airworthiness of the installation. Since the inlet usually is associated with a propulsion or thrust device, disastrous consequences can result from ingestion of a metal object. The recording equipment must be compatible with the sensors both in characteristics and in recording capability. Any ground recording equipment to be used during static tests must be coordinated with the flight recording installation. The historical background of inlet probes shows much to be desired in all aspects, and considerable efforts need to be made to improve the inlet instrumentation. There are techniques used in wind tunnels and research laboratories that might be suitable for flight test applications.

The sensor probes usually are mounted on a rake and installed within the inlet. A typical installation is shown in Fig. 13-1.

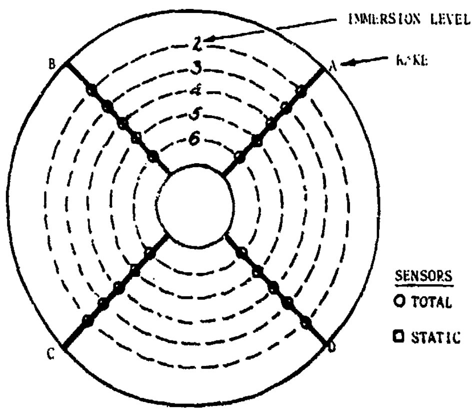


Figure 13-1. Inlet Rake

The location of this rake should be as near the compressor as feasible. Prior to installation, the rake must be tested to insure that it safely can withstand the dynamic pressure, vibrations, and temperatures that are anticipated. The number of probes needed will vary depending upon the data requirements; three should be considered a minimum unless the particular inlet and/or engine installation has been well defined by previous data. The fewer the number of probes, the more critical are the locations. Pressure profiles are very unpredictable both with station and immersion depth. A poor location thus can yield a very nonrepresentative value. The engine and aircraft manufacturer will have a certain amount of data on either the engine or the inlet which should be used when possible. However, data on the total installation may not be available which dictates that an estimation be made as to the best location.

During the engine calibration, the test cell usually is equipped with an inlet bellmouth that has extensive instrumentation. This instrumentation is generally more complete and more accurate than that available during the flight test. Thus, every opportunity should be taken to gather the maximum data possible during the calibration. This information serves as a valuable guide in determining the best location for the flight test probes. Any data from the tests with flight equipment installed can be compared to the data with the bellmouth to show installation losses. A total system calibration also can be determined.

As previously discussed, when averaged inlet performance will suffice, a minimum of three probes is adequate if suitably located. The inlet probes are usually total pressure sensors referenced to the free stream static pressure. When profile information is desired, the minimum installation is a rake of four azimuths with four immersion depths on each arm. For these data to be most useful, the rake should be located as close to the compressor as feasible or at some representative position based on previously recorded data.

The common practice for obtaining averaged data is to average the data from the three individual sensors. The data can be presented on a visual indicator and recorded manually or by automatic means. In the former case, transients cannot be defined; while in the latter, a high recording speed is necessary to produce statistically valid data. An analog recording will show best the average and the exact nature of the variations. This recording method requires a channel for each sensor and, when a great many probes are used, the recording capability may be insufficient. For this case, scani-valves and temperature steppers may be a practical means to obtain a time dependent sampling. The sampling rate for a given parameter will vary inversely with the number of recorded parameters. Sufficient analog channels should be used to allow correlation of the transients and the time dependent digital values. During the static ground tests, the flight instrumentation may be supplemented with engine analyzer equipment or fuselage mounted sensors near the inlet. This type of instrumentation is best suited for ground station recording equipment if compatible with the sensors and data requirements.

The calibration and preflight procedures are critical. A total system calibration is required since the individual sensor calibration may not include all system losses even when a null balance or compensation device is used. The preflight check must include data from each sensor as well as an appropriate ambient reading in order to provide correction or bias values. In particular, the temperature must be recorded precisely. Avoid the practice of taking preflight instrumentation readings and listing them as corresponding to an ambient reading at a remote location. Such things as open hangar doors, light sources, overhead hangar heaters, operating equipment, or wind can cause small local temperature changes of sufficient magnitude to invalidate the data. In the case of engine inlets, local conditions may dictate that the ambient readings be taken in the immediate vicinity of the sensors.

13-4 TEST PROCEDURES

The inlet performance tests should start with the static test cell calibration of the engine. The engine calibration specification should have included provisions for the various inlet and exhaust configurations. To achieve the full range of airflows within the inlet, the engine should be operated over the speed range from flight idle to the maximum during calibration. The airflow conditions must be stabilized prior to recording data. The baseline data usually are recorded during steady-state operation. Additional data should be obtained for acceleration and deceleration conditions. Of primary interest are operations that might introduce compressor stall or flameout conditions. Any unusual items concerning the inlet—such as inlet guide vanes, variable geometry, interference, blockage or distortion—will require additional tests. The extent of these can be determined only on an individual basis. All configurations should be well documented with photographs to assist in later correlation and interpretation of flight test results.

The installation losses of the particular configuration must be defined precisely. The engine calibration is the best opportunity to obtain basic data under optimum operating conditions with maximum instrumentation facilities. The calibration configuration should be the particular inlet to be used during the flight tests. When compared to the optimum, these data will show losses and provide correction factors for later test results. The final configuration then consists of the test inlet with the test instrumentation. Comparison of these results will show the losses introduced by the flight test instrumentation and provide a total system calibration.

During the ground run, engine rigging, thrust stand, and hovering tests, inlet data can be recorded at no expense in flight or operation time. These data will correlate with the engine calibration data and will include any contributions from the vehicle such as downwash, recirculation, or fuselage interference. The test procedures detailed in Chapters 4 and 5 must be integrated with the operation and inlet data requirements. Any ground recording equipment or additional sensors independent of the flight instrumentation must be correlated with respect to data recording and identification.

The flight test data should be representative of the complete test envelope. Data should be recorded for all the power required performance tests previously discussed. The conditions and procedures specified for the performance tests will provide a suitable test environment for collecting the inlet performance data. Traditionally, the inlet data have assumed a secondary priority in these tests and the true importance must be emphasized to the flight crew. The more critical data are for conditions of unusual airflow around the aircraft. Specifically, these conditions are in-ground effect hover, takeoff, and sideward and rearward flight. For these tests, pressure losses can be expected, distortion usually is increased, temperature rises, and fluctuations are common and may be aggravated by any possible engine exhaust reingestion.

13-5 DATA REDUCTION

The data reduction may be accomplished either with automatic or manual methods. When a great deal of data is recorded automatically, it is advisable to use similar reduction methods. Some of the difficulties to be encountered are nonuniform sampling times and rates, malfunctioning sensors, and erroneous recordings. These have a tendency to cause computer difficulties unless minutely detailed instructions are included to cope with any contingency. Particular attention should be given to include preflight and postflight biases obtained from the instrument calibrations. Resistance calibrations or other in-flight checks should be considered.

The data reduction form shown in Table 13-1* for a single parameter can be expanded to include any number of probes or locations. The form is for flight test data and can be modified to accommodate data from engine calibrations or other types of operations.

*The data reduction forms (tables) are located at the end of each chapter.

13-6 DATA PRESENTATION

The inlet performance data are presented as shown in Figs. 13-2 through 13-15.

TABLE 13-1

DATA REDUCTION FORM FOR INLET PERFORMANCE

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1	Ambient total pressure	P_{T_0}		psi	From free airstream sources
2	Static probe pressure	P_{s_2}		psi	From static probes or rake or manifold
3	Average static probe pressure	$P_{s_{2\,AVG}}$	$P_{s_{2\,AVG}} = \frac{\sum_{i=1}^{n} P_{s_{2_i}}}{n}$	psi	
4	Inlet total pressure	P_{T_2}		psi	
5	Average inlet total pressure	$P_{T_{2\,AVG}}$	$P_{T_{2\,AVG}} = \frac{\sum_{i=1}^{n} P_{T_{2_i}}}{n}$	psi	
6	Maximum static pressure	$P_{s_{2\,MAX}}$		psi	Maximum recorded during point on each sensor
7	Minimum static pressure	$P_{s_{2\,MIN}}$		psi	Minimum recorded during point on each probe
8	Static pressure distortion	L	$L = \frac{P_{s_{2\,MAX}} - P_{s_{2\,MIN}}}{P_{s_{2\,AVG}}}$		(8) = [(6) – (7)] / (3)
9	Maximum inlet total pressure	$P_{T_{2\,MAX}}$		psi	Maximum total pressure recorded on each sensor

TABLE 13-1 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
10	Minimum inlet total pressure	$P_{T_{2_{MIN}}}$		psi	Minimum total pressure recorded on each sensor
11	Total pressure distortion	K	$K = \dfrac{P_{T_{2_{MAX}}} - P_{T_{2_{MIN}}}}{P_{T_{2_{AVG}}}}$		$(11) = [(9) - (10)] / (5)$
12	Total pressure recovery factor	η_R	$\eta_R = \dfrac{P_{T_{2_{AVG}}}}{P_{T_a}}$		$(12) = (5) / (1)$
13	Inlet Mach number	M_2	$M_2 = \sqrt{\dfrac{2}{\gamma - 1}\left[\left(\dfrac{P_{T_{2_{AVG}}}}{P_{s_{2_{AVG}}}}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]}$		$(13) = \sqrt{5\left\{\left[\dfrac{(5)}{(3)}\right]^{0.286} - 1\right\}}$
14	Inlet static temperature	T_{s_2}		°K	From individual sensors or rake
15	Inlet total temperature	T_{T_2}	$T_{T_2} = T_{s_2}\left[1 + \left(\dfrac{\gamma - 1}{2}\right)M^2\right]$	°K	$(15) = (14) \cdot [1 + 0.2\,(13)^2]$

