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LEVEL II

LIFT PERFORMANCE INDICATOR SYSTEM FEASIBILITY STUDY

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Final Report

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Prepared for

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Fort Eustis, Va. 23604

APPLIED TECHNOLOGY LABORATORY POSITION STATEMENT

The work reported herein is part of a continuing effort of the Applied Technology Laboratory to conduct investigations directed toward advancing the state of the art of cargo handling for Army aircraft. The object of this particular effort was to investigate the feasibility of a system for determining helicopter lift performance. Such a system would be used to inform a helicopter pilot of a lifting capability of his aircraft prior to flying so that he can be assured that his cargo load does not exceed the capabilities of the helicopter. Several previous research efforts have been sponsored by the Applied Technology Laboratory on possible input parameters for a lift performance system. This current effort includes an assessment of the results of prior efforts.

The results of this study have shown that helicopter lift performance indicator systems that would inform pilots of the likelihood of successful takeoffs and landings before committing the aircraft to flight are feasible for the aircraft addressed in the study, except that in the case of the UH-1, significant redesign of the skid-landing gear would be required to accommodate and facilitate incorporation of weight measurement instrumentation. Of the helicopters surveyed, the CH-47C shows the greatest need for a lift performance indicator system based on aircraft configuration, load capacities, and typical mission and cargo types. An in-house investigation which was conducted concurrently with this contractural effort revealed that a relatively small number of helicopter accidents related to overgross condition operations occurred with cargo helicopters (i.e., CH-47).

However, as pointed out in the report, there appears to be some validity in the development of a weight measurement and center-of-gravity indicating system for use on cargo helicopters. The state of the art of such systems is such that it should not require further research efforts but rather could be developed with little difficulty. The decision to develop such a system is considered to be one for the appropriate aircraft project manager. It is therefore concluded that no further research work is justified on this subject by this Laboratory unless a change in the requirements develops.

The technical monitor for this contract was Mr. G. William Hogg, Aeronautical Systems Division.

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This report presents the results of a study that was conducted to determine the feasibility of a helicopter lift performance indicator (LPI) system that would inform the pilot of the likelihood of a successful takeoff and landing before committing the aircraft to flight. The study includes the determination of the functional requirements (including accuracy) for an LPI system. Four specific helicopters were considered: CH-47C, CH-54B, UH-1H, and UH-60A (UTTAS).

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No single performance criterion or measure conveys the capability of the helicopter to execute all of the various takeoff and landing maneuvers. The LPI system should therefore be capable of informing the pilot of the vehicle's hovering capabilities (both in and out of ground effect), vertical and best-airspeed climb capabilities, and several other performance measures (essentially the same type of information as contained in the performance section of the operator's manual for the vehicle). This information should be selectable for real-time and remote-site conditions (input by the pilot), for both single and dual engine operation, and for maximum and normal rated power conditions. Wind velocity effects should be included. Generating this information in advance of lift-off requires that actual vehicle weight be measured. The system should also compute center of gravity from measured weight because of its impact on performance capability and its importance with respect to cargo positioning.

A system that will provide the above information is feasible. In the recommended approach, helicopter takeoff and landing performance capabilities are calculated from stored vehicle performance characteristics using estimated power available for the measured ambient conditions and measured gross weight. Vehicle performance characteristics are based on standard flight test results and do not require modification to reflect rotor degradation. Power available is calculated from standard engine power available functions modified by a calibration factor extracted from the standard topping check. Vehicle weight is obtained by measuring oleo strut pressures. An antistiction technique is recommended for maximum accuracy. In the case of skid-type landing gear (UH-1H), direct weight measurement is not feasible without significant redesign of the landing gear to facilitate sensor installation.

A program for development of the LPI system is recommended and outlined.

PREFACE

This study was conducted for the Applied Technology Laboratory, U.S. Army Research and Technology Laboratories (AVRADCOM), Fort Eustis, Virginia, under Contract DAAJ02-76-C-0036.

AVRADCOM technical direction was provided by Mr. G. W. Hogg.

The study was performed by the Applied Systems Research Group of the Electronics department of AiResearch Manufacturing Company of California. The following personnel participated in the program: W. J. Harris, Dr. J. D. Chang, R. E. Vesque, Dr. J. Kukel, J. E. Minnear, W. H. McCormack, and B. L. Mellinger.

In conducting interviews with U.S. Army helicopter pilots, the assistance of the following personnel was essential and is gratefully acknowledged:

Lt. Colonel S. D. Arnold, U.S. Army Aircraft Development and Test Agency, Fort Rucker, Alabama

Major R. Kerrigan, Commanding Officer, 179th Aviation Company, Fort Carson, Colorado

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1. INTRODUCTION

1.1 BACKGROUND

In a review of U.S. Army operations in Vietnam, Lieutenant Colonels Watson and Dunham discussed problem areas and suggested a number of improvements for Army helicopters that would "enhance operational effectiveness and safety."[1] Foremost among suggested improvements for future Army aircraft was the following:

> A performance indicating device that would provide the crew with accurate and dependable information as to predicted performance under existing gross weight and atmospheric conditions. The system should utilize a lightweight computer that would correlate information received from a gross weight measuring device with atmospheric sensors and stored helicopter performance data to determine and transmit to an instrument in the cockpit the helicopter's predicted performance.

The effort reported here was addressed to determining the feasibility of just such a device--a helicopter lift performance indicator (LPI) system that would inform the pilot of the likelihood of a successful takeoff and landing before committing the aircraft to flight.

Although this is the first program to explore the feasibility of the above type of system, several previous efforts sponsored by the Army have dealt with important elements of the problem. In 1966, a requirement was generated for an integral gross weight measurement system for the CH-47 under the provisions of AR 71-1's Expedited Nonstandard Urgent Requirements for Equipment (ENSURE). This resulted in the side-by-side test in 1967 of two different weight and balance systems on a CH-47 at the U.S. Army Aviation Test Facility at Edwards Air Force Base, California.[2] One system ("STAN", supplied by Fairchild) was based on oleo pressure measurement, the other on strain gage sensing of axle deflections ("STOW", supplied by National Water Lift). Both systems failed to meet performance objectives, the principal deficiency being excessive error under dynamic weighing conditions (rotors turning at flight idle power level).

The Army apparently concluded that a development effort would be required to obtain a weight and balance system with the desired performance characteristics and, for that reason, the ENSURE requirement could not be met. There is no evidence of any direct follow-up action to eliminate the deficiencies noted in the referenced test.

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[1]LTC William R. Watson, Jr., and LTC John R. Dunham, Jr., "Resume of U.S. Army Helicopter Operations in Vietnam," Proceedings, American Helicopter Society 24th Annual National Forum, May 8-10, 1968.

[2]Allyn E. Higgens, et al, Engineering Flight Test of the CH-47 (Chinook) Helicopter Integral Weight and Balance Systems (ENSURE), U.S. Army Aviation Test Activity, Edwards AFB, California, March 1968.

The Applied Technology Laboratory, U.S. Army Air Mobility Research and Technology Laboratories (AVRADCOM), has sponsored several contractual efforts directed toward establishing the feasibility of obtaining the various necessary parametric inputs to an LPI system. In perhaps its earliest involvement, the Laboratory sponsored a design study that resulted in the adaptation for the CH-47 of a production weight and balance system used on the USAF C-130 (National Water Lift's System for Takeoff Weight--STOW).[3] This formed the basis for the system that was later tested in the ENSURE program.

A more recent effort sponsored by the Applied Technology Laboratory was directed at solving the principal weight and balance measurement problem noted in the earlier ENSURE effort; namely, dealing with the residual rotor thrust developed by the helicopter on the ground operating at flight idle power (that is, rotor speed at 100 percent or less, collective at minimum or greater, and cyclic at any position).[4]

In this effort, an experimental weight and balance system was developed for the CH-47B. Basically, this system was an oleo-pressure-type weight and balance system with provision for residual thrust compensation based on measurement of the strain in the transmission covers produced by rotor lift forces. In tests of the installed system in 1972, static weight measurement accuracy was adequate, but large errors occurred in the rotor lift (strain gage) measurements that were attributed to thermal stresses and extraneous forces in the dynamic condition (that is, production of deformations unrelated to those produced by the lifting force). No further contractual efforts were expended to improve the rotor lift measurement.

One of the reasons for the residual thrust measurement approach pursued in the above effort was the conclusion that estimates of residual thrust based on rotor aerodynamics, an approach implemented in the earlier ENSURE program, cannot be made with sufficient accuracy. Indeed, test results from the earlier program showed that sometimes the aerodynamic estimate produced good results and sometimes it did not, with no clear correlation of the degree of error with any of the test condition variables.

In reviewing the above programs, an apparently important deficiency in the test procedures in both programs was identified. At worst, this deficiency may have led to incorrect conclusions regarding test results;

3 Stuart L. Varner, Design Analysis of Integral Weight and Balance System for Army Cargo Helicopters, USAAVLABS Technical Report 67-50, U.S. Army Aviation Materiel Laboratories, Fort Eustis, Virginia, August 1967, AD 664644.

4 Richard L. Dybvad, Helicopter Gross Weight and Center of Gravity Measurement System, USAAMRDL Technical Report 73-66, Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, August 1973, AD 771955.

at best, it prevents accepting those questionable conclusions (see Weight Measurement, Section 3). It is possible, for example, that the efficacy of the theoretically valid method of aerodynamic estimation of residual lift may have been incorrectly assessed.

The Applied Technology Laboratory has also sponsored efforts to develop a technique for predicting the maximum power available from a helicopter's gas turbine engine prior to liftoff, taking into account the effects of engine performance degradation as well as the effects of ambient conditions.[5, 6] The original objective was to obtain this adjusted estimate of maximum power while the aircraft is in the loading process by measuring the required engine variables while the engine is at a low power level, nominally 30 percent. It was found that it is impossible to obtain sufficient accuracy at that power level. The objective was then modified to include higher power level operation of the engine as might be attained enroute to the load pickup point or in the last flight.

Theoretical performance of the technique for the higher power level measurements appeared good, but in applying the technique to test cell data acquired from engines when they were new (or newly overhauled), and from the same engines after they had been returned from the field at various stages of degradation, the accuracy of the resulting maximum power available estimates for the engines (taken as a whole) was relatively poor, even for measurements taken at 90 percent power. The poor results were attributed primarily to inadequate instrumentation; but additionally, the contractor recommended further analytical investigation of a modification of the original technique that would eliminate, it was thought, error caused by an unforeseen characteristic of the data.

The results of the present study show that the above technique is not requisite to the development of the LPI system.

The Applied Technology Laboratory has participated directly and through sponsorship[7] in the development and evaluation of direct density measurement devices. Two different approaches have been evaluated, both employing

[5] Joseph M. Kos, et al, Feasibility Investigation for Determining Army Helicopter Gas Turbine Engine Maximum Power Available, USAAMRDL Technical Report 72-58, Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, February 1973, AD 758461.

[6] Edward V. Fox, et al, Advanced Feasibility Investigation for Determining Army Helicopter Gas Turbine Engine Maximum Power Available, USAAMRDL Technical Report 74-49, Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, August 1974, AD 786546.

[7] Donald W. Blincow, Nuclear Helicopter Air Density Indicating System Flight Test Program, USAAMRDL Technical Report 74-19, Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, May 1974, AD 786565.

radioactive isotopes. Density of the local atmosphere is sensed in terms of backscatter (in one approach) and absorption (in the other) of radiation. Based on flight tests of experimental units for the two concepts, ATL concluded that better accuracy is afforded by calculation of density from measurements of the parameters that significantly influence it. The findings of this study are that a pressure-temperature model for density produces variations in the resulting calculated performance capabilities of the helicopter no larger than about ± 0.4 percent over the applicable flight envelope, compared to using a perfect density measurement.

Another U.S. Army organization, the Avionics Laboratory of the Army Electronics Command, has been involved for several years in the development of a Helicopter Lift Margin System (HLMS) through its participation in the Joint Army Navy Aircraft Instrumentation Research (JANAIR) Program sponsorship of that development effort.

The approach implemented by the HLMS depends on measuring the effective gross weight of the aircraft. This is obtained by hovering the helicopter out-of-ground effect and converting the power required to sustain the helicopter to an equivalent weight using a nominal lift vs power characteristic for the helicopter. This effective gross weight is stored so that, thereafter, a "lift margin" can be calculated by estimating the maximum power available for the current ambient conditions, converting this to maximum available lift by using the same relationship used in the effective weight calculation, and then subtracting the effective gross weight (that was stored earlier). The resulting lift margin is applicable only to hover out-of-ground effect (HOGE) conditions.

The key aspect of this approach is the weighing maneuver. The effective weight that is obtained corresponds to actual weight to the extent that the maneuver achieves a steady HOGE with zero relative wind and zero rate of climb.

The HLMS program culminated in the flight test of the system on a UH-1M helicopter by the U.S. Army Aviation Engineering Flight Activity at Edwards Air Force Base, California.[8] Based on the evaluation of system operation and test results, it was concluded that (1) addition of a suitable low airspeed measurement to improve the accuracy of the weighing maneuver, and (2) elimination of the requirement to HOGE prior to having lift margin information available would result in an operationally suitable system.

No further development of this system has taken place.

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8 Daumants Belte, et al, Helicopter Lift Margin System and Low Speed Performance Evaluation, USAAEFA Project No. 73-01, U.S. Army Aviation Engineering Flight Activity, Edwards Air Force Base, California, August 1977.

One other Army-sponsored program of note was sponsored by the Land Warfare Laboratory, Aberdeen Proving Ground, Maryland.[9] This study resulted in the development and test of a manually operated, slide-rule-type device for determining power margin for the UH-IH helicopter. The concept that was implemented is a somewhat more sophisticated version of the power margin check that is currently prescribed in the UH-IH Operator's Manual and that is implemented by means of an instruction placard in the UH-IH cockpit. (Review of this procedure, incidentally, shows that it overstates performance capabilities at density altitudes below about 10,000 ft, and that is probably virtually worthless at a place such as Fort Rucker where the geometric altitude is less than 500 ft.)

Commercial and private helicopters share a similar lack of instrumentation in the lift performance area. There are no production gross weight measurement systems on any helicopters. Aerospatiale installs a collective pitch indicator-computer (a manual, slide-rule-type device) on some of its helicopters that functions as a sort of power margin and effective gross weight calculator.

In the fixed-wing area, weight and balance measurement systems have become a standard option for large jet aircraft (e.g., 747, DC-10, L-1011), and hundreds of oleo pressure weight and balance systems have been retrofit to other aircraft (DC-8, DC-9, 707, 737, Vanguard, Falcon 20, Gulfstream II, DH-C5). Presently flying production weight and balance systems include both oleo pressure and strain gage deflection measurement systems.

Compared to weight measurement for fixed-wing aircraft, helicopters do not present any unique design requirements except where skid-type landing gear are used and except for handling residual thrust.

1.2 APPROACH

Earlier efforts at definition and development of the equivalent of an LPI system have been based on the premise that a particular item of information (usually some measure of capability over the requirement connected with HOGE) is a necessary and sufficient indication relative to takeoff and landing capabilities. Having set up what appeared to be an arbitrary criterion, success was then measured in terms of the ability to meet that goal.

In contrast, one of the objectives of this study was to define the requirements of a system that would provide an indication of the potential success of takeoff and landing in advance of liftoff. This objective was pursued by analyzing the various helicopter takeoff and landing modes to identify the key performance capabilities required and the effect of variations in these capabilities on potential takeoff and landing success. This

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[9] E. Kisielowski and E. Fraundorf, Helicopter Payload Capability Indicator, Technical Report No. LWL-CR-02M69, U.S. Army Land Warfare Laboratory, Aberdeen Proving Ground, Maryland, March 1971, AD 723436.

resulted in the definition of basic information that should be furnished to the pilot as well as fundamental accuracy requirements.

It was found that no single performance criterion or measure can convey the capability of the helicopter to execute all of the various takeoff and landing maneuvers. For example, where it is necessary to perform a vertical takeoff, the performance capability in question is vertical climb rate capability (a capability of at least 300 fpm is normally required). In a different situation, a 15-percent power margin at a 5-ft hover might indicate adequate capability.

The result was the definition of requirements as reflected in the system illustration in Figure 1. It was found that the LPI system should be capable of informing the pilot of the vehicle's hovering capabilities (both OGE and IGE), climb capabilities (both vertical and at best airspeed), and several other performance measures--in short, the system should provide essentially the same types of performance information that is contained in the performance section of the operator's manual for the vehicle. Additionally, the system should provide the measured gross weight of the vehicle and the computed c.g. Both of these items are critical to performance capability and both are limited irrespective of the performance capability of the helicopter.

In the recommended approach, lift performance is calculated in two basic steps. First, maximum available power is computed from a relatively simple nominal schedule of normalized power vs ambient temperature that describes the action of the engine controls. This value is adjusted by a simple calibration constant to account for trim variation from the nominal engine power schedule and is multiplied by the measured ambient pressure to obtain actual engine power. This power level is limited by a fixed-value transmission power or torque limitation. Alternative single-engine and normal rated power display modes are obtained by simply using different multiplicative constants.

The power computed in the first step is used in the second step to compute the desired performance capabilities. The basis of these computations is a set of performance characteristics that have been derived from flight tests of a representative aircraft. The typical characteristic consists of a nonlinear curve relating nondimensional parameters and allowing the desired performance capability to be computed based on inputs of power, air density, and vehicle weight.

Vehicle weight is obtained by measuring oleo strut pressure. An antifriction technique is used for maximum accuracy. Other required sensors include ambient pressure and temperature. Measurement of cargo hook loads is also recommended. Engine torque is input to the system for use in approximating fuel used for calculating performance capabilities just before landing.

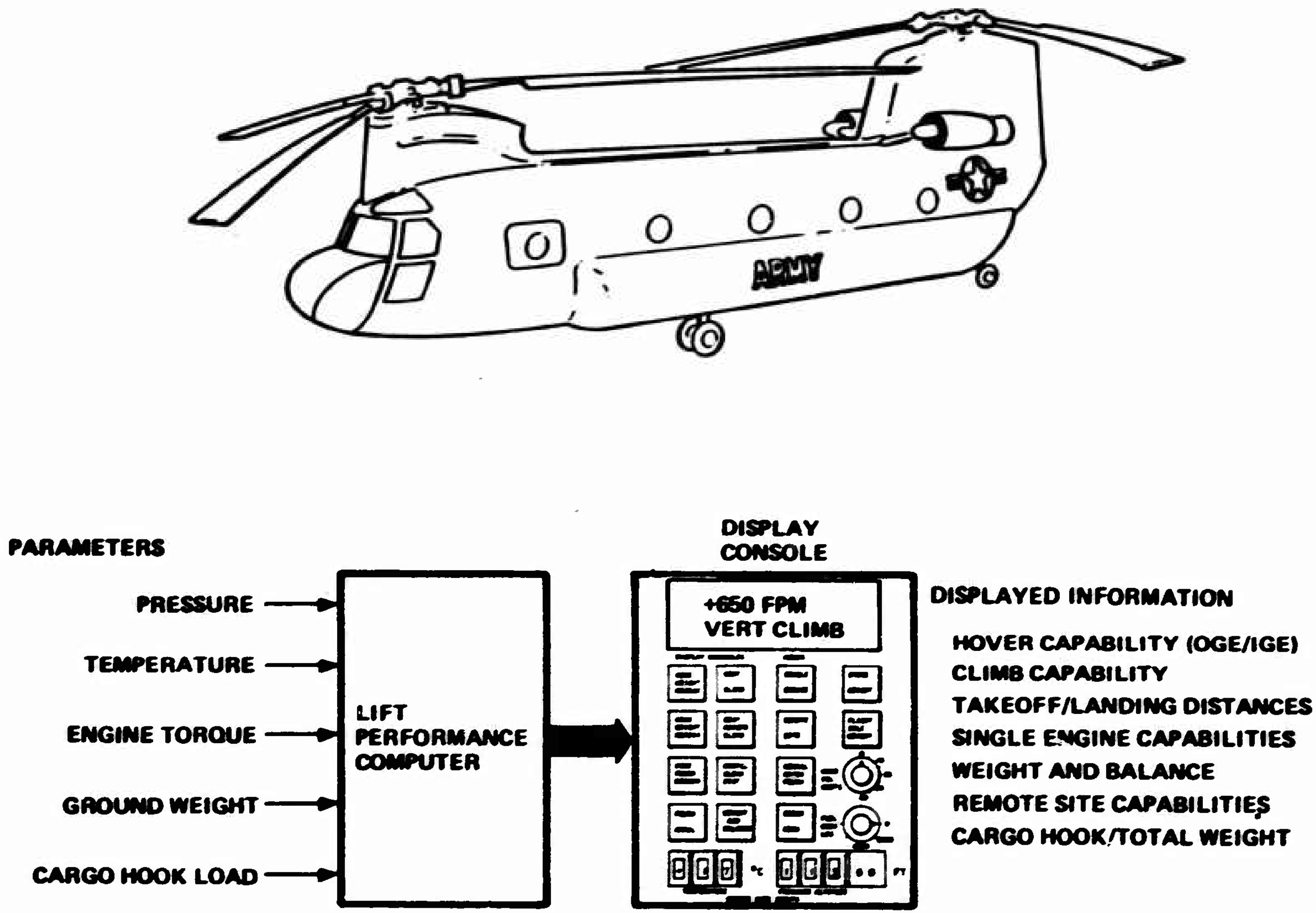

Figure 1. Lift Performance Indicator System Concept.

The lift performance computer would consist of a digital computer synthesized from large scale integrated circuit (LSI) microcomputer set components. The system would be accurate to within approximately 3 percent (in terms of vehicle gross weight), which is considered a minimum acceptable level.

2. FUNCTIONAL REQUIREMENTS

One of the fundamental tasks of this study was to derive realistic functional requirements for a lift performance indicator (LPI) system. This was accomplished primarily by dissecting the various helicopter takeoff and landing modes to identify the key performance capabilities required and the effects of variations in these capabilities on potential takeoff and landing success. This resulted in the definition of basic information that should be furnished to the pilot as well as fundamental accuracy requirements.

Factors affecting helicopter performance capabilities were analyzed to determine measurement and computational requirements. The analytical effort was supplemented by interviews of pilots to gain their perspective and views of the functional requirements of an LPI.

2.1 SUMMARY OF REQUIREMENTS

The intended function of the LPI system is to furnish the pilot with information that will enable him to determine the likelihood of a successful takeoff and landing before committing the helicopter to flight. This objective rules out power margin approaches where the relative capability of the helicopter is estimated while airborne by comparing actual power usage to maximum available power. Actual weight measurement is required.

Approaches considered in the past for this type of system have generally been aimed at supplying a single performance capability index, such as the excess of maximum available lift over vehicle weight for HOGE, termed "lift margin". Use of a single index seems to ignore the diverse modes of takeoff and landing that are performed with the helicopter. With the aid of supplementary charts, a single index could be extended to cover other situations, but this is counter to the objective of the system.

The LPI system should be capable of informing the pilot of the vehicle's hovering capabilities (both OGE and IGE), climb capabilities (both vertical and at best airspeed), and several other performance measures--in short, the system should provide essentially the same types of performance information as contained in the performance section of the operator's manual for the vehicle. Additionally, the system should provide the measured gross weight of the vehicle and the computed c.g. Both of these items are critical to performance capability and both are limited irrespective of the performance capability of the helicopter.

For the purpose of discussing LPI functional requirements, a display and control panel is shown in Figure 2. The panel illustrated contains 13 latching, lit-when-selected pushbuttons for selecting information to be displayed by the system and one pushbutton for storing measured gross weight in the system memory (for later use with weight-off struts).

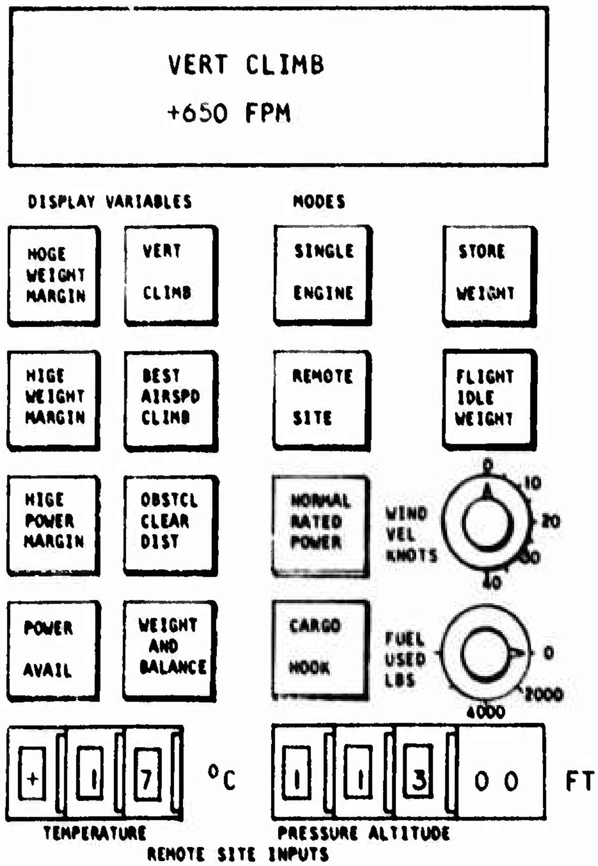

Figure 2. Illustrative LPI Control/Display Panel.

2.1.1 Display Variables

The performance capabilities of the helicopter are computed from its measured gross weight, estimated power capabilities for the ambient temperature and pressure, and stored performance characteristics. The latter includes the entire range of performance characteristics that determine the vehicle's capability to execute the various possible modes of takeoff and landing. A particular capability is selected for display by the pilot by actuating the applicable pushbutton. A selected pushbutton is backlighted and remains in effect until another display variable is selected.

The performance variables or margins selected for display are described below. The type of information that would be displayed is illustrated in Figure 3.

2.1.1.1 HOGE Weight Margin

The HOGE weight margin is equal to the maximum vehicle gross weight that can be supported out-of-ground effect minus the actual, measured weight of the vehicle. A negative weight margin indicates insufficient power available for hovering and also indicates the amount of weight that would have to be removed to achieve hovering capability. HOGE capability is required for nap-of-the-earth (NOE) operation and can also be necessary for vertical takeoffs and landings in confined areas (e.g., a clearing within a forest).

In Figure 3, the illustrative display for HOGE weight margin shows that the helicopter has enough power to support an additional 3173 lb at HOGE (but at zero vertical climb capability).

2.1.1.2 HIGE Weight Margin

The HIGE weight margin is equal to the maximum gross weight that can be supported at the typical hovering height of the helicopter at which the "before takeoff check" is performed (10 ft for the CH-47 and CH-54) minus the actual weight of the helicopter. This margin could be calculated for a variable height input by the pilot, but that appears to be an unnecessary complication. HIGE weight margin is potentially very useful as a criterion for normal and maximum performance takeoffs because it can directly relate excess power requirements to vehicle gross weight, indicating the additional weight that could be added, or the weight that should be removed. If used, it would replace HIGE power margin. The illustrated display shows that an additional 6345 lb could be supported at a 10-ft hover but with zero power reserve.

2.1.1.3 HIGE Power Margin

This is the power equivalent of the preceding HIGE weight margin. This performance characteristic is very attractive for several reasons, (1) it has been in use a long time so it is readily accepted, (2) it can be related to several takeoff mode power requirements (and is widely used for that),

VARIABLE SELECTED	MODE SELECTED	SAMPLE DISPLAY
HOGE WEIGHT MARGIN	(NONE)	HOGE WT MARGIN +3173 LBS
HIGE WEIGHT MARGIN	(NONE)	HIGE WT MARGIN +6345 LBS
HIGE POWER MARGIN	(NONE)	HIGE PWR MARGIN +19 PCT TORQ
POWER AVAIL	(NONE)	MAX TORQ 156/78 +17C 3200 FT
VERT CLIMB	REMOTE SITE	REMOTE 11300 DA VERT CLIMB -322 FPM
BEST AIRSPD CLIMB	SINGLE ENGINE	1 ENG BEST CLIMB -200 FPM 70 KIAS
OBSTCL CLEAR DIST	(NONE)	T/O DISTANCE ZERO FT
WEIGHT AND BALANCE	(NONE)	GW 32470 LBS CG 335 IN.

Figure 3. LPI Display Formats.

and (3) it is the only performance measure that can actually be verified by the pilot before executing a takeoff profile (before entering a hazardous region of the height velocity graph). Thus, the pilot can obtain a fair measure of verification of all the LPI performance estimates by verifying the HIGE power margin. This verification would be accurate to within about ± 5 percent under most conditions--not sufficient to check the accuracy of the system, but sufficient to detect problems. Power required for HIGE is somewhat insensitive to wind velocities below about 20 knots, so the verification would be applicable to the basic zero wind estimates.

2.1.1.4 Power Available

Available power is computed in order to compute performance capabilities. The capability of displaying this variable is desired for two reasons. First, the pilot must presently consult charts in his flight manual in order to determine maximum available power, except insofar as he is able to recognize those combinations of ambient temperature and pressure for which engine power exceeds the transmission limitation (below about 7000 ft density altitude for the CH-47C and CH-54B). For these conditions, the maximum power is equal to the transmission limitation (red-lined on the torquemeters). For a given helicopter, the dual-engine limitations are different from single-engine limitations. Second, in performing the topping check (maximum power check) for the engines, special charts must again be consulted to determine the values of torque that should be obtained. The display of power available would provide the information needed for both uses.

The illustrative display is applicable to the CH-47C. The maximum torque is equal to the transmission limitation of 78 percent for each engine for a total of 15 percent. Also displayed are the ambient temperature and pressure altitude used in the computation. This is merely a convenient mode in which to display the latter data which are used for every display variable. If the single-engine mode had been selected, a single torque value would have been displayed equal to the actual maximum torque available from the engine (for the displayed ambient conditions, approximately 87 percent).

2.1.1.5 Vertical Climb

Vertical climb rate capability is easily computed from HOGE weight margin, and can be estimated in the same way. The criterion applied to vertical climb rate appears to be quite uniform: a minimum capability of 300 fpm climb rate is advised for this mode of takeoff.

The sample display also illustrates the use of the REMOTE SITE mode. In this mode, the temperature and pressure altitude values input by the pilot are used to compute the performance capabilities. The equivalent density altitude is displayed for reference (11,300 ft DA in the figure). The displayed climb capability of -322 fpm shows that if the pilot attempted to HOGE under those conditions, he would sink at a rate of 322 fpm.

2.1.1.6 Best Airspeed Climb

This selection would provide the airspeed at which maximum climb rate is obtained and the magnitude of that rate. In the sample display format, the output for single-engine mode is shown. This particular mode of operation illustrates the possible use of the LPI system in an emergency; namely, engine failure. For this condition, the system informs the pilot of the best airspeed and resulting climb rate to expect for the prevailing ambient conditions. (If the helicopter were at altitude when the failure occurred, the climb rate would improve as the vehicle descended.) The 179th Aviation Company uses this capability as a safe-flight criterion: the single-engine climb rate must be no lower than -500 fpm for the CH-47C. The CH-47C operator's manual specifies that at least a 500 fpm climb rate at best airspeed is required for rolling takeoffs.

2.1.1.7 Obstacle Clearance Distance

This selection would provide the takeoff distances necessary to clear a 50-ft obstacle in the maximum performance takeoff mode. Two distances would be provided: (1) the approximate distance required to achieve rotation speed (distance needed to accelerate to that speed--specified at 23 knots IAS for the CH-54B and 35 to 40 knots for the CH-47C), and (2) the total distance needed to clear the 50-ft obstacle. In the example display, the distance shown is zero feet, the standard method of showing that the helicopter is capable of vertical takeoff.

Obstacle clearance distances are not always included in the performance section of the helicopter operator's manuals (none for the CH-47C). Also, these distances are strong functions of wind velocity, but wind velocity correction data is not normally available. Pilot technique is also important. The display of obstacle clearance distances could therefore prove to be controversial.*

2.1.1.8 Weight and Balance

Weight and balance are important in their own right. Gross weight is limited irrespective of performance capabilities, and c.g. is limited as a function of gross weight. Exceedance of a c.g. limit should probably be indicated in some manner by the display. It may prove desirable to be able to manually input gross weight to the LPI to provide preloading planning capability.

In the CARGO HOOK mode, the measured cargo hook load would also be indicated and the c.g. display logic would be suppressed. (If the c.g. is within limits without the cargo hook load, it will remain within limits as loads are added to the cargo hook.)

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*Obstacle clearance landing distances, where defined, should also be considered for display.

2.1.2 Display Modes

The mode pushbuttons cause the performance capabilities to be computed for special conditions. The mode selections are not mutually exclusive, and one or more can be selected at the same time. A common selection would be single-engine operation for the remote site. If a mode pushbutton is not selected, the basic operating mode is selected by default.

2.1.2.1 Basic Operating Mode

The basic operating mode is as follows:

(1) Two-engine operation (as applicable)

(2) Present measured ambient pressure and temperature

(3) Maximum power

(4) Cargo hook signal locked out

2.1.2.2 Special Operating Modes

The purposes of the special modes are described below.

2.1.2.2.1 Single Engine

Applicable to two-engine helicopters, this mode causes the performance characteristics to be computed for single-engine operation. These characteristics are applicable to emergency operation, and are therefore applicable to in-flight operation as well as flight planning (these are the hash-marked pages in the performance section).

2.1.2.2.2 Remote Site

This mode causes the LPI computer to use the manually selected temperature and pressure altitude values representing conditions at the landing site. Seventy percent of Army helicopter accidents occur during landings. The pilot can also estimate fuel use and input that to the system for the remote-site computation.

2.1.2.2.3 Normal Rated Power

This is the maximum power level that can be used continuously (as opposed to the maximum or takeoff ratings that generally correspond to maximum available power levels that are time-limited). Performance capabilities computed for this power rating cannot be characterized as vital information, yet this information is invariably included in the operator's manual.

2.1.2.2.4 Cargo Hook

Cargo hook load measurements would be locked out unless this mode is activated. The principal consideration is accuracy and display formatting. The cargo hook signal could probably be locked out automatically.

2.1.3 Other Controls and System Functions

2.1.3.1 Remote-Site Inputs

To determine performance capability at a remote site (or at the same site later in the day), the pilot must manually supply the applicable temperature and pressure by means of thumbwheel inputs, for example.

2.1.3.2 Fuel Used

This is also provided for the remote-site computation, the idea being that the present weight minus the estimated amount of fuel to be used would yield the weight at the remote site. The fuel-used dial would be spring-loaded to zero pounds. It might prove convenient to have a more generic capability, either a plus and minus change-of-weight dial or thumbwheel inputs.