TABLE 13-1 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
16	Airflow	W_a	$W_a = P_{S_{2_{AVG}}} \times A_S \left\{ \frac{2\gamma}{\gamma-1}\left[\left(\frac{P_{T_{2_{AVG}}}}{P_{S_{2_{AVG}}}}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]\left(\frac{P_{T_{2_{AVG}}}}{P_{S_{2_{AVG}}}}\right)^{\frac{\gamma-1}{\gamma}} \times \frac{g}{RT_{T_2}}\right\}^{1/2}$	lb/sec	$(16) = (3) \cdot A_S \sqrt{\frac{7 \times 32.17}{R \cdot (15)}\left\{\left[\frac{(5)}{(3)}\right]^{0.286} - 1\right\}\left[\frac{(5)}{(3)}\right]^{0.286}}$ A_S = engine inlet area, ft^2
17	Mass flow	$\dot{\mu}$	$\dot{\mu} = W_a/g$	slug/sec	(17) = (16) / 32.17
18	Inlet pressure ratio	$P_{2_{PR}}$	$P_{2_{PR}} = \frac{P_{T_{2_{AVG}}}}{P_{S_{2_{AVG}}}}$		(18) = (5) / (3)
19	Inlet temperature ratio	θ_{T_2}	$\theta_{T_2} = T_{T_2}/T_o$		T_o = 288.15° K (19) = (15) / 288.15
20	Standard inlet air pressure ratio	δ_{T_2}	$\delta_{T_2} = P_{T_2}/P_o$		P_o = 14.7 psi (20) = (5) / 14.7
21	Engine speed	N_e		rpm	From recording time correlated with other data

Table 13-1 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
22	Corrected engine speed	N_{e_C}	$N_{e_C} = N_e / \sqrt{\theta_{T_2}}$	rpm	(22) = (21) / $\sqrt{(19)}$
23	Corrected airflow	W_{a_C}	$W_{a_C} = \dfrac{W_a \sqrt{\theta_{T_2}}}{\delta_{T_2}}$	lb/sec	(23) = (16) $\cdot \sqrt{19}$ / (20)
24	Gear height above the ground	H_G		ft	From photograph or cable length recording
25	Angle of attack	α		deg	From A/C attitude or free stream reference
26	Angle of sideslip	β		deg	From A/C or free stream reference
27	Free stream total temperature	T_{T_a}		°K	From free stream reference system
28	Inlet temperature rise	ΔT	$\Delta T = T_{T_2} - T_{T_a}$	°K	(28) = (15) – (27)