2.1.3.3 Automatic Weight Updating

Vehicle gross weight changes as fuel is used. To be able to estimate performance margins just prior to landing, the system integrates fuel flow rate using fuel flow signals (if available) or torque. Torque is proportional to fuel flow and can yield a reasonably accurate estimate of fuel usage for this purpose. Since vehicle weight is measured as the force exerted on the landing struts or axles, initial gross weight must be stored in memory before the helicopter leaves the ground. This is accomplished by means of the store weight control.

2.1.3.4 Store Weight Control

The measured gross weight of the vehicle would be smoothed or filtered to eliminate minor fluctuations for display. This smoothed value would be stored in memory upon actuation of the pushbutton labeled STORE WEIGHT. In the cargo hook mode, actuation of that button would store the current value of the measured cargo hook load.

2.1.3.5 Flight Idle Weight Correction

This is a mode selection provision applicable only to weight measurement. Normally, gross weight would be measured with the engines off. The flight idle mode introduces a correction to account for the thrust of the rotor(s) at the flight idle power level for weight measurement. The method of correction is discussed in Section 3. The correction procedure requires that the pilot position the collective at minimum, maintain rotor rpm at 100 percent, and center the cyclic. This conforms to currently prescribed procedures.

2.1.3.6 Wind Velocity Correction

A dial is shown for manually inserting wind velocity as estimated by the pilot or read from a low airspeed sensor. The dial is spring loaded to zero. Wind velocity affects all the performance variables except best airspeed climb; however, wind velocity correction data is not normally available for all the performance margins, in particular the HIGE margins and obstacle clearance distance. Where wind corrections are not available or policy prevents their use, a message could be displayed to indicate that. In other cases, the pilot would observe the change in capability as he dialed in the wind velocity.

2.1.4 Measurements and Accuracy

Overall LPI accuracy required is to within ±3 percent. This is equivalent to a 3-percent error in the gross weight measurement or a 3-percent error in the computed maximum available lift. Performance margin equivalencies are described later.

Center of gravity is computed from the vehicle weight measurements. Desired c.g. accuracy is to within ±1 in. (±3 in. could be tolerated). This imposes an accuracy requirement on gross weight to within ±1 percent (this is target accuracy, somewhat larger errors could be tolerated).

Required measurements and corresponding maximum allowable errors are listed below:

Measurement	Max. Allowable Error
Gross weight	±1 percent (target)
Ambient temperature	±1°C
Ambient pressure	±0.5 percent
Cargo hook load	±2 percent
Fuel flow*	±5 percent
Pitch angle**	1°C

*If available; otherwise use fuel flow analog (e.g., torque)
**Angle formed by longitudinal reference axis of vehicle with respect to the gravimetric horizontal plane.

The ambient temperature and pressure accuracies are moderate with respect to currently available aircraft instrumentation, but are sufficiently high that they can almost be neglected in a root-of-summed-squares (RSS) summation of errors.

2.1.5 Engine Maximum Power Calibration Input

As discussed in Section 4, all helicopter engine controls provide for adjusting the maximum power output of the engine, i.e., the power developed for maximum throttle input demands. This adjustment is checked when the engine is installed by causing the engine to deliver maximum power (under the condition that transmission limitations are not exceeded). This topping check is performed periodically thereafter, or whenever the controls are suspected of being out of tolerance. Tolerances on the maximum power adjustment range from ±2 to ±4 percent. To avoid this error, a calibration input for the maximum power point is required.

As noted earlier, the LPI display of power available can be used in performing the topping check. The procedure would consist of setting the calibration input for the engine to zero (nominal). The topping check would then be made, using the power available display as a guide. If the engine maximum power is within limits, then the calibration input would be adjusted until the power available display agreed with the actual power (torque) indication. This could be done in real time or following the flight by using the remote-site mode.

2.1.6 Supplementary or Growth Functions

Several additional functions should be considered for the LPI system. These are not required to fulfill the basic system objectives, but represent suitable extensions of system capabilities on the basis of their functional and technical relationship to baseline LPI functions and capabilities. Functions recommended for consideration are described below.

2.1.6.1 Engine Pressure Ratio (EPR) for CH-54B

For the CH-54B, maximum power is controlled by the pilot by adhering to an EPR schedule provided in the operator's manual. Computation of this EPR value, which is a function of ambient temperature and pressure, is obviously suited to the LPI system. This variable would be displayed for the power available selection.

2.1.6.2 Range and Endurance

All of the variables required to compute range and endurance information for the aircraft are available within the LPI system. With the relatively simple addition of the required performance characteristics, range and endurance computation could be added to the LPI system. With this addition, the entire performance section of the operator's manual would be incorporated within the LPI system and would be available to the pilot at the touch of a button--the electronic performance library.

2.1.6.3 Engine Performance Tracking

Engine degradation can produce significant changes in maximum available power. If the changes exceed the tolerances on maximum power, then according to present maintenance practice, the engine controls require adjustment. Techniques for predicting the maximum power available following degradation, based on measurements made at part-power conditions, have been investigated; but the prediction errors are larger than the tolerances prescribed for topping checks. In lieu of an automatic technique, the most reasonable approach is to track engine performance and check maximum power available (topping) whenever significant variations in performance occur. The present Health Inspection Test (HIT) check or the types of operations performed by an Automatic Inspection, Diagnosis, and Prognosis System (AIDAPS) could be incorporated within the LPI system with the addition of the appropriate, already instrumented, engine parameters. This function is not considered critical to implementation of the LPI concept. Methods of establishing and checking maximum available power already exist and appear sufficiently accurate in terms of the procedures and tolerances used to support accurate LPI operation. Moreover, the LPI system would provide outputs that would indicate the need to recheck maximum power. This function is therefore considered a worthwhile growth function. (This topic is discussed further in Section 4.)

2.1.6.4 V_{ne} Computation

With no additional variables, the LPI system could perform the computation of the never-exceed airspeed presently performed with a manual slide-rule-type device by CH-47 pilots. With the accurate gross weight and c.g. available to the LPI, this V_{ne} computation might also be competitive with the cruise-guide-indicator system that is proposed for the CH-47C. (This topic is considered further in Section 2.8.)

2.1.7 Typical Operation of the LPI System

During the course of loading the helicopter, the crew would have the LPI display weight and balance to observe weight and c.g. limitations. At the completion of loading (or earlier), the system would be interrogated for the desired takeoff and landing performance estimates. For these estimates to be made, the store weight pushbutton would be actuated to store the measured weight in the LPI computer memory. Performance variables would then be selected as desired for computation and display.

The particular variables and modes selected would depend on the particular situation (e.g., terrain, vehicle weight, and unit policy). Gross weight, ambient conditions, and a key performance estimate might be recorded in a log. The pilot might also wish to dial in the estimated wind velocity and observe the increase in performance margins, especially if the zero-wind values were marginal.

Estimated performance for the landing zone might also be obtained, especially if the takeoff performance were marginal or if the landing zone were in a confined area or at a higher density altitude, or both. For this estimate, the pilot would dial in the estimated temperature and pressure altitude for the landing zone and select the remote-site mode. Here again, the increased performance available due to headwinds could be observed if the wind velocity at the landing site were known. For a closer estimate of landing performance, the pilot could dial in an estimate of the amount of fuel that will be used before landing. The pilot can observe the effects of inaccurate estimates for all of these variables by varying the appropriate input values (e.g., the reduction in vertical climb capability for a 5-degree rise in ambient temperature).

Initial, before takeoff gross weight is stored in the LPI memory. During the flight, this value is updated for fuel used. Landing performance estimates can then be updated near the point of landing with an accurate measure of gross weight and with current ambient conditions.

During the flight, the LPI system could be set to the single-engine mode to provide emergency information if needed. Other tasks, such a V_{ne} computation and display, could also be considered.

Even in its most basic form, the LPI system is multifunctional. Before the flight, it is a flight planning aid, indicating the takeoff and landing modes that are available on the basis of required performance margins. During cargo loading, it is a real time loading aid, showing the position of vehicle c.g. as loads are positioned and secured. Following loading, it is a safety device, allowing comparison of takeoff and landing performance margins with safe-range criteria.

In performing these basic functions, the system generates information that can be used for several special purposes. Engine topping check information and single-engine emergency information are the prime examples. With the parameters that the system uses and with its computational capabilities, several other functions can be considered for the system as a natural outgrowth.

2.2 HELICOPTER TAKEOFF AND LANDING PERFORMANCE MARGINS

2.2.1 Multivariable Display Capability Needed

In contrast with fixed-wind aircraft, helicopters can take off and land in a variety of ways. As a result, no single performance criterion or measure can convey the capability of the helicopter to execute all of those maneuvers. For example, where it is necessary to perform a vertical takeoff, the performance capability in question is vertical climb rate capability (a capability of at least 300 fpm is normally required). In a different situation, a 15-percent power margin at a 5-ft hover might indicate adequate capability.

One performance margin descriptor could be made to serve, however, since all possible margins are functions of the same variables (ignoring wind and ground effects). For example, if the variable were lift margin for

HOGE for the CH-47, then a margin of 1500 lb might be required for vertical takeoff, whereas -3000 lb might be sufficient for a rolling takeoff. The obvious problem with this approach is the confusion that could be caused by limits that are both variable in magnitude and sign. This would also tend to obscure the physical significance of the limit. Moreover, the relationship between the various modes is altered by the effects of wind and ground proximity.

It appears, therefore, that in order to use a single performance margin for all takeoff and landing modes, it would be necessary to employ supplemental charts and graphs to interpret the display. But this sort of dependence on handbook material, which is inconvenient for cockpit use, is one of the factors that contributes to the need for an LPI system. This is, therefore, an undesirable solution, and it is concluded that the LPI system should have multivariable display capability that will provide directly usable and physically significant performance margin data.

2.2.2 Types of Performance Margins

Several types of performance margins can be used (in at least some instances) to convey the capability of the helicopter to execute a given maneuver. The possibilities are briefly described below.

2.2.2.1 Lift Margin

This is a term of relatively restricted applicability, since it has physical significance only in relation to the hovering capability of the helicopter (OGE or IGE). In that context, it is equal to the maximum weight that could be supported by the helicopter (at maximum power for the given ambient conditions) minus the actual weight of the helicopter.

2.2.2.2 Weight Margin

This measure has somewhat broader applicability and is equal to the maximum gross weight at which a given maneuver could be performed (for maximum power at given ambient conditions) minus the actual weight of the helicopter. With respect to hovering capability, weight margin would be equal to lift margin (given the same source of actual weight). But unlike lift margin, one could speak of the weight margin for a 300 fpm vertical climb.

2.2.2.3 Power Margin

Power margin is the reserve power that would be available when performing a specific maneuver. For example, if a 500 fpm climb at best climb airspeed required 2000 shp and the maximum power available for the current ambient conditions were 2500, then the power margin would be 500 shp. Power margin has broad applicability and is similar to weight margin.

2.2.2.4 Performance Margins in Terms of Absolute Capabilities

The margin can be expressed in terms of the capability in question. For example, in the case of vertical climb capability, the performance margin is directly reflected in the estimate of the absolute capability, because the excess of lift capability over gross weight determines the vertical climb capability. (HOGE weight margin expressed as a percentage of gross weight can be directly converted to vertical climb capability.) Other performance capabilities that are direct performance margins are maximum climb rate at best airspeed, minimum takeoff and landing distances for obstacle clearance, and hover ceilings.

2.2.3 Comparison of Performance Margins

There is no clearly superior single way of expressing performance margins for the various operating modes. Lift margin seems to convey a physically significant quantity with respect to hover capability. Weight margin corresponds closely to the flight planning process where charts in the operator's manual are used to determine the maximum gross weight at which a required capability can be achieved. Power margin most closely corresponds with present procedures in the cockpit and is also attractive because it is one of the few measures that can be easily verified in selected instances. For example, an estimate of the power margin for a 10-ft hover could be checked fairly accurately for most conditions, whereas checking the accuracy of a lift margin estimate would require converting an observed power margin to lift margin.

Power margin is viewed as a necessary display capability that would be used primarily to check the validity of LPI system operation. A typical operating procedure is envisioned in which the pilot obtains, among other measures, the estimated power margin at HIGE (10-ft hover for CH-47 or CH-54) and maximum power capability for the measured ambient conditions. Then, during his hover check just prior to takeoff, he observes the power level of the engines and the corresponding actual power margin. For proper LPI system operation, the observed power margin should agree with that margin predicted by the LPI (which can be displayed at the same time that the pilot makes the hover check). This simple check would provide a test of overall LPI system operation and assurance that the remainder of the LPI variable displays are within tolerance.

Power margin is therefore considered a display variable of general utility. The remainder of the performance margins are considered useful insofar as they relate to the specific takeoff and landing modes.

2.3 CRITERIA FOR ASSURING SUCCESSFUL TAKEOFFS AND LANDINGS

The operator's manual for each helicopter contains nominal procedures for the various possible takeoff and landing modes and conditions. Each manual also includes criteria and guidelines for determining the capability of the helicopter to execute the various maneuvers. Most of this information is provided for flight planning (such as the maximum gross weight that can be supported in HOGE), but some operational guidelines are also provided such as power margin checks.

For the CH-47C and CH-54B, a "Takeoff and Landing Data Card" is published as part of the Operator's and Crew Member's Checklist. In the performance data section of the operator's manuals for those aircraft, the pilot is advised to fill out the data card (i.e., a local reproduction of it) in the course of his analysis of the flight for mission planning. The data card can then be used for reference prior to takeoff and landing.

Figures 4 and 5 reproduce those data cards. They are of interest here insofar as they summarize an official U.S. Army view of information needed in the cockpit for reference prior to takeoff and landing.

Interviews of pilots (Section 2.9) indicate, however, that these data cards are not used. For most missions in the typical aviation unit, where loads do not come close to the maximum capability of the helicopter, ample performance margins are assured. Even when the information is needed, there are several drawbacks that discourage the use of the data card:

(1) It takes considerable time to extract the information from the performance charts.

(2) Changes in ambient conditions can invalidate the results.

(3) Actual gross weight estimates are unreliable and inaccurate.

Figure 6 shows a checklist that is actually used by an operational unit for CH-47 operations. The unit is the 179th Aviation Company and is located at Ft. Carson, Colorado, nearly 6000 ft above sea level. The safety criterion that this unit uses pertains to single-engine emergency operation, for which they wish to limit the sink rate at best climb airspeed to no more than 500 fpm. The use of a single criterion strikes a balance between the quantity of information that would be useful and the quality of information that this based on changeable ambient conditions and unreliable gross weight information. It will be shown later that rate of climb at best airspeed is least sensitive to measure errors, so it represents a good choice from that point-of-view.

To focus further on what is needed, Table 1 presents criteria for assuring successful takeoffs for the various takeoff modes used by Army helicopters. (The information also applies to landing insofar as the types of modes and performance data of interest.) The table lists the general conditions that prompt the use of a particular mode, the nominal procedures followed in executing the takeoff, and the information and nominal limits that can be used to estimate the capability of the helicopter to execute the takeoff. The information listed is a composite of the procedures for the helicopters reviewed in the study.

TM 55-1520-217-CL-2

TAKE OFF AND LANDING DATA CARD

TAKE OFF DATA

NORMAL AND MAXIMUM PERFORMANCE TAKE-OFF:

	TAKE-OFF SITE	LANDING SITE
MAXIMUM GROSS WT	______ LBS	______ LBS
INDICATED AIRSPEED	______ KNS	______ KNS
ACCELERATING DISTANCE	______ FT	______ FT
CLEAR 50 FOOT OBSTACLE	______ FT	______ FT

ROLLING TAKE-OFF:

MAXIMUM GROSS WT	______ LBS	______ LBS
INDICATED AIRSPEED	______ KNS	______ KNS
ACCELERATION DISTANCE	______ FT	______ FT
CLEAR 50 FOOT OBSTACLE	______ FT	______ FT

VERTICAL TAKE-OFF:

MAXIMUM GROSS WT	______ LBS	______ LBS
MAXIMUM RATE-OF CLIMB	______ FPM	______ FPM

LANDING DATA

APPROACH TO HOVER AND VERTICAL LANDING:

APPROACH IAS	______ KNS	______ KNS
HOVER ALTITUDE	______ FT	______ FT

ROLLING LANDING:

APPROACH IAS	______ KNS	______ KNS
DECELERATE ALTITUDE	______ FT	______ FT
MAXIMUM GROUND SPEED	______ KNS	______ KNS

(FRONT SIDE)

TM 55-1520-217-CL-2

CONDITIONS

	TAKE-OFF SITE	LANDING SITE
DENSITY ALTITUDE	______ FT	______ FT
PRESSURE ALTITUDE	______ FT	______ FT
TEMPERATURE	______ °C	______ °C
DEW POINT	______ °C	______ °C
WIND DIRECTION	____ / ____ KNS	DIRECTION ____ / ____ KNS
RUNWAY HEADINGS	____ / ____ FT	HEADING ____ / ____ FT
GROSS WEIGHT (TAKE-OFF)	______ LBS	(LANDING) ______ LBS
GROSS WEIGHT (TAKE-OFF)	______ LBS	(LANDING) ______ LBS
CG (FWD) ____ IN (AFT)	____ INS	(FWD) ____ IN (AFT) ____ INS
CG (FWD) ____ IN (AFT)	____ INS	(FWD) ____ IN (AFT) ____ INS
CG LIMITS (TAKE-OFF)	______ INS	TO ______ IN
CG LIMITS (LANDING)	______ INS	TO ______ IN

EMERGENCY OPERATION (ONE ENGINE)

MAXIMUM GROSS WEIGHT FOR HOVERING ______ LBS

BEST RATE OF CLIMB AT GROSS WEIGHT ______ FPM

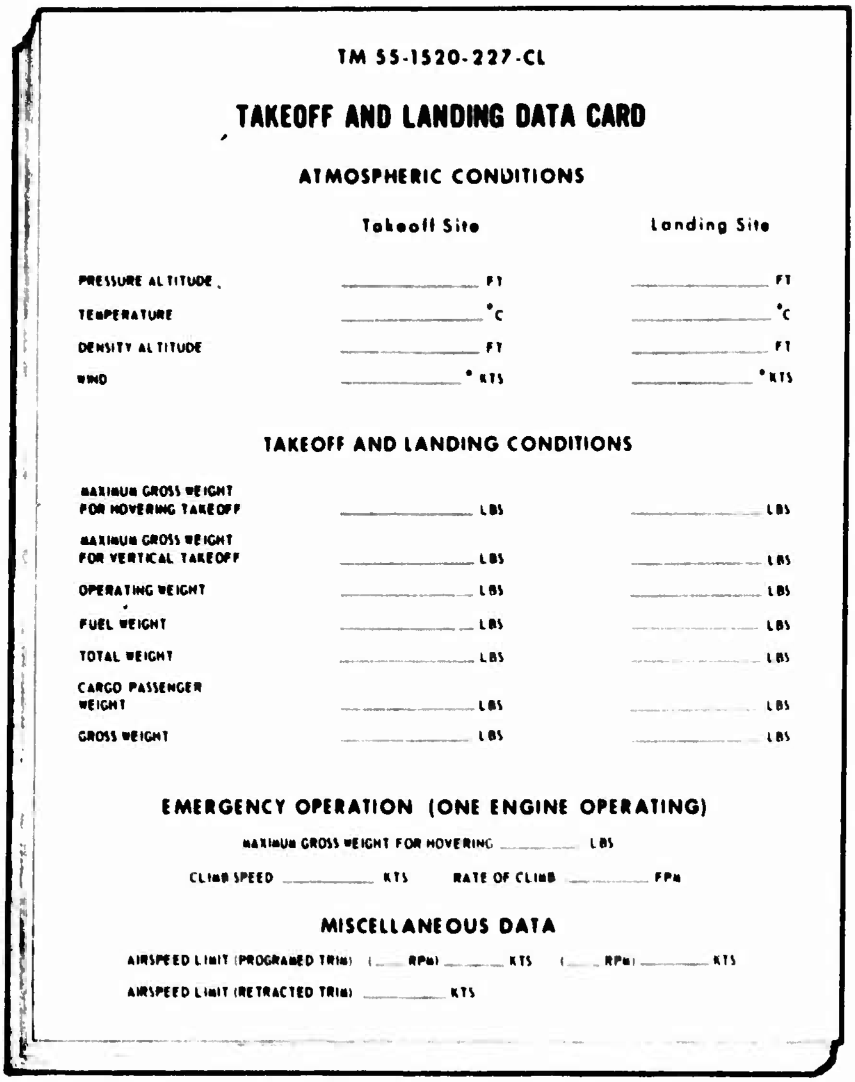
SINGLE ENGINE AIRSPEED ______ KNS

REMARKS:

(REAR SIDE)

SPA 4628-20

Figure 4. Takeoff and Landing Data Card for CH-54B.



TM 55-1520-227-CL

TAKEOFF AND LANDING DATA CARD

ATMOSPHERIC CONDITIONS

	Takeoff Site	Landing Site
PRESSURE ALTITUDE	______ FT	______ FT
TEMPERATURE	______ °C	______ °C
DENSITY ALTITUDE	______ FT	______ FT
WIND	______ ° KTS	______ ° KTS

TAKEOFF AND LANDING CONDITIONS

MAXIMUM GROSS WEIGHT FOR HOVERING TAKEOFF	______ LBS	______ LBS
MAXIMUM GROSS WEIGHT FOR VERTICAL TAKEOFF	______ LBS	______ LBS
OPERATING WEIGHT	______ LBS	______ LBS
FUEL WEIGHT	______ LBS	______ LBS
TOTAL WEIGHT	______ LBS	______ LBS
CARGO PASSENGER WEIGHT	______ LBS	______ LBS
GROSS WEIGHT	______ LBS	______ LBS

EMERGENCY OPERATION (ONE ENGINE OPERATING)

MAXIMUM GROSS WEIGHT FOR HOVERING ______ LBS

CLIMB SPEED ______ KTS RATE OF CLIMB ______ FPM

MISCELLANEOUS DATA

AIRSPEED LIMIT (PROGRAMED TRIM) (___ RPM) ______ KTS (___ RPM) ______ KTS

AIRSPEED LIMIT (RETRACTED TRIM) ______ KTS

Figure 5. Takeoff and Landing Data Card for CH-47C.

AIRCRAFT PERFORMANCE CHECKLIST

	GROUND	CRUISE
PRESSURE ALT	______	______
TEMPERATURE	______	______
DENSITY ALT	______	______

CORRECTED NI (L-7C) NO. 1 ____ NO. 2 ____

GROSS WEIGHT ______ LBS

VNE AT CRUISE ALTITUDE:

PROGRAMED ____ KTS RETRACTED ____ KTS

"SINGLE ENGINE OPERATION"

BEST CLIMB SPEED ______ KTS

RATE OF CLIMB ______ FPM

SPA 4628-30

GO/NO-GO CRITERION:

SINK RATE FOR SINGLE ENGINE OPERATION AT BEST CLIMB SPEED NO GREATER THAN 500 FPM

(RATE OF CLIMB > -500 FPM)

Figure 6. Aircraft Performance Checklist Used by 179th Aviation Company (Applicable to the CH-47C).

TABLE 1

CRITERIA FOR ASSURING SUCCESSFUL TAKEOFFS

Mode and Conditions	Standard Procedure	Performance Data and Criteria
Vertical takeoff: confined area—no room for horizontal acceleration	Vertical climb until obstacles cleared. Transition to forward flight—accelerate horizontally, maintaining altitude, until transitional lift attained; then begin climb.	Vertical climb capability. (300 fpm climb capability)
Obstacle clearance—max performance takeoff: confined area	5-ft hover (CH-47C; 10 ft for CH-54). Accelerate horizontally until 35 to 40 knots. Begin climb maintaining airspeed until clearance; then obtain best climb speed.	Power margin for 5-ft or 10-ft hover. Distance to clear 50-ft obstacle. (15 percent power margin, adequate distance)
Obstacle clearance—constant angle takeoff: confined area	Initiate takeoff from ground or hover. Control adjustments as necessary to maintain a flight path along line from takeoff point to above obstacle.	Power margin for 5-ft or 10-ft hover. (Margin at least as great as for maximum performance takeoff)
Rolling takeoff: weight too great for hovering takeoff	Apply collective to accelerate during ground roll. At 20 to 30 knots, increase collective: lift off. Continue acceleration to attain best climb airspeed; then adjust to climb attitude.	Climb capability at best airspeed. Distance to clear 50-ft obstacle. (500 fpm climb capability) (adequate distance)
Normal hovering takeoff: unconfined area, typical loads	Establish 10-ft hover (CH-47, CH-54). Accelerate horizontally until translational lift is obtained; then begin to climb. Continue to increase airspeed and altitude simultaneously until desired quantities attained.	Power margin for 10-ft hover. (7 percent power margin)
Instrument takeoff: rain, snow, dust, etc., obscures visual references	Initiate from ground or hover. Engine power at 15 percent greater than hover power. Leave ground with positive rate of ascent. Maintain "wings level" until 30 to 40 knots; then transition to coordinated flight. (Similar procedure for "forward flight" takeoff.)	Power margin for 10-ft hover. (15 percent power margin)

Table 1 and the takeoff and landing checklists clearly indicate the types of information that should be furnished to the pilot--hover capabilities, climb capabilities, obstacle clearance distances, and power margins. Required accuracy is examined next.

2.4 ACCURACY REQUIREMENTS

None of the previous efforts described in Section 1 include any derivation of the accuracy that should be required of a weight and balance or LPI system. In most cases, however, a target or "required" accuracy of ± 1 percent is quoted. Usage of the 1 percent number appears to be a rule of thumb; if accuracy is within 1 percent, it is probably more than adequate; if it is not within 1 percent, then someone will have to determine if it is sufficient.

In this study, an attempt was made to derive a realistic accuracy requirement for the LPI system. LPI performance was considered in its relation to safety and cargo transport efficiency. In considering safety, the relation between LPI error and predicted performance in the various takeoff and landing modes was derived. The results show that errors as large as about ± 3 percent* can be tolerated without making special allowances such as increasing required performance margins.

In considering cargo transport efficiency, the approach taken was to assume that a cargo allowance would be made equivalent to the maximum possible LPI error. The results show that the present 37 percent efficiency level can be raised to 95 percent for LPI accuracy to within 3 percent.

Finally, the accuracy required for the c.g. computation, based on c.g. limits, was used to set an independent requirement on weight measurement accuracy. A required c.g. accuracy between ± 1 and ± 3 in. is recommended for helicopters in the CH-47 and CH-54 class. To achieve this accuracy, gross weight measurement accuracy to within ± 1 to ± 3 percent is required. A target accuracy of ± 1 percent is recommended.

It is concluded that LPI system accuracy to within ± 3 percent is consistent with application objectives and that gross weight measurement accuracy to within ± 1 percent is desirable, although degraded accuracy to ± 3 percent may be acceptable.

2.4.1 Accuracy Levels Commensurate With Takeoff and Landing Performance Margins

Each of the various takeoff and landing modes of the helicopter has, in general, a designated performance margin that is a criterion for the adequacy of performance capability relative to that mode. For example, for the instrument takeoff mode for the CH-47C, the operator's manual specifies

*All errors and tolerances described here are equivalent to percentages of vehicle maximum gross weight, unless stated otherwise.

a 15-percent power margin at a 10-ft hover for executing that takeoff. Similarly, the equivalent of a 15-percent power margin at a 2-ft hover is specified for confined area takeoffs of the UH-1H, whereas only about a 7-percent margin is designated for a normal takeoff. Errors in the LPI system can be related to the above performance margins to determine accuracy levels that are commensurate with those margins.

Table 2 lists the effect of errors in the LPI system on resulting performance capability estimates. For example, the vertical rate of climb capability of the helicopter is calculated from maximum power available for the measured ambient conditions, the measured gross weight of the vehicle, and the vertical rate of climb performance characteristics of the vehicle. If the measured gross weight were in error by 1 percent, then the calculated vertical rate of climb would be in error by 70 to 100 fpm, depending on the helicopter type. (The tabulated effects were developed from data for the UH-1, CH-47, and CH-54.) The last two entries in the table are for reference only and show the 1 percent values of takeoff weights of the study helicopters in pounds, and the changes in lift capability produced by various wind velocities.

In the basic lift performance calculation procedure, performance capability is calculated from measurements of vehicle gross weight and ambient pressure and temperature. In Table 2, the effects of LPI errors are tabulated for a 1 percent error in weight measurement, or its equivalent. The relative effects of ambient temperature and pressure errors vs gross weight errors are given below:

	$\frac{\Delta P}{P}$	$\frac{\Delta T}{P}$	$\frac{\Delta GW}{GW}$
In-torque limited regime	1/3	-1/3	1.0
Out-of-torque limited regime	1.0	-1.1	1.0

where P = ambient pressure
T = ambient temperature (in absolute units)
GW = measured gross weight

In the torque-limited regime (transmission torque limit), maximum available power is fixed at the transmission torque limit and is, therefore, not a function of ambient conditions. The result is that in this regime, errors in pressure and temperature have a relatively small effect on the performance estimate. The above tabulation shows that a 1-percent error in pressure has only one-third the effect of a 1-percent error in the weight measurement in that regime.

Outside that regime, where maximum regime power is less than the transmission limit, maximum available power is a function of ambient conditions and a 1-percent error in pressure or temperature is approximately equivalent to a 1-percent error in gross weight. For the CH-47C and

TABLE 2

EFFECTS OF LPI ERROR ON PERFORMANCE ESTIMATES

Performance Variable	Effect or Equivalent of 1 Percent LPI Error*
Vertical rate of climb	70 to 100 fpm
Rate of climb at best airspeed	40 to 55 fpm
Single-engine rate of climb at best airspeed	25 to 40 fpm
Distance to clear 50-ft obstacle (max performance takeoff)	1.6 to 3.2 percent change in distance
HOGE capability	1 percent change in weight capability
HOGE ceiling capability	620 to 850 ft change below 7000 ft density altitude, 290 ft change above 7000 ft density altitude
HIGE capability	0.6 to 1.3 ft change in height capability at 5 ft
Power margin	1.5 percent change (for 1 percent weight error)
Takeoff weight (maximum)	UH-1H: 95 lb; UTTAS: 200 lb; CH-47C: 460 lb; CH-54B: 470 lb
Wind velocities for 1 percent, 5 percent, and 10 percent changes in lift capability (for HOGE)	UH-1H: 5, 13, and 20 knots CH-54B: 7, 15, and 20 knots CH-47C: 8, 19, and 28 knots

---

* 1 percent error in weight measurement or in estimation of basic lift capability.

CH-54B, the torque-limited range extends up to about 7000-ft density altitude, so most of the time temperature and pressure errors have a relatively small effect.

Table 3 lists LPI accuracy levels that appear consistent with the performance margins designated for the various takeoff and landing modes. The first column in the table lists the performance capability and the typical performance margin prescribed for that mode (based on procedures for the study aircraft). The second column lists the LPI error equivalent to the performance margin. Finally, the third column lists a suggested maximum allowable LPI error that appears consistent with the extent to which the performance margin could vary without compromising safety.

The most stringent requirement appears to be the vertical takeoff and landing modes. The normal minimum performance margin standard for vertical takeoff is 300 fpm climb capability, and 100 fpm is defined as the overload limit (i.e., if a vertical climb rate of 100 fpm cannot be achieved, the aircraft is overloaded for that mode). This implies a tolerance of 200 fpm which translates to an LPI tolerance of ±2 to ±3 percent, depending on the specific helicopter.

Also listed in Table 3 is the single-engine sink rate limit used by the 179th Aviation Company as a takeoff criterion for the CH-47C and a corresponding suggested LPI tolerance based on that limit. The sink rate criterion appears to be matched to the landing gear impact limitation of sink rate no greater than about 450 fpm (this limit approximates the limits for all the study helicopters). Here again, vertical landings impose the more stringent requirement, a maximum LPI tolerance of ±2 to ±3 percent appearing consistent with the condition.

Reviewing the tolerances developed in Table 3, it appears that the LPI system could be used without making allowance for error if the total system error were limited to approximately ±2 to ±3 percent. With larger errors (or with a more conservative approach), some allowance for error could be required in using the LPI system to determine loading capability. In other words, some degree of cargo carrying capability, or efficiency would be sacrificed to compensate for LPI errors in order that safety would not be compromised.

2.4.2 Maximum Utilization of Helicopter Cargo Capacity

Applied Technology Laboratory analysis of statistics on U.S. Army helicopter cargo operations shows that the average helicopter cargo load weighs about 37 percent of the helicopter's maximum payload weight capacity. Installation of lift performance indicators will not necessarily improve cargo carrying efficiency, but one can derive the degree of accuracy that would be required in the indicator system to support a certain level of efficiency. This is done below.

TABLE 3

LPI ACCURACY LEVELS COMMENSURATE WITH TAKEOFF AND LANDING PERFORMANCE MARGINS

Performance Capability and Nominal Margin	Equivalent LPI Error*	Suggested Max Tolerances: Margin and LPI Error**
Normal hovering takeoff, 7 percent power margin	7 percent (power available)	±3.4 percent power margin ±2.5 percent LPI error
Obstacle clearance takeoff, 15 percent power margin	15 percent (power available)	±4.5 percent power margin*** ±3 percent LPI error
Instrument takeoff, 15 percent power margin	15 percent (power available)	±4.5 percent power margin*** ±3 percent LPI error
Rolling takeoff, 500 fpm climb	1.8 to 2.5 percent per 100 fpm (climb at best airspeed)	±120 to 165 fpm ±3 percent LPI error
Vertical takeoff, 300 fpm climb	1 to 1.5 percent per 100 fpm	±200 fpm ±2 to ±3 percent LPI error
One engine climb rate, 500 fpm sink rate limit****	2.5 to 4 percent per 100 fpm (climb at best airspeed)	±100 fpm ±2.5 to 4 percent LPI error
Landing gear limitations:	ground contact at sink rate <450 fpm	
Vertical landing	1 to 1.5 percent per 100 fpm	±200 fpm ±2 to ±3 percent LPI error
Normal landing	1.8 to 2.5 percent per 100 fpm	±200 fpm ±3.6 to ±5 percent LPI error
GCA landing mode:	500 fpm rate of climb desired for missed approach (see rolling takeoff)	

NOTES:

* Where LPI error considered as gross weight measurement error or error in estimation of basic lift capability, unless otherwise indicated.