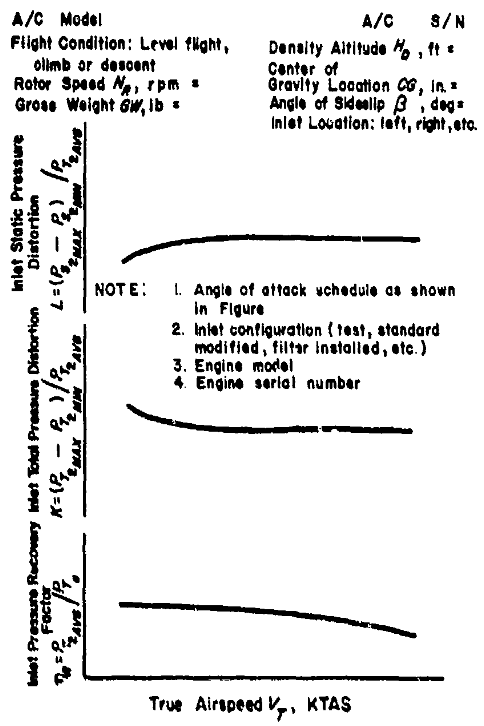


Figure 13-2. Inlet Performance Variation With Airspeed

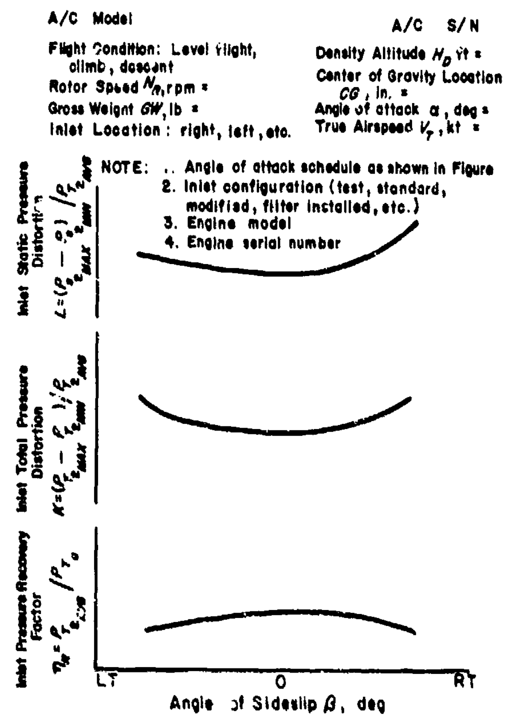


Figure 13-4. Inlet Performance Variation With Angle of Sideslip

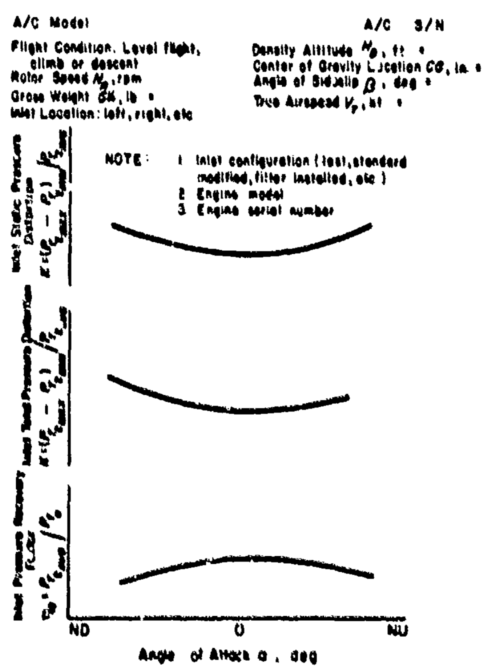


Figure 13-3. Inlet Performance Variation With Angle of Attack

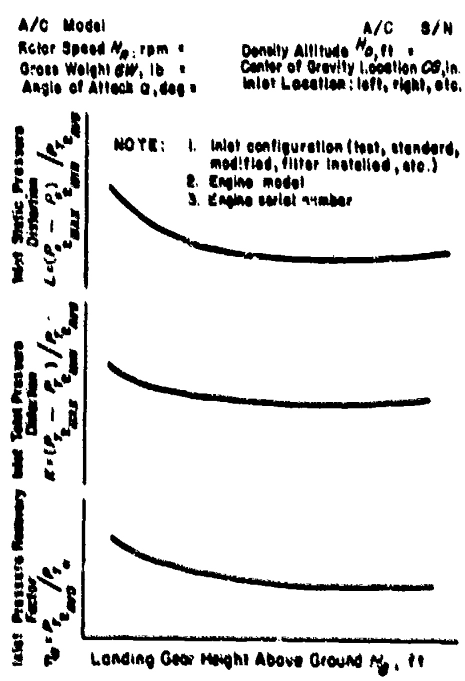


Figure 13-5. Inlet Performance Variation With Hover Height

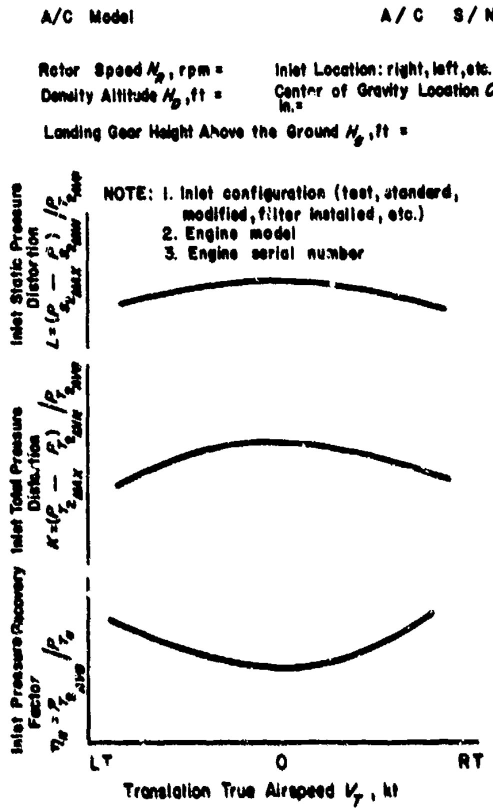


Figure 13-6. Inlet Pressure Characteristics During Translation

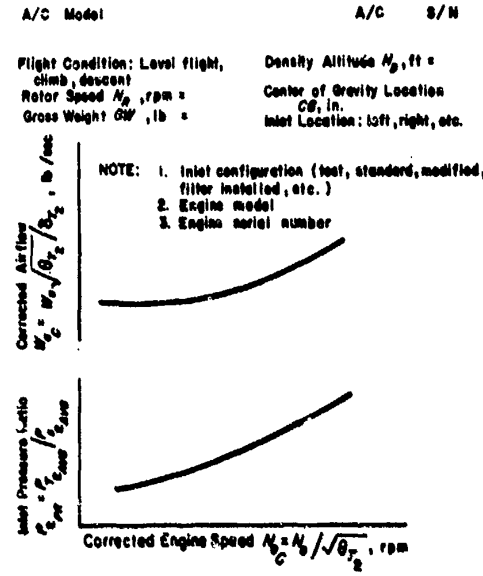


Figure 13-8. Inlet Performance Variation With Corrected Engine Speed

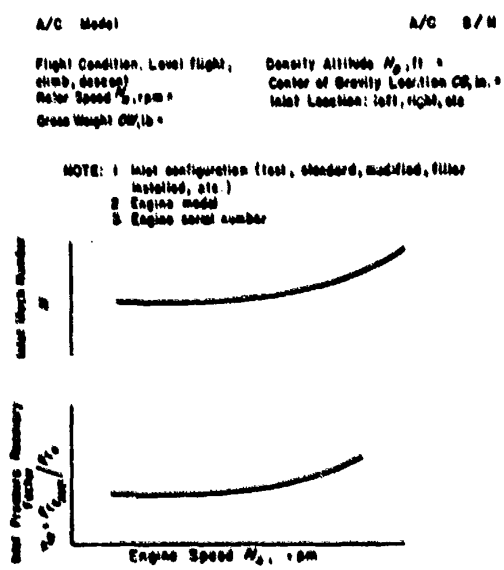


Figure 13-7. Inlet Performance Variation With Engine Speed

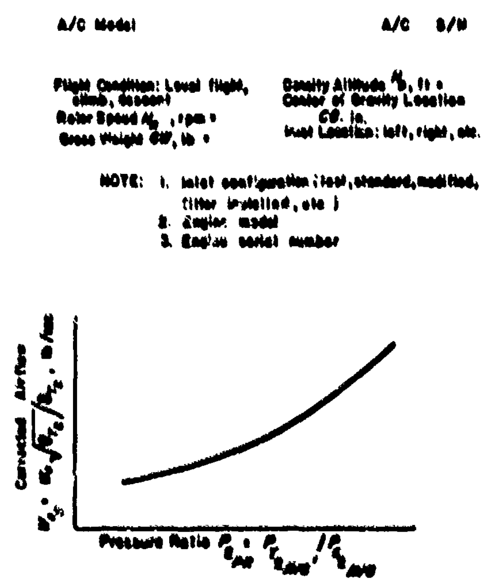


Figure 13-9. Inlet Airflow and Pressure Characteristics

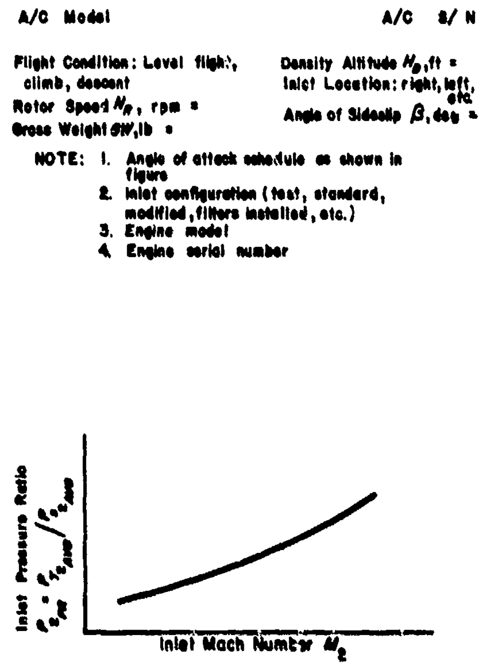


Figure 13-10. Inlet Airflow Characteristics

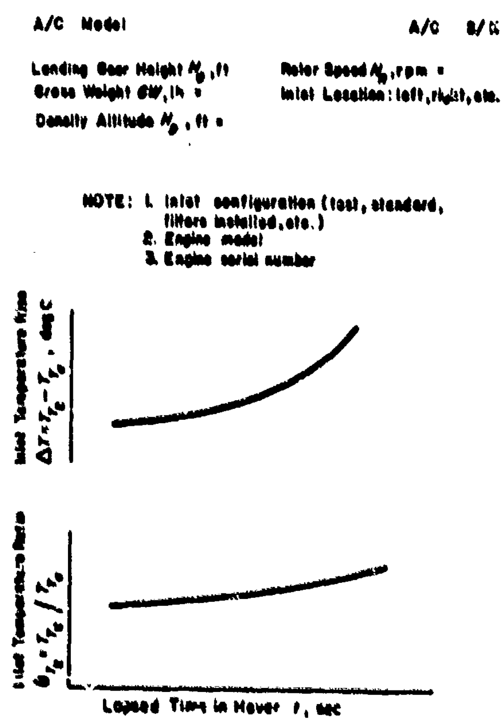


Figure 13-11. Inlet Temperature Variation During Hover

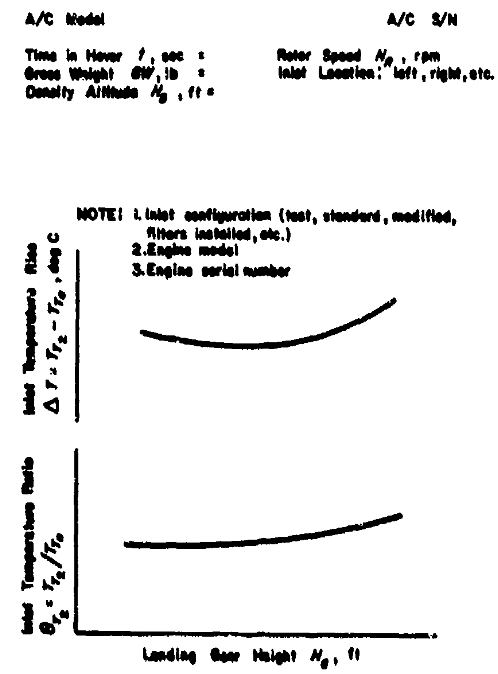


Figure 13-12. Inlet Temperature Variation With Hover Height

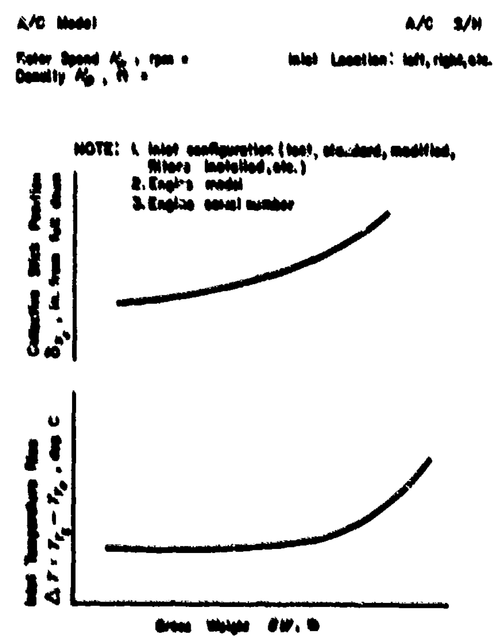


Figure 13-13. Inlet Temperature Variation With Hover Thrust

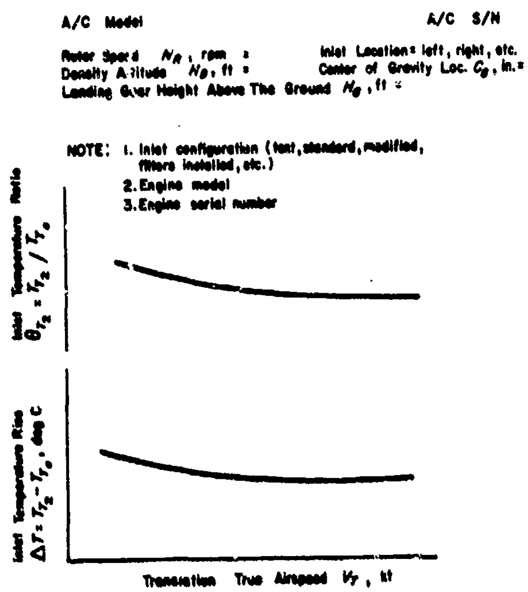


Figure 13-14. Inlet Temperature Rise During Translation

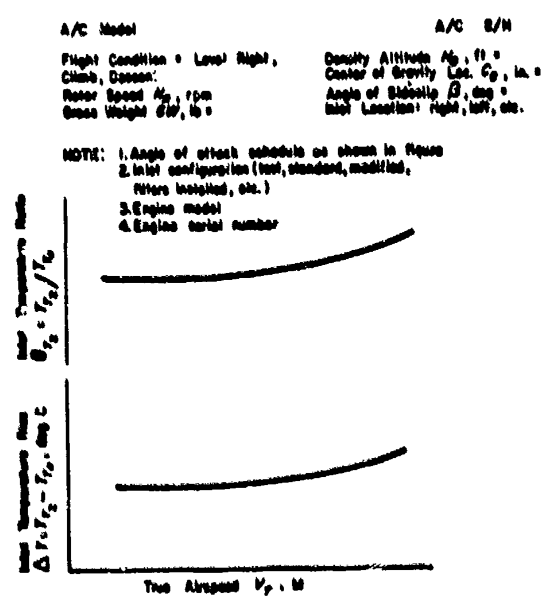


Figure 13-15. Inlet Temperature Variation With Airspeed

CHAPTER 14

ENGINE PERFORMANCE

14-1 GENERAL

The tests discussed in previous chapters are intended primarily to determine the power required characteristics for the aircraft in various flight regimes and operating conditions. The total performance evaluation also requires determination of power available and operating characteristics of the propulsion system. Engine theory and power charts often are used to determine the power that a given engine will deliver under certain conditions. These data are usually generalized by the engine manufacturer with little regard for any particular application or installation. Power charts ususally are based on known tolerances or deviations in a series of engines or power indicating devices. For these charts, an accuracy of ±5 percent is not uncommon—which is inadequate for flight test applications. Other means are needed to obtain an accurate measure of engine power. Engine calibrations are costly and time-consuming particularly when various environmental conditions are included.

Propeller and torque shafts are often difficult to instrument because of weight and space limitations, as well as mechanical and installation problems. However, this is the best measure of the power being developed. Since reciprocating engines are the exception rather than the rule, this discussion will be devoted to contemporary free turbine jet engines.

14-2 PLANNING

The purpose of the engine performance test is to evaluate the engine with respect to operating conditions and the particular installation or application. Engine peculiarities caused by the installation or accessories must be evaluated in terms of basic engine performance. Primary variables include flight regime, altitude, fuel flow, and pressure. The flight test data are used also to assess engine specification compliance for comparison with design criteria, and to calculate total aircraft performance. The engine response characteristics are evaluated with respect to mission demands and pilot suitability. The performance is defined over the total flight envelope and the data are generalized for all altitudes, engine speeds, and power levels. In many cases, there is a mismatch between the engine limits and those imposed by the aircraft power drive system.

Thus, flight data alone may be insufficient to define engine performance, and importance is lent to calibration and ground run data. This information can increase confidence in extrapolating the data to the appropriate limits. Helicopter performance is very sensitive to power; accordingly, the measurements must be made with a high degree of accuracy. Sufficient data must be obtained to define clearly the operating characteristics. Seemingly small items such as inlet, accessory, or installation losses are often quite significant in the final analysis.

Engine performance data are needed for all flight regimes and can be obtained during the standard aircraft performance tests, which greatly reduces the flight requirements. Most of these data is for the purposes of determining power required and not the maximum available. However, this is not of concern since (1) all of the data are used in developing the generalized engine operating curves, and (2) a certain amount of data will be at full power and should define the upper limits adequately. Careful planning is needed to incorporate the engine test requirements into the aircraft performance tests.

The instrumentation system must be designed to measure mechanical items such as engine control positions, as well as pressures, temperatures, and fuel flow. Many of these values are small in magnitude and transient in nature.

14-3 INSTRUMENTATION

The instrumentation needed to measure engine performance consists of devices for monitoring the mechanical, aerodynamic, and thermodynamic parameters. The instrumentation requirements are prepared in the form of an instrumentation specification, as discussed in Chapter 2. The primary mechanical measurement is of compressor and power turbine speeds, and should be quite accurate. These data are displayed visually on the photo panel and in the cockpit, and recorded by automatic devices. Inlet guide vanes, bleed valves, anti-icing devices, or throttle positions are recorded in the most suitable manner. The measurement of the operation of fuel control mechanisms normally is not required, but may be recorded if needed. If possible, the fuel flow rate should be recorded on a volumetric and time basis. Mass flow systems are susceptible to temperature changes and are not sufficiently accurate for flight test work. The fuel flow and temperature values should be sensed as near the fuel control as possible to obtain the most accurate parameters for determination of fuel mass flow. The system should be examined carefully for any fuel return or bypass features and, should they exist, the proper measuring sensors should be installed. Consideration should be given to instrumenting the engine or mounts for vibration—although not classified as a performance item, it can shed light on general engine operation and performance deterioration.

The free stream operational environment of the engine exists only at the compressor face and at the exhaust. This environment is measured in terms of total and static pressures, and temperatures. The inlet instrumentation has been discussed previously in par. 13-3, and the exhaust conditions are measured in the same manner. The magnitude of the instrumentation required will vary with the installation, however, long or complex exhausts can introduce sizable losses.

In addition to engine speed and fuel flow, the primary measures of engine performance are pressures, temperatures, and torques. Various methods are used to determine power; the most common are based on some combination of torque, turbine inlet temperature, compressor discharge pressure, exhaust gas temperature, and fuel consumption. Torquemeters are an integral accessory supplied with most contemporary free turbine engines and are calibrated as a part of the engine calibration. The decision to install an additional test system usually is based on historical data or the evaluation of the performance of the equipment during the calibration. Space permitting, the instrument should be installed to obtain the torque measurement on the engine output shaft. If the engine calibration did not include calibration of the total torquemeter and indicator system, additional calibrations probably will be required. Common practice is to parallel the standard torque indicator system with a more sensitive test system.

14-4 TEST METHODS

14-4.1 ENGINE CALIBRATION

The engine should be calibrated for the various configurations that will be used during the test program. These data will provide an early estimate of the performance that will be obtained during the flight test. The performance of an engine can be described by a function of speed, pressure, fuel flow, and temperature, i.e.,

$$f = \frac{SHP}{\delta\sqrt{\theta}}, \frac{N_1}{\delta\sqrt{\theta}}, \frac{W_f}{\delta\sqrt{\theta}} \qquad (14\text{-}1)$$

These relationships will form generalized curves only if changes in the efficiency or characteristics of the engine are negligible. Otherwise, the actual engine performance with altitude and temperature will not be the same as the predicted values. To determine accurately this variance, it is necessary to conduct calibrations at more than one set of atmospheric conditions.

During the planning phase, the author of the calibration specification should have considered the facilities, scope of test, and the data range and accuracy required. The first calibration should be at sea level standard day conditions for a basic engine with optimum inlet and exhaust configurations. The calibration should include all data that will aid in comparing the engine performance with the basic design points. In all the calibrations, engine speed is varied incrementally through the full operating range and data are recorded at each point. The time required to stabilize will vary with each engine and can be determined by monitoring the operating parameters. Recording data during nonstable conditions will introduce scatter in the data and can jeopardize the end result. At a given engine speed—usually 100%—the effects of engine trim, inlet guide vanes, and other auxiliary items should be evaluated. For this configuration, the requirement for an atmospheric calibration should be based on design considerations since the data are not directly applicable to flight test.

Repeat calibrations should be made with the engine configurations that will be flight tested—such as inlet instrumentation and exhaust hardware. The atmospheric calibration is more important here since it will provide an early indication of the estimated flight test performance. The auxiliary items should be well defined during this calibration since they often will form the basis for correction factors. Consideration also should be directed toward engine trim features, fuel control operation, and how they will influence the resultant performance. Output torque is measured both in absolute values and in terms of the engine torque indicator. When a special torque sensor is being used, it should be installed and calibrated at this time.

14-4.2 GROUND TESTS

Ground tests provide data on the total power plant installation at a zero airspeed condition. The downwash environment may cause deviations from the engine calibration results. In addition, at this time the engine performance may be related to the thrust producing devices (rotors, propeller, fans, etc.). The data requirements are incorporated into the maintenance ground runs when possible to increase productivity. By taking into account installation losses, the data obtained internally from the engine will compare with those obtained during the calibrations and will give an indication of the engine condition. Extreme care must be taken to insure correlation of the ground and onboard data recording equipment.

14-4.3 FLIGHT TESTS

The engine performance parameters are recorded during the previously discussed power required tests for the different operating conditions. For these steady-state conditions, power is set for a given test point and engine data are recorded. The engine must be allowed to stabilize at each test point and sufficient time must elapse to obtain a representative fuel flow rate. All parameters should be monitored for any ususual changes during the recording interval. Pertinent comments for recording engine data during each test and at each condition are presented in the appropriate chapters.

Although omitted from the previous discussion, the engine operating characteristics with respect to mission requirements and/or pilot compatibility are extremely important. Engine control and response often are neglected because of the lack of well defined specifications or requirements and the lack of appreciation of their overall importance. In the early development stages of an airframe/engine configuration, considerable attention is devoted to the mechanical aspects of the power transfer system—i.e., natural frequency of components, dynamic loading capability, and system rigidity. On either end of these systems are the engine fuel control and the rotor. The rotor is controlled through the cyclic and collective pitch which in turn transmits power demands through the system to the fuel control. The fuel control then adjusts the engine to operate in a manner that will comply with the pilot's needs and demands. Thus, an evaluation should be made as to the suitability of the total system while performing maneuvers related to mission accomplishment.

Characterisitically, helicopter operations place many demands on the propulsion system. Power demands are often, rapid, and vary greatly in magnitiude. The basic engine has several operating conditions such as steady-state, acceleration, and deceleration. There are fuel/engine speed areas where the engine will attempt to accelerate so rapidly that the engine will surge. Thus, an acceleration schedule must be incorporated in the fuel control to prevent this over-fuel condition. An alternate method is to provide inlet guide vanes that adjust the airflow with engine speed. Here over-fueling is considered to be the difference between the transient fuel flow demand and the steady-state requirement at that engine speed. Since rotor loading conditions will influence power turbine speeds, this should be a variable during the tests. Initial tests often are conducted with the aircraft on the ground and the rotor unloaded. From ground idle, the engine condition lever is placed rapidly in the "fly" position and the engine parameters are recorded. An automatic analog type data recording device is desirable, however, transient speeds and times will give an indication of the lag and acceleration rates. The results will vary with altitude and temperature; accordingly, the tests should cover the widest range feasible.

For an in-flight condition, the engine will be at some given speed under some applied load from the rotor. Thus, the engine will accelerate to supply the changing power demands from the rotor caused by collective or cyclic pitch change inputs. The tests are conducted for realistic or mission maneuvers—such as acceleration during takeoff, climb initiation—and during quick stop or power recovery. Both the rate and amount of collective input are varied incrementally to encompass the range of acceleration and loading conditions. Deceleration of the system is evaluated by lowering the collective to reduce the load and engine power requirements. This is done from a stabilized condition at various speeds and power settings. Both the rate and amount of collective change are the input variables.

The stability of the system can be evaluated by disturbing the rotor through pulse type inputs from the cockpit controls. The size, duration, and method of input will vary with each aircraft. the only stipulation being that they are of sufficient magnitude to cause an engine speed change. Particular emphasis should be placed on types or combinations of inputs the pilot will make during mission performance.