** LPI error expressed as percent of gross weight.

*** Equivalent to 5- to 10-percent increase in distance to clear 50-ft obstacle.

**** Single engine failure emergency. Sink rate limit is used as takeoff criterion by 179th Aviation Co., USA.

If the LPI system were used as the basis for loading the helicopter, then it would be possible to assure safe operation by allowing for the maximum possible error in the indication. For example, if the maximum error possible were 500 lb, then the safe maximum load would equal the maximum load capacity minus 500 lb. Figure 7 summarizes this approach and defines an effectiveness factor equal to the percentage of the maximum load that could be carried after allowing for LPI errors. Maximum payloads are listed for the helicopters, and the effectiveness factors for the helicopters are plotted as functions of LPI error in percent.

Under these ground rules, it can be seen that the 37 percent loading could be raised to 95 percent with LPI errors limited to no greater than about 3 percent. Achieving high levels of cargo carrying effectiveness does not appear to impose a severe accuracy requirement.

2.4.3 Weight Measurement Accuracy Required for C.g. Calculation

The general form of the calculation of the longitudinal location of the vehicle c.g. is

$$X_{cg} = \frac{X_1(W_1 + W_2) + X_2(W_3 + W_4)}{W_1 + W_2 + W_3 + W_4} \qquad (1)$$

where X_{cg} is the location of the c.g. (from the longitudinal reference point), X_1 and X_2 are the distances to the forward and rear struts, W_1 and W_2 are the measured weights on the forward struts, and W_3 and W_4 are the measured weights on the rear struts.

Ignoring the independent effect of ground slope on the c.g. measurement, errors in c.g. are obviously related to errors in the separate weight measurements. If a distribution of errors in the individual measurements is assumed, then c.g. error can be related directly to gross weight error. This has been done for the CH-47 and CH-54 as shown below:

Error Distribution	C.g. Error/GW Error (in./percent)
CH-47C	
All error in main (fwd) struts	0.75
All error in rear struts	1.95
Equal percentage errors	1.0 to 1.15
CH-54B	
All error in main (rear) struts	0.55
All error in nose strut	2.4
Equal percentage errors	0.8 to 1.05

MAXIMUM PAYLOAD POTENTIAL

= MAXIMUM GROSS WEIGHT

- OPERATING WEIGHT

- MINIMUM FUEL LOAD

ACHIEVABLE PAYLOAD POTENTIAL

= MAXIMUM PAYLOAD POTENTIAL

- LPI ERROR

LPI EFFECTIVENESS

$$\frac{\text{ACHIEVABLE}}{\text{MAXIMUM}} \times 100 \text{ (PERCENT)}$$

	MAX. PAYLOAD	1 PCT. ERROR
UH-1H	3,720 LB	95 LB
UTTAS	9,500 LB	220 LB
CH-47C	23,300 LB	460 LB
CH-54B	25,000 LB	470 LB

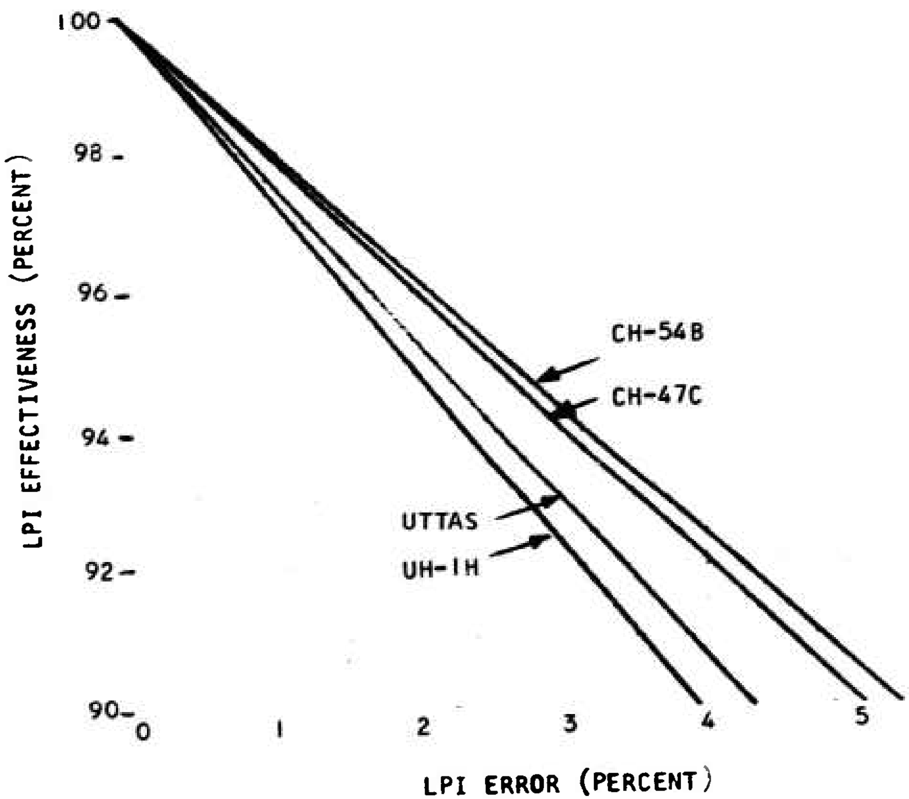


Figure 7. Payload Capability vs LPI Accuracy.

The equal percentage error distribution is most likely, so the c.g. error sensitivity is taken as 1.0 in. per percent error in gross weight.

The c.g. limit ranges at minimum weight for the CH-47C and CH-54B are 48 in. and 26 in., respectively. At maximum weight, the ranges are 15 in. and 18 in. Maximum errors in c.g. beyond about 7-1/2 and 9 in. (respectively) at maximum weight conditions appear to make the measurement of questionable value because one could not be absolutely sure that the c.g. was within limits. If the errors were distributed normally and the actual c.g. locations were distributed uniformly, then 13 percent of the c.g. locations actually within limits would appear out of limits, and 13 percent of the out-of-limit locations within 3σ of the c.g. limit boundaries would appear within limits. Thus, although this degree of accuracy is not as bad as it might first appear, it is still probably intolerable.

It appears that an accuracy on the order of ± 3 in. is tolerable. This would cause about 4 percent of acceptable c.g. locations to appear unacceptable. Viewed another way, the limit could be contracted an inch on either side to restrict exceedances of the original limit to no more than about 1 in.

A target accuracy of ± 1 in. is recommended, at maximum weight with degraded accuracy at lower weight levels consistent with the expansion of the allowable c.g. range, as illustrated below for the CH-47C and CH-54B.

	Weight Range (lb)	Allowable C.g. Location (in.)	Range (in.)	Target Accuracy (in.)
CH-47C	< 28,550	301 - 349	48	3.2
	33,000	310 - 338	28	1.9
	44,800	319 - 336	17	1.1
	≥46,000	320 - 335	15	1.0
CH-54B	<30,000	323 - 349	26	1.5
	38,000	326 - 346	20	1.1
	>42,000	328 - 346	18	1.0

For equal weight measurement error distribution (the most likely distribution), c.g. error sensitivity is approximately 1 in. per percent error in gross weight. Therefore, based on required c.g. accuracy, the commensurate gross weight accuracy required is to within 1 to 3 percent, with 1 percent a desirable target accuracy.

2.5 LIMITATIONS AND CONSTRAINTS

Significant limitations and constraints applicable to the use of the LPI are identified here. In general, constraints are factors such as geographic location of an aviation unit that affect the potential value of the information provided by the LPI system. These factors do not have a direct influence on the design of the LPI, but they could influence Army policy regarding implementation.

Limitations are factors (other than technological) that affect the accuracy to which helicopter performance capability can be estimated or that affect the extent to which the performance capability is effectively utilized. Effective utilization is principally a function of pilot technique.

The limitations and constraints, although significant, do not compromise the potential utility of the LPI system concept.

2.5.1 Constraints Imposed by Mission, Terrain, and Policy

The LPI system will be most useful under the following conditions:

Mission--long range, heavy cargo

Region--high density altitude (e.g., high desert)

Takeoff and landing sites--confined area (e.g., mountainous terrain)

Unit policy--maximum load per LPI

Mission determines the type of load, and the type of load determines the need for LPI, at least to a certain extent. Cargo loads, particularly internal cargo loads, present a more challenging situation for the helicopter pilot for the following reasons:

(1) Gross weight and c.g. can be estimated more accurately for personnel or armament than for cargo.

(2) Personnel loads are standard and require only superficial weight and balance checking.

(3) Personnel loads can exceed the gross weight capabilities of the vehicle only at extreme density altitudes (e.g., CH-47C two-engine HOGE capability equals maximum gross weight with troop load at about 15,000-ft density altitude; with one engine out, sink rate is within acceptable limits up to about 15,000-ft density altitude).

The LPI system will be of more value in regions of high density altitudes because of the diminished performance margins available. For example, there is a 10,000-lb difference between HOGE capabilities at sea level and 12,000 ft density altitudes for the CH-47C.

Local terrain conditions also exert a strong influence on performance capability because they determine the takeoff or landing modes that can be used. For example, with no operational restrictions, the CH-47C can take off at its top gross weight up to a density altitude of a little more than 8000 ft, where some 6000 to 7000 lb would have to be removed to perform a vertical takeoff.

Aviation unit or Army policy could obviously delimit the usefulness of the LPI system. The average cargo helicopter mission is performed at 37 percent load capacity. This means that at sea level standard conditions, the average CH-47C is operating with a HOGE lift margin of about 14,000 lb.

2.5.2 Limitations due to Pilot Technique

Pilot actions, broadly considered here as technique, can influence the maximum available capabilities of the helicopter and the extent to which those capabilities are effectively utilized. The most significant examples are:

Obstacle Clearance Takeoff--Poor technique can double distance to clear obstacles.

Wind Effects--Orientation of vehicle in wind is important during transition and in obstacle clearance takeoffs.

Takeoff and Landing Modes--Choice of mode, where possible, can produce large differences in performance margin.

Vehicle Loading--At the extremes are inefficiency and hazardous operation.

Maximum Available Power--For most conditions, maximum power is under pilot control--± 5 percent tolerance with respect to nominal maximum power estimated.

Pilots vary in their ability to make maximum use of performance capabilities. For example, Schmitz[10] and others have shown that in obstacle clearance takeoffs, wide variations in performance (distance to clear 50-ft obstacle) occur with variations in technique. Distances nearly double with moderate changes from optimal vehicle height above the ground during horizontal acceleration and with changes in airspeed at which rotation occurs. (The distances to clear 50 ft obstacle range typically from near zero--at which vertical takeoff is possible--to a maximum of about a quarter mile.)

The heavy dependence of obstacle clearance distance on pilot technique might make the LPI display of this variable controversial. Wind also has a

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10 Frederic H. Schmitz and C. Rande Vause, "Near-Optimal Takeoff Policy for Heavily Loaded Helicopters Exiting from Confined Areas," J. Aircraft, May 1976.

dramatic effect on this variable. In the applicable situation, takeoff distance is an important determinant of potential takeoff success; for that reason, evaluation of its display in any test version of an LPI system is warranted.

Pilot technique can cause significant variations in performance with respect to wind effects. Again, the largest variation possible is in obstacle clearance takeoffs. A moderate head wind can halve the zero-wind takeoff distance, but the same magnitude tail wind can double it; so orientation of the vehicle into the wind is critical for this type of takeoff or the similar landing mode. For hovering and vertical takeoffs, vehicle orientation is less critical until transition. For the hovering helicopter, wind always aids lift no matter what the orientation; however, the effect is not uniform with azimuth. A CH-47C pilot stated, for example, that in a demonstration of the maximum lift capability of the CH-47 to Brazilian Air Force officers, he was unable to take off until orienting the helicopter so that the wind was at a 45° azimuth.

Pilots can profoundly influence takeoff and landing success by their choice of basic modes and by the way they load their vehicles. The choice of modes is largely constrained by external factors, and loading is probably limited by policy and practice. Notwithstanding these constraints, the LPI should influence these processes by providing quantitative descriptions of the differences in modes and quantitative loading information. The LPI should provide a basis for pushing overly conservative pilots toward higher efficiency, and it can provide a quantitative restraint for overly enthusiastic pilots.

The pilot also influences the maximum power available in the regime where transmission torque limitations are applicable. Variations up to about ± 5 percent with respect to the nominal power available in the torque-limited regime can occur due to torque measurement and display accuracy, actual rotor rpm set by the pilot, and the precision with which the pilot observes the torque limitation.

All helicopter engines are derated for operation at low density altitudes. For example, at density altitudes below approximately 7000 ft, the engines of the CH-47C and CH-54B can provide power in excess of power train limitations. Observance of these limitations is solely a pilot function. The limitation is imposed as a fixed torque limit (for example, 50 psi torque pressure for the UH-1, 78 percent torque per engine for dual-engine operation of the CH-47C).* How well pilots adhere to these limitations is unknown. How to treat this particular determinant of maximum available

*At sea level standard conditions, the CH-47C engines can deliver 98 percent torque, a 25 percent increase over the 78 percent torque limit. The engine power capability drops off with altitude and is equal to 78 percent at about 8000 ft.

power--which may be the limiting factor 70 to 100 percent of the time depending on the location of the aviation unit--could be a tough philosophical question, but it is avoided here by accepting the torque limitation at face value.

The accuracy of torque indication that the pilot is presumed to monitor is no better than about ±2 percent (±5 percent for the UH-1). Variations in rotor rpm also enter the process, since the rotor thrust is a function of power (torque times rotational frequency). The allowable variations in rotor rpm typically span about 5 percent or more, but pilots tend to operate in the high end of the range with variations probably no greater than ±1 to ±2 percent.

Where maximum available engine power is less than the transmission limitation, maximum available power is controlled exclusively by the engine controls for most helicopters (including UH-1H, CH-47C, and UTTAS). In this regime, rotor rpm variations have very little effect due to the power extraction characteristics of the free turbine: a 4 or 5 percent variation in rotor rpm would change the power input to the rotor by only about 1/2 percent. The power plant in the CH-54B is an exception. There the pilot also controls the maximum power available by adhering to an engine pressure ratio (EPR) schedule provided in the operator's manual. (An LPI for the CH-54B should provide the EPR value as an auxiliary function.) So in the case of the CH-54B, the pilot controls the maximum available power for all conditions.

2.5.3 Other Limitations

Other factors imposing limitations are (1) measurement and/or prediction of effective wind velocity, (2) prediction of lift performance for external cargo loads, and (3) variations in engine maximum power due to engine performance degradation and engine control drift or malfunction.

With state-of-the-art techniques, wind velocity can be measured on the helicopter to within about ±5 knots, and effective accuracy will be further dependent on variations in wind with time and position. Additionally, it is not always possible to orient the vehicle into the wind during takeoff or landing, especially in confined areas. Thus, although wind can significantly boost performance under certain conditions, it is likely to be treated conservatively in policy and in practice; that is, much as it is now.

External cargo loads cannot be weighed by the LPI system while the helicopter is on the ground, and pilots explain that the estimated weights of these loads provided by ground personnel can be very inaccurate, with some errors exceeding 5 percent of vehicle gross weight. Cargo hook load instrumentation, therefore, is required to be used in conjunction with the stored pre-takeoff weight of the helicopter measured by the LPI system while the helicopter is on the ground. The final decision as to whether to commit the aircraft to flight must then be delayed until acquisition of the external load and measurement of its weight by the cargo hook load sensor. (Note that advanced versions of the CH-47 and future large cargo helicopters are being designed with tandem cargo hooks for carrying loads such as the MILVAN container.)

Variations in scheduled engine maximum power can occur due to engine performance degradation and engine control drift or malfunction or maladjustments. This topic is discussed in detail in Section 4. The principal limitation arising from this area is that there is no way to ensure that the maximum power of the helicopter turboshaft engine is within tolerance except by actually operating it at its maximum power level. It is generally possible to determine that an engine has suffered no significant performance deterioration by checking its operation at lower power levels, but it is not possible to verify correct action of the engine controls at maximum power without actually operating the engine at maximum power.

Maximum power, as used here, denotes the scheduled power output of the engine as limited by the engine controls for maximum demand. Below 5000 to 10,000 ft density altitude, the actual maximum power used would be lower than available engine power due to transmission power or torque limitations. Thus, most pilots of UH-1H, CH-47C, and CH-54B aircraft never see their engines top out, except when they perform special topping maneuvers designed for checking the proper calibration of the engine controls.

This limitation is applicable, therefore, to high density altitude operations where the power available from the engines is less than the transmission limitation. One means of insuring that this has no effect on safety is to provide an LPI output that the pilot can check to insure that no significant change has occurred in the engine controls. For example, the LPI can provide a selectable output of the maximum available power output of the engines in terms of torque pressure, and the pilot, where performance margin is marginal, can check this output against the actual torque that the engines are supplying. The output could be used in the topping check to indicate the amount of power that should be obtained, rather than using the complicated charts provided for that purpose in the organizational maintenance manuals. An attractive alternative is to have the LPI system provide a continuous output of maximum engine power (torque) available to a torque indicator that incorporates a maximum torque available "bug". The transmission limitation would be indicated by a red line, as usual.

2.6 CORRECTIONS FOR WIND EFFECTS

2.6.1 Magnitude of Wind Effects

Relative wind exerts a strong influence on helicopter performance. In general, the effect is positive, except that in nonvertical takeoffs and landings and in transitions, head winds are beneficial, but tail winds are detrimental. Mathematical modeling of wind effects is described in Appendix B (UH-1, CH-47, and CH-54).

The effects on HOGE performance are typified by the following equation:

$$\frac{\Delta T}{T} (\%) \approx 0.02 \ V_W^2 \qquad (2)$$

where $\frac{\Delta T}{T}$ is the percentage change in lift capability and V_W is the relative wind velocity in knots. It can be seen that the effect is relatively small until 10 knots is reached. The increase in lift at 5 knots is 1/2 percent; at 10 knots, 2 percent; at 15 knots, 4.5 percent; at 20 knots, 8 percent; and at 30 knots, 18 percent. Beyond 30 to 40 knots, the above model is no longer accurate. It is also not applicable to hover in ground effect, as explained later.

For low rates of vertical climb (V_V within about $\pm$500 fpm), vertical climb capability bears a constant relationship to excess lift. A representative effect of wind on vertical climb capability is given by a modification of the above equation:

$$\Delta V_V \text{ (fpm)} \approx 2 \ V_W^2 \qquad (3)$$

Thus, vertical climb capability increases by about 200 fpm at a wind velocity of 10 knots compared to the zero wind capability. Or approximately the same vertical climb capability can be achieved at an increased weight equal to the change in HOGE lift capability.

Helicopter climb capabilities at best climb airspeed are, of course, unchanged by wind, so the criteria for one-engine-out emergencies and rolling takeoff capability would remain unchanged.

Maximum performance (i.e., nonvertical) takeoff and landing distances are strong functions of wind velocity. Tests with a UH-1 showed that moderate winds could halve (headwind) or double (tailwind) takeoff distance. Data for quantitatively evaluating this effect is not included in operator's manuals and would require empirical development in order to provide a wind correction for takeoff distance.

Wind has a significant effect on performance for essentially all takeoff and landing modes, and the effect is positive provided the aircraft's nose is oriented into the wind. The increases in hover and vertical climb capability are only slightly affected by wind direction, but direction would become very important when transitioning from one of those modes to forward flight, or vice versa.

2.6.2 Treatment of Wind Effects

Three ways of handling wind effects were considered relative to the LPI system:

(1) Takeoff and landing success should not be predicted on performance margins attributed to wind effects: zero-wind performance quantities must be calculated and displayed.

(2) Relative wind velocity is measured for input to the LPI system. Performance estimates for wind effects are automatically adjusted assuming the vehicle is headed into the wind.

(3) Manual input of measured or estimated wind velocity should be provided. Zero-wind performance estimates are corrected for manually inserted wind velocity. (Input device returns to zero value when not held to a finite value of wind velocity.)

The third alternative is recommended on the basis of consideration of the following factors: wind velocity measurement, wind variability and pilot technique, estimates for remote sites, measurement verification, Army policy, and wind velocity distributions.

2.6.3 Wind Velocity Measurement

The Army has embarked on a program to develop omnidirectional low airspeed measurement and display capabilities for its future helicopters. This capability is considered necessary for: (1) navigation and flight control at very low altitudes (where Doppler returns are often poor), (2) fire control to adjust for the effects of wind on trajectories, and (3) improvement of instrument flight capability.

With the vehicle stationary, the measurement of low airspeed yields wind speed, and this measurement could therefore form the basis of a wind velocity input to the LPI system.

Table 4 summarizes the characteristics of seven approaches that have been pursued for low airspeed measurement. Prototype models of the first six techniques have been evaluated by the U. S. Army Aviation Engineering Flight Activity at Edwards Air Force Base.* Later paragraphs briefly describe the operating principles of the technqiues. All of the techniques except one (Elliott LASSIE II) require mounting outside the rotor downwash (best location found is above rotor hub) in order to achieve relatively accurate measurements in the low speed range. (For high forward speeds they could be located forward of the main rotor mast where they would be out of the downwash.)

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*Test reports are listed in the bibliography.

TABLE 4

LOW AIRSPEED SENSING TECHNIQUES

Developer and Technique	Basic Output	Mounting Location	Accuracy*	Complexity	Comments
Airometric (Aeroflex)—servoed turbine flow synchronization	TAS magnitude and azimuth	Above rotor	±5 knots	Medium	
Elliott-swivelling Pitot-static probe	Longitudinal and lateral IAS	Below rotor	±5 knots	High	Only sensor adequate below rotor
Honeywell-ultrasonic wave transit time	3-axis TAS	Above rotor	±3 knots (claimed)	High	No moving parts
J-TEC--vortex shedding frequency	Longitudinal and lateral TAS	Above rotor	±5 knots	High	No moving parts
Pacer—rotating venturi tubes	Longitudinal and lateral TAS	Above rotor	±5 knots	Medium	
Rosemount—orthogonal, bidirectional impact pressure	Longitudinal and lateral IAS	Above rotor	±5 knots	Low	
Bolt, Beranek and Newman--electro-optical	Longitudinal (positive) TAS	Above rotor	(unidirectional only)	Medium	No moving parts

*For relative wind sensing 0 to 40 knots in any direction.

Accuracy capabilities of the various approaches appear to be approximately the same--not much better than to within about ±5 knots.* For comparison, most of the 21 pilots who were interviewed thought that they could estimate wind velocity (from the cockpit while on the ground) to within ±5 knots for winds up to 15 to 20 knots and within ±5 to ±10 knots for higher velocity winds. One of the principal problems with respect to accuracy is the turbulence created by the rotor (worst case is in close proximity to the ground). The pressure based approaches yield direct outputs of IAS while the other approaches measure TAS directly. Either form would be adequate for a wind velocity measurement.

All of the approaches reflect the complicated problem of measuring low airspeed magnitude and direction, and all represent fairly substantial systems. The Rosemount approach appears most simple, but still requires a sensor mounted above the main rotor hub with pressure lines running down the rotor mast.

It is obviously desirable that the LPI approach not depend on a relative wind measurement for its implementation. The achievable accuracy for relative wind measurement is apparently not much better than to within about ±5 knots. At 25 knots, ±5 knots is equivalent to a ±5 percent change in lift capability. In the effective weight measurement approach, this would be the variation observed in effective weight.

2.6.3.1 Airometric Systems Corp. (Aeroflex) True Airspeed Vector System

The sensor in this system employs two servo loops, one for airspeed sensing and the other for direction sensing. The speed sensor consists of a stream tube, straightener vanes, a turbine, and a hot wire anemometer bridge network. When the turbine is synchronized with the airflow, its outflow is axial. Departure from synchronized speed (as when airspeed changes) adds an angular component to the outflow that is sensed by the anemometer bridge and causes the servo to act to resynchronize the turbine. The result is that turbine speed is proportional to airflow through the sensor.

The speed sensor is mounted on a swivel, and its angular orientation is controlled by a second servo. This servo acts to follow airflow by cancelling the error signal provided by a hot wire anemometer bridge network mounted on the stream tube so that angular flow with respect to the stream tube axis causes unequal cooling of the anemometer and resulting imbalance in the bridge network to create the error signal. Rotation of the stream tube drives a synchro-transmitter.

Mounted above the main rotor hub, tests of this sensing system indicated accuracy to within ±5 knots at low airspeeds. With its servomotors and hot wire anemometer bridge networks, the reliability of this type of sensor could be expected to be poorer than the approaches having no moving parts.

*Better accuracy could be expected, however, with the rotor stopped.

2.6.3.2 Elliott Low Air Speed Sensing and Indicating Equipment (LASSIE II)

The omnidirectional sensor consists of a swivelling Pitot-static probe with a guide vane. Mounted beneath the rotor, the vane aligns the probe with the resultant airstream (horizontal airspeed plus rotor downwash), allowing measurement of the magnitude of the flow and the angles between the airstream and the fuselage longitudinal axis. An airspeed computer unit resolves the data to obtain the horizontal components of airflow, that is, forward and lateral airspeeds.

This is the only omnidirectional airspeed sensor that can operate reasonably accurately beneath the rotor (this is its principal advantage). To achieve reasonably good accuracy (OGE) requires that fairly complicated characterization data be stored for the probe, including data to correct for the discontinuity that occurs when the probe transitions from the rotor wake to the free stream (at about 20 knots forward airspeed in the UH-1 test installation). Accuracy to within about ± 5 knots has been demonstrated for OGE conditions. Error increases in ground effect (below 10 ft skid height in test) due to expansion of the rotor slipstream and turbulence (5 to 10 knots lower readings). This error in ground effect appears repeatable; therefore, a correction could probably be introduced for wind measurements with the vehicle on the ground (sensed by a squat switch).

2.6.3.3 Honeywell Ultrasonic Wind Vector Sensor

This approach is based on the characteristics of ultrasonic signal transmission through a moving air mass. The relative wind is resolved into three orthogonal components, based on measurements of ultrasonic wave transit times between three ultrasonic transmitter-receiver pairs. A temperature sensor is included in the assembly to compensate for variations in the speed of sound in air due to temperature changes.

The sensor has been tested on the AH-1G where it was found that mounting above the main rotor hub is required for high accuracy. The manufacturer claims that accuracy to within ± 3 knots was demonstrated in the above testing, except that performance for negative longitudinal velocities was not that good due to interference of the sensor mounting structure with the flow.

Obtaining the velocity components from the wave transit time measurements requires iterative solution of a set of four equations involving all four basic math operations. Both analog and digital processors have been mechanized for that purpose.

2.6.3.4 J-TEC Associates VT-1003 Omnidirectional Airspeed System

The J-TEC sensor is based on the aerodynamic phenomenon of vortex shedding from a bluff body, where the frequency of the shed vortices is proportional to the velocity of the fluid, regardless of the fluid density. A cylindrical post is used to generate the vortices which travel with the airflow through the sensor between an ultrasonic transmitter-receiver pair. The acoustic signal is modulated by a vortex pair, and the resulting frequency is proportional to velocity. To achieve omnidirectional capability, velocity sensors are mounted in each end of three tubes that are mounted at 120° angular separation. A computer resolves the velocity signals from the six sensors into longitudinal and lateral airspeed components.

A sensor assembly of the above type was tested at Edwards on an AH-1 helicopter. The unit was mounted at about 1 ft above the main rotor hub. Test data indicate that accuracies to within ±5 knots in longitudinal and lateral airspeeds are probably achievable. Since the vortex generation rate is inversely proportional to the diameter of the cylindrical post, the post must be kept clean and wear-free.

2.6.3.5 Pacer Systems Low Range Airspeed System (LORAS II)

The sensor unit consists of two diametrically opposed venturi tubes mounted at the ends of radial, tubular arms extending from a hub that is mounted to the base of the sensor. The hub, arms, and venturis form a rotating assembly (13 in. diameter) that is driven at a constant speed (720 rpm) in a horizontal plane. The total velocity at each venturi consists of a steady component due to the constant rotational speed and a sinusoidal component at rotational frequency due to the relative wind velocity. The venturi tubes are connected to a differential pressure sensor through the radial tubes. The signal from the pressure sensor is resolved into longitudinal and lateral velocity components that are filtered to remove the modulation frequency.

An advantage cited for this approach is that due to the physics involved, the differential pressure generated is directly proportional to relative wind velocity, thereby giving constant sensitivity to velocity changes even at zero airspeed in contrast to the typical Pitot-static arrangement where sensitivity is proportional to velocity and is therefore zero at zero airspeed. Thus, it is reasoned, LORAS II should display a lower threshold and better accuracy in the low airspeed range.

Accuracy to within ±5 knots is indicated by test data for the sensor mounted above the main rotor hub. Locations beneath the rotor appear unacceptable (discontinuities are present due to rotor wake transition and also ground proximity effects).

2.6.3.6 Rosemount Orthogonal Low Airspeed System

This system extends the conventional dynamic pressure approach to unidirectional indicated airspeed measurement to the orthogonal, bidirectional case. The sensor is a hemispherically tipped cylinder with four internal chambers running the length of the cylinder. Each chamber has a set of pressure sensing ports located on the cylinder portion of the probe. Mounted on the aircraft, the cylinder is aligned vertically with two chambers in the fore and aft directions and the other two in left and right orientations. The pressure difference between the fore and aft chambers is the algebraic dynamic pressure in the longitudinal direction and defines IAS for the longitudinal axis. Lateral airspeed is obtained similarly.

The Rosemount approach appears to be the simplest workable approach to omnidirectional low airspeed measurement. To achieve accuracies to within ± 5 knots, however, requires mounting above the main rotor hub. In that position, the measurement does not appear sensitive to ground effect.

2.6.3.7 Bolt, Beranek and Newman Optical Convolution Velocimeter

Development of this sensor is being sponsored by USAF Flight Dynamics Laboratory. It has not been flight tested on a helicopter.

In the sensor, a collimated infrared output from a light-emitting diode is projected across a flow section where the air has been heated slightly to "mark" it. The locally heated region refracts the light beam, producing a shadow graph (a pattern of light and dark regions) on an optical grating. The light transmitted by the grating is reflected onto a photodetector that produces a signal whose frequency is directly proportional to the airspeed (measured by a special electronic circuit called a correlation discriminator).

The sensor has been tested in wind tunnels and on a Cessna 172. Wind tunnel test data indicate that a sensor of this type might be accurate to within ± 5 to ± 2 knots. The sensor is inherently unidirectional. It could be servoed to track relative wind direction, or an angular array of sensors (as many as six) could be constructed to obtain omnidirectional capability. The developer considers the sensor to be in an early research and development phase.

2.6.4 Wind Variability and Pilot Technique

The influence of wind on helicopter performance margin can be accurately estimated, given wind velocity and direction; but having measured and computed the effect on performance margin of a 25-knot wind, a change in the wind velocity of 5 knots at the time of takeoff is no different in effect than a 5-knot error in the original measurement--both can lead to a 5-percent error in lift capability. Wind direction changes are also important relative to nonvertical takeoffs and transitions. For steady wind, the pilot controls the orientation of the vehicle with respect to the wind and thereby also enters into the determination of wind effects. Thus, one could think of an effective accuracy for wind velocity measurement that would include wind characteristics and pilot technique in addition to measurement accuracy.

It appears that the pilot should be aware of the extent to which the estimated performance margin depends on wind effects. One way of accomplishing this is to have him dial in the wind velocity, so that he observes the change in performance margin.

2.6.5 Estimates for Remote Sites

Wind conditions at the terminus of a flight can be quite different from those at the beginning due to time and geographic changes. For the remote-site mode, therefore, there should be provision for optional input of wind velocity in addition to the required inputs of pressure altitude and temperature.

The LPI system can also be used to estimate landing performance just before an approach. This would be an update of the estimate made previously in the remote-site mode, with updated quantities including actual pressure altitude, actual temperature, and takeoff weight compensated for fuel used. For this use, the system would be operated in a real-time mode. With the helicopter in flight, however, the airspeed measurement would not be equal to the wind velocity; so this represents a second case where a manual input of wind velocity is needed.

2.6.6 Verification and Army Policy

The basic, zero-wind LPI performance estimate can be verified in part by the pilot by hovering in ground effect (a maneuver almost always possible). The pilot can have the LPI system estimate maximum power available and estimate power margin for HIGE (in addition to vertical climb capability). Then in the normal takeoff procedure where the helicopter is hovered IGE while controls and instruments are checked, the actual power margin can be estimated compared to the predicted power margin.

Because of the interaction of ground effect and wind effects, however, HIGE performance is comparatively unaffected by wind in the low speed range. Thus, while a 10 percent increase in HOGE capability might occur due to wind, only a 2 or 3 percent increase might occur in HIGE. This means that the zero-wind performance capability can be fairly accurately checked even in the presence of a moderate wind, but wind effects will be difficult to verify. This could be an inhibiting factor.

Army policy may also enter into the treatment of wind effects. In present operator's manuals (CH-47, CH-54, UH-1), for example, wind corrections are given only for hovering OGE performance--not for vertical climb, maximum performance takeoff distance, or any other performance capabilities. This seems to reflect a defacto policy that might also apply to an LPI system.

2.6.7 Wind Velocity Distributions

It was noted earlier that variations in wind velocity can lower the effective accuracy of wind velocity measurement. Wind velocity varies with time (e.g., gusts) and position. Skelton[11] presents data showing that wind

11 Grant B. Skelton, "Investigation of the Effects of Gusts on V/STOL Craft in Transition and Hover", AFFDL-TR-68-85, Air Force Flight Dynamics Laboratory, Wright Patterson AFB, Ohio, October 1968.

may be divided into a slowly-time-varying mean-wind component and a rapidly-time-varying gust component. From the point-of-view of the LPI system, the slowly-time-varying component is sufficiently slow to be considered invariant. The higher-frequency energy representing the gust component, however, is typically centered at a frequency of about 1 cpm. This frequency is unfortunate, because it is so slow that it can cause error in estimating the steady wind component, yet it is sufficiently high that it will cause a wind velocity change before or during takeoff. Moreover, Skelton shows that the gust component usually contains about as much energy as the steady wind component. A steady wind component of 10 knots with gusts to 15 or 20 knots would approximate this model.

Skelton also presents a model for the wind profile with altitude. For average conditions of surface roughness and turbulence, the probability distribution of wind speeds at a 30-ft altitude is represented by a straight line on log-normal graph paper between 3 m/s (meters/second) (5.8 knots) at 50 percent probability and 12 m/s (23 knots) at 1 percent probability. The distribution for 100 m (328 ft) altitude is again a straight line between 4 m/s (7.8 knots) at 50 percent probability and 16 m/s (31 knots) at 1 percent probability. The ratio of these velocity distributions is proportional to the 1/8th power of the altitude ratio. This means that one can expect about a 50-percent change in velocity measured at ground level and at an altitude a few hundred feet above the ground.

The above information shows that about half the time wind velocity at ground level will be negligible in terms of its effect on lift performance, and that when the wind is significant its gust content and variation with location are also significant.

2.7 AIR DENSITY MEASUREMENT

Efforts directed at development of direct air density measurement sensors have been justified in large part by the belief that air density calculated from pressure and temperature measurements is not sufficiently accurate for estimating lift performance. Appendix A examines this question in detail and demonstrates that direct air density measurement is not required.

2.8 CENTER OF GRAVITY MEASUREMENT

2.8.1 Need for Center of Gravity Measurement

Center of gravity (c.g.) measurement and display was not specified by the Army as a function to be considered for the lift performance indicator, but it has been found essential for satisfying the Army's objectives. The objective use of the device is to provide the pilot with information that he can use to judge the probable success of takeoff and landing prior to committing the aircraft to flight. The system would allow the aircrew to control the loading of the vehicle based on quantitative information--the actual gross weight of the helicopter vs its performance capabilities. An essential ingredient of load control is c.g. Moreover, even with gross weight well within limits, c.g. can be outside limits and produce a hazardous condition.

Out of balance makes control more difficult and decreases maneuverability. The condition can be noted after coming to a hover position. With the c.g. forward of balance, the nose of the helicopter tilts down; aft of balance causes the nose to tilt upward. To counter these effects requires use of the cyclic controls (c.g. forward, cyclic aft, and vice versa), with the result that cyclic stick travel is restricted in the direction opposite the c.g. location. This reduces maneuverability. With the c.g. forward, for example, it might be impossible to flare properly in an autorotational landing. With the c.g. aft, the higher allowable airspeeds might be unattainable due to insufficient forward cyclic displacement.

In pilot interviews conducted in this program, it was found that most pilots believe that the LPI system should include c.g. measurement and display. The pilots consider weight and balance as their foremost instrumentation needs in this area.

All methods of actual helicopter weight measurement that have been considered in this study can yield horizontal c.g. location (lateral c.g. measurement is not needed) as the result of a simple calculation using the weight measurement inputs (weight on struts or wheels). Additionally, a measurement of the longitudinal inclination of the fuselage is needed that requires a relatively simple, inexpensive transducer. This is used to correct for variations in c.g. due to slope, amounting to about 1 in. per degree for the CH-47 and CH-54, and to correct for gross weight variations of about 1-1/2 percent per 10 degrees.

Thus, c.g. measurement is essentially a fallout of actual weight measurement. It is an important variable that can adversely affect flight safety. A system that measures and displays c.g. can also save time spent in calculating c.g. location and in relocating cargo after out of balance is discovered in hover. C.g. measurement and display is therefore considered an LPI functional requirement.

2.8.2 V_{ne} Computation Alternative

The LPI system gross weight and c.g. measurements, together with ambient pressure and temperature measurements, offer an alternative means of computing the never-exceed airspeed limit, V_{ne}. Many current cargo helicopters include cruise guide indicators that are supposed to show the pilot when his airspeed is approaching the point where main rotor stall may occur. These indicators sense the loading in rotor pitch linkages. As the rotor approaches stall, the loads on the pitch linkages increase. By observing limits on these loads, or stresses, main rotor stall conditions can be avoided.

A cruise guide indicator system proposed for the CH-47C is based on sensing the loads on the pivoting actuator and fixed link of the aft rotor system by means of two strain gage bridges. These loads would be increased by a bias in the cyclic to counteract aft c.g. The prescribed pilot action is to limit airspeed to stay within the limits indicated by the cruise guide indicator. Without the indicator, the pilot is supposed to compute V_{ne} by means of a slide-rule-type device that computes the never-exceed IAS based on inputs of pressure altitude and temperature (limit based on density altitude), gross weight at takeoff, and rotor rpm. With an out-of-balance condition, the aft rotor collective differs significantly from nominal, invalidating the nominal V_{ne} limit schedule.

This situation, plus the lack of accurate gross weight measurement and the inconvenience of the V_{ne} computer, very likely prompted the introduction of the cruise guide indicator concept. The lift performance computer is an obvious alternative to both devices. With the added measurement of rotor rpm, the LPI would have all the information needed to compute an accurate and convenient V_{ne}. Moreover, this would provide the LPI with an inflight function to perform where otherwise it would not normally be in use (except when called on to provide engine-out performance information and landing capabilities). The V_{ne} computation might require adjustment in the case of external loads (cargo hook). In this case, the load on the helicopter varies due to aerodynamic loads on the sling load. This increased loading, however, would be reflected in the cargo hook load measurement and could likely be included in the computation without undue complexity. The inclusion of the V_{ne} computation function is recommended for evaluation in an LPI test system.

2.9 PILOT INTERVIEWS

Army helicopter pilots were interviewed at the Aircraft Development Test Activity, Cairns Field, Fort Rucker, Alabama, and at the 179th Aviation Company, Fort Carson, Colorado. Twenty-one pilots were interviewed, with predominant flight experience for each pilot averaging 1015 hours CH-47 flying time and 880 hours UH-1 flying time. Interviews were conducted verbally and also by means of questionnaires.

The purpose of the interviews was to obtain the pilots' assessment of problems related to determining the probable success of takeoffs and landings and their opinions on the functions that should be performed by an LPI system. Some of the more important findings are summarized below.

2.9.1 Pilot Experience with Lift Performance Limitations

All of the pilots interviewed had experienced hazardous situations due to overgross conditions (i.e., insufficient power relative to the gross weight of the vehicle). Removing cargo to enable takeoffs and aborted landings were common experiences. It was evident that the most serious situation is where lift capability is sufficient for takeoff but is inadequate for landing at a higher density altitude. One pilot related experiencing this situation. He started to "fall through" during an attempted landing in mountainous terrain; he returned to the lower elevation takeoff point where he found that his "6000 lb" sling load weighed 8500 lb. A former

CH-54 pilot related that one of the CH-54's in his outfit in Alaska "fell through" while attempting to land on a ledge of a glacier at high altitude (it was carrying a cargo/personnel pod at the time). The result was one death and destruction of the aircraft.

2.9.2 Pilot Evaluation of Current Procedures

The consensus was that present procedures are inadequate, and all of the pilots except two indicated that a lift performance indicator would both speed operations and improve safety. One of the two exceptions believes that those advantages would accrue to this sort of functional aid, but questions the need for an electronic system to provide it. That is, he would prefer that performance information be generated by some sort of slide-rule device, but advises that a weighing system could be very useful. The other excepting pilot stated that an indicator was not needed, but that "familiarity with performance charts" was a needed improvement. It is noteworthy that the first pilot is a maintenance officer and the second is an instructor pilot with extensive flight experience.

The pilots appear to accept the performance estimation procedures and data in the operator's manual as valid; however, they do not appear to use them extensively. This is likely due, in part, to the common distrust of cargo weights provided by ground personnel. It was also indicated that it is impractical to use the operator's manual in the cockpit. In effect, the present lift performance criterion (especially at remote sites) is that if the helicopter will lift off the ground, it is committed to flight. This check is adequate with respect to safety except for the following potential conditions, (1) landing at a higher density altitude, (2) shutdown of one engine, (3) HOGE maneuver for NOE profile, and (4) confined area takeoff (obstacle clearance).

As shown in Figures 4 and 5, checklists provided in the operator's manuals give criteria for all the common takeoff and landing modes. It appears that the uncertainty with respect to actual weight, the changeable ambient conditions, the inconvenience of using the checklists and manual in the cockpit, and perhaps the unfamiliarity with the charts (because they are not used?) combine to eliminate the usefulness of the checklists.

2.9.3 Pilot Views on Instrumentation Needs

Responses showed that most pilots consider weight estimation the most serious problem (especially for remote sites) and believe that a weight measurement system is needed, including cargo hook loads. One pilot stated: "At a minimum, the system should be able to give a readout of total aircraft weight on the ground with the rotors stopped, plus some type of cargo hook strain system to show weight on hook." It would be desirable, they noted, to be able to obtain an accurate weight measurement with the rotors turning. One pilot stated that during unloading and loading of cargo at remote sites, it is normal to shut down engines; but it is possible, especially for combat situations, that the engines would be operated at flight idle or even at a higher power level. From other pilot interviews, however, it appeared that the more typical procedure at a remote site is to leave the engine condition lever at the flight position and to position the thrust control rod (collective) at the 3° detent.

Collectively, the pilots are less positive with respect to the computation and display of performance capabilities or margins. While nearly all the pilots expressed the need for weight measurement, only about half the pilots felt that performance capabilities display is needed. An apparently relatively common viewpoint was that with the addition of a weight measurement, the remainder of the problem could be solved adequately by the pilot using present procedures (i.e., the operator's manual). As one pilot put it, "A computerized system that would include engine performance and outside air conditions (density altitude, wind, etc.) seems like a luxury and Army pilots are not used to luxury." Thus, several pilots thought that it would be desirable to have the performance computation feature, but wondered if it could be justified.

One pilot expressed the concern that with a lift performance computer available, "the pilots would not maintain a knowledge of aircraft performance characteristics or have any idea of what the aircraft can do until getting in the seat and having everything turned on." On the other hand, as another pilot stated, "It should replace as much as possible the performance charts in the operator's manual that are not always practical for use in the cockpit. The performance charts should be retained, however, for preflight planning."

Several pilots, especially those whose desk assignments involve maintenance, expressed concern for the problems that would be caused by the addition of another black box. One of these men thought that the idea of performance indicators being more readily available in the cockpit was excellent; however, he would opt for some sort of manual slide-rule-type aids that wouldn't stop working when contaminated with hydraulic oil. He also stated that even though a weighing system could be very useful, he is opposed to adding another automatic system to the aircraft.

With respect to specific display quantities, the pilots mentioned hover (IGE and OGE) capabilities, vertical climb capability, and single engine rate of climb (with best airspeed for that condition). Gross weight was actually most frequently mentioned, but additionally almost half the pilots specifically mentioned c.g. calculation and display as a needed or essential system function. Little interest was expressed for display of distance to clear 50-ft obstacles, perhaps because that is not presently in the CH-47C operator's manual.

2.9.4 Other Pilot Comments

Some additional miscellaneous findings that bear on the problem are as follows:

(1) Takeoffs at gross weights in excess of HOGE capability are commonplace.

(2) Sling load operations comprise a significant percentage of operations for all helicopters considered in this study.

(3) Aborted takeoffs and landings due to insufficient lift performance occur infrequently during peacetime operations, but are relatively common occurrences in combat operations.

(4) The decision to initiate a flight operation can be based on factors in addition to the likelihood of successful takeoff and landing. For example, the likelihood of executing a successful landing following a single-engine failure is a criterion used by the 179th Aviation Company at Fort Carson, Colorado.

(5) The personal data base of even the most experienced pilots interviewed does not appear to include sufficient data to make a qualitative comparative evaluation of actual helicopter performance vs predicted performance from the operator's manual.

(6) NOE doctrine requires HOGE capability. This tends to limit the effective performance capability of helicopters (since they can take off IGE) and enhance the potential usefulness of an LPI system.

2.9.5 Influence of Geographic Location of the Aviation Unit

The elevation of Fort Rucker, Alabama, is less than 500 ft, while the elevation of Fort Carson, Colorado, is nearly 6000 ft. This difference in elevation was reflected in the responses of the pilots interviewed at the two locations. The pilots at Fort Carson seemed much more aware of potential problems associated with lift performance and the consensus favored an LPI system with broad capabilities. At Fort Rucker, the general opinion was that a weight measurement system would suffice.

The reason for this difference in viewpoint is evident in the performance charts for the CH-47. The maximum gross weight capability of the CH-47C (T55-L-11 engines) is 46,000 lb, exclusive of atmospheric effects. The pressure altitude at Ft. Rucker seldom exceeds 1000 ft; the temperature seldom exceeds 35°C. For the typical worst-case condition, the maximum (though nominal) gross weight for hovering OGE would be about 44,000 to 45,000 lb. For HIGE below about 25 ft (aft wheel height), the maximum capability is pushed up to 46,000 lb. Thus, at Ft. Rucker, except for unusually long sling lengths, the gross weight capability remains essentially constant at 46,000 lb. Moreover, the flights do not terminate at locations of higher elevations. With essentially constant maximum load capability, the Ft. Rucker pilots could get along nicely with only weight and balance information.

In sharp contrast, the airfield at Fort Carson, Colorado is nearly 6000 ft above sea level with nearby mountain passes above 8000 ft and mountain peaks (like Pike's) reaching up to 14,000 ft. Takeoffs and landings at density altitudes exceeding 12,000 ft are not uncommon in mountain exercises. Maximum gross weight for vertical takeoffs could drop to as low as about 32,000 lb (more than the weight of two fully fueled UH-1's below the maximum gross weight capability of the CH-47C of 46,000 lb)--hence, the difference in attitude between Fort Rucker and Fort Carson pilots.

3. WEIGHT MEASUREMENT

Weight measurement was found to be the key technological issue with respect to the feasibility of the LPI system--system requirements are easily met provided accurate gross weight measurement is feasible. It is desirable that accurate weight measurement be feasible for dynamic conditions (main rotors turning) as well as for static conditions. Dating back to 1967, tests of several developmental weight and balance systems for the CH-47 were not considered entirely successful, the principal deficiency being degraded accuracy with the main rotors operating at minimum thrust levels as opposed to being stopped.

3.1 COMPARISON OF WEIGHT MEASUREMENT APPROACHES

Table 5 provides a comparison summary of five weight measurement approaches: one strain gage technique and four oleo pressure measurement approaches which differ only in the means of compensation for the effects of strut friction.

Briefly, the measurement approaches are:

In-Axle Strain Gage--Strain gage deflection sensors mounted inside each landing gear axle measure the vertical shear deflection of the axle due to the weight supported.

Oleo Pressure Measurement--Pressure sensors measure the internal pressure in each oleo which is a function of the weight supported by the strut. Four approaches are evaluated which differ only in the way that stiction is handled:

(1) Uncompensated: This is the baseline pressure approach consisting only of the oleo pressure measurement with no compensation for friction.

(2) Functional Unsticking: Using the same hardware as in (a) above, "unsticking" of the oleo struts is accomplished by operating the helicopter to impose dynamic loads on the struts sufficient to free them (that is, by taxiing or applying lift and then reducing power to ground idle or lower).

(3) Pneudraulic ΔV Pressure Averaging: A hydraulic or pneumatic-hydraulic system causes the pressure inside each strut to cycle through the maximum and minimum pressures possible (the strut is "unsticked" in both directions). These two pressures are measured and averaged for an estimate of the frictionless pressure level.

(4) Zero-Friction Oscillating Swivel: An electromechanical or hydraulic actuator built into the strut torque arm assembly causes the piston to angularly oscillate slightly within the strut cylinder. After about 10 oscillations, the frictional force acting in the axial

TABLE 5
COMPARISON OF WEIGHT MEASUREMENT APPROACHES
OLEO PRESSURE WITH COMPENSATION FOR STRICTION

Evaluation Factor	In-Axle Strain Gage	Uncompensated	Functional Unsticking	Pneudraulic ΔV Pressure Ave.	Zero-Friction Oscillating Swivel
1. Static accuracy (error band)	± 1 percent	± 2 to ± 3 percent	± 1 to ± 2 percent	± 1 percent	± 1 percent
2. Required developments	Sensor-axle for each aircraft	None	None	Pneudraulic ΔV design analysis	Swivel actuator design analysis
3. Development risk	Low to moderate	None	None	Low	Low
4. Helicopter applicability	All except UH-1	All except UH-1	All except UH-1	All except UH-1	All except UH-1
5. Maintainability	Poor	Very good	Very good	Fair	Good
6. Calibration required	Yes	No	No	No	No
7. Complexity	Medium	Very low	Very low	Med. to high	Low
8. Size and weight	Low	Low	Low	High	Low
9. Operational convenience	Good	Good	Poor	Fair	Good
10. Sensitivity	Excellent	Poor	Good	Good	Good
11. Cost units (excluding electronics)	5	1	1	15	5
12. C.g. measurement capability	Good	Fair to poor	Fair	Good	Good

direction is reduced to a negligible proportion and the pressure in the strut is at its (essentially) frictionless value.

Table 5 shows that the zero-friction oscillating swivel approach provides superior accuracy and good maintainability characteristics at reasonable cost and low development risk. For future helicopters (not yet off the drawing board), the strain gage approach is more competitive, but it should be compared with an oleo pressure approach wherein an antifriction technique is incorporated within the struts. The latter possibility should be explored in a design-feasibility study.

3.1.1 Evaluation Factors

Explanatory and supplementary remarks regarding the evaluation factors in Table 5 are given in the following paragraphs.

3.1.1.1 Static Accuracy (Error Band)

The accuracy of the strain gage technique is based on the performance levels achieved by production weight and balance systems on the 747 and L-1011. Accuracies of the uncompensated and functional unsticking pressure approaches are based on test results of experimental systems installed and tested on the CH-47. Accuracies for the other compensated pressure approaches are based on reported results for installed production systems and on laboratory test results. The zero-friction technique appears to have the highest accuracy capability.

3.1.1.2 Required Developments

For the strain gage approach, the axle and strain gage deflection sensor together constitute a weight transducer and therefore represent a design and development problem requiring separate solutions for each aircraft.

The pressure sensors required to implement an oleo pressure approach are available as off-the-shelf items. However, the system elements necessary to implement the pneudraulic or zero-friction compensation approaches would require design effort for each selected helicopter type.

3.1.1.3 Development Risk

Risk is associated with any required developments. Based on state of the art, one can confidently expect the strain gage approach to be ultimately successful. The low to moderate risk is associated with the possibility of the development cycle extending significantly beyond the time of installation of production units in order to achieve a drift-free sensor intallation.

The pneudraulic and zero-friction antistiction techniques represent straightforward design problems and are considered low risk efforts.

3.1.1.4 Helicopter Applicability

All of the techniques are applicable to all helicopters with oleo shock struts. The UH-1 with its skid-type landing gear is the only helicopter considered in the study for which the techniques are not applicable. Static weight measurement for the UH-1 does not appear feasible without significant redesign of the landing gear. (As an example, an experimental design modification of the commercial counterpart of the UH-1 incorporated "liquid springs" (the oleo principle) in the landing gear and allowed weight measurement by the pressure approach.)

3.1.1.5 Maintainability

The "poor" ranking for the strain gage approach is due principally to the conclusion that special tools and test equipment would be required to replace a sensor in an axle and accomplish recalibration. This would likely prohibit this function from being accomplished by organizational maintenance (most likely a general support item) and might restrict sensor replacement to periodic aircraft inspection intervals (i.e., 100-hr intervals). This is a serious drawback since this type of situation supposedly led to the removal of the production in-axle strain gage measurement system that was installed in the USAF C-130 fleet. (This subject is discussed later.)

Maintainability is assessed as fair for the pneudraulic antistiction approach because of its relatively high complexity.

3.1.1.6 Calibration

Calibration is required for the strain gage approach and requires precision weighing equipment to accurately determine the weight on each wheel. Calibration could possibly be required for the uncompensated and functional unsticking pressure approaches to achieve slightly better accuracy, depending on accuracy requirements for both weight and center of gravity.

3.1.1.7 Complexity

Complexity also relates to reliability and maintainability. The basic oleo pressure sensing approach is very simple, consisting essentially of a pressure transducer connected to each strut. The strain gage approach is easily twice as complicated, with about twice the number of transducers required and a much more sensitive sensor-measurement interface.

With a pneudraulic antistiction system, the pressure approach becomes quite complicated. The zero-friction antistiction approach, in contrast, is much simpler in concept and implementation. Although the zero-friction technique involves more hardware than the strain gage technique, the interfaces involved are not nearly as tenuous; therefore, the zero-friction technique complexity is ranked as being lower.

3.1.1.8 Size and Weight

All of the techniques have comparable sizes and weights except the pneudraulic technique in which those two factors are comparatively much larger.

3.1.1.9 Operational Convenience

Ideally, operation of the LPI system should require no special procedures and should not interfere with operation of the helicopter in any way. The functional unsticking approach is ranked poor in this respect because it would require that a specific sequence of operations be followed for maximum weighing accuracy.

The pneudraulic approach is ranked only fair because the weighing procedure would require several minutes of elapsed time. The remaining approaches are judged not to have a significant impact in this area.

3.1.1.10 Sensitivity

The uncompensated oleo pressure technique is judged to have poor sensitivity because conceivably the helicopter's load could be changed by several percent without any change in indication given by that type of system.

3.1.1.11 Cost Units

The simple, uncompensated oleo pressure approach is by far the least expensive. With the cost of instrumentation (pressure transducers) and fittings for that approach taken as one cost unit, the approximate comparative costs of the other approaches are given.

3.1.1.12 Center-of-Gravity Measurement Capability

All actual weight measurement approaches considered provide the information needed to compute longitudinal c.g. location. Whether a particular approach is suitable for providing c.g. location is a question only of accuracy. Based on c.g. limits for the aircraft considered (for example, at maximum gross weights the allowable c.g. ranges for the CH-47C and CH-54B span 15 and 18 in., respectively), the desired accuracy is to within a very few inches.

The weight measurement accuracy needed to accurately compute c.g. location closely matches the accuracy desired for weight measurement on its own merits. For example, gross weight accuracy to within 1 percent provides a c.g. accuracy to within 1 or 2 in. The c.g. indication becomes marginally useful at a gross weight accuracy of around 3 percent. This topic was examined in detail in Section 2.4.3.

3.1.2 Other Factors Bearing on Weight Measurement

The only other major factor of interest is weight measurement during dynamic conditions, that is, with the rotor turning. This area is examined in Section 3.4. Although it is an important consideration, it does not appear to affect the comparative merits of the weight measurement approaches.

3.1.3 Summary of Basic Advantages and Disadvantages

Table 5 lists the major advantages and disadvantages of the two basic measurement approaches. Basically, the oleo pressure approach is superior in every respect except for the very important disadvantage that to reach the full potential of the approach--to be accurate to within ± 1 percent--it is necessary to compensate in some manner for oleo stiction. This shortcoming tends to promote the strain gage approach to the level of being a viable alternative.

3.2 OLEO PRESSURE MEASUREMENT APPROACHES

3.2.1 The Oleo Shock Strut as a Weight Sensor

The landing gear struts of both large fixed-wing aircraft and helicopters are telescoping hollow cylinders which act as shock absorbers. In the large fixed-wing aircraft, the bottom part of each cylinder is filled with oil and the top is pressurized with air. In the helicopter, an additional floating piston may be present separating the air and oil, as in the case of the struts on the CH-47 and CH-54 helicopters. This approach allows greater flexibility in the design of damping orifices used in the shock strut. In the static condition, the pressure inside the shock strut balances the weight supported by the strut and is therefore a measure of that weight.

Figure 8 is a cutaway illustration of the forward landing gear shock strut assembly for the CH-47C helicopter. Pressurized air resides beneath the floating piston. Hydraulic fluid is contained above the floating piston and both below and above the piston head (containing the variable orifice assemblies). As load is applied to the strut (i.e., increased weight), the piston tube assembly tends to move upward within the cylinder. This compresses the fluid above the piston head to a pressure greater than that beneath the head until the compression damping valve unseats, allowing fluid to flow through its orifice and into the chamber beneath the piston head; the resultant pressure increase causes the floating piston to move downward, increasing the pressure in the air chamber. In this way, the strut assembly compresses with increased load until equilibrium is attained, where the force on the piston head supplied by the internal pressure balances the force on the other end of the piston.

Pressure inside the strut is measured by simply replacing one of the oil filler plugs at the top of the strut by a "T" connection, using one port to provide servicing access and the other for input to a pressure transducer.

TABLE 5

OLEO PRESSURE VS STRAIN-GAGE APPROACHES

Oleo Pressure Measurement Approach	Strain-Gage Measurement Approach
Advantages	**Advantages**
• Off-the shelf pressure sensors	• Excellent sensitivity
• Simple installation and replacement of components	• Oleo stiction not a factor
• No calibration required	• No special procedures for static weight measurement
• Accuracy to within ± 1 percent achievable with compensation or antistiction)	• Accuracy to within ± 1 percent achievable
• Excellent long-term stability	• Low power requirement
• Low development risk	• No impact on aircraft vulnerability
• One sensor per strut (vs one per wheel)	
Disadvantages	**Disadvantages**
• Some form of compensation or antistiction necessary to achieve accuracy to within ± 1 percent	• Sensor and sensor installation design required for each aircraft (probably including new axle design for performance assurance)
(All other disadvantages relate to the above factor)	• Development cycle likely to extend significantly beyond installation of production units.
	• Complicated installation and calibration requiring special tools and test equipment that are likely to require high echelon maintenance
	• One sensor per wheel

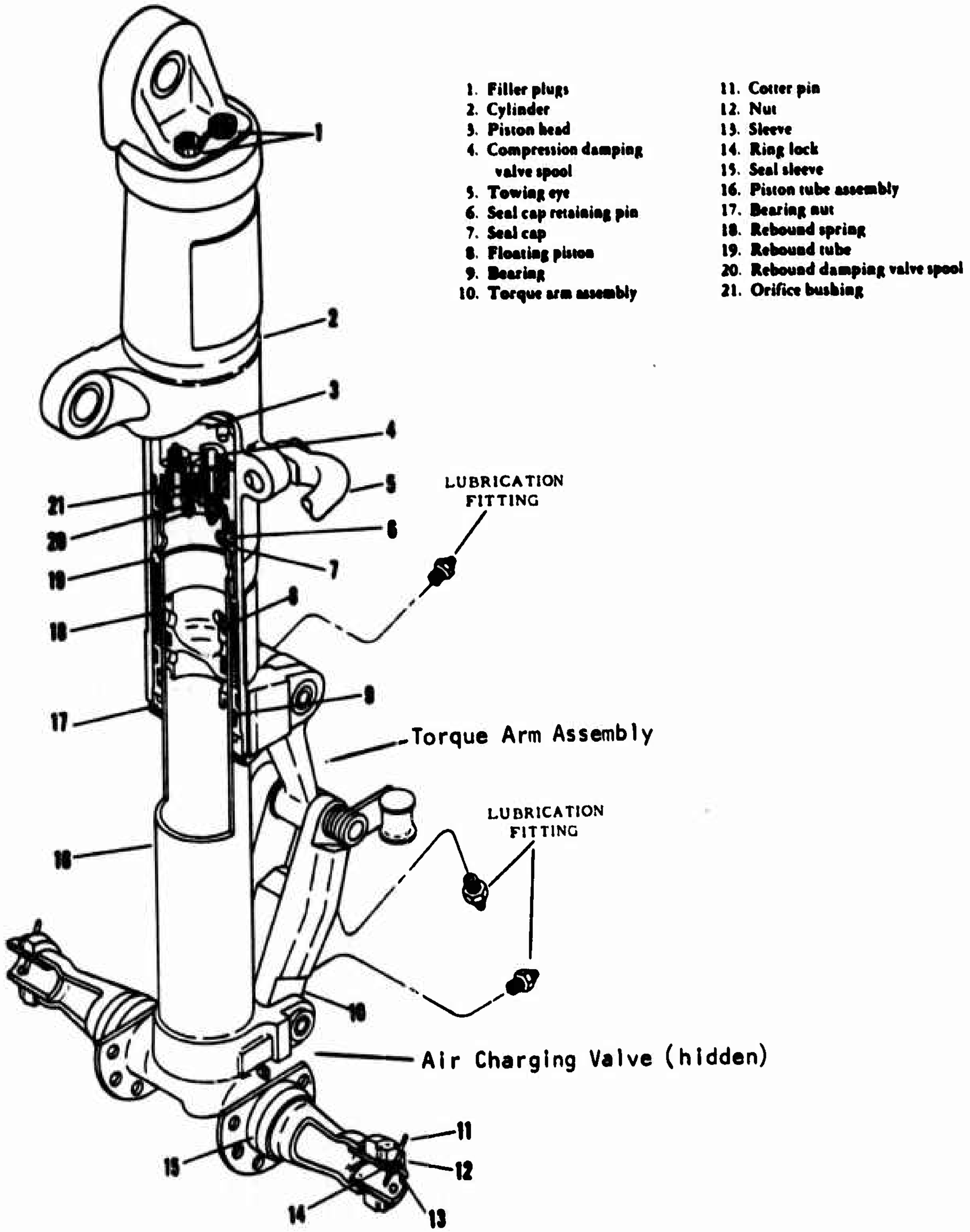

Figure 8. CH-47C Forward Landing Gear Shock Strut (From CH-47C Organizational Maintenance Manual).

3.2.2 The Most Widely Used Technique

Oleo pressure measurement is the technique most widely employed in production weight and balance systems for fixed-wing aircraft. (There are no production weight and balance systems of any type on any helicopters.) Hawkins[12] states that about 300 aircraft have been equipped with the STAN* system which employs the strut pressure measurement approach, including the following aircraft types: 707, 737, DC-8, DC-9, C-130, KC-135, Gulfstream II, and Falcon 20. Users include a number of domestic and international airlines, corporate aircraft, and the USAF.

All of the oleo pressure measurement based systems have been installed on a retrofit basis and represent the only successful retrofit application of weight and balance systems.

3.2.3 Stiction: Principal Limitation on Accuracy

All aircraft landing gear exhibit friction between the oleo piston and cylinder. The magnitude of the frictional force varies widely among the landing gear types, depending, for example, on the design of bearing surfaces and seals, materials, and tolerances. Sensitivity to factors such as temperature, contamination, and strut inclination will similarly vary widely. Even for an individual landing gear strut, the magnitude of the maximum static friction can be expected to vary significantly due to such factors as actual component tolerances; mating surface finishes; lubrication; temperature; attitude of the landing gear; and the degree of swell, wear, and previous motion of its elastomeric seals. Moreover, this frictional force can act in the upward or downward direction, with the exact magnitude and direction depending on the historical state of the oleo.

Figure 9 illustrates the stiction characteristics of an oleo strut. The pressure level E would be required to support the load on the strut in the absence of friction. With friction present, the pressure can vary between levels A and C. Suppose that the strut were initially at level E and fluid was added slowly to the strut. Then the pressure would increase until at level A the pressure overcame the static friction. At this point, the strut would move in a direction to relieve the excess pressure (that is, it would elongate) and the pressure would drop to approximately level B, representing a lower dynamic friction level still acting in the same direction (always opposite the direction of the pressure change). If we now began to withdraw fluid from the strut, the pressure would drop from B, through E, all the way down to C, at which point the static friction (now acting in the oppposite direction) would again be overcome. The strut would compress a bit, and the pressure would essentially instantaneously increase to level D, representing the magnitude of the dynamic frictional force opposing motion of the piston within the cylinder.

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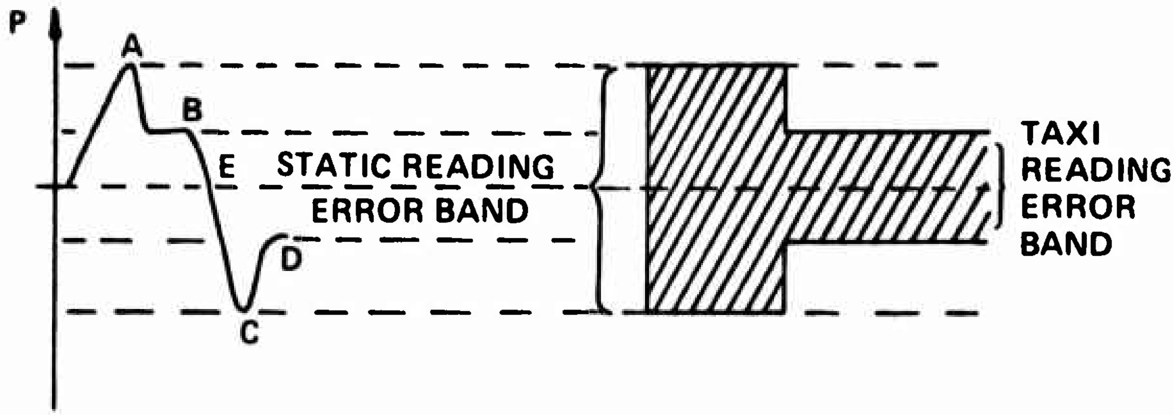
*STAN is a registered trademark of Fairchild Instrument and Camera Corporation.

[12] B.J. Hawkins, "STAN Development and Applications," SAWE Paper No. 1073, May 1975.



- BY TAXIING AIRCRAFT
- BY INCREMENTING VOLUME
  $$P = K\left(\frac{T}{V}\right)$$
- BY INCREMENTING TEMP
  $$P = K\left(\frac{T}{V}\right)$$
- BY MECHANICAL ACTUATION
  - YAW MOTION
  - UP-DOWN MOTION

Level E: Frictionless pressure level

Levels B and D: Dynamic friction error band

Levels A and C: Static friction error band

Figure 9. Oleo Pressure Characteristic and Means of Achieving "Unsticking:.

Typical values of oleo friction for fixed-wing aircraft expressed as a percentage of acting weight are ±4 percent static and ±2 percent dynamic. It can therefore be expected that the static reading error band (see Figure 9) in almost all cases will be larger than can be tolerated for an acceptable weighing system, at least for fixed-wing aircraft.

The oleo stiction characteristics must be established empirically for each specific case. The C-130 represents a worst-case example where errors up to ±20 percent are possible due to stiction.

Reference 4 includes test data that enable estimation of the stiction characteristics of CH-47 landing gear struts. Based on that data, the static error band for the forward struts is ±1.2 percent (full scale) and for the rear struts is ±4.1 percent (full scale). In a normal loading configuration, the rear struts carry less than half the load carried by the forward struts so that the combined static error band is about ±2 percent. The dynamic friction band would be about half this amount, or ±1 percent (test data in Reference 2 support this estimate). Errors due to friction can be expected to increase with variations in inclination (more than 5 degrees tilt from vertical) and at temperature extremes.

For the CH-47, an oleo pressure-based weight measurement approach could be expected to exhibit marginally adequate accuracy for static weight measurement where the principal error source would be the ±2 percent stiction error band. Given the acceptability of compensatory measures to reduce the principal error source to the dynamic friction band of ±1 percent accuracy would be more than acceptable.