The governing capability and speed control authority provided the pilot is evaluated through the range of the engine speed select device. The operation of the device through full range should be timed and a judgment given of its suitability. During power changes, the speed control should be used to determine that the authority is sufficient to cope with any deviations. The extremes are for a sudden autorotational entry from a full power condition and for a rapid maximum power climb from a hover. A check should be made to insure the power available does not change as the speed select position is varied through its range. This is accomplished by varying rotor speed while at a full power condition. There will be a torque increase as rotor speed is reduced, however, power available should not decrease. For dual-engine aircraft, the previous tests are considered in view of relative engine performance and load carrying characteristics.

14-5 DATA REDUCTION

A general data reduction form for engine performance is shown in Table 14-1*. This will need modification for the various engine

*The data reduction forms (tables) are located at the end of each chapter.

types and models currently in use, and as such should be used primarily as a guide. Note that a great many items are not included and must be added for specific performance requirements. These, in addition to the information from Chapter 13 and other preceding portions, then can be used to accomplish a propulsion system evaluation.

In a flight test application, it is very useful to determine the power for any particular test point rapidly and easily. This contributes to the flight planning, test accomplishment, and greater flight safety. Since test vehicles usually have calibrated torquemeters, this can be done best by using a torquemeter constant that is developed as follows.

Power P is defined as the time rate of doing work w

$$P = dw/dt \tag{14-2}$$

Thus, for a rotating shaft with force F in pounds, radius R in feet and N revolutions per unit time, the power P developed is

$$P = 2\pi FRN \tag{14-3}$$

Defining a shaft horsepower SHP as 33000 lb-ft/min and torque $Q = F \times R$ (lb-ft), then

$$SHP = \frac{2\pi Q N_e}{33000} \tag{14-4}$$

When Q is the engine torque, the engine torquemeter calibration curve shown in Figure 14-1 can be entered to obtain the torque conversion factor.

In terms of flight parameters, the torque ΔP in psi is converted to torque Q in lb-ft by

$$Q = \Delta P \times K_{TM} \tag{14-5}$$

Since rotor speed N_R is more precisely measured and is better suited for calculations than engine speed, it should be used in conjunction with the gear ratio, K_{GR}, to obtain engine speed N_e

$$N_e = N_R \times K_{GR} \tag{14-7}$$

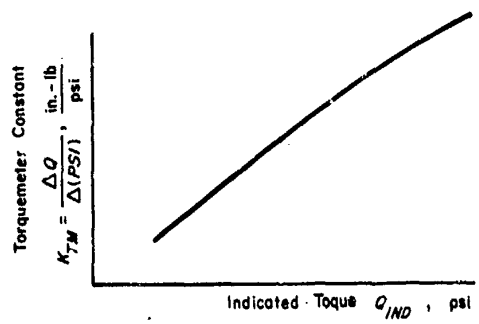


Figure 14-1. Torquemeter Constant as a Function of Power Setting

Substituting these values with their dimensions into the basic Eq. 14-4

$$SHP = \frac{2\pi\left(\frac{1}{\text{rev}}\right)\Delta P(\text{psi}) \times K_{TM}\left(\frac{\text{in.-lb}}{\text{psi}}\right) \times N_R\left(\frac{\text{rev}}{\text{min}}\right) \times K_{GR}}{12\left(\frac{\text{in.}}{\text{ft}}\right) \times 33000\left(\frac{\text{ft-lb}}{\text{min}}/SHP\right)} \tag{14-8}$$

The use of numerical values for the particular installation then will yield a simple equation for calculating SHP based on two indicated values

$$SHP = K_{TM} \times N_R \times \Delta P \tag{14-9}$$

As discussed in Chapter 9, currently available mass flowmeters are not of sufficient accuracy, which necessitates the volumetric measurement of fuel flow. Converting this to mass flow requires a fuel temperature measurement since specific gravity is a function of temperature. Since the fuel temperature changes during the flight, the fuel mass flow must be calculated at each test point to obtain accurate engine performance data. Fuel normally is obtained from large tanks that were filled with various production runs of fuel, each with a slightly different specific gravity. The preflight specific gravity of the fuel is measured. Extensive tests have shown that for a particular grade of fuel, such as JP-4, the variation in specific gravity is essentially the same regardless of the value for an individual sample as shown in Fig. 14-2.

The preflight value is plotted as shown in Fig. 14-2. The variation with temperature is assumed to follow the average which allows the anticipated variation curve to be drawn. The test fuel temperature then is used to enter this figure and obtain the specific weight for the individual test condition. The curve is linear so that in Fig. 14-2,

$$\tan\theta = \frac{\Delta(FSW)}{\Delta T_f} = \frac{(FSW_2 - FSW_1)}{(T_{f_2} - T_{f_1})} \qquad (14\text{-}10)$$

where subscripts 1 and 2 denote preflight and test conditions, respectively.

Then

$$FSW_2 = FSW_1 + (T_{f_2} - T_{f_1})\tan\theta \qquad (14\text{-}11)$$

To illustrate the procedure, two high and low averages at two temperatures of 30 fuel samples of JP-4 are

SPECIFIC GRAVITY SG at 0°F	SPECIFIC GRAVITY SG at 240°F	$\Delta SG/\Delta T$
0.8265	0.7260	0.00041878
0.7990	0.6955	0.00043128
0.7865	0.6810	0.00043962
0.7775	0.6710	0.00044379
	AVG	0.00043336

$$\tan\theta = \frac{\Delta SG}{\Delta T} = 0.00043336 \qquad (14\text{-}12)$$

$$\text{volume of fluid} = 7.481\ \text{gal/ft}^3 \qquad (14\text{-}13)$$

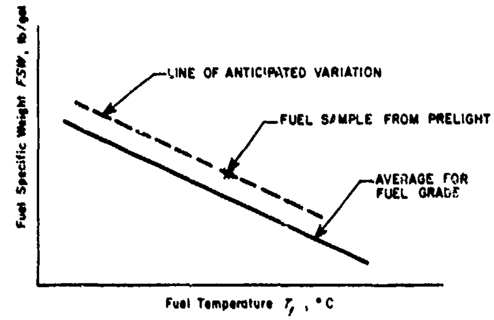


Figure 14-2. Fuel Specific Weight Variation With Temperature

$$\text{Water specific weight} = 62.4\ \text{lb/ft}^3 \text{ at } 4^\circ\text{C} \qquad (14\text{-}14)$$

$$\text{Water specific weight} = \frac{62.4}{7.481} = 8.3411\ \text{lb/gal} \qquad (14\text{-}15)$$

Then

$$FSW_2 = 8.341 \times SG_1 - 4.3336 \times 10^{-4}(T_{f_2} - T_{f_1}) \qquad (14\text{-}16)$$

The engine data are most conveniently defined by assigning subscripts which denote engine stations. This is nonstandard throughout the industry; however, in this document, Fig. 14-3 is applicable.

14-6 DATA PRESENTATION

The engine performance data are presented as shown in Figs. 14-4 through 14-24.

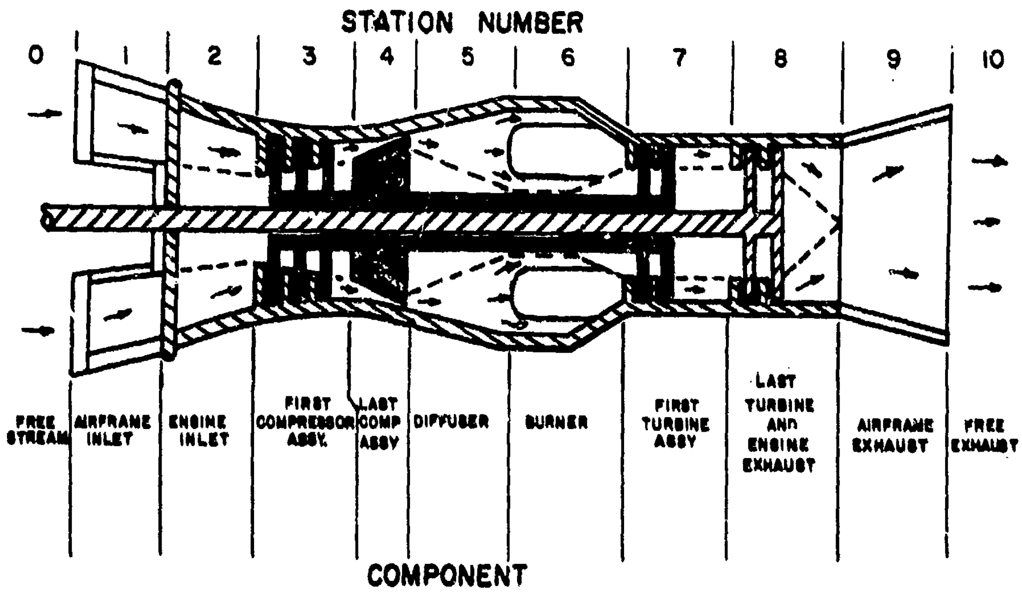


Figure 14-3. Engine Stations and Terminology

TABLE 14-1

DATA REDUCTION FORM FOR ENGINE PERFORMANCE

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
1-28					From Table 13-1
29	Inlet guide vane position	IGV		% from full open	
30	Throttle position	δ_T		% from full open	
31	Indicated torque	ΔP		psi	From standard and test systems in appropriate units
32	Actual torque	Q		lb-in.	From test stand or power shaft instrumentation
33	Compressor discharge pressure/turbine inlet total pressure	P_{T_4}		psi	From test stand or aircraft instrumentation
34	Engine pressure ratio	PR_e	$PR_e = P_{T_4} / P_{T_{2\,AVG}}$		(34) = (33) / (5)
35	Turbine inlet temperature	T_{T_7}		°C	From test stand or aircraft instrumentation
36	Exhaust gas temperature	EGT		°C	From test stand or aircraft instrumentation