3.2.4 Stiction Compensation and "Unsticking" Techniques

In most cases, the static friction error band as defined in Figure 9 can produce unacceptably large errors. Several techniques are used or have been advanced to compensate for the error or to reduce the magnitude of the possible error. Apart from redesigning the oleo strut to reduce friction, there are only two basic ways of reducing the error:

(1) Cause the frictional force to be limited to the dynamic band (that is, overcome static friction by some means to "unstick" the strut).*

(2) Cause the internal pressure to trace through a pressure cycle (starting at initial pressure and cycling through levels A-B-E-C-D in Figure 3), while at the same time measuring the oleo pressure. Pressures at levels A and C (or B and D) are then averaged to estimate the level E pressure.

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*Rotational unsticking motion of the piston relative to the cylinder is a special case that reduces the axial frictional force to negligible proportions.

3.2.4.1 Unsticking by Taxiing

Several techniques for implementing the above approaches are listed in Figure 9. Taxiing the aircraft breaks the stiction and causes the oleo pressure to settle within the dynamic friction band. Presently, this is the only approach employed by production oleo pressure weight and balance systems (with the exception of a single pneumatic system under current evaluation, which will be described below). There is a tendency of the oleos to settle near one of the dynamic friction boundaries. This has been used to advantage by incorporating this offset into the calibration of the systems. This enables achieving a higher average accuracy level than would otherwise result from the dynamic friction band. Clark[13] notes, however, that the oleos do not always settle at the (calibrated) dynamic friction boundary (that might be viewed as the mode of the distribution). Thus, the calibration for the predominant mode of behavior admits the possibility of infrequent, but larger, errors.

3.2.4.2 Equivalent of Taxiing for the Helicopter

Operations for helicopters that would have the same effect as taxiing for the fixed-wing aircraft include taxiing, followed by setting the engine(s) to ground idle power (negligible lift), or simply running the main rotor up to flight idle followed by ground idle, or less. Operating the main rotor(s) at flight idle will free the struts, but compensation for the lift provided by the rotors is required (this will be examined in Section 3.4). It is also possible that the vibration present during ground idle operation may be sufficient to "unstick" the struts, but this is uncertain at the present time.

3.2.4.3 Unsticking by Incrementing Specific Volume

Oleo strut pressure can be cycled as shown in Figure 9 by adding to or subtracting from the air or hydraulic fluid in the strut. This could be open loop (meter set amount into and out of strut) or closed loop (sense strut motion) operation and can either be half cycle (unstick at level A and read resultant pressure at level B) or full cycle with pressure averaging (average of pressure at levels A and C, for example).

Fairchild has done the most work in this area with the development of their C-130 STAN "S" system ("S" for stimulation). Figure 10 shows the portion of the system for strut pressure cycling, and Figure 11 diagrams typical operation for a strut. As shown in Figure 11, pressure is cycled to obtain measurements of maximum and minimum pressures. According to Fairchild friction can vary significantly, but the maximum and minimum pressure values produced are always approximately equally displaced from the average, or frictionless, value. The two pressures are averaged to obtain an estimate of the average or frictionless pressure.

The equipment required to exercise the strut pressure cycle is shown in Figure 10 (not shown is the Electronic Sequencer and Control Unit). The

[13] D.K. Clark, "An Oleo Settling Weight and Balance System," SAWE Paper No. 1033, May 1974.

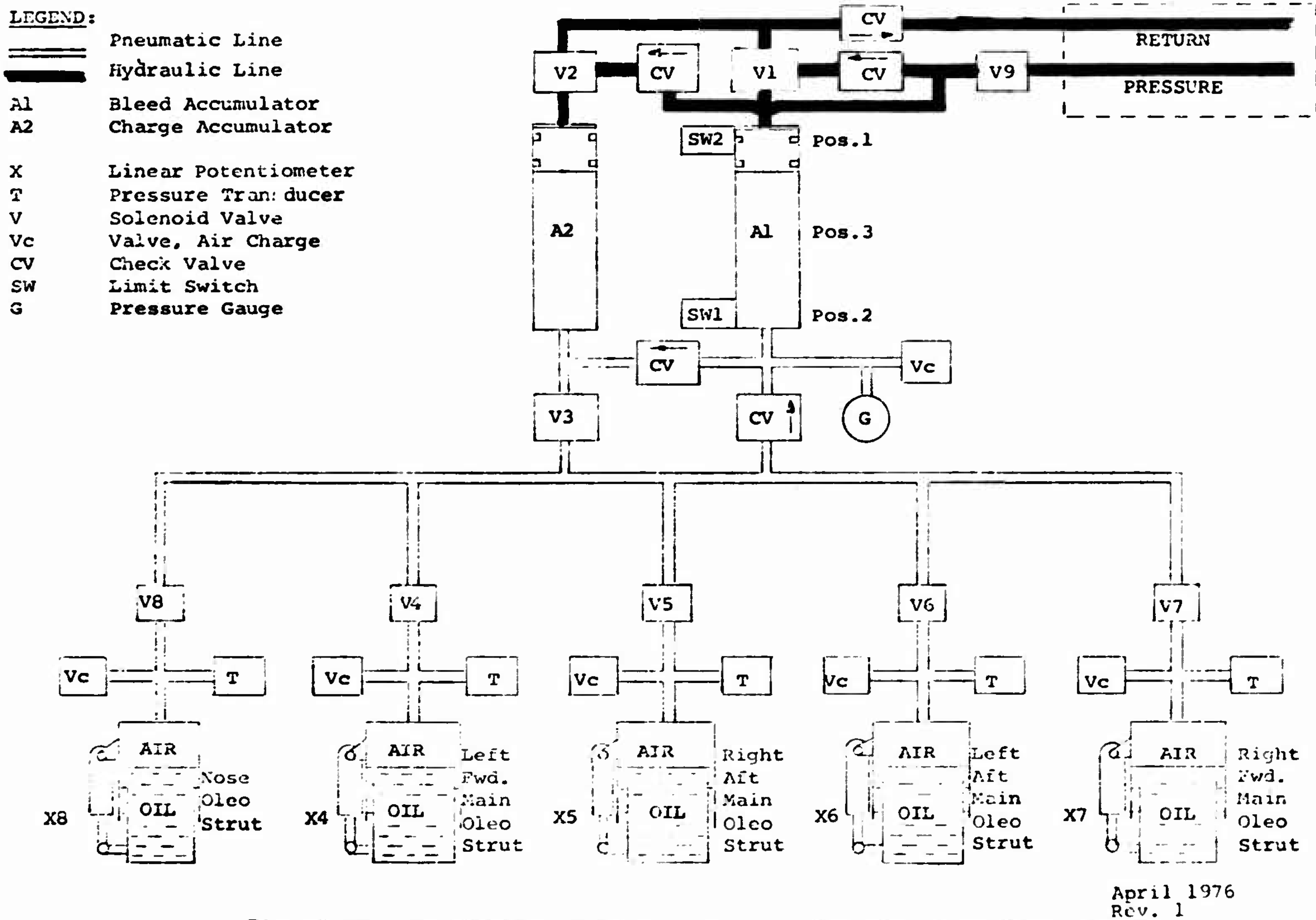
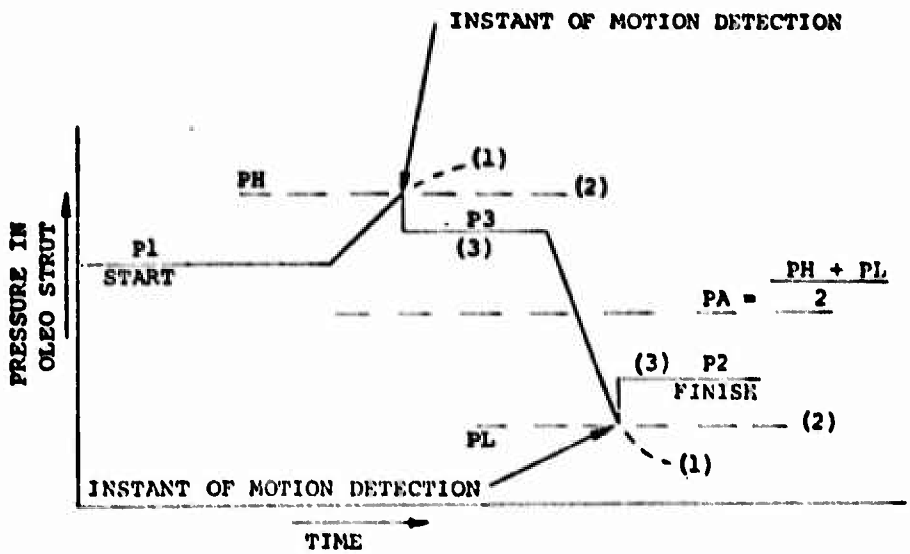
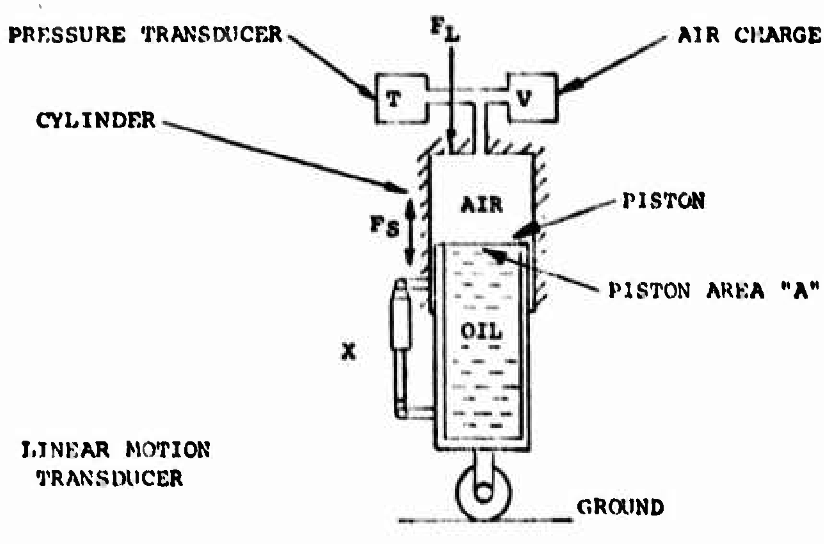


Figure 10. Schematic of Fairchild's C-130 Gear Stimulating System (Courtesy of Fairchild Industrial Products).



PEAK PRESSURE AVERAGING DIAGRAM



LANDING GEAR OLEO STRUT

Revised 3/26/75

Note: Fairchild Camera and Instrument Operation holds U.S. Patent No. 3, 581, 836 on the Peak Pressure Averaging Technique.

Figure 11. Fairchild's Peak Pressure Averaging Approach (Courtesy of Fairchild Industrial Products).

system is powered by the aircraft's auxiliary hydraulic system. A complete cycle of the stimulating system consists of the following steps:

(1) Setup: The piston in Accumulator A1 is actuated from position 1 to position 2 charging Accumulator A2.

(2) Charge: Air is channeled into a particular strut until stiction is overcome and motion occurs as sensed by the strut position sensor. Peak pressure is determined at the moment of motion of the strut.

(3) Bleed: Air is channeled out of the strut. Accumulator A1 piston starts at position 2 and is driven by the air from the strut to some new position, labeled position 3. If A1 is driven all the way to position 1 before strut motion, then A1 is recycled to position 2, charging A2, and then further air is bled from the strut. This is repeated until strut motion occurs, at which time minimum pressure is determined.

The remainder of the sequence for a strut is dedicated to automatic servicing of the strut air charge. This is accomplished by a sequence of operations involving measurements of the strut pressure, air volume (in effect), and ambient temperature to control the mass of air inside the strut.

This cycle is repeated for each strut individually. Then the whole sequence is repeated again. The first round of stimulation accomplishes servicing of the oleos. The second round provides the peak pressure data from which weight and balance is later computed. (The complete cycle is accomplished for the main gear and then the nose gear.)

The Fairchild gear stimulation system is sophisticated. A significant part of its functions can be associated with its automatic strut air pressure servicing, but all of the system components are required irrespective of that function. (It might be possible to dispense with the linear motion transducer and sense motion indirectly through the resultant pressure change--without the servicing function.) It might be argued that the system is an automatic strut servicing system that also facilitates strut pressure measurement for weight and balance computation.

It may be possible to design a practical open loop type system, but much effort has gone into the design of the C-130 system and it has been well thought out and engineered. It therefore probably represents the best engineering solution for a hydraulic-pneumatic antistiction system, at least for the C-130.

The configuration of helicopter shock struts as on the CH-47 and CH-54 may favor the use of hydraulic fluid as the stimulation medium instead of air because of the location of hydraulic and pneumatic access ports (hydraulic fluid filler ports at the top of strut and air charging valve at the bottom of strut) which are opposite to the locations on fixed-wing aircraft struts such as the C-130

3.2.4.4 Variable Temperature Antistiction Technique

Figure 12 illustrates an ingenious method for overcoming stiction in a shock strut that was recently under development by Canadian Marconi Company (CMC).[13] The idea is based on the pressure vs temperature relationship of a confined gas. If the temperature of the air inside the oleo strut is increased, its pressure will rise until the pressure increase overcomes stiction. Then the strut will move in a direction to relieve the pressure. Thus, the oleo pressure cycle can be traversed by heating and cooling the air inside the oleo.

When energized, the CMC antistiction unit circulates air from the oleo chamber through the heater for 15 seconds. This heats the air to 100° to 200°F higher than its initial temperature (adequate for stiction levels of up to about 15 percent). Maximum pressure is measured and stored. The gas in the oleo is then allowed to cool and within about 45 seconds reaches the minimum pressure value.

Advantages of the temperature approach are:

- Comparatively small and lightweight.
- Electrical power more convenient.
- Independent system on each oleo.
- Simultaneous operation for all oleos possible.

The approach is not as attractive for helicopters having air chambers on the wheel-end of the strut because of the pneumatic tubing that must be added. Other, less visible problems could be encountered with the heating function.

Development of the CMC antistiction device appeared well underway in 1974. Reference 13 shows an installation of it on a 707-320C landing gear strut (on the aircraft) and alludes to its availability as a part of a production weight and balance system (Canadian Marconi Company, "CMA-721 Integral Weight and Balance System"). However, no test results for the installed system are presented in the reference. Currently, CMC states that it is no longer in the weight and balance instrumentation business and declines to discuss past efforts.

At the moment, therefore, the variable temperature approach must be judged experimental and unproven, though potentially very attractive.

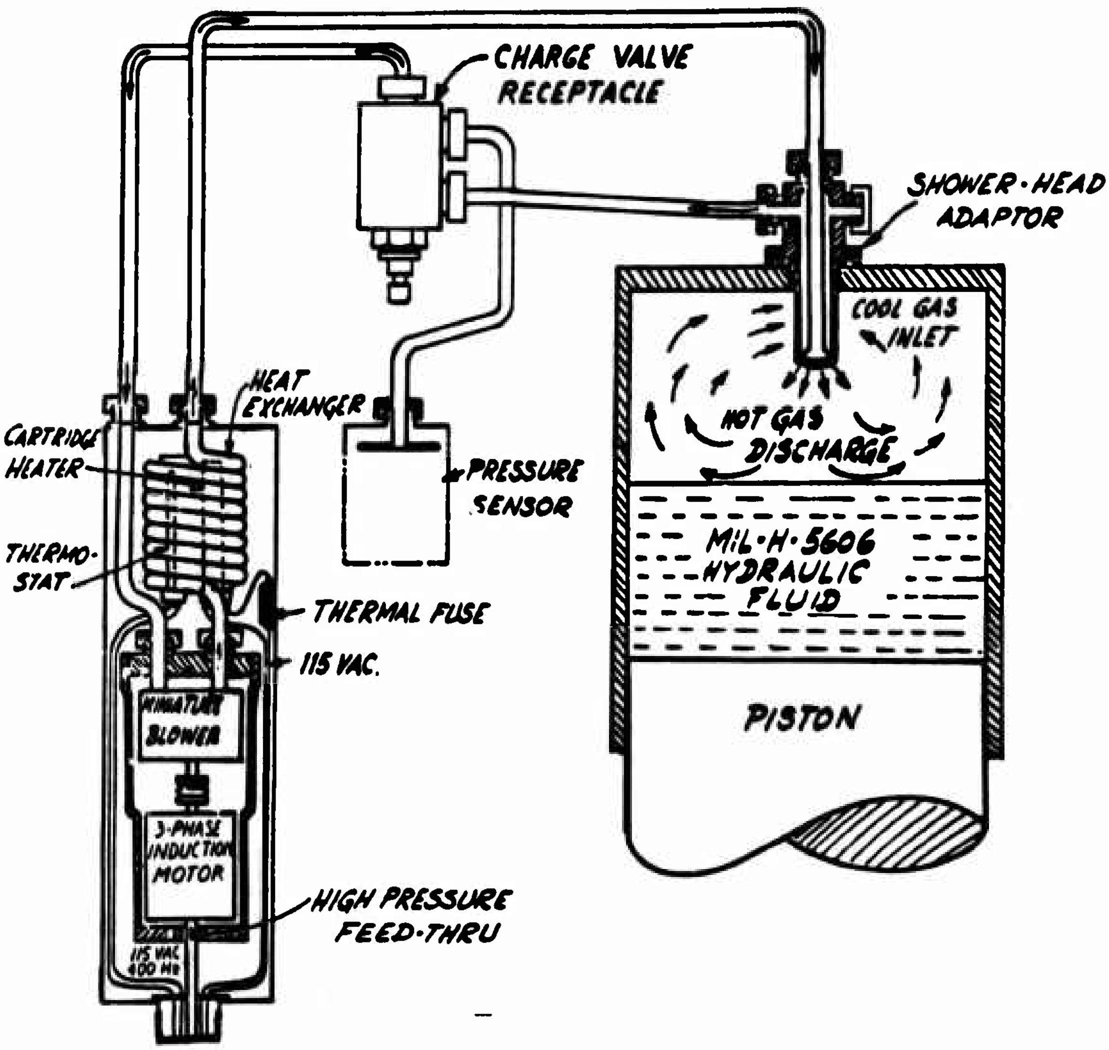


Figure 12. Variable Temperature Antistiction Approach
Reprinted from Reference 13 by permission
of the Society of Allied Weight Engineers, Inc.

3.2.4.5 Unsticking by Mechanical Means--The Zero-Friction Technique

Unsticking requires relative motion between the piston and cylinder of the oleo strut. The previous two techniques (incremental volume and variable temperature) achieved unsticking by operating on the internal state of oleo gas, varying its pressure sufficiently to overcome stiction. External means must overcome stiction by somehow applying a force between the cylinder and piston. (Taxiing imposes a forcing function on the wheels which in turn acts on the piston. Main rotor rotation imposes a forcing function on the fuselage which acts on the cylinder.)

The relative motion between piston and cylinder need not be axial; it can also be rotational (picture rotational oscillation of the piston within the cylinder). The latter idea has been investigated at the National Water Lift Company whose sister division, the Cleveland Pneumatic Tool Company, is a manufacturer of landing gear (C-130 and C-141, for example).[14]

Implementation of the technique requires redesign of the torque arm assembly on the landing gear strut. (See Figure 8 for a view of the torque arm assembly on the CH-47 main landing gear strut.) The torque arm controls the angular orientation of the piston with respect to the cylinder (i.e., it prevents the piston from rotating within the cylinder). To achieve unsticking, the torque arm must be replaced with a device that will cause the piston to undergo angular oscillation. The torque required to accomplish this is reported to be less than 10 percent of the scrubbing torque.

Hydraulic or electromechanical actuation schemes can be used. For example, one hydraulic solution would resemble a power steering design. Reference 14 shows an electromechanical solution for a C-141 oleo that appears quite attractive (see Figure 13). The CH-47 torque arm (Figure 8) could be similarly modified by replacing the cylindrical elbow joint by a spherical joint, where the lower arm contains the cupped retainers holding the sphere and the upper arm is connected directly to the sphere. The latter connection is made eccentric such that if the arm is rotated about its longitudinal axis (that axis remaining stationary with respect to the cylinder), the sphere is caused to drive the other torque arm in an oscillating manner.

One of the attractive features of the angular oscillation approach is that it reduces the axial stiction error to a theoretically negligible percentage. While the piston is being rotated within the strut, static friction cannot exist. But dynamic friction force acts in a direction to oppose motion. Thus, with no motion in the axial direction, there can be no dynamic friction force in that direction and stiction error is eliminated.

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14 Loren Isley and Erwin Hartel, "A New Approach to the On-Board Weight and Balance System," SAWE Paper No. 748, May 1969.

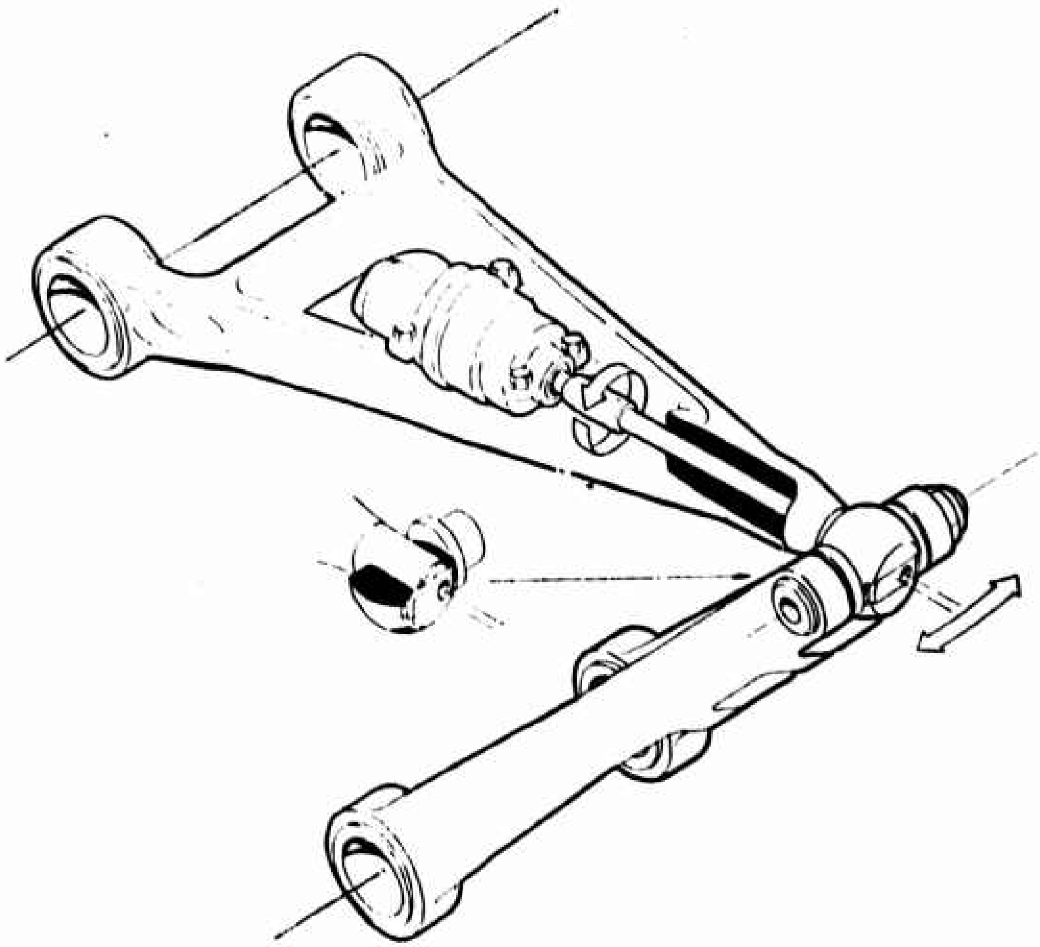

Figure 13. Incorporation of Electromechanical Actuator in C-141 Main Landing Gear Strut Torque Arm Assembly for Producing Rotary Oscillation of Piston. Reprinted from Reference 14 by permission of the Society of Allied Weight Engineers, Inc.

Reference 14 shows that on the order of 10 oscillations are required to reduce the stiction error to negligible proportions. The reference also includes laboratory test data for a C-141 landing gear. Before "stiction removal," errors as large as 5.7 percent are noted with an average error magnitude of 3.3 percent. After "stiction removal" (10 angular oscillations), the average error magnitude is 0.3 percent. In a personal communication, the author states that the measured error was within the tolerance of the test instrumentation (i.e., it could easily be lower than reported). In addition to the C-141, gears were tested for the C-130, 727, and the CH-47.

Mechanical devices that would act in the axial direction represent a more difficult design problem, especially in the case of retrofit application. One problem is that the axial displacement of the piston with respect to the cylinder is a variable. Another problem is that a single actuator acting on the cylinder and piston will produce a torque tending to bind the piston within the cylinder. No solutions have been advanced along this line.

3.2.4.6 Design of Unsticking Devices Integral to the Shock Strut

For future aircraft designs, and perhaps even for present-day aircraft, methods of incorporating the antistiction device within the oleo should be considered. The advantages are that the integral unsticking device would be sturdier, less subject to sand and dirt contamination, probably simpler and more reliable, and would reduce the exposure of hydraulic or electrical lines.

Integral designs have already been considered by some. For example, Boeing obtained a patent for some sort of integral approach for the SST, and Cleveland Pneumatic did some work along these lines several years ago.

It appears that the simplest integral approach is an incremental volume technique in which a piston would be driven by hydraulic or electrical means to displace a variable amount of hydraulic fluid or air within the oleo. Somewhat more difficult would be an integral drive to produce angular oscillation of the piston with respect to the cylinder, or vice versa.

3.3 STRAIN GAGE WEIGHT MEASUREMENT APPROACHES

3.3.1 Unsuccessful Retrofit Applications

Strain gage weight and balance systems failed to live up to their potential until they were incorporated into the aircraft in the design stage. No examples of successful retrofit application of strain gage approaches for integral aircraft weight measurement have been found.

In a side-by-side test of the STOW system (a strain gage system built by National Water Lift Company and incorporating strain gage sensors in the axles) and the STAN system (Fairchild's oleo pressure system) installed in the CH-47, the strain gage system appeared significantly inferior in accuracy, installation, calibration, and maintainability.[2] Similar results appeared for a side-by-side test of the STAN system and an in-axle strain gage system built by BLH Electronics.[15]

Several years ago, the USAF had its C-130 fleet equipped with a production strain gage weight and balance system ("STOW", System for Takeoff Weight) manufactured by National Water Lift Company. The weight sensors consisted of strain gage sensors mounted in each axle. According to engineers who worked for National at the time, the combination of maintenance problems and the not-absolutely-essential character of the system caused the USAF to remove the systems from the C-130 fleet. One problem was that if an axle was replaced or a sensor failed, the weight and balance system (WBS) would not be put into working order until periodic inspection (which could represent an elapsed time of several months to over a year) because of the skill levels and equipment required.

Another problem was that calibration of the system was quite complicated, requiring variable weight on the gears to obtain the required multipoint calibration. It appears that part of this problem stems from the sensor design as illustrated in Figure 14. The sensor consists of two separate parts; one part consists of a cantilever beam and mounting collar. The beam is instrumented with strain gages and the tip of the beam rests on an "anvil," which is physically separate and attached to its own mounting collar. The beam and anvil are mounted inside the axle. The cantilever beam is designed to measure the deflection due to shear displacement and the vertical component of the bending displacement. As load is applied to the landing gear, the beam and anvil will tend to separate. Therefore, it is necessary to preload the beam against the anvil so that at maximum loading the beam is still deflected by some amount. The sensor was installed to obtain a preload in terms of a specific sensor output, but one would expect that the accuracy of this adjustment would not be very good.

ELDEC Corporation and BLH Electronics both maintain that an axle deflection sensor that senses bending deflection will suffer inaccuracies due to the fact that bending stresses are also induced by side and drag loads and that bending stresses depend on the moment arm of the load as well as its magnitude. Their sensors are therefore designed to sense only shear deflection.

3.3.2 Current Production System for the DC-10, 747, and L-1011

Currently the strain gage systems on the DC-10 and 747 (built by BLH Electronics) and the L-1011 (built by ELDEC Corp.) are the only weight and balance systems offered on off-the-assembly-line aircraft (though optional equipment). For both the 747 and L-1011, the specification accuracy is ± 1 percent. Published test data and contact with the airframe manufacturers show that these systems are achieving this level of performance.

15 Edward Low, "Testing of Two Integral Weight and Balance Systems on the C-7A," SAWE Paper No. 881, May 1971.

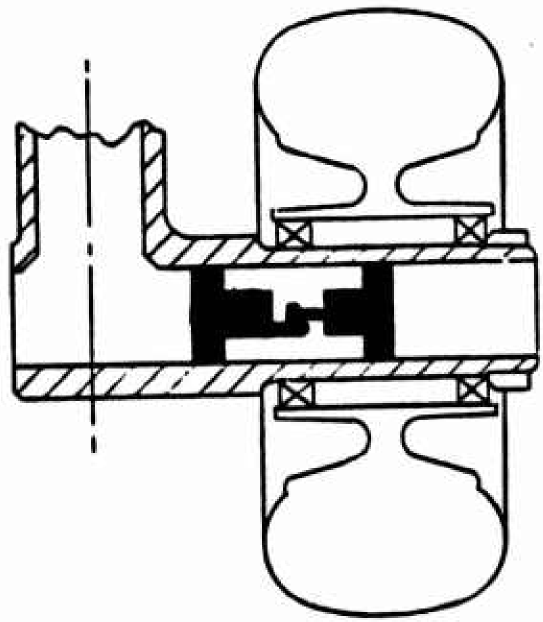

Figure 14. Configuration of Strain Gage Deflection Sensor Used in the STOW Weight and Balance System (cantilever deflection beam is on right half of sensor; "anvil" is on the left half of sensor).

The apparent dichotomous results between the retrofit systems and original equipment systems are due to two main factors:

(1) The sensor-landing gear matching problem is greatly simplified by designing the landing gear to facilitate strain gage sensor installation.

(2) Achieving a reliable, maintainable, stable, accurate strain gage sensor installation is an art demanding an evolutionary design cycle for each aircraft type.

The first characteristic is most aptly demonstrated by the L-1011 system where the strain gage sensors on the main landing gear are mounted to lugs on the bogey beams that are part of the original castings. A Lockheed representative states that the ELDEC system exhibits excellent stability (long-term repeatability) after several landings during which the readings "settle in".

The system on the 747 employs strain gage sensors mounted on the inside of each axle (for a total of 18 sensors). The design artistry required in achieving successful system operation is demonstrated by the evolution of the 747 strain gage sensor installation design. Stability of the original sensor installation was poor, but after a period of time the problems were isolated in full-scale laboratory tests of the landing gear sensor installation. Boeing and BLH engineers arrived at a combination of materials and procedures that produced a stable, drift-free installation. It is interesting to note that with a change in the design of the landing gear for a new short-body version of the 747, problems are again being experienced with the weight and balance system and Boeing engineers speculate that a change in surface finish of the axle interior may be responsible. Thus, another design-development iteration may be required to achieve within-specification performance of the system for this new model of the same aircraft.

The fact that the strain gage systems are capable of meeting performance objectives (after some growth pains) is nowhere better illustrated than in the "primary certification" of the 747 WBS by the German equivalent of the FAA. This denotes recognition of the accuracy and repeatability of the system and allows its use to satisfy weight and balance checking of the aircraft with the system in place of manifest calculations.

Some insight into the design considerations for strain gage weight sensors may be obtained by a simple comparison of several characteristics of the BLH and ELDEC approaches:

Mounting--In the BLH approach, the strain gage transducer is mounted inside axle by means of expansion collets. One transducer for each wheel (axle). Optional equipment for Boeing 747 and DC-10 aircraft. In the ELDEC approach, the strain gage transducer is mounted to attachment lugs that are integral to each horizontal bogey beam (or nose wheel axle). One transducer for each wheel. Optional equipment for Lockheed L-1011 aircraft and similar to C-5A systems.

Installation--Installation of ELDEC transducer is simpler. BLH sensor requires special tools, and installation factors (torques, alignment, etc.) are probably more critical and more difficult to control.

Reliability--BLH sensor is in more protected location. Damage to ELDEC sensor (in some cases leading to moisture-induced problems) due to abuse has been a problem in the past (substantially solved now). BLH sensor more susceptible to damage during wheel/axle changes and during installation. Wiring is about the same for both.

Long-Term Stability--BLH sensor appears to be more susceptible to design-induced problems (e.g., very much an empirical process to arrive at a drift-free installation; also subject to greater temperature cycling); whereas ELDEC sensor appears more susceptible to damage-induced problems. Both airframe manufacturers currently claim adequate (good) stability (i.e., sufficient to support 6 months or greater calibration intervals).

Calibration--Should be essentially the same.

Accuracy--Reported test data for both systems indicate apparent capability to meet specification accuracy ± 1 percent for gross weight and (less certain) ± 1 percent for MAC.

Complexity--The sensing elements of the two sensors are foil strain gages and therefore of equal complexity. The BLH sensor is geometrically more complex due to the method of mounting (expansion collets).

Maintainability--ELDEC sensors appear to be much more easily replaced. BLH sensor location in axle could mean that its installation would be affected by tire, brake, and certainly axle changes.

3.3.3 Outlook for Helicopters

Because of the successful application of strain gage sensors to modern-day aircraft, it can be concluded that this approach can also succeed for helicopters provided that the flexibility of redesigning landing gear axles to accommodate or facilitate installation of strain gage sensors is allowed. Even with this provision, it must be recognized that the state of the art is such that some degree of redesign and development is to be expected following installation of the first production prototype.

3.4 RESIDUAL THRUST

With the main rotor(s) turning at minimum collective, sufficient thrust is generated to introduce sizable error into the measurement of gross weight in terms of weight on wheels. To enable gross weight measurement in this dynamic condition, the residual thrust must be measured. There are two ways of doing this: direct measurement of the rotor thrust (by means of strain gages installed on load carrying structure, for example) or by estimating the rotor thrust by means of its aerodynamic characteristics.