TABLE 14-1 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
37	Fuel flow	W_f		lb/hr	From test stand or aircraft instrumentation
38	Engine thrust	F_g		lb	From test stand instrumentation
39	Exhaust gas static pressure	P_{s_8}		psi	From test stand or aircraft instrumentation
40	Engine inlet temperature ratio	TR_e	$TR_e = \dfrac{T_{T_1}}{T_{T_{1_{AVG}}}}$		(40) = (35) / (15)
41	Test cell shaft horsepower	SHP_{TC}	$SHP_{TC} = \dfrac{Q \times N_{SHAFT}}{500}$	HP	Calculated from measured torque at (32)
42	Torquemeter conversion constant	K_{TM}	$K_{TM} = \dfrac{SHP_{TC} \times 12 \times 33000}{2\pi \times \Delta P \times N_R \times K_{GR}}$	in.-lb/psi	Calculated from test cell K_{GR} = gear ratio $(42) = \dfrac{(41) \cdot 12 \cdot 33000}{6.28 \cdot (31) \cdot K_{GR} \cdot (44)}$
43	Test shaft horsepower	SHP_t	$SHP_t = \Delta P \times K_{TM} \times N_R \times K_{GR}$	HP	Calculated from test instrument indicated torque
44	Rotor speed	N_R		rpm	Stabilized at some test condition

TABLE 14-1 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
45	Power turbine speed	N_{II}	N_{II} = total gear ratio x N_R	rpm	From test cell data or calculated from flight test data; (45) = K_{GR} · (44)
46	Corrected engine speed	N_{e_C}	$N_{e_C} = \frac{N_e}{\sqrt{\theta_{T_2}}}$	rpm	(46) = (21) / $\sqrt{(19)}$
47	Corrected shaft horsepower	SHP_C	$SHP_C = \frac{SHP_t}{\delta_{T_2}\sqrt{\theta_{T_2}}}$	HP	(47) = (43) / [(20) · $\sqrt{(19)}$]
48	Corrected fuel flow	W_{f_C}	$W_{f_C} = \frac{W_f}{\delta_{T_2}\sqrt{\theta_{T_2}}}$	lb/hr	(48) = (37) / [(20) · $\sqrt{(19)}$]
49	Corrected airflow	W_{a_C}	$W_{a_C} = W_a\sqrt{\theta_{T_2}}/\delta_{T_2}$	lb/sec	(49) = (16) · $\sqrt{(19)}$ / (20)
50	Ideal mass flow	W_{a_i}		lb/sec	Use equation shown in Table 13-1, step 16
51	Interstage bleed mass flow	W_{IB}		lb/sec	Use equation shown in Table 13-1, step 16

TABLE 14-1 (Continued)

STEP NO.	DEFINITION	SYMBOL	EQUATION	UNITS	REMARKS
52	Corrected interstage bleed airflow	W_{IB_C}	$W_{IB_C} = W_{IB}\sqrt{\theta_{T_2}}/\delta_{T_2}$		$(52) = (51)\cdot\sqrt{(19)}/(20)$
53	Interstage bleed airflow ratio	$W_{IB_{AR}}$	$W_{IB_{AR}} = W_{IB}/W_a$		(53) = (51) / (16)
54	Differential time	Δt	$\Delta t = t_2 - t_1$	sec	Time from start of a maneuver to the test point in time
55	Rotor speed static droop	ΔN_{R_s}	$\Delta N_{R_s} = N_R - N_{R_T}$	rpm	rpm difference from initial stabilized rotor speed to stabilized value after the maneuver
56	Rotor speed transient droop	ΔN_{R_T}	$\Delta N_{R_T} = N_R - N_{R_T}$	rpm	Maximum rpm difference from initial stabilized value and maximum recorded during maneuver
57	Rotor speed time lag	Δt_R	$\Delta t_R = t_2 - t$	sec	Time from start of maneuver to change in rotor speed

A/C Model A/C S/N

NOTES: 1. Engine model
2. Engine serial number
3. Torque measuring system (test, standard, etc.)

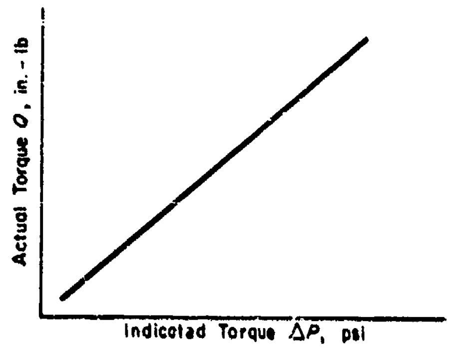


Figure 14-4. Torquemeter Calibration

A/C Model A/C S/N

Rotor Speed N_R, rpm =

Density Altitude H_D, ft =

NOTE: 1. Engine model
2. Engine serial number
3. Engine location: right, left, etc
4. Inlet configuration (test, standard, modified, filter installed, etc.)
5. Zero ram airflow

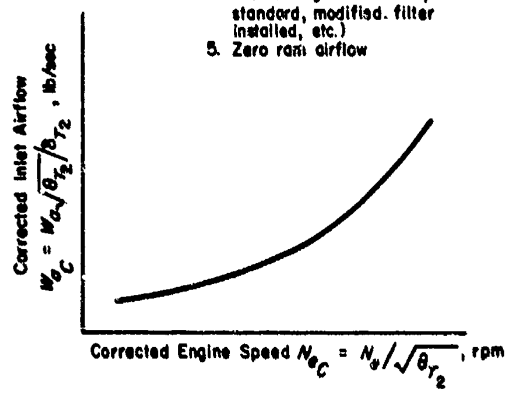


Figure 14-6. Inlet Airflow and Engine Speed

A/C Model A/C S/N

Rotor Speed N_R, rpm

Density Altitude H_D, ft =

NOTE: 1. Engine model
2. Engine serial number
3. Engine location: left, right, etc.
4. Inlet configuration (test, standard, modified, filter installed, etc.)
5. Ideal airflow calculated from

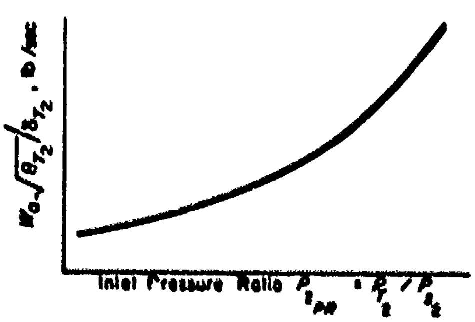


Figure 14-5. Airflow Variation With Inlet Pressure Ratio

A/C Model A/C S/N

Rotor Speed N_R, rpm =

Density Altitude H_D, ft =

NOTE: 1. Engine model
2. Engine serial number
3. Engine location: right, left, etc
4. Inlet configuration (test, standard, modified, filter installed, etc.)
5. Zero ram airflow

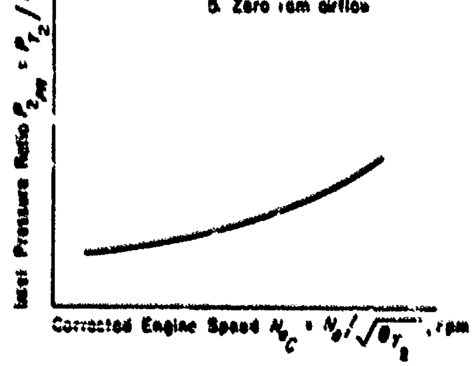


Figure 14-7. Inlet Pressure and Engine Speed

A/C Model A/C S/N

NOTES:
1. Engine model
2. Engine serial number
3. Engine location: left, right, etc.
4. Inlet configuration (test, standard, modified, filter installed, etc.)

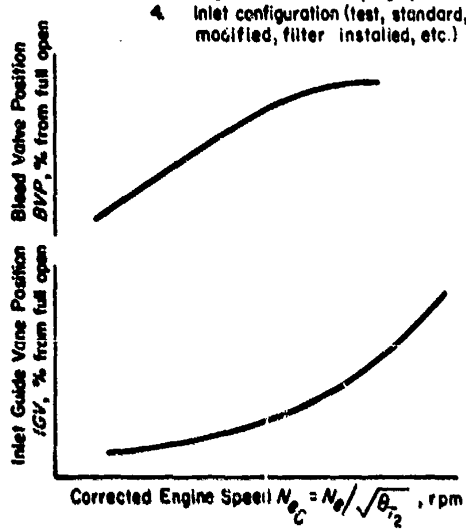


Figure 14-8. Inlet Guide Vane and Bleed Valve Position With Engine Speed

A/C Model A/C S/N

Rotor Speed N_R, rpm =

Density Altitude H_D, ft =

NOTES:
1. Engine model
2. Engine serial number
3. Inlet configuration (test standard, modified, filter installed, etc.)
4. Engine location: left, right, etc.
5. Exhaust configuration

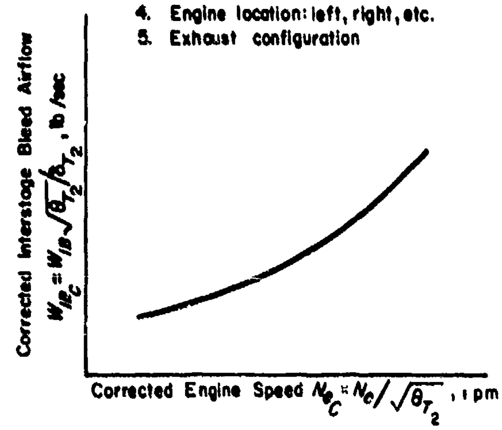


Figure 14-10. Interstage Bleed Airflow and Engine Speed

A/C Model A/C S/N

Rotor Speed N_R, rpm =

Density Altitude H_D, ft =

NOTES:
1. Engine model
2. Engine serial number
3. Engine location: left, right, etc.
4. Inlet configuration (test, standard, modified, filter installed, etc.)