Both of these approaches have been tried. Higgens[2] reports the results of tests of the STAN and STOW systems (see Section 1.1). These developmental weight and balance systems installed on a CH-47 used aerodynamic estimation of residual thrust. Dybvad[4] reports the results of testing another developmental weight and balance system for the CH-47. This system, manufactured by ELDEC Corporation, employed strain gages installed on transmission housings to attempt to measure the forces developed by the main rotors.*

Based on review of the above efforts, analysis of the aerodynamic relationships involved, and review of operating procedures, it is concluded that the rotor lift estimation approach is a sufficiently accurate and viable method of computing residual lift under restricted application conditions (collective and cyclic controls placement and wind velocity). It appears to offer better achievable accuracy than the strain gage rotor force measurement pursued by ELDEC in Reference 4. However, no matter what approach is used to measure or estimate rotor residual thrust, there is an additional problem associated with the weight on wheels measurement for this dynamic condition. That is, it appears that large errors may have occurred in the weight on wheels measurement during dynamic conditions (rotors turning on ground), particularly in the case of the oleo pressure measurements. In the ELDEC program, the reported test data may indicate the occurrence of large error in the aft gear weight measurement (oleo pressure in aft gear shock struts). In neither of the two cited CH-47 weight and balance system test programs is the cause of errors adequately pinpointed. In both programs, accuracy was found to be adequate in the static mode but was unacceptable in the dynamic mode; in both programs the measurement(s) causing the unacceptable error were not sufficiently diagnosed, but unpredictable residual thrust does not appear to be the culprit.

3.4.1 Residual Thrust Characteristics

During unloading or loading operations on the ground, the helicopter's rotor can be turning at up to full speed. The collective control will be at minimum during the condition, but the thrust generated (the "residual" thrust) is significant. For example, the nominal thrust generated by the CH-47C with the collective in the 3° detent is 6,000 lb. This is 13 percent of the maximum gross weight capability of 46,000 lb.

*These are the only known attempts to deal with residual thrust

Discussions with operational personnel have established both the routineness of the condition and the desirability of gross weight measurement capability in the presence of residual thrust. The attractiveness and usefulness of an LPI would be diminished if it provided inaccurate information during residual thrust conditions.

For a given collective pitch angle, the nondimensional thrust of a rotor is constant at hover. That is,

$$C = \frac{T}{\rho A (\Omega R)^2} = \text{constant for fixed collective pitch} \qquad (4)$$

Therefore, at minimum collective if rpm (Ω) is relatively constant, the residual thrust is a function only of atmospheric density.

The residual thrust for the CH-47C, for example, is approximately

$$T_R = 6000 \; \sigma \text{ lb} \qquad (5)$$

where σ is the density ratio (= 1.0 at sea level).

Moreover, the overall effect of this thrust on the weight supported by the landing gear is constant with moderate variations in the cyclic control as shown in the test results reported by Higgens.[2] Movement of the cyclic, however, has a strong influence on c.g. location as calculated from weight-on-wheels measurements. The sensitivity is about 15 in. c.g. movement per inch of cyclic stick position movement from neutral for the CH-47. Desired accuracy for c.g. computation is on the order of ± 1 in., and no more than about ± 5 in. can be tolerated.

Obtaining repeatable values of the cyclic stick position does not appear to be a problem. For example, on the CH-47 there is a cyclic stick position indicator. Initial calibration of the system for both minimum collective and neutral cyclic effects might be necessary to achieve high accuracy.

3.4.2 Means of Handling Residual Thrust

Table 6 summarizes the effects and measurement possibilities at various rotor conditions. The static condition presents no problem.

The ground idle condition does not present much of a problem either, but this mode is transitory (limited duration) and it is probably not reasonable to assume that the vehicle could be forced into this mode for weight measurement. The flight idle mode is common to present operational procedure. That is, under normal operating conditions, if the vehicle is powered on the ground during loading or unloading (or during gross weight measurement by an LPI system), the flight idle power level would be used. It therefore appears reasonable to assume the existence of this mode for the purpose of aerodynamic weight measurement. The last condition covers

TABLE 6

SUMMARY OF WEIGHT AND C.G. MEASUREMENT FEASIBILITY
AT VARIOUS ROTOR CONDITIONS

1. STATIC

Condition: Rotors stopped

Residual thrust: None

Compensation: None required

2. GROUND IDLE

Condition: Engine power lever at ground idle
Collective at minimum
Cyclic and pedals at neutral

Residual thrust: Very low (about 1 to 3 percent of maximum thrust)

Compensation: Nominal compensation adequate. Can probably tolerate some degree of off-centered cyclic (limited experimental evidence).

3. FLIGHT IDLE

Condition: Engine power lever at flight mode position
Collective at minimum
Cyclic and pedals at neutral

Residual thrust: Significant (greater than 10 percent of maximum thrust).

Compensation: Aerodynamic compensation technique is theoretically adequate, but was unsuccessful in test case. Direct measurement is a difficult empirical task. For aerodynamic case, cyclic must be within $\pm 1/4$ in. of neutral position for adequate c.g. accuracy.

4. NONMINIMUM COLLECTIVE

Condition: Engine power lever at flight mode position
Collective at any position

Residual thrust: Any magnitude

Compensation: Aerodynamic compensation is not feasible. Feasibility of direct measurement requires experimental determination.

the situation where the collective can be at any value on the ground. Aerodynamic compensation is not feasible for this condition (and direct rotor thrust measurement is also likely to be impossible).

Thus, there are three ways of handling residual thrust:

(1) Exclude the residual thrust mode from the permissible modes of system operation.

(2) Compensate for variations introduced by residual thrust by direct sensing of rotor thrust effects.

(3) Compensate for variations introduced by residual thrust by estimating the residual thrust based on rotor aerodynamic relationships.

The impact of simply ignoring residual thrust would significantly decrease the desirability of the system. It is however not considered an intolerable alternative. For example, under present normal operating procedures, the engines are shut down during unloading and loading operations; of course, combat operations could easily change this.

The ELDEC program[4] is the only reported attempt to measure rotor residual thrust (as opposed to aerodynamic estimates). The approach taken was to attempt to measure structural stresses induced by the rotor in transmission components. The attempt was unsuccessful, with the average error in gross weight introduced by the rotor lift sensor exceeding 10,000 lb. The errors were primarily associated with temperature variations of the structures instrumented; with a calibrated compensation for temperature effects, the errors were greatly reduced. The temperature effects were large, time varying, and not exceptionally repeatable. It was concluded that oleo pressure sensing worked fine for static conditions, but for dynamic conditions, "accurate rotor lift measurement remains an elusive goal."

It is likely that successful residual thrust measurement would also depend on repeatable positioning of the cyclic to neutral position, since a thrust vector change would alter the stress pattern. Also, with a single-rotor helicopter, it would probably be necessary to assume a fixed direction of the residual thrust vector with respect to the fuselage (measuring magnitude has proved very difficult; adding direction as an additional requirement would appear untimely). Thus, repeatable positioning of the cyclic control appears necessary for the measurement approach.

Successfully estimating residual thrust requires that the thrust be a precise, repeatable function of a minimum number of measurable variables. The rotor nondimensional thrust equation given earlier is a valid function and identifies several principal variables. In order to limit the required number of measurements for an accurate estimate, the conditions under which residual thrust would be estimated must be controlled:

(1) Collective at minimum

(2) Cyclic at neutral

(3) Rotor rpm at 100 ± 1 percent

(4) Winds less than 20 to 25 knots

The rotor thrust measurement approach of Reference 4 was apparently predicated on the observation that "Prior development programs have shown that lift cannot be predicted accurately on the basis of pitch settings and rotor rpm." This statement is apparently based on the test results reported by Higgens (side-by-side test of two weight and balance systems for the CH-47). After analyzing those test results, it is difficult to conclude that they show that residual thrust cannot be accurately estimated. The results show loss of system accuracy under dynamic conditions (larger gross weight and c.g. errors), but the source of the inaccuracy is not identified. Moreover, the errors were associated with operation in winds. With the rotors turning and at zero wind, the accuracy of each of the two systems was within 1 percent of actual gross weight. Figure 15 shows a portion of applicable test results from the subject test. In the illustrated case, the cyclic stick position was varied. Note the results for the Fairchild system (oleo pressure sensing approach). With the lift correction applied (unblackened circle vs blackened circles), the gross weight indication is very close to the actual weight throughout the cyclic variation. Computed c.g., on the other hand, varies widely because there is no correction for cyclic position. This figure illustrates the accuracy of the residual lift estimation, the insensitivity of the gross weight measurement to cyclic variations, and the high sensitivity of the c.g. computation to cyclic variations.

With an accurate residual thrust estimate, the accuracy of an oleo pressure sensing approach should actually improve with residual thrust because the dithering provided by the rotor forces should free the oleos from stiction effects. This appears to have been observed in the above-mentioned test program.

The principal deficiency noted in the test program was that the accuracies of both systems degraded severely in the presence of moderate winds of 5 to 20 knots. The oleo pressure system tended to be in error on the low side, but not consistently. The strain gage system exhibited more random and less severe variation. There was no overall pattern suggestive of a rotor thrust variation with wind direction and velocity.

The wind, of course, would have affected rotor thrust. Appendix B shows the combined effects of ground proximity and relative wind on rotor thrust. When the helicopter is close to the ground, the combined effects of the wind and ground proximity remain relatively constant up to a certain velocity. This behavior has been observed for the UH-1 (see Appendix B). Wind effects are independently quite large, but in the vicinity of the ground as wind increases, the helicopter tends to move off the "bubble"

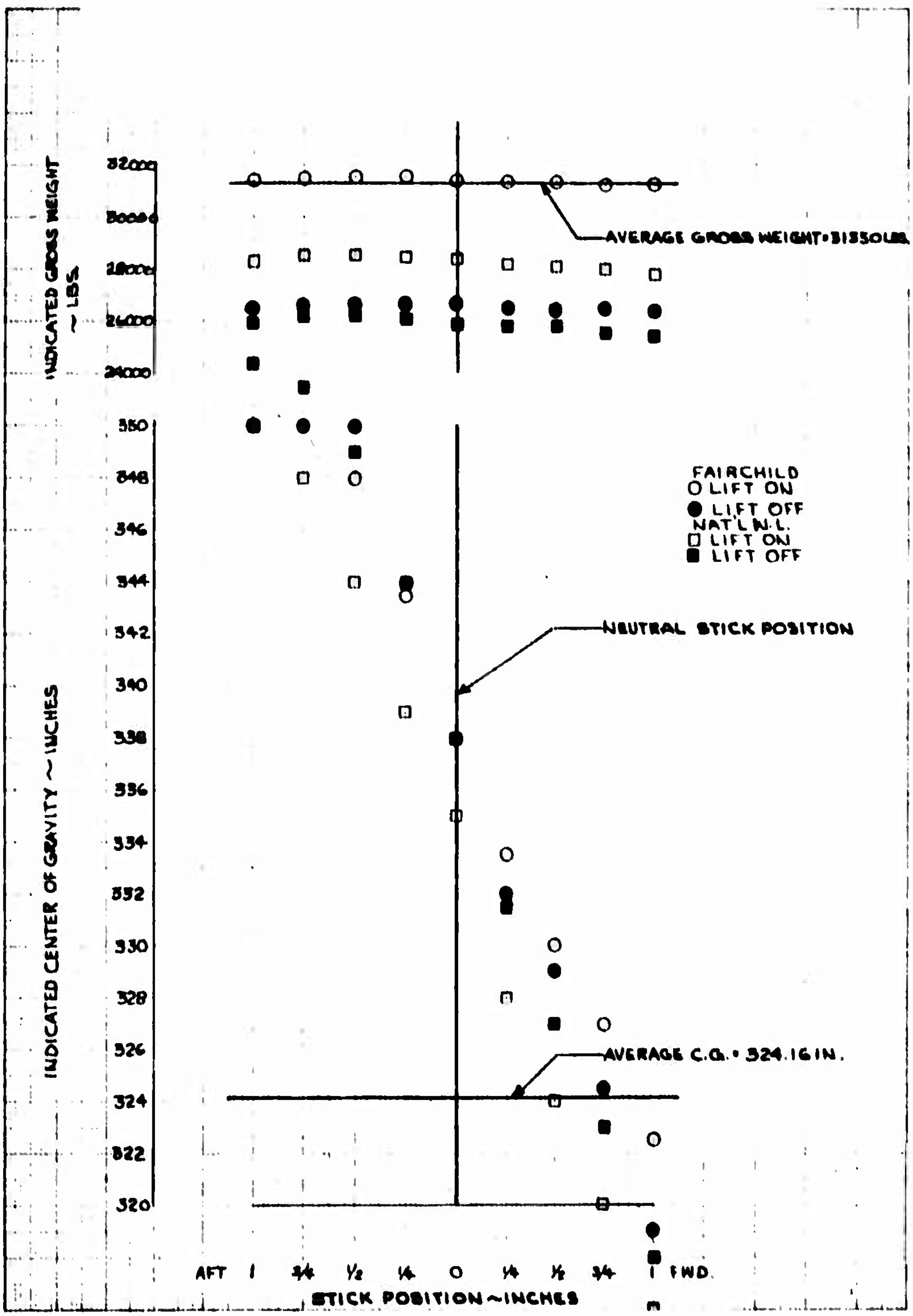

Figure 15. Variations of Indicated Gross Weight and Center of Gravity With Longitudinal Stick Position.

with a consequent reduction in ground effect tending to balance the increased thrust due to the wind increase.

In the case of residual thrust, wind effects would be further scaled down, and there would be little change in the residual thrust with wind variations below a certain magnitude (10 to 20 knots, perhaps more).

In the tests of "dynamic" accuracy of the ELDEC weight and balance system reported in Reference 4 after correcting the rotor thrust measurements for errors due to temperature variations, the indicated gross weight was still in error by an average of -4500 lb (with an average residual thrust after correction of 5900 lb). It was concluded that the remaining, still large error was due to some other unknown variation in the rotor thrust measurements. Another explanation is that a problem was experienced with the aft landing gear weight measurement during dynamic conditions. This explanation is suggested by comparing the indicated aft gear load that would be experienced if the two rotors each supplied half of the nominal residual thrust. There were seven test cases at different gross weights and c.g.'s. All of the forward gear loads were within 1000 lb of predicted, but the aft gear loads were as much as -5600 lb from predicted (and averaging -4000 lb different). Moreover, the pattern of variation from predicted matched the pattern of variations of the gross weight and c.g. errors. The predicted values of the aft gear loads were substituted in place of the measured load, and the gross weight and c.g. errors recalculated. With the measured load the errors were (average ±standard deviation) gross weight -4500 ±1100 lb, c.g. -36 ±15.7 in. With the predicted aft gear load, the errors were gross weight -500 ±600 lb, c.g. -5.4 ±10.6 in. The fact that the average and standard deviations of the errors are substantially reduced supports the contention that a problem was experienced with the aft gear measurement.

In summary, estimation of rotor residual thrust based on rotor aerodynamics appears feasible. The relationship employed is the same as that implemented in the Aerospatialle manual collective pitch computers for the Alouette, Lama, and Puma helicopters (see Appendix C). There appears to be other problems associated with the "dynamic" mode that may prevent accurate gross weight and c.g. measurements during that mode. In past test programs involving the CH-47 the problems were apparently not sufficiently diagnosed. It is likely, therefore, that in any further pursuit of weight measurement for the CH-47, the problems will be encountered again. It is recommended that the resolution of this question be included as an objective of any LPI system development and test program involving the CH-47.

4. CALCULATION OF PERFORMANCE CAPABILITIES

4.1 SUMMARY

This section develops procedures recommended for estimating helicopter performance capabilities for lift performance indication. Equations for calculating vehicle weight and c.g. from weight-on-wheels measurements are described in Appendix D.

The LPI system will generate the same information as that in the performance section of the operator's manual relative to takeoff and landing capabilities (basically hover, climb, and obstacle clearance capabilities). In the operator's manual, nominal engine and aircraft characteristics are combined into relatively complex family-of-curve-type graphs. In the LPI system, engine and vehicle characteristics will be stored separately in the form of normalized and nondimensional functions (from which the operator's manual charts are derived). This method results in more simple representation and computational flexibility and allows the use of calibration constants to adjust nominal characteristics to a specific vehicle as necessary. The LPI performance estimates are based on the measurement of ambient conditions and gross weight.

The key technical issues are the methods of dealing with variations in engine and rotor performance characteristics due to factors such as deterioration. The most practical method of adjusting stored engine performance characteristics to account for such variations is to employ a calibration constant derived from the topping (maximum power) check presently performed routinely and on condition. It appears that the installed average performance characteristics of the representative main rotor may be used without incurring excessive error and that significant performance deterioration is both correctable and signaled in advanced by vibration and lowered V_{ne}.

4.2 BASIC PERFORMANCE COMPUTATION PROCEDURES

4.2.1 Operator's Manual Approach

It is instructive to consider the performance computation approach in the typical operator's manual for a U.S. Army helicopter (the approach is basically the same for all helicopters). Consider the method employed to determine hover capability. Figure 16 presents the chart used to determine maximum hover capability for the CH-47B for dual-engine operation. (This chart is similar to the one of the CH-47C and includes an example of its use).

The hover capability chart is used to determine the maximum gross weight at which the vehicle will hover. The use of the chart in Figure 16 is illustrated for the following conditions:

Model: CH-47B
Data Basis: Flight Test
Date: August 1967
Engines: (2) T55-L-7B
Fuel Grade: JP-4
Fuel Density: 6.5 lb Gal

HOVER CAPABILITY (230 RPM)

Notes: 1. Use military power for takeoff.

2. This chart suitable for operation at 225 rotor rpm.

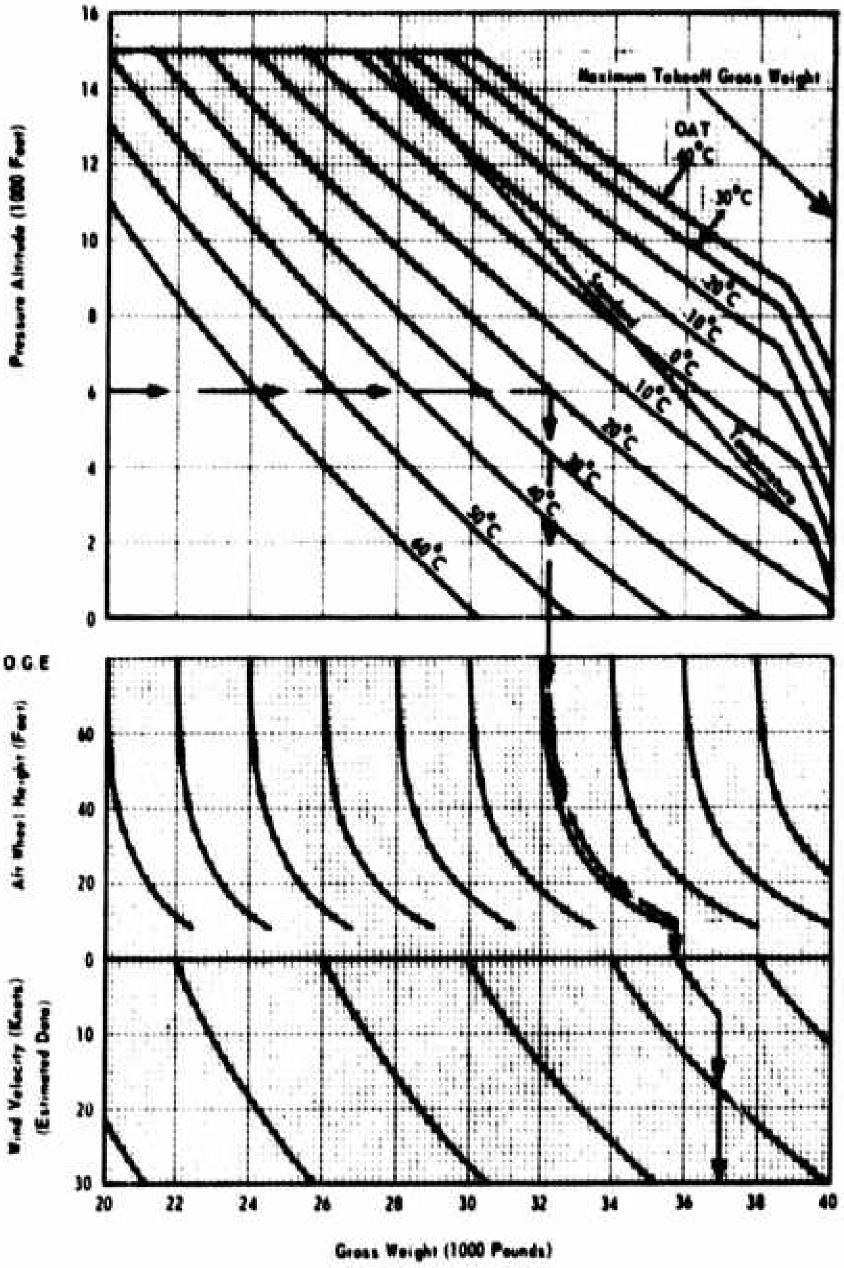

Figure 16. Hover Capability Chart for CH-47B.

6000-ft pressure altitude

20°C ambient temperature

10-ft aft wheel height above ground

18-knot wind velocity

Beginning on the left side of the chart at the indicated pressure altitude, trace right until the curve for the correct ambient temperature is encountered. Then trace downward to the ground effect curves. Then trace parallel to the nearest ground effect curve until the correct aft wheel height is reached. The example shows that this same paralleling procedure is followed with respect to wind velocity to arrive at the final gross weight capability.*

Having determined the maximum gross weight that can be sustained in hover, the pilot can compare this to the estimated or planned actual gross weight to get an idea of the relative capability of the vehicle to execute that maneuver. This procedure is representative of the general flight planning process for takeoff and landing. For each of the key maneuvers involved in various takeoff and landing modes, there is a performance chart chart for determining the maximum gross weight at which the maneuver can be executed for the prevailing ambient conditions. For some maneuvers, most notably obstacle clearance takeoffs, the chart shows the capability of the vehicle in a different form; for example, the distance required to clear a 50-ft obstacle for the estimated vehicle weight and ambient conditions.

The above procedures are applicable to all Army helicopters. The fact that they are not used very much is due to several factors (discussed in Section 2.9), including the lack of reliable weight information and the inconvenience of the performance charts in the cockpit.

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* This procedure is invalid. Appendix B shows that the effects of ground proximity and wind velocity are not independent. Indeed, the combined effect tends to remain constant for low vehicle heights up to a wind velocity of about 20 knots where the effect of wind velocity begins to predominate. Thus, the procedure in Figure 16 yields an additional 1000 lb hover capability that very likely does not exist. Moreover, due to the interaction of wind and ground effect, the wind could have a negative contribution for the illustrated conditions. This situation, however, is not a typical flight test objective and no data for the CH-47 can be referenced to confirm the above. Note that the wind velocity correction in Figure 16 is denoted "estimated data." This is an example of an area where additional flight test data are needed to support detailed design of LPI functions. The continued existence of the chart in Figure 16 with its erroneous procedure may indicate infrequent use of these charts.

4.2.2 Lift Performance Indicator Approach

In terms of the information to be provided, there is no difference between the LPI approach and that of the operator's manual. To meet the basic objectives for the system, it is necessary to provide the same information with regard to takeoff and landing capabilities that is provided in the performance section of the operator's manual. In the case of hover capabilities, for example, the system will compute the hover gross weight capability for the measured ambient conditions and will measure actual gross weight. These quantities can be displayed directly, or a weight margin can be displayed consisting of the gross weight capability minus the actual weight, with a separate mode for the display of measured gross weight and c.g. The information is the same in both cases (display design was not a part of this study, and material relating to that area in this report is for illustration only). The form in which the information is displayed is immaterial to the system design at this point.

Thus, the method of derivation of the information, not the type of information, distinguishes the LPI system from the operator's manual approach for generating lift performance information.

There are basically two ways to mechanize the calculation of the performance capabilities for hover such as was illustrated in Figure 16. One approach is to directly program the graphs from the operator's manual so that in effect those procedures are exactly duplicated. This has been done on a limited basis in a test case.[16] This approach is unattractive because it is inflexible and unnecessarily complicated. It is much more convenient to use the nondimensional performance characteristics of the airframe and engine from which the operator's manual graphs are derived.

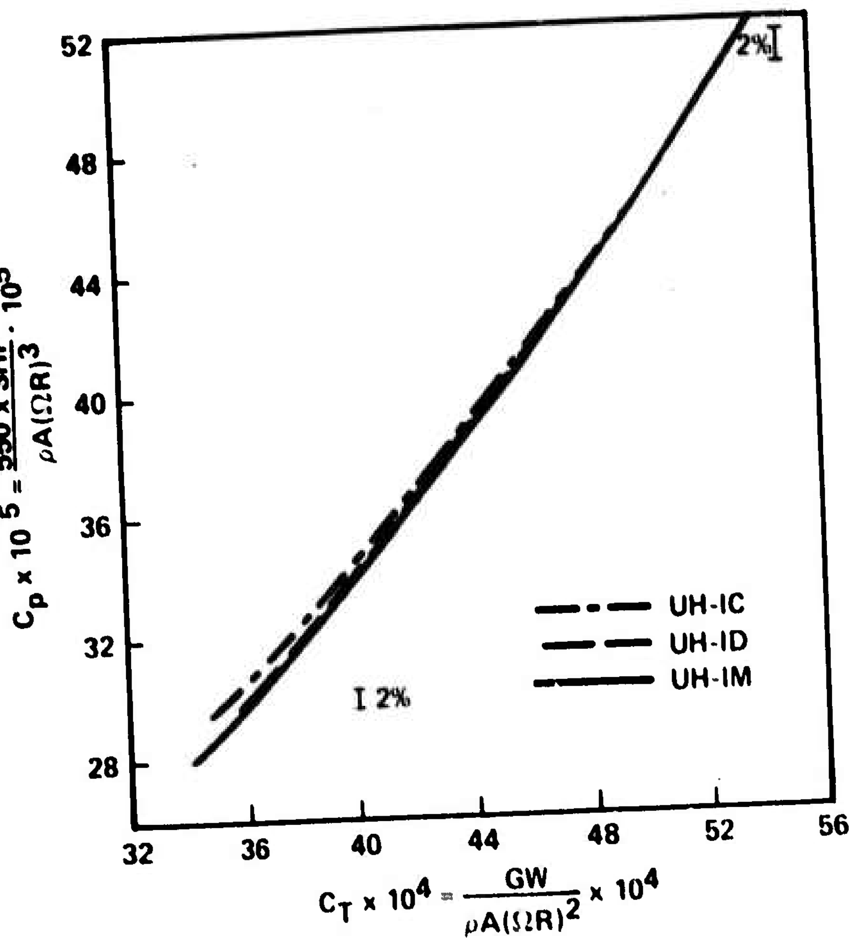
Figure 17 shows the nondimensional HOGE performance characteristics for three UH-1 models. For a given rotor speed, the nondimensional power coefficient is a function only of shaft horsepower and density and determines a corresponding value of the nondimensional thrust coefficient. Thus, for a specific combination of maximum power available, rotor speed, and air density the nondimensional characteristics in Figure 17 define the HOGE gross weight capability. With the addition of a relationship defining maximum power available, the curve in Figure 17 replaces a family of curves similar to the ones shown in Figure 16. (Nondimensional HOGE characteristics for the UH-1H, CH-47C, and CH-54B are shown in Appendix A.)

The form of the required maximum power available relationship is shown in Figure 18 for the helicopter-engine combinations reviewed in this study:

UH-1H: T53-L-13B engine

CH-47C: T55-L-11A engine

[16] E. E. Eloe and R. T. Scott, "A Helicopter Flight Performance System Using An LSI Processor," Naval Postgraduate School, June 1973.



C_T = Nondimensional Thrust Coef.

C_p = Nondimensional Power Coef.

A = Rotor Disk Area

ρ = Air Density

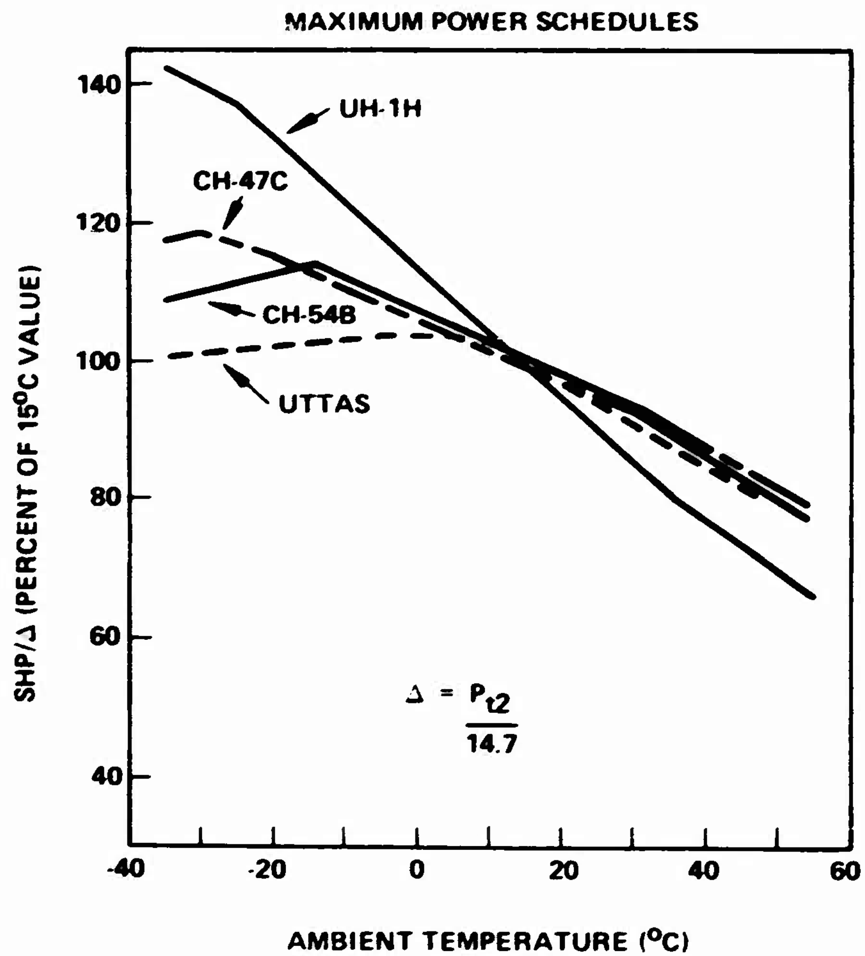
Ω = Rotor Rotational Velocity

R = Rotor Radius

SHP = Shaft Horsepower

GW = Gross Weight

Figure 17. Nondimensional HOGE Performance Characteristics for Three UH-1 Models.



TRANSMISSION LIMITS TORQUE TO CONSTANT VALUE BELOW APPROXIMATELY

10,000 FT* UH-1H

7,000 FT* CH-47C & CH-54B

* DENSITY ALTITUDE

AT CONSTANT TORQUE SHP VARIATIONS WITH ROTOR RPM: LESS THAN ± 1%

WHERE LIMITED BY SCHEDULE, PRESSURE AND TEMPERATURE ERROR EFFECTS

PRESSURE 1%/%

TEMPERATURE 1%/°C UH-1H

0.5%/°C OTHERS

SCHEDULE ACCURACY: UH-1H = ± 2%, CH-47C = ± 4%, CH-54B = ± 3% (REPRESENTS TRIM OR TOPPING TOLERANCE)

Figure 18. Maximum Power Scheduling by Engine Controls.

CH-54B: T73-P-700 engine

UTTAS: T700-GE-700 engine

The power schedules shown in Figure 18 are implemented by the respective engine fuel controls indirectly through their speed governors. Adherence to these nominal schedules depends on several factors that will be considered later. As indicated in the figure, when the shaft horsepower output is normalized by the engine inlet pressure, the action of the engine controls can be represented by a single curve.

An additional limitation of maximum available power that must be considered is the transmission power or torque limit. The relation of this limit to the otherwise available power is illustrated in Figure 19, which is a maximum available power chart for the CH-47C (T55-L-11A engines). In Figure 19, the family of curves with temperature as a parameter is the equivalent of the single curve for the CH-47C in Figure 18 (and again illustrates the simplicity possible using normalized or nondimensional characteristics). The curves show the power output for one engine. Superimposed on the chart is the transmission limit for dual-engine operation. Note that this is a simple fixed limit. The second, higher limit on engine output is the engine torque limit which is applicable in the case of single-engine operation of the CH-47C.

This transmission limitation situation is applicable to all helicopters, since it is a general design trade-off. The affect of this limitation on performance capability produces a distinctive characteristic that can be seen in Figure 16. Almost all of the curves in the figure have "knees" at about 39,000 lb weight. To the left of the knees, gross weight capability increases relatively rapidly with reduced altitude at a given temperature (due to increased pressure). At the knee, engine power is exactly equal to the transmission limit, so no further increase in power occurs to the right of the knee as altitude is reduced; the result is that the gross weight capability increase is less rapid, since it is due to the increased air density only. Figure C-3 in Appendix C illustrates this characteristic for the UH-1H.

4.2.3 LPI Calculation Routine for HOGE Capability

Figure 20 illustrates the incorporation of these characteristics into a calculation routine for an LPI system. The principal parameter measurements are shown on the left side and are ambient temperature (T), ambient pressure (P), and vehicle gross weight (W). Maximum available power from the engine is calculated in a two-step process. The action of the engine controls is represented by the schedule of normalized power vs ambient (engine inlet) temperature. Actual maximum available engine power is then formed by multiplying by measured ambient (engine inlet) pressure. A calibration constant is introduced at this point to account for variations of the engine from its nominal trim schedule (the rationale for this procedure is described later). The transmission torque limit is represented in the figure by a graph indicating that maximum available power is equal

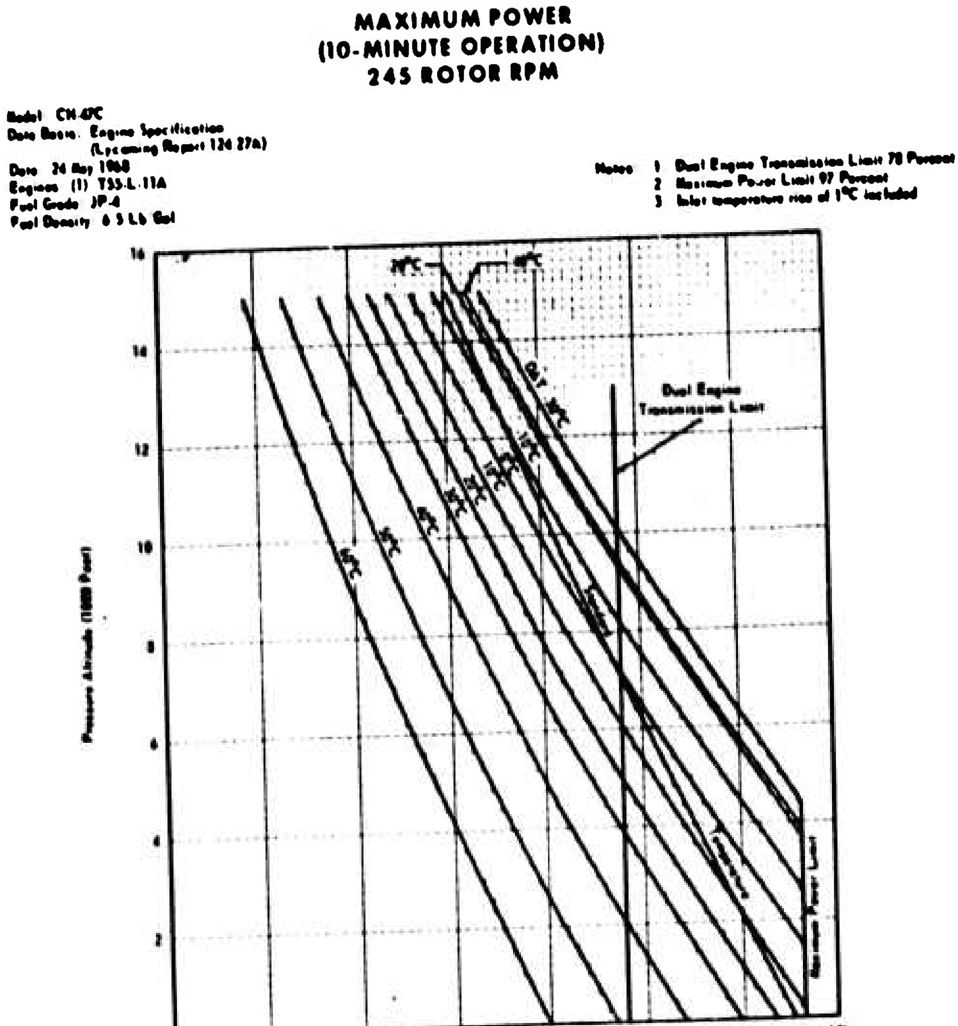
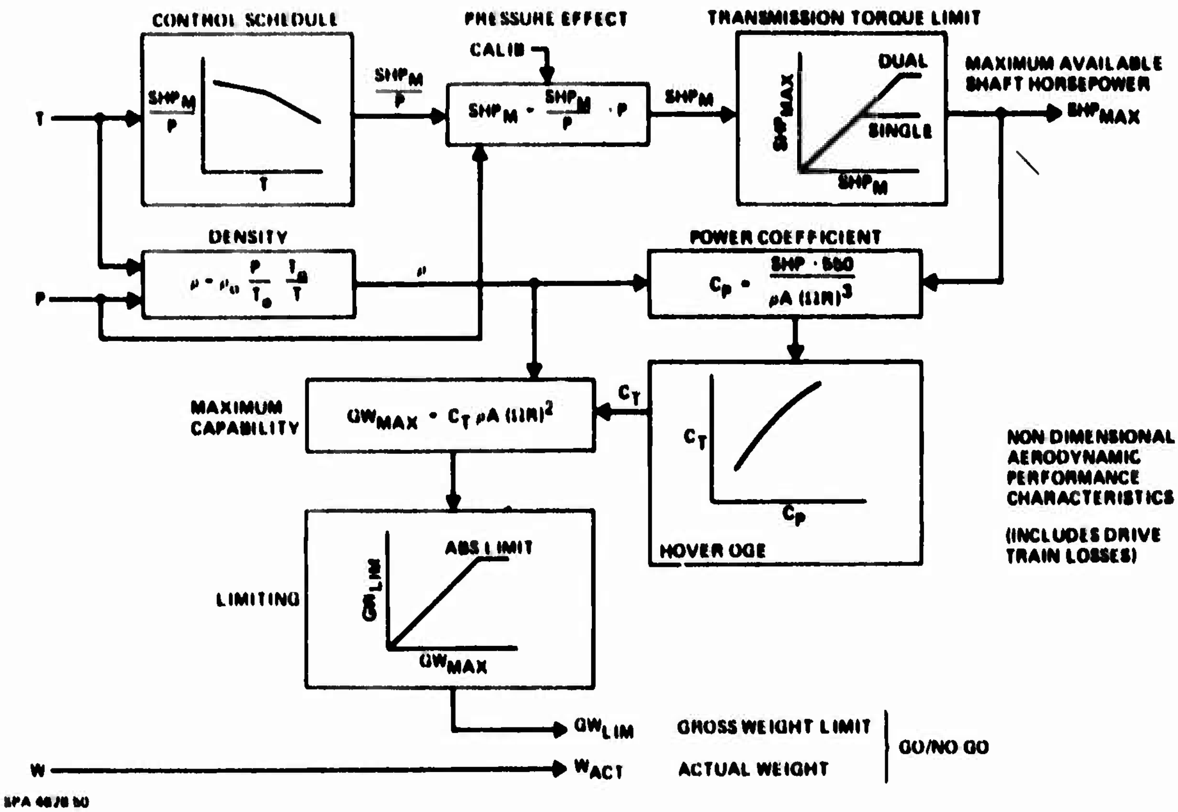


Figure 19. Maximum Available Power Chart for CH-47C, T55-L-11A Engines (From CH-47C Operator's Manual).



SHP_M = Maximum Available Engine Shaft Horsepower

C_T = Nondimensional Thrust Coefficient

C_P = Nondimensional Power Coefficient

A = Rotor Disk Area (a constant)

R = Rotor Diameter (a constant)

Ω = Rotor Rotational Speed (constant for nominal, standard speed)

Figure 20. LPI Calculation of HOGE Capability.

to maximum scheduled engine power up to a fixed limit depending on whether it is single or dual operation. Power contributed by the two engines would differ only by the difference in calibration constants with respect to maximum power capability. Therefore, the summation of power can be handled in the calibration constant that is equal to actual power output divided by standard output at a reference point (or an equivalent quantity). The constants can therefore simply be added to sum power outputs (and this sum halved to predict single-engine power capability).

Air density is calculated as shown from measured temperature and pressure, and together with maximum available power determines the nondimensional power coefficient, and in turn the nondimensional thrust coefficient from which maximum gross weight capability is calculated. Gross weight capability of the helicopter is subject to a maximum limit as indicated in the figure, irrespective of maximum capability. This can be seen in Figure 16 where the hover capability curves terminate on the right side at the maximum takeoff gross weight limit (the limit for the CH-47C is 46,000 lb compared to 40,000 lb shown for the CH-47B).

The resulting gross weight capability is compared to actual weight to determine the relative capability of the vehicle to hover OGE. The pilot can make this comparison, or the LPI system can do part of it for him by computing a weight margin for the maneuver (capability minus requirement).

The vehicle gross weight limit causes a small dilemma: on the one hand, the limit denotes a prescribed limitation on the capability to increase vehicle gross weight; that is, cargo cannot be added beyond this limit. On the other hand, as a measure of capability the absolute gross weight limit is fictitious and should be ignored, since it is lift capability that is of interest. For example, the vehicle could be at its gross weight limit but have significantly higher lift capability; this excess capability is of vital interest. This problem is of no consequence with respect to judging feasibility, but must be resolved at some point in the detailed design of a specific LPI system.

4.2.4 Single-Engine Capabilities*

The above procedure covers the basic computation of HOGE capability for dual-engine operation. Single-engine capability for this and all other maneuvers is obtained by simply halving the calibration constant. The aerodynamic performance characteristics, being nondimensional, are unaffected and are equally applicable to the single-engine mode. This simple procedure for obtaining single-engine performance capability is an obvious advantage of using nondimensional performance characteristics. Capabilities for normal rated power (or any other power level) can also be easily calculated since the lower power level can be represented as a fixed percentage of the maximum power level.

*This discussion applies to two-engine helicopters only.

4.2.5 HIGE and Vertical Climb Capabilities

The calculations of HIGE capability and vertical climb capability are similar, since both are obtained from modification of HOGE performance characteristics. As shown in Appendix B, ground effect can be represented as an increase in capability of the following form:

$$GW_{HIGE} = GW_{HOGE} \left(1 + K_1 e^{-K_2 H/D}\right) \qquad (6)$$

where the second factor in parentheses defines the fractional increase in gross weight capability. K_1 and K_2 are constants, H is height of wheels or skids above ground, and D is rotor diameter. If HIGE capability is calculated for a fixed nominal height above ground as is recommended in Section 2, then the above equation shows that HIGE capability is equal to a fixed percentage increase over HOGE capability.

Appendix B shows that vertical climb capability is a linear function of hover weight margin for low to moderate rates of climb; that is,

$$V_V = K \frac{\Delta GW}{GW} \qquad (7)$$

where ΔGW is the HOGE weight margin and V_V is vertical climb capability. The above equation also applies to negative weight margins which produce negative rates of climb, or positive sink rates.

4.2.6 Wind Velocity Corrections

Appendix B also includes the development of wind velocity corrections applicable to HOGE operation of the helicopters. The corrections also apply to vertical climb capability, since the latter can be expressed as a function of HOGE weight margin. Wind corrections are also needed for HIGE operation and for the calculation of obstacle clearance distance, but data for defining these functions are not readily available. This situation can be handled either by acquiring the required data through flight test or by restricting the LPI display to zero wind conditions for those performance capabilities where the wind velocity corrections are undefined.*

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*Theoretical estimates represent another alternative.

4.2.7 Other Performance Capabilities

It can be seen in Figure 20 that maximum power available is generated in the course of computing any of the performance capabilities, so it is a simple matter to display it. This value is also changed directly by selection of the single-engine mode or the normal rated power mode.

Power margin predictions are obtained by reversing the computation of gross weight capability. That is, beginning with measured vehicle weight, the applicable nondimensional performance characteristic is used to solve for the power needed to support that weight. Then this required power is compared to maximum available power to find power margin, the excess of available power over required power.

Remote-site capabilities are obtained by simply replacing the measured pressure and temperature by manually input values of those variables and by altering the gross weight by a fuel-used input.

Other performance capabilities that require computation are climb capability at best airspeed and obstacle clearance distance (both for takeoff and landing, as applicable). These would be computed in the same manner as HOGE capability; that is, the aerodynamic performance capability would be calculated from the applicable performance characteristic for the same maximum power available as previously described. Examples of nondimensional performance characteristics applicable to these modes have not been derived for this report, and in some cases the data required to develop the functions do not appear to be available (in particular, takeoff and landing distances for the CH-47C). Additionally, wind corrections for takeoff and landing distances appear to be universally unavailable. Such performance characteristics are difficult to estimate and may require empirical definition (i.e., flight test).

4.2.8 Illustrative Computational Forms

The expressions for the nondimensional power and thrust coefficients in Figure 20 are shown in generic form and are applicable to all helicopters. The expressions become greatly simplified for a specific helicopter. For example, for the UH-1H at standard rotor speed the expressions are

$$C_P = 2.369\ (10^{-7})\ SHP/\sigma \tag{8}$$

$$C_T = 3.5074\ (10^{-7})\ GW/\sigma \tag{9}$$

where σ equals actual air density divided by standard sea level air density, or

$$\sigma = \frac{P}{P_0}\frac{T_0}{T} = \frac{\delta}{\theta} \tag{10}$$

The maximum power schedules shown in Figure 18 are composed of linear segments and can therefore be represented by a few simple linear equations. The nondimensional performance characteristics are nonlinear and in general require multisegment representation (table look-up functions). Appendix C presents a linear approximation for the UH-1H nondimensional HOGE characteristic and also an equation for the maximum power available schedule. Together with the material in Appendix B, relatively complete mathematical representations of the performance capability computations for the UH-1H are available as follows:

(1) Maximum Available Power:

$$Q_{max} = K(3.651-2.651\ \theta)\ \delta \tag{11}$$

where K is the ratio of actual to nominally scheduled torque at topping (from topping check)

(2) HOGE Gross Weight Capability (Zero Wind):

$$GW_{HOGE} = 574.13\ Q_{MAX}^{0.72}\ \sigma^{0.285} \tag{12}$$

(3) HIGE Gross Weight Capability (at 2 ft, Zero Wind):

$$GW_{HIGE} = 1.18\ GW_{HOGE} \tag{13}$$

(4) Wind Correction:

$$\Delta GW_{WIND} = 0.000647V^{1.70}\ (GW) \tag{14}$$

$$GW_{HOGE \cdot WIND} = GW_{HOGE} + \Delta GW_{WIND} \tag{15}$$

(5) Vertical Rate of Climb:

$$V_V = 7500\ (GW_{HOGE \cdot WIND} - W)/W\ \text{fpm} \tag{16}$$

where W = actual weight (measured)

(6) HIGE (2 FT) Power Margin:

$$\Delta Q = Q_{MAX} - \left(\frac{W}{677.5}\right)^{1.389}\sigma - .396 \tag{17}$$

The above equations cover all of the required performance capabilities except takeoff distance and best-airspeed climb (data are also available for these characteristics, but no attempt was made to reduce them to analytic representations).

Appendix C shows how several of the UH-1 performance characteristics can be incorporated into a simple, manual, slide-rule-type lift performance computer.

can be incorporated into a simple, manual, slide-rule-type lift performance computer.

4.2.9 Variations from Basic Procedures

Recapping the above procedures, it is seen that lift performance is calculated in two basic steps. First, maximum available power is computed from a relatively simple nominal schedule of normalized power vs ambient temperature that describes the action of the engine controls. This value is adjusted by a simple calibration constant to account for trim variation from the nominal engine power schedule and is multiplied by the measured ambient pressure to obtain actual engine power. This power level is limited by a fixed-value transmission power or torque limitation. Alternative single-engine and normal rated power display modes are obtained by simply using different multiplicative constants.

The power computed in the first step is used in the second step to compute the desired performance capabilities. The basis of these computations is a set of performance characteristics for the aircraft that have been derived, in general, from flight test data; the typical characteristic consists of a nonlinear curve relating nondimensional parameters and allowing the desired performance capability to be computed based on inputs of power, air density, and vehicle weight.

Vehicle weight and c.g. are calculated from measured variables as shown in Appendix D. This basically consists of simply summing the measured forces on struts to obtain weight and calculating c.g. according to simple fixed relationships based on the relative locations of the struts.

These procedures, particularly the use of the vehicle performance characteristics, are applicable to all four study aircraft and to all helicopters in general. However, these procedures provide only nominal performance to the extent that a specific aircraft's performance characteristics can differ from those of the flight test aircraft (and can vary with time), also to the extent that maximum power available for the aircraft can vary from its nominal schedule and with time. The question is whether the basic procedures provide sufficiently accurate results or are the following refinements necessary:

(1) Adjustment to maximum available power calculations to account for variations due to engine performance degradation, engine control degradation, or engine control schedule tolerances.

(2) Adjustment to performance characteristics to account for rotor degradation and variation of rotor characteristics from nominal.

4.3 DEFINITION OF MAXIMUM POWER AVAILABLE (MPA)

Before considering MPA computational schemes, it is important to have a reasonably clear understanding of what constitutes maximum available power for an operational helicopter, to be aware of the mechanisms that act to delimit MPA, to appreciate the nature and extent of variations in expected MPA due to degradation and other factors, and to have an approximate idea of the engine calibration and checks performed in the operational unit with respect to MPA. The following paragraphs outline these considerations.

4.3.1 Effects of Operating Limitations on MPA

The maximum power that an installed engine will deliver is subject to automatic engine controls and operating limitations. Observance of the latter is the responsibility of the vehicle operator. In some cases (for example, the T700 automatic temperature limiting and the T53 manual emergency control), an automatic control function can be overridden by the operator. Since the purpose of emergency override provisions is to counter malfunctions of the automatic controls, such features can be ignored in considering MPA computation algorithms (they allow achieving or exceeding normal maximum power).

Engine operating limits that can affect maximum power available generally exist for engine torque or power output, turbine temperature, and spool speed. If these variables remain within operating limits, then MPA is determined by the engine controls as shown earlier in Figure 18.

The principal operating limitation is maximum allowable torque or shaft horsepower. Helicopter engines are typically derated for operation at standard sea level conditions. The helicopter's structure and power train are sized for a given loading. The engine is sized to provide the power required to sustain that loading at a given temperature-pressure condition (say 5000 ft, 95°F). The result is that for many commonly encountered pressure altitude and temperature conditions, the engine(s) can provide power in excess of power train limitations and the responsibility of observing those limitations falls to the operator. This condition is applicable to all four helicopter-engine combinations considered in this study. For example, at standard temperature conditions, the CH-47C and CH-54B aircraft are transmission-torque limited up to a pressure altitude of about 7000 ft.

Turbine interstage or exhaust gas temperature operating limits are common to all helicopter gas turbine engines. As will be demonstrated later, these limits do not generally limit maximum available power, even for degraded engines. One reason for this is that exceedance of the temperature limit generally denotes a condition that must be corrected. The T700 engine is unique among the engines reviewed in this study because it incorporates automatic temperature limiting (that can be overridden by the pilot).

Although operating limits exist for spool speed, it is very rare for overlimit gas generator spool speed to be encountered. Most helicopter gas turbine engines are free turbine engines in which maximum engine power corresponds to maximum gas generator speed which is controlled by a speed governor. So in most cases, an overspeed condition can only result from malfunction or misadjustment of the engine controls. (As an engine's power output falls off with age, the engine's gas generator speed is retrimmed to a higher value and can approach its operating limit; exceedance of that limit is the boundary beyond which further adjustment is not allowed.)

4.3.2 Effects of Engine Controls on MPA

All of the engines considered in this study incorporate the same basic control scheme implemented by hydromechanical fuel controls. Free turbine and gas generator speed governors measure and control those speeds according to a lowest-wins approach. In operating the helicopter, the gas generator speed control is set to maximum and the free turbine speed control is set to obtain a speed of 100 percent. This speed is maintained as collective pitch is increased, causing an increase in gas generator speed to provide the additional power, until the gas generator speed reaches its maximum value. At this point, no further increase in power will be allowed by the fuel control and any increase in load will cause the free turbine speed to fall off or droop.

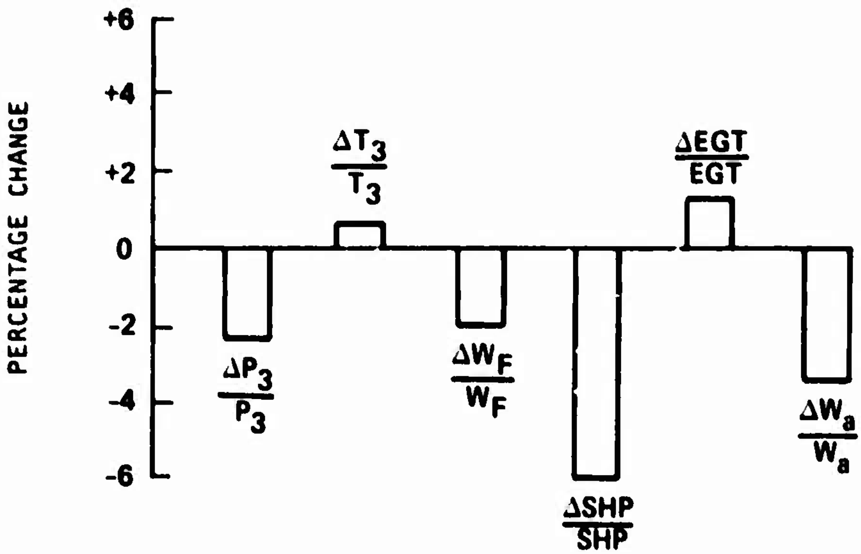
For the T53 (UH-1) and T55 (CH-47) engines, this is the point at which maximum available power is obtained subject to torque and engine temperature limitations. This is also the scheme for the T700 (UTTAS) engine, but additionally, automatic turbine interstage temperature limiting is accomplished by the electrical control unit, an adjunct to the hydromechanical fuel control. The temperature limiting function can be overridden, however, by advancing the power lever beyond the normal maximum power point. The T73 (CH-54) engine also employs the basic hydromechanical speed control approach, but maximum power is selected by the pilot according to an engine pressure ratio (EPR) schedule.

For all four engines, the gas generator speed at which maximum power is developed is a function of compressor inlet temperature (CIT) as measured by the fuel control. Figure 18 shows the resultant variation in shaft horsepower (SHP) with CIT or ambient temperature.

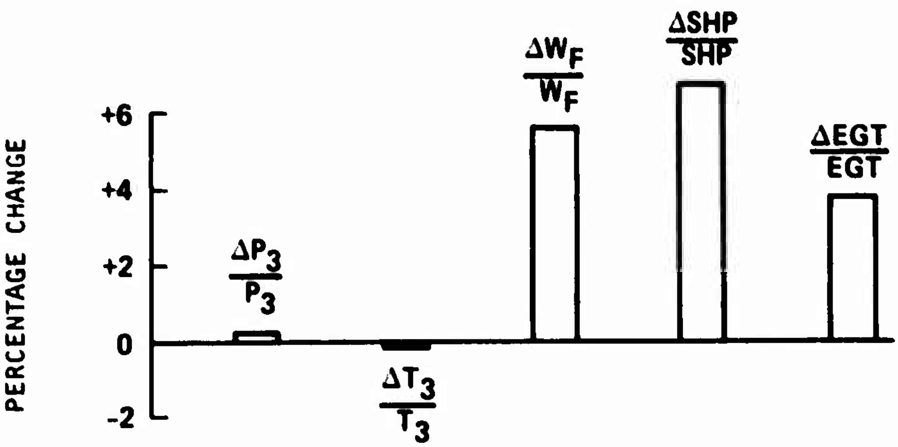
4.3.3 Effects of Engine Degradation on MPA

The order of magnitude of engine degradation effects is relatively well defined. Two specific types of degradation will be considered here to illustrate the character and magnitudes of internal performance changes.

Figure 21 illustrates typical changes that would be observed in engine performance parameters compared at constant gas generator speed due to two major forms of engine degradation: compressor erosion and gas generator turbine degradation due to temperature erosion (burning, sulfidation). The



A. Eroded Compressor



B. Gas Generator Turbine Degradation

Figure 21. Engine Degradation Effects at Constant Gas Generator Spool Speed.

changes shown are actual test results from the Automatic Inspection Diagnostic and Prognostic System (AIDAPS) program.* Since the changes are shown for constant referred speed, they illustrate the variations that would be observed at maximum power conditions. Note that engine power output, shaft horsepower can increase as well as decrease due to engine degradation.

The variation in maximum power observed by the pilot will not be as large as that indicated by the above type of performance comparison in the case of the typical hydromechanical fuel control. This is due to the droop characteristic of the spool speed governor of the control. Figure 22 illustrates the mechanism involved. With degradation of the compressor section, loss of SHP output is due to diminished pumping capacity of the compressor; i.e., less airflow at the same spool speed. This condition also causes a reduction in fuel flow of about 1/3 to 3/4 (the SHP drop depending on the exact nature of the degradation. Because of the droop characteristic of the fuel control speed governor, the rpm for a particular power demand changes as shown in Figure 22. (The new operating point is found by constructing the droop line to pass through the initial operating point; the intersection of the droop line and the new fuel flow characteristic determines the new operating point.)

With a negative change in fuel flow, as in the case of compressor erosion, the new operating point is at a higher rpm. Increased rpm means increased power output. The net change in SHP can be visualized by considering the curves in Figure 22 to represent SHP instead of fuel flow. Then the change in SHP due to performance is the difference between the curves at the same value of spool speed. The net change is the difference in SHP for the two operating points (i.e., new rpm value).

For the T53 engine, the droop slope is -6.5 according to References 5 and 6. Based on the T53 model specification, the slope of the SHP vs N^1 characteristic at maximum power conditions is about five. The net change in SHP can be expressed as the sum of the change due to performance and the change due to the new operating point:

$$\frac{\Delta SHP'}{SHP'_{Net}} = \frac{\Delta SHP'}{SHP'_{Perf}} \quad \frac{\Delta W_f'}{W_f'} \cdot \frac{5}{6.5} \tag{18}$$

If we take as typical a change in fuel flow at slightly over one-half the change in SHP namely 13/20, then the result is

$$\frac{\Delta SHP'}{SHP'_{Net}} = \frac{1}{2} \; \frac{\Delta SHP'}{SHP_{Perf}} \tag{19}$$

Thus, the effect of performance degradation on maximum available power is substantially lessened by the rebalancing action of the fuel control. Moreover, the fuel control can be re-rigged to cause the engine to deliver

*Test results were obtained by installing degraded parts in otherwise normal engines and obtaining "before" and "after" data.

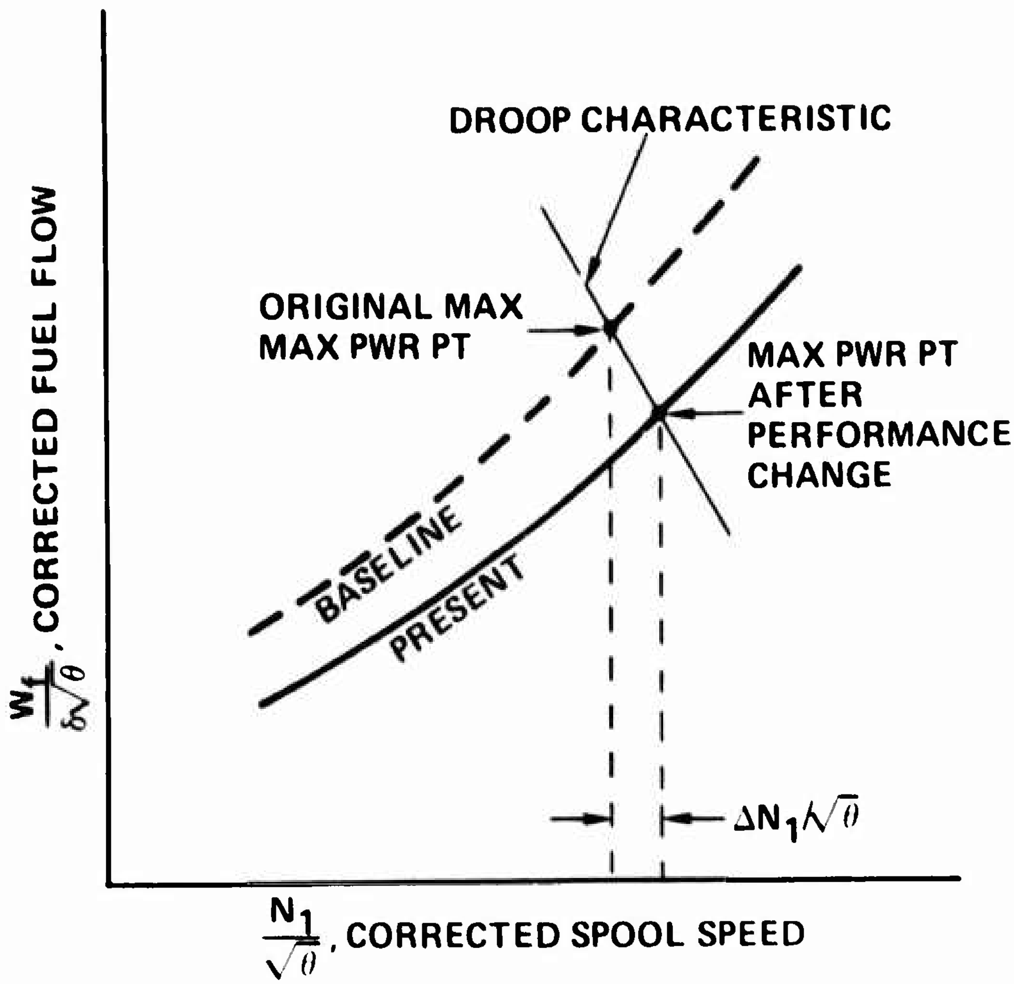


Figure 22. Interaction of Performance Change and Fuel Control Droop Characteristic.

maximum prescribed power irrespective (within limits) of the internal condition of the engine. In general, this requires increasing the spool speed of the gas generator to make up for internal performance loss. Spool speed can be increased in this manner up to a limit designated for the engine.

Based on AIDAPS experience, it appears that SHP losses (considering the SHD vs N_1 characteristic only) due to severe compressor foreign object damage (FOD) or erosion probably range between 3 to 12 percent. The effect on maximum available power without readjustment of the fuel control is expected to be in the range of 2 to 8 percent reduction. If the engine fuel control has been readjusted since suffering the performance decrement (e.g., minor or moderate erosion), the maximum available power is independent of the performance level.

Figure 21 shows the effects of turbine degradation at constant gas generator spool speed. The pattern of changes shown is typical; fuel flow and SHP changes are similar in magnitude and about twice as great as exhaust gas temperature (EGT). With the Army's current engine health tracking procedure, EGT variations are limited to about 2 percent. Corresponding variations in fuel flow and SHP would be about 4 percent.

The fuel control rebalancing mechanism is equally applicable to this mode. Since the fuel flow and SHP values are relatively close, the effect would be more pronounced; as a result, it is expected that maximum available SHP variations due to this mode of engine deterioration would be limited to about 2 percent.

The foregoing discussion is applicable to the typical fuel control where, from the standpoint of automatic control, maximum available power is scheduled and limited in terms of gas generator speed. It is less applicable to the UTTAS power plant (T700) where in addition to rpm scheduling, integral temperature limiting is incorporated.

If the preceding example of turbine degradation occurred in an engine whose fuel control incorporated automatic temperature limiting, and if the temperature limit were exceeded by the increase in temperature shown in Figure 21, then the results with respect to maximum power available would be far different. For example, suppose that the temperature level of the engine were at its controlled limit preceding the turbine degradation. Then as the degradation occurred, the engine would be controlled at a lower power level to to maintain the same temperature level. If the results in Figure 21 had been plotted for constant EGT instead of current N_1, then one would observe a decrease in N_1 of about 2 percent and a decrease in SHP of 6 percent or greater. Thus, the specific control actions of an engine's fuel control make a great difference on the effects of degradation on maximum power available. We see in the case of turbine degradation that it is possible to observe an increase in SHP of at least 3 percent or a decrease in SHP of at least 6 percent for the same degradation, depending only on the specific control functions of the fuel control (for selected initial temperature conditions). If the type of degradation does not tend to raise the temperature level in the engine (such as compressor erosion), or if compressor erosion), or if the increased temperature level is still within limits, then the results will not be affected by the temperature limiting function of the fuel control.

4.3.4 Effects of Engine Trim on MPA

For the typical free turbine engine, as more power is demanded from the engine the gas generator speed increases until finally it reaches a limiting value--it tops out (any further demand causes the free turbine speed to fall off or droop). Tests for within-tolerance operation of this mechanism are therefore called topping checks. The action of adjusting the engine control to obtain prescribed power at topping is called trimming.

Figure 23 illustrates the topping check procedure for the T55. The family of curves in Figure 23, incidentally, reduce to the single normalized curve for the CH-47C in Figure 18. This illustrates the use of the LPI system as an aid for performing topping checks. This check is performed at engine installation and periodically thereafter.

Maximum power available is a specified quantity that must be achievable by an engine in correct working order. Performance degradation can alter the maximum power available; but if this alteration exceeds the tolerance on specified maximum power, then the control must be readjusted to the specified limit.

The tolerance on engine trim is one measure of the accuracy that would be needed to track maximum power available variations (such as those due to engine degradation). For example, the trim tolerance for the T53-L-13 engine in the UH-1H is ± 1 psi torque pressure, or about ± 2 percent SHP. If this tolerance is exceeded, an appropriate fuel control adjustment is made to achieve specified performance. If an engine is out of trim, it is out of adjustment and requires maintenance. In practice, topping checks are performed periodically or whenever a problem in that area is suspected.

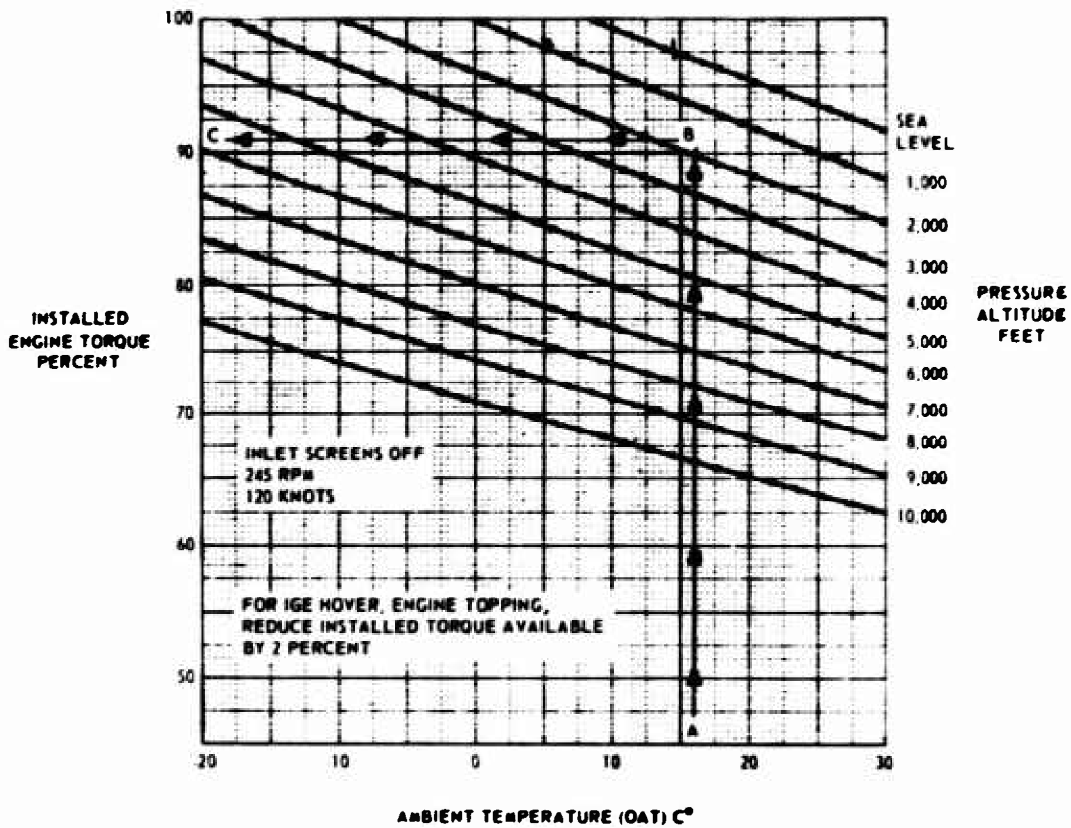
A device or technique that would estimate the MPA variations for an engine would also check the trim status of the engine. Thus, any sizable change in maximum available power would indicate an out-of-trim condition and a need for maintenance action.