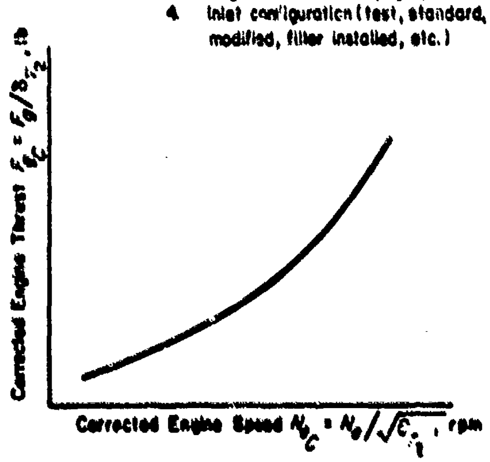


Figure 14-9. Engine Thrust and Engine Speed

A/C Model A/C S/N

Rotor Speed N_R, rpm =

Density Altitude H_D, ft =

NOTES
1. Engine model
2. Engine serial number
3. Engine location: left, right, etc.
4. Inlet configuration (test standard, modified, filter installed, etc.)
5. Exhaust configuration

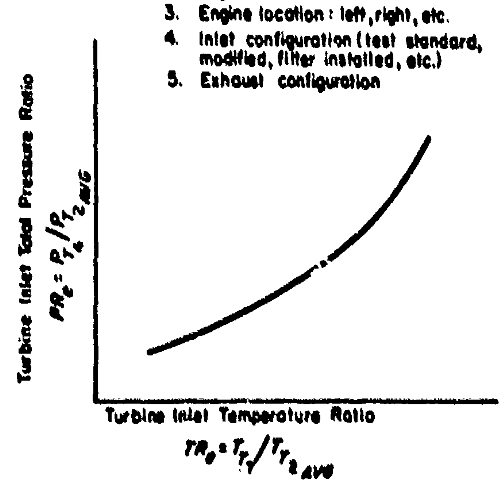


Figure 14-11. Turbine Inlet Pressure and Temperature Characteristics

A/C Model A/C S/N

Rotor Speed N_R, rpm =

Density Altitude H_D, ft =

NOTES: 1. Engine model
2. Engine serial number
3. Engine location: left, right, etc.
4. Inlet configuration (test, standard, modified, filter installed, etc.)

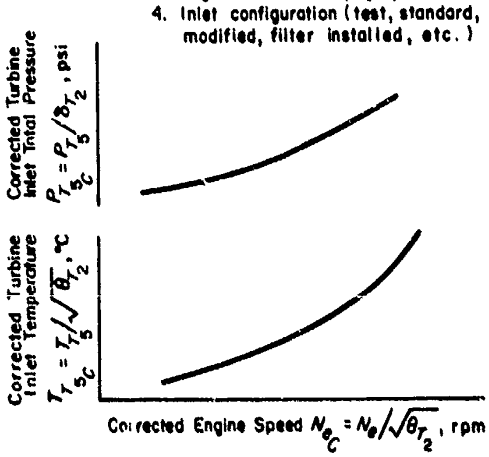


Figure 14-12. Pressure and Temperature Variation With Engine Speed

A/C Model A/C S/N

Rotor Speed N_R, rpm =

Density Altitude H_D, ft =

NOTES: 1. Engine model
2. Engine serial number
3. Engine location: left, right, etc.
4. Inlet configuration (test, standard, modified, filter installed, etc.)
5. Exhaust configuration

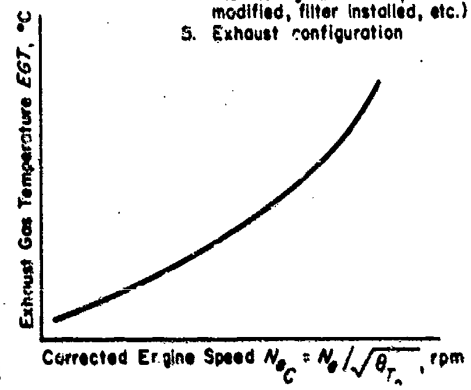


Figure 14-14. Exhaust Gas Temperature and Engine Speed

A/C Model A/C S/N

Rotor Speed N_R, rpm =

Density Altitude H_D, ft =

NOTES: 1. Engine model
2. Engine location: left, right, etc.
3. Engine serial number
4. Inlet configuration (test, standard, modified, filter installed, etc.)
5. Fuel grade (JP-4)
6. Fuel specific weight at ___°C

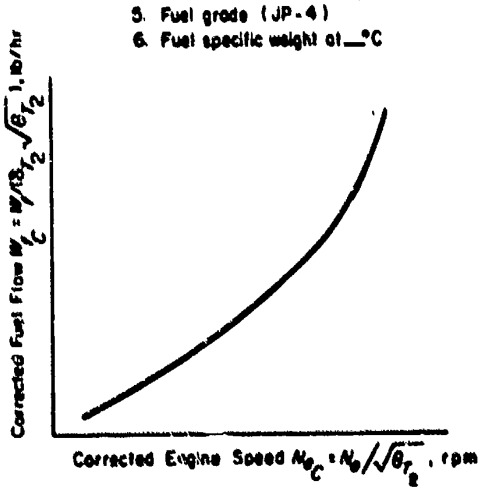


Figure 14-13. Fuel Flow and Engine Speed

A/C Model A/C S/N

Rotor Speed N_R, rpm =

Density Altitude H_D, ft =

NOTES: 1. Engine model
2. Engine serial number
3. Engine location: left, right, etc.
4. Inlet configuration (test, standard, modified, filter installed, etc.)

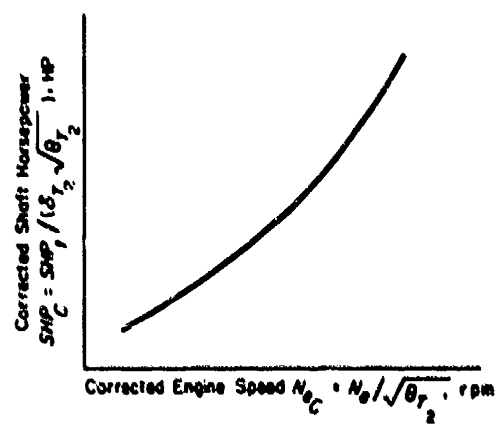


Figure 14-15. Shaft Horsepower and Engine Speed

A/C Model A/C S/N

Rotor Speed N_R, rpm =

Density Altitude H_D, ft =

NOTES:
1. Engine model
2. Engine serial number
3. Engine location: left, right, etc.
4. Inlet configuration (test, standard, modified, filter installed, etc.)
5. Flight condition: (hover, climb, level flight, descent, etc.)
6. Fuel grade (JP-4)
7. Fuel specific weight at ___°C

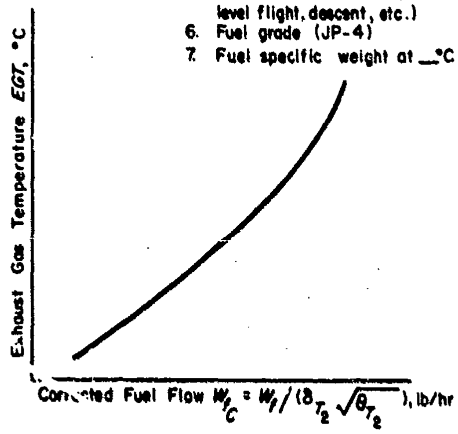


Figure 14-16. Exhaust Temperature Variation With Fuel Flow

A/C Model A/C S/N

Rotor Speed N_R, rpm =

Density Altitude H_D, ft =

NOTES:
1. Engine model
2. Engine serial number
3. Engine location: left, right, etc.
4. Inlet configuration (test, standard, modified, filter installed, etc.)
5. Fuel grade
6. Fuel specific weight at ___°C

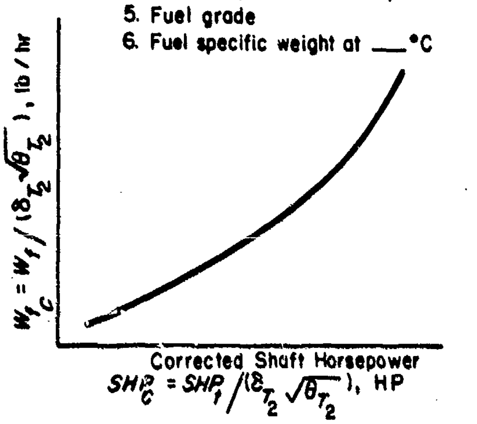


Figure 14-18. Airflow and Shaft Horsepower

A/C Model A/C S/N

Rotor Speed N_R, rpm =

Density Altitude H_D, ft =

NOTES:
1. Based on inlet recovery characteristics as shown in Fig. 13-2
2. Engine model
3. Engine serial number
4. Engine location: left, right, etc.
5. Inlet configuration (test, standard, modified, filter installed, etc.)

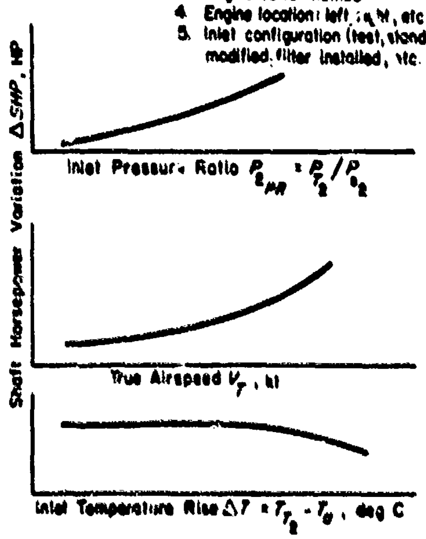


Figure 14-17. Shaft Horsepower Variation With Inlet Pressure and Temperature

A/C Model A/C S/N

Rotor Speed N_R, rpm =

Density Altitude H_D, ft =

NOTES:
1. Engine model
2. Engine serial number
3. Engine location: left, right, etc.
4. Inlet configuration (test, standard, modified, filter installed, etc.)
5. Fuel grade
6. Fuel specific weight at ___°C

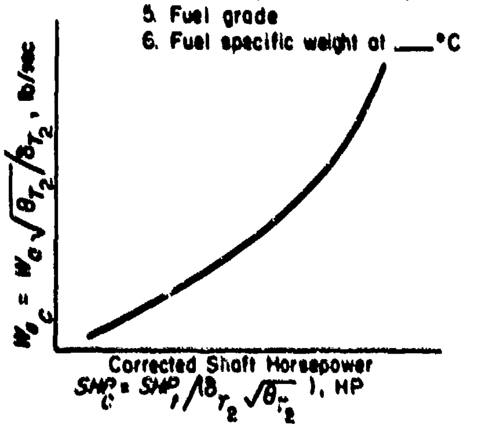


Figure 14-19. Fuel Flow and Shaft Horsepower

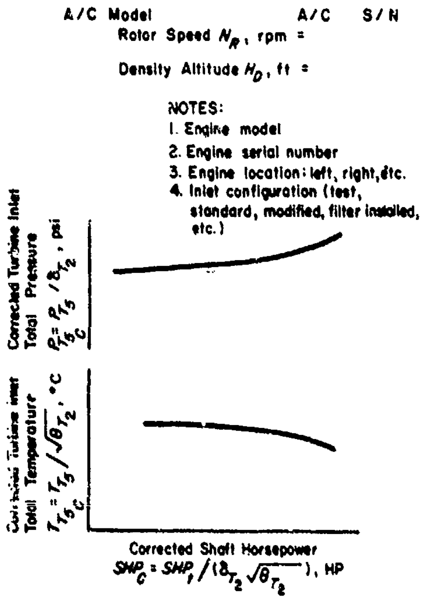


Figure 14-20. Turbine Inlet Total Pressure and Temperature Variation With Shaft Horsepower

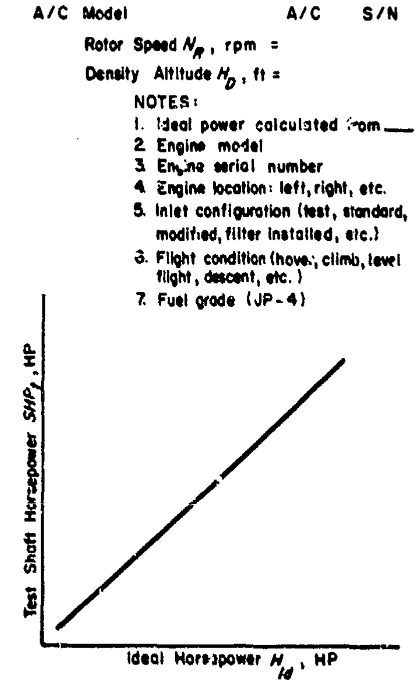


Figure 14-22. Total Engine Efficiency

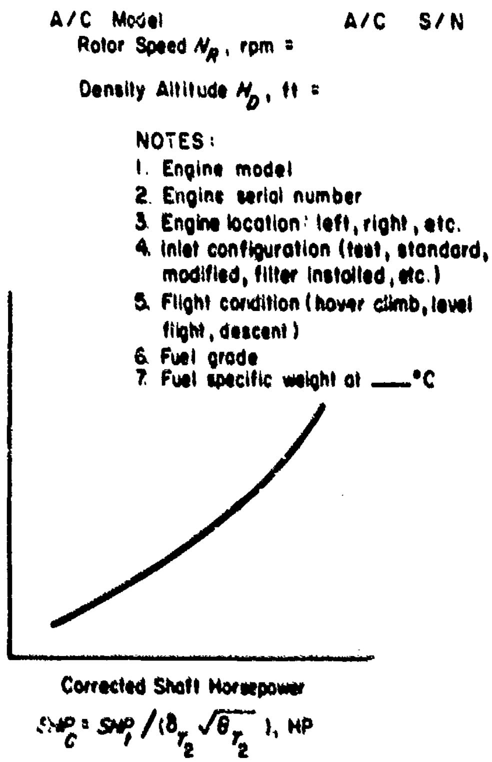


Figure 14-21. Exhaust Gas Temperature and Shaft Horsepower

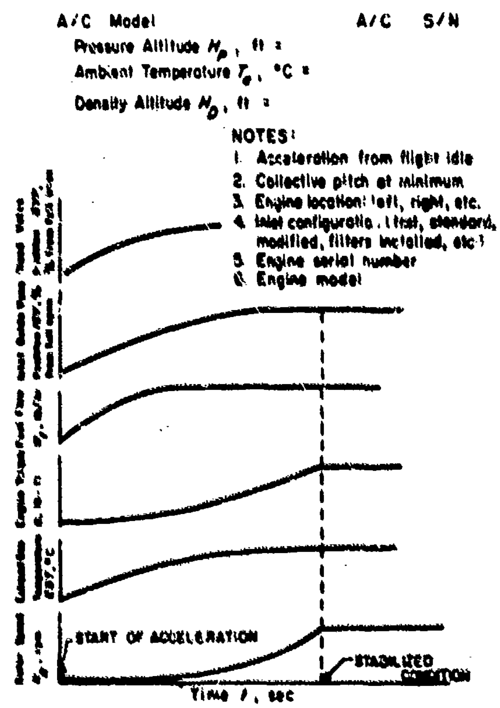
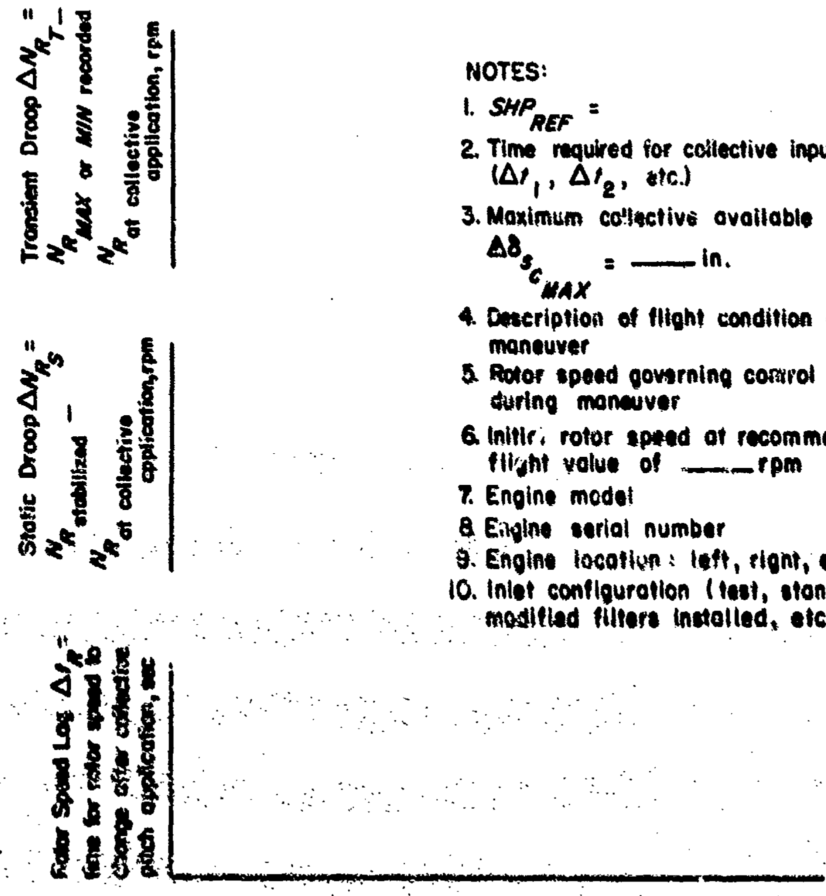


Figure 14-23. Engine Acceleration Characteristics

A/C Model | A/C S/N
Pressure Altitude H_p, ft = | Density Altitude H_D, ft =
Ambient Temperature T_o, °C = | Gross Weight GW, lb =



NOTES:

1. SHP_{REF} =
2. Time required for collective input (Δt_1, Δt_2, etc.)
3. Maximum collective available $\Delta\delta_{s_{c_{MAX}}}$ = ____ in.
4. Description of flight condition or maneuver
5. Rotor speed governing control not used during maneuver
6. Initial rotor speed at recommended flight value of ____ rpm
7. Engine model
8. Engine serial number
9. Engine location: left, right, etc.
10. Inlet configuration (test, standard, modified filters installed, etc.)

Incremental Longitudinal Control Input $\Delta\delta_{s_e} = \delta_{s_{e_{MAX\ RECORDED}}} - \delta_{s_{e_{REF}}}$, in.

Incremental Lateral Control Input $\Delta\delta_{s_a} = \delta_{s_{a_{MAX\ RECORDED}}} - \delta_{s_{a_{REF}}}$, in.

Incremental Directional Control Input $\Delta\delta_{s_r} = \delta_{s_{r_{MAX\ RECORDED}}} - \delta_{s_{r_{REF}}}$, in.

Incremental Collective Control Input $\Delta\delta_{s_c} = \delta_{s_{c_{MAX\ RECORDED}}} - \delta_{s_{c_{initial}}}$, in.

Incremental Power Change $\Delta SHP_t = SHP_{MAX\ RECORDED} - SHP_{REF}$, HP

Figure 14-24. Droop Characteristics

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