With a maximum power available estimation technique with accuracy to within ± 2 percent, for example, one could not be certain that the T53 engine was in trim, and an excursion of ± 4 percent would be required to ensure that the engine was out of trim.

Using this potential capability as a criterion, desired accuracy would be to within $\pm 1/2$ percent or better.

4.4 COMPUTATION OF MAXIMUM POWER AVAILABLE

In the basic approach recommended for computing MPA (Section 4.2), the nominal maximum power schedule for the engine is used in combination with a calibration constant derived from the topping check that adjusts the nominal



- ENGINES CHECKED SEPARATELY
- MAX POWER OBTAINED BY REDUCING POWER ON ONE ENGINE, INCREASING POWER ON OTHER
- TOPPING CONDITION:
  - NI STABILIZED AT MAX
  - ROTOR RPM DROOP FROM 250 TO 245 RPM
  - 120 KNOTS AIRSPEED
- TOLERANCE:
  INDICATED TORQUE WITHIN ± 4% OF CHART VALUE
- TORQUE LIMITS:
  SINGLE: 100%
  DUAL: 78%

SPA 4628-29

Figure 23. Topping Check Procedure for CH-47C With T55-L-11A Engines.

schedule to represent the actual schedule for a specific engine. No provision is incorporated to automatically compensate for changes in the actual power available function for the engine due to degradation or other causes on a day-to-day basis. Instead, it is assumed that some method of tracking engine performance variations is employed (such as the presently implemented HIT check or an equivalent procedure performed by the LPI system--identified in Section 2 as an auxiliary or growth function). It is also assumed that in the event of significant performance variations and periodically following inspections, a topping check is performed (defining a new value of the calibration constant for the LPI system) so that the maximum variation in MPA compared to the LPI schedule would be about the same as the tolerance on engine trim, or about ± 2 percent.

In evaluating the adequacy of this approach it can be compared with a more sophisticated MPA prediction technique in which MPA predictions are automatically altered to reflect the effects of changes in engine performance levels that are also automatically calculated. This section will review this sophisticated alternative technique in an effort to provide that comparison. In addition, the discussion reviews the interaction between available power and engine performance and places the recommended specific calculation techniques within the context of a generic treatment of fuel control functions that determine maximum power available.

4.4.1 Compensation for Engine Degradation Effects

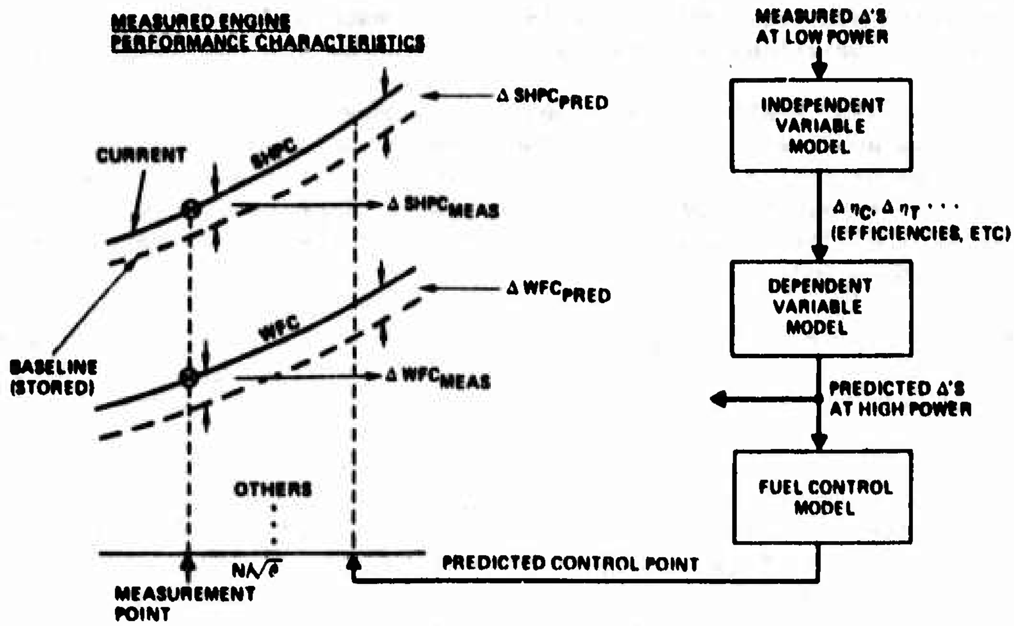
MPA is obtained when a controlled variable, such as N_I, reaches its limiting value. If more than one variable is automatically controlled or limited, then MPA occurs at the lesser relative limit. If functions are available relating the engine power output to the controlled variables, then the power level that would be obtained at the limit for each of the controlled variables can be computed. The MPA is the minimum of those values. Referred SHP vs referred gas generator speed is an example of one of the required functions.

Engine deterioration alters the relationships among the engine performance variables, so in order to accurately predict MPA following significant engine degradation (and without the engine having been retrimmed), it is necessary to reestablish the performance variable relationships or to alter the original functions to reflect the changes.

4.4.1.1 Thermodynamic Model Approach

A sophisticated technique for doing this is described by Fox in Reference 6. The method is designed to predict MPA using measurements made at low or intermediate power levels that might occur before takeoff of the helicopter. Figure 24 illustrates the basic method.

Initially, baseline performance characteristics (gas generator curves) for the specific engine are measured and stored in the memory of the onboard system. In the figure, baselines for referred shaft horsepower (SHPC) and referred fuel flow (WFC) are plotted as functions of referred gas generator



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SHPC = Referred Shaft Horsepower, $\frac{SHP}{\delta\sqrt{\theta}}$

WFC = Referred Fuel Flow, $\frac{WF}{\delta\theta^K}$

η_C = Compressor Efficiency

η_T = Turbine Efficiency

δ = (Engine inlet pressure)/14.7 psi

θ = (Engine inlet temperature)/518.7°R

MEAS = Measured

PRED = Predicted

Figure 24. Thermodynamic Model Maximum Power Available Prediction Technique.

speed, $N1/\sqrt{\theta}$. Actually, baselines for a minimum of 5 and preferably 6 or more variables are required in the technique described by Fox. (The process of forming the baselines and the required accuracy are not described.)

With no deterioration, the baseline gas generator curves define the relationships among all the engine performance variables, and with the fuel control model determine the engine operating point for maximum power. This point can be represented by a particular value of referred speed, which in turn, determines the values of all the other referred performance parameters, including SHP.

When an engine suffers internal deterioration, the relationships among its performance variables change. This is indicated in Figure 24 by the difference between the engine's current gas generator curves vs its baseline curves.

The change in performance can alter the maximum power available in two ways: the power output for a particular operating point can change and the operating point itself can change. Both mechanisms can be visualized in Figure 24 where the engine variables are plotted as a function of referred gas generator speed, the value of the latter governed by the fuel control. If the current variable relationships in the region of maximum power are known, then the new maximum power operating point and corresponding power output can be calculated.

The procedure Fox develops provides an estimate of the relationships at maximum power based on measurements made at low power level. The variables are measured at a low power level and compared to their baselines. These changes are used to compute the variations in component characteristics (such as efficiencies) that are descriptive of the degradation that has taken place in the engine. These variations are computed by means of a mathematical model of the engine, termed the "independent variable model" in Figure 24. These are considered independent variables in the sense that they cause the changes in the measured performance variables. Assuming that these changes remain constant over the operating range of the engine, the variations in the dependent variables, such as SHP, can be computed for any power level using the inverse form of the same mathematical model. Thus, the changes can be computed for the maximum power level. Based on the predicted changes at maximum power (which when added to the baselines determine the absolute values of the variables at maximum power), the operating or control point of the engine for maximum power is computed by means of a mathematical model of the fuel control limiting functions. The operating point in combination with the revised performance characteristics (baselines plus changes) determines the maximum power estimate.

The measurements required to implement this technique and the definitions of the dependent and independent variables are listed below.

(1) Measurements Required for Thermodynamic Model Approach:

Compressor inlet pressure, P_2

Compressor inlet temperature, T_2

Compressor discharge pressure, P_3

Compressor discharge temperature, T_3

Gas generator spool speed, N_1

Free turbine speed, N_2

Shaft horsepower, SHP

Fuel flow, WF

Interstage turbine temperature, T_7

Interstage turbine discharge pressure, P_7

(2) Referred Performance Variables:

Referred shaft horsepower, $\frac{SHP}{\delta\sqrt{\theta}}$

Referred fuel flow, $\frac{WF}{\delta\,\theta^{K}}$

Referred compressor discharge pressure, $\frac{P_3}{\delta}$

Referred compressor discharge temperature, $\frac{T_3}{\theta}$

Referred turbine interstage pressure, $\frac{P_7}{\delta}$

Referred turbine interstage temperature, $\frac{T_7}{\theta}$

Referred gas generator spool speed, $\frac{N_1}{\sqrt{\theta}}$

(One of the above is selected as the independent variable, usually compressor discharge pressure or gas generator spool speed, and the remainder are treated as functions of it.)

(3) Component Performance Characteristics:

Compressor efficiency, η_c

Gas generator turbine efficiency, η_{GT}

Power turbine efficiency, η_{PT}

Turbine nozzle area, A_5

Power turbine nozzle area, A_7

Engine airflow, $\frac{W_a\sqrt{\theta}}{\delta}$

4.4.1.2 Evaluation of Thermodynamic Model Technique

The unique feature of the thermodynamic model technique described by Fox is its use of a complex mathematical model to extrapolate performance changes measured at low power levels to predict the changes that would be observed at high power levels. The model used is the same as that used in the Army's AIDAPS program to diagnose the cause of internal engine degradation causing performance change.

The model consists of a set of differential equations that are based on the engine internal flow process and that relate the changes in measurable variables to changes in component performance characteristics such as efficiencies and effective flow areas, as previously identified. The coefficients in these equations are variable and are complicated functions of the measured engine variables. In order to restrict the relationships to a manageable number (6 equations in 6 unknowns), a number of simplifying assumptions are made such as combustion efficiency and pressure drop remaining constant. The resulting model is an approximation, but one that works well in diagnosing engine degradation.

The equations are treated as linear algebraic expressions relating the changes in the dependent variables. In order to calculate the changes in the performance characteristics, it is necessary to calculate the coefficients for the particular operating state of the engine, then simultaneously solve the 6 equations for the 6 unknowns. To simplify this process for an onboard computer, the coefficients are precalculated based on average engine characteristics at selected operating points (indexed in terms of the independent, measured engine variable), and corresponding solutions are obtained that consist of algebraic equations. Each equation relates the change in a component performance coefficient to the changes in the measured engine variables multiplied by fixed coefficients. Each set of solutions is applicable to a particular operating point defined by the value of the independent measured variable (e.g., Fox uses compressor discharge pressure). Interpolation is used to obtain the solutions for intermediate operating points.

Since the coefficients are computed from average engine characteristics, the precalculated solution approach produces, in effect, a solution for a nominal engine at the same operating point as the actual engine and displaying the same changes from baseline. This is another approximation.

The changes in the component performance characteristics calculated in the above manner are assumed to be the same at all power levels. To predict the changes in the engine variables at high power, the above process is simply reversed: the calculated changes in efficiencies are substituted into the original model equations (using precalculated coefficients there also), and the changes in the engine variables are computed. These changes plus the baseline values provide the magnitudes of the variables and enable calculation of the control point and corresponding maximum power output.

Why employ complex mathematical models? Fox reasons that the component performance changes behave predictably with changes in power level (it is assumed the changes are unaffected by power level), whereas the changes in the engine variables behave dependently. Supposing that the component performance changes do remain constant (e.g., that the percentage change in compressor efficiency compared to its baseline level is the same at high power levels as at low power levels). How much error is introduced by assuming, instead, that the engine variable changes remain constant (e.g., that the percentage change of SHPC from its baseline is the same at high power levels as at low power levels)?

This question was answered in the following way. Using the thermodynamic model for the T53 engine from the AIDAPS program, applicable influence coefficients were obtained for a moderately low power level (referred compressor discharge pressure of 70 psi) and for a high power level (100 psi). Then the change in each component performance characteristic required to separately produce a 1 percent change in SHP at the low power level was calculated. Next, the amount of change in SHP that each one of those variations would produce at the high power level was calculated. Comparing the variations in SHP produced at the high power level to the original 1 percent variation at low power indicates the amount of error that could be introduced by simply assuming that SHP variations would be the same at the two power levels.

It was found that the maximum variation would occur for gas generator turbine nozzle degradation (area change). For this case, a 1.2 percent variation in SHP occurred at high power compared to 1 percent at low power. (Equivalently, we would find 6 percent at high power vs 5 percent at low power.) The average comparison was much closer: 1.07 percent vs 1.0 percent. Corresponding differences in fuel flow and EGT (the only other dependent variable of direct interest) were 0.18 and 0.17 percent maximum, and 0.07 percent average for both. These errors do not appear to be very significant.

Even in the case of the generic fuel control (to be described later), only variations from baseline of SHP, fuel flow, and turbine temperature directly affect MPA. The variations from baseline of compressor discharge temperature, and turbine discharge pressure are used only to compute the variations in the component performance characteristics. It appears that the potential gain in accuracy afforded by this expenditure is not significant compared to desired accuracy levels and expected degradation effects. Moreover, the potential gain accuracy depends on the validity of the assumption that the changes in component performance characteristics remain constant, irrespective of power level. The assumption appears tenuous, if only because of extrinsic effects such as instrumentation characteristics. Fox, in reviewing disappointing test results, questions his own assumption and recommends pursuit of a modified technique in which measurements would be made at two power levels to allow for linear variation with power level of the component performance changes. But if a two-point method is used, one might as well deal directly with variations of the engine variables: why not simply assume that these variations from baseline are linear with power level?

A separate problem area is associated with attempting to predict MPA on the basis of a single set of measurements taken at low power. There are actually three problems here: obtaining data for stabilized operation, the degraded accuracy of engine instrumentation at low power levels, and random fluctuations due to extraneous effects. These factors combine to make it difficult to achieve high accuracy at low power levels. The situation improves for higher power levels. In Reference 6, using before and after test cell data for degraded engines, the error in predicted MPA decreased roughly in a linear manner with increases in the power level of the measurement point; but even at 90 percent power levels, desired accuracy of 1 percent could not be met for some of the test case engines. (At 90 percent, standard deviation of the error in predicted MPA was 0.8 percent of the 10 test case engines, yielding a 3σ error of about 2.5 percent.)

Why base the MPA prediction on a single set of measurements made at low power conditions? The reason is that it is desired to have the power estimate be as up to date as possible in terms of possible engine degradation effects. But the internal degradation that gives rise to engine performance changes can be characterized as resembling a wear process (for example, sand, dirt, and temperature erosion). It occurs slowly.

Therefore, if it is necessary to assess engine performance for modifying MPA predictions, the assessment could be based on multiple measurements of engine performance made at stabilized flight conditions where both external factors and engine power levels are more suitable. Variations from baseline performance levels can be either combined into an average for the preceding flight or can be incorporated into a moving average. With the measurements made at moderate to high power levels, it can be assumed that the variations at maximum power are the same as observed at the lower levels. That is, there is no need to employ a complicated procedure for extrapolation where the potential gain in accuracy is only a few tenths of a percent.

Another factor that needs to be considered is the specific trim procedure for the engine. The foregoing discussion has referred to "automatic" compensation for the effects of engine degradation; for that reason one might infer that no manual adjustments to an LPI system employing such a technique would be required--a definite advantage over the recommended technique where manual input of a calibration constant based on topping check results is required. In general, however, both techniques require manual input of essentially the same calibration information. The reason for this is that all fuel controls have trim adjustments that alter their nominal limiting functions. All four engines considered in this study have trim adjustment provisions. Even in the case where a fuel control adjustment might be used to adjust limiting to a fixed designated limit, there is a trim tolerance to be considered. Future electronic and digital fuel controls will also have trim provisions.

Army policy with regard to engine trim for its helicopters has been to adjust engine controls such that a specified output power is obtained. This adjustment is checked periodically (or on condition) in the topping check, and if not within tolerance the engine is retrimmed. This would appear to limit the increase in accuracy that can be achieved by a technique that automatically compensates for engine degradation, because if the engine output is not within the topping tolerance, then retrimming would appear to be in order. Of course, this policy can change. A likely situation, for example, is that an engine would be required to deliver at least minimum specified power with the maximum limited only by operating limitations (both manually and automatically controlled).

Thus, under present circumstances the "automatic" technique would also require the use of a calibration constant input; this constant would change, in general, with each topping check.

Finally, it should be noted that tracking engine degradation effects does not provide any information about the condition of the engine controls. There is no way to automatically detect variations due to engine controls, which are traditionally a major maintenance item. Presently, the topping check is the only means of checking correct operation of the engine controls as well as the engine with respect to maximum power.

4.4.2 Treatment of Fuel Control Limiting Mechanisms

The second major area with respect to MPA computation is the treatment of fuel control limiting mechanisms. Specific considerations are:

(a) Relation of limiting mechanisms to maximum power available.

(b) Relation of manually imposed limits (such as torque) to automatically imposed limits.

(c) Variations in the limiting functions among engines of the same type.

4.4.2.1 Generic Fuel Control Model

References 5 and 6 present a generic model of engine control limiting functions applicable to MPA determination. That model is reviewed below in relation to the specific characteristics of the engines considered in this study. It will be shown that more simple control representations are possible for the T53, T55, T73, and T700 engines.

The generic fuel control is considered to determine MPA through application of three control limits: a turbine discharge temperature limit, a gas generator speed limit, and a metered fuel limit. The lowest value of maximum power derived from the three limits corresponds to maximum available power. Figure 25 is a graph of three functions of ambient temperature (F_N, F_T, and F_W) that define the limiting mechanisms for the T53-L-13B engine.

The function, F_N, represents the normalized power output of the engine at constant maximum gas generator speed as ambient temperature is varied. F_T and F_W are similarly defined for turbine discharge temperature and fuel flow, respectively. F_W is also a function of ambient pressure and can be represented by a family of curves with altitude as a parameter.

The functions are equal to the normalized output of the engine when the respective variables are held constant at their limiting values, as follows:

$$F_N = \frac{(SHP/\delta)}{SHP_o} \text{ at } N_1 = 25{,}400 \text{ rpm} \tag{20}$$

$$F_T = \frac{(SHP/\delta)}{SHP_o} \text{ at } T_7 = 1840^{o}R \tag{21}$$

$$F_W = \frac{(SHP/\delta)}{SHP_o} \text{ at } W_f = 820 \text{ pph} \tag{22}$$

The functions are normalized by correcting the engine power output to standard sea level pressure (that is, SHP/δ) and by ratioing the corrected output to the power obtained at standard sea level pressure and temperature (SHP_o). The functions would be more properly plotted on separate graphs since the values of SHP_o are different in each case (that is, at standard sea level conditions a different power output would generally be obtained if operating at the spool speed limit vs operating at the temperature limit). The nominal maximum power output at standard sea level conditions is 1400 SHP.

Although the above functions are appropriate for the T53 engine, they represent a hypothetical model of the fuel control functions, and the differences between the model and actual engine should be noted. The actual T53 engine control, for example, does not exercise any direct control over engine temperature. However, if the control did incorporate direct temperature limiting, F_T would represent that function. There is a maximum temperature limit for the engine, but observing that limitation is the

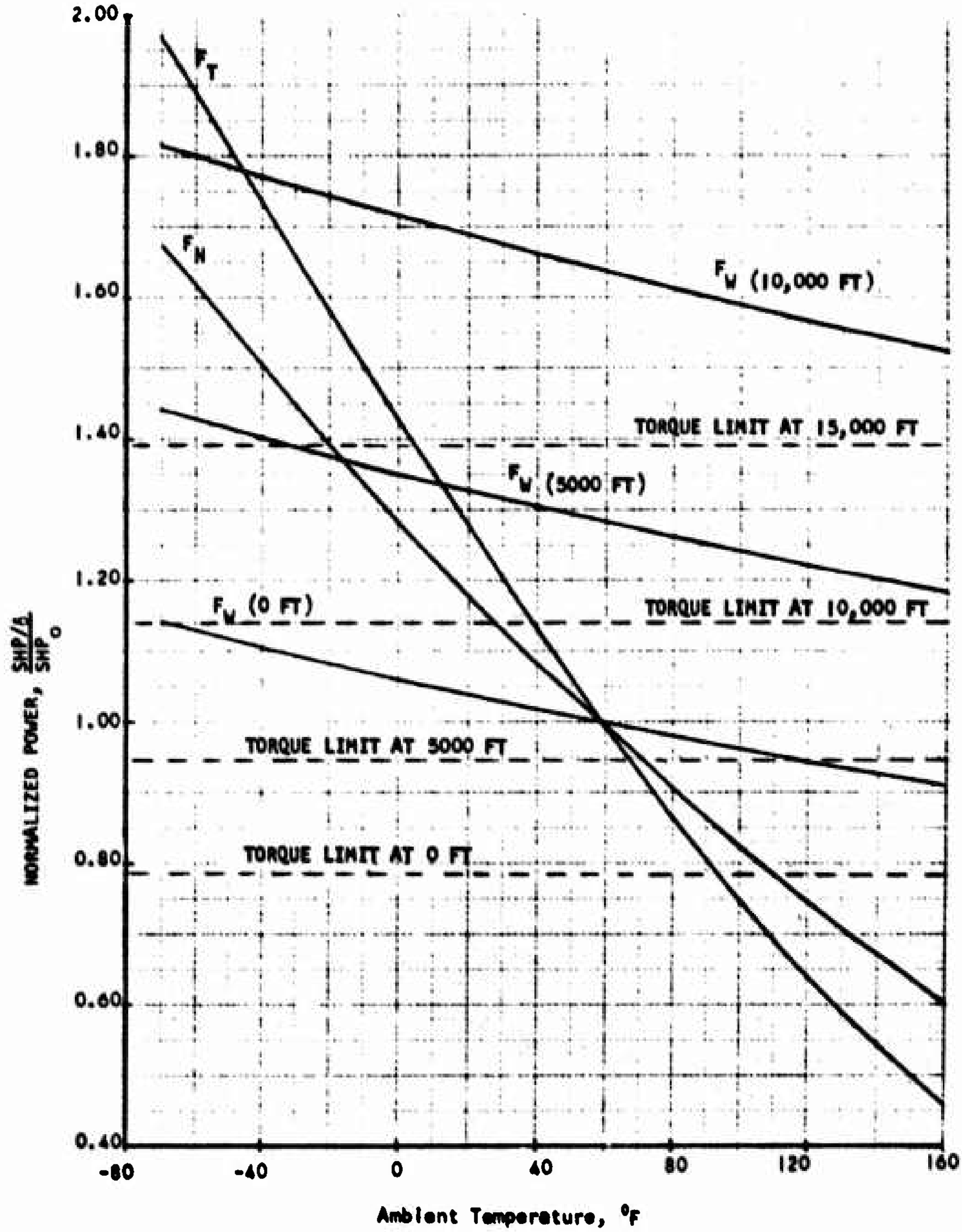

Figure 25. Fuel Control Limit Functions (Transmission Torque Limit Shown for Reference).

responsibility of the pilot (EGT indicator red line). The F_T function can therefore be used to represent the action of the pilot.

The actual engine control indirectly controls engine temperature through the F_N function. Note that the F_T and F_N schedules are quite similar in Figure 25. In the figure, the two schedules are identical at standard temperature because they are normalized by their respective standard day output values, SHP_o; but in terms of uncorrected SHP, the schedules would be offset from one another.

The spool speed function, F_N, closely corresponds to the actual control action of the T53, except that in practice, the T53 is trimmed to produce rated SHP (1400 hp at sea level conditions) for military power by varying the rpm limit through adjustment of the fuel control*. Also, over the range of ambient temperature, actual spool speed is varied slightly by the control (as opposed to being held at a constant value).

The fuel flow function, F_W, is representative of the maximum fuel flow capability of the control, but in practice this limit has no effect on MPA (shown later in this section).

Transmission torque limit curves are superimposed on the graphs of the fuel control functions to show where the automatic functions are applicable. For example, at sea level pressure torque limiting prevails up to an ambient temperature level of about 95°F. Since the transmission torque or SHP limit is a fixed value, the limit in terms of $SHP/_{\delta}$ varies with altitude as illustrated.

The functions shown in Figure 25 provide a convenient representation of control effects on MPA because they can be plotted on one chart to illustrate comparative effects and also the effects of ambient conditions on MPA. However, this representation is an approximation of the actual case for a specific engine because it is based on average engine performance characteristics in addition to the fuel control limiting mechanisms.

As an example, the F_N function is derived from the referred SHP vs referred gas generator speed characteristics; i.e.,

$$\frac{SHP}{\delta\sqrt{\theta}} \text{ vs } \frac{N1}{\sqrt{\theta}}$$

The maximum speed that the fuel control implements as a limit corresponds to the value of referred speed at which referred SHP is equal to 1400 hp for the T53 (nominally). To calculate the values of F_N vs ambient temperature for plotting, the procedure is to (1) calculate for the temperature,

*One-eighth (1/8) turn of the fuel control's military power trim screw changes the maximum rpm limit by about 1 percent, producing about a 5 percent change in MPA.

(2) calculate referred speed using the N_1 limit value and θ, (3) find the corresponding value of referred SHP, (4) multiply this value by $\sqrt{\theta}$, and (5) divide the result by SHP_o (nominally 1400 hp) to get F_N.

The curves for F_T and F_W are similarly calculated using the following engine performance curves:

$$F_T: \quad \frac{SHP}{\delta\sqrt{\theta}} \text{ vs } \frac{T_7}{\theta}$$

$$F_W: \quad \frac{SHP}{\delta\sqrt{\theta}} \text{ vs } \frac{W_f}{\delta\sqrt{\theta}}$$

The functions plotted in Figure 25 are based on performance curves for an average T53 engine. The accuracy of this representation of the control actions on MPA depends on two factors (engine degradation effects considered separately), (1) how well the control performs its scheduling function (e.g., for the T53, the degree of variation of the maximum speed point from expected with variations in temperature and with time), and (2) how well the average engine performance characteristics represent the characteristics of the specific engine. In the case of the latter factor, the variation in the slopes of the performance relationships determines the amount of error introduced. To visualize the source of error, trace through the first three steps of the calculation of the F_N characteristic for several different ambient temperatures. It will be shown later in a specific example for the T53 that the amount of error is tolerable.

This calculation approach is employed by Fox in Reference 6, but the error introduced by it is not specifically considered. It is clear, however, that in any approach that requires formulating specific performance variable baselines, it would be better (measurably improved accuracy) to use the baselines directly (same steps as in calculating the F_N characteristics) rather than using relationships that are based on average engine characteristics.

Where actual engine characteristics are not acquired, accuracy is not a factor in the representation of control effects, and the F_N approach is attractive in its convenience.

4.4.2.2 Adjustments for a Specific Engine

The fuel control functions typified in Figure 25 are formulated for a given model engine based on averaged performance characteristics and are applicable to all engines of that model because the functions are normalized. In order to use the functions, however, it is necessary to define the sea level standard output values; that is, the value of SHP_o for each of the applicable functions.

These constants can be obtained in several ways. For a controlled variable that has a fixed maximum limit (such as a fixed maximum temperature limit), the sea level standard output value can be conveniently obtained by reading the value of referred SHP at the referred value of the controlled variable numerically equal to its limit from the appropriate gas generator characteristic as available from acceptance test data. An alternative method is to assume that the slope of the applicable gas generator characteristic is constant; then any test point can be extrapolated to obtain SHP_O at the limit value of the controlled variable.

In the case of a variable that is adjusted to trim the engine, the SHP_O value should be obtained from topping check data, because the value will depend on both the performance of the engine and the fuel control adjustment (e.g., the nominal maximum speed for the T53 is 25,400 rpm but an engine may be trimmed to limit the speed to 25,000 rpm to obtain the sea level rated power of 1400 SHP). In this case (using spool speed as an example), the value of SHP_O can be found directly from the F_N characteristic SHP_O is simply equal to the quantity required to cause the measured SHP/δ to fall on the standard F_N curve at the correct ambient temperature. This is equivalent to the calibration constant identified in the recommended approach described earlier.

When an engine's performance changes or it is retrimmed the effect on the fuel control functions is a change in the SHP_O value(s). Thus, it is only necessary to adjust those value(s) to update the MPA characteristics of the engine. The periodic topping checks that are a part of present Army maintenance practice can provide timely, sufficient data for this purpose.

4.4.2.3 Fuel Flow Limit Can Be Ignored

Although each helicopter engine has a maximum fuel flow limit, it has no bearing on maximum power available and can be ignored. The fuel flow limit curves in Figure 25 were calculated for this report based on an average linearized SHPC versus WFC relationship as illustrated in Figure 26. In this figure, an average characteristic is shown as well as one derived from the T53 model specification.

The average characteristic was calculated using 35 engine samples from the AIDAPS program, including engines in various stages of deterioration (from minor to severe cases of compressor erosion, FOD, and turbine nozzle and rotor blade erosion due to burning). The numbered curves are specific engine samples showing the largest variations from average. The relatively small variation between the various curves shows that for this engine, specific fuel consumption does not change very dramatically for its common modes of deterioration.

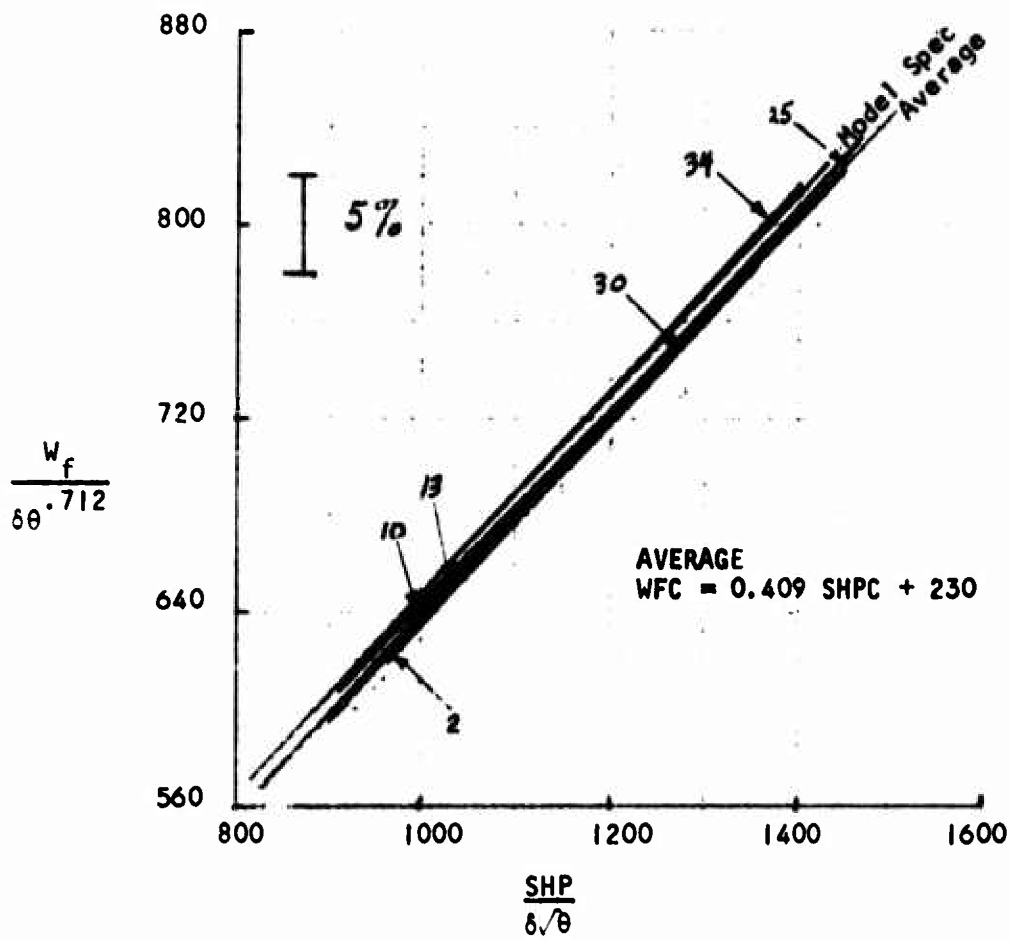

Figure 26. Referred Fuel Flow vs. Referred Shaft Horsepower for T53 Engines at Various Stages of Deterioration.

The equation for SHPC as a function of WFC for the average T53 is

$$SHPC = 2.445\ (WFC - 230) \tag{23}$$

where $SHPC = \frac{SHP}{\delta\sqrt{\theta}}$ and $WFC = \frac{W_f}{\delta\theta^{0.712}}$

The F_W curves are calculated by using a fixed value of fuel flow and values of δ and θ corresponding to various pressure altitudes and temperatures. For the average engine at sea level standard conditions the fuel flow required to produce rated power (1400 hp) is 802.6 pph.

Transmission power limitation curves are also plotted in Figure 25. Comparing these with the F_W curves, it is seen that the fuel flow limiting function can be ignored for the T53 in the UH-1 because for all combinations of pressure and temperature the torque limit is always lower than the fuel flow limiting function. This is also true irrespective of the condition of the engine. Moreover, considering that the maximum fuel flow capability of the fuel control is 900 pph (as opposed to the 802.6 pph required for rated power for the average engine), it is seen that the fuel flow limiting function would have no effect even if the SHP limit were raised to 1485 which is the engine SHP limit (as opposed to the transmission limit of 1100 SHP).

Thus, the fuel flow limiting function can be ignored for the T53 and for the other engines as well, since those engines are also derated (SHP or torque limits substantially lower than the maximum capability of the engine at sea level standard conditions).

4.4.2.4 Treatment of Temperature Limit

The T53, T55, and T73 engines do not have automatic temperature limiting, but what might be viewed as the equivalent exists in the form of operating limits on tailpipe or turbine interstage temperature. On the other hand, the spool speed schedules that are implemented in the controls for those engines are designed to limit internal temperatures. Moreover, the control schedules and temperature limits exist so that a temperature margin is provided to be "eaten up" as the engine degrades. The question is whether the equivalent of the F_T function should be considered when computing MPA for these engines.

The conclusion reached is that an exceedance of a temperature limit at military power for these engines represents an abnormal condition requiring investigative and corrective action by maintenance, but that the temperature limitation has no effect on MPA for normal operation or during an emergency. Therefore, the temperature limiting function can be ignored in computing MPA for these engines.

The above conclusion is supported by test data for the T53 engine. Figure 27 is a plot of 35 samples of temperature levels from test cell tests of T53 engines. Samples are from new, newly overhauled, aging, and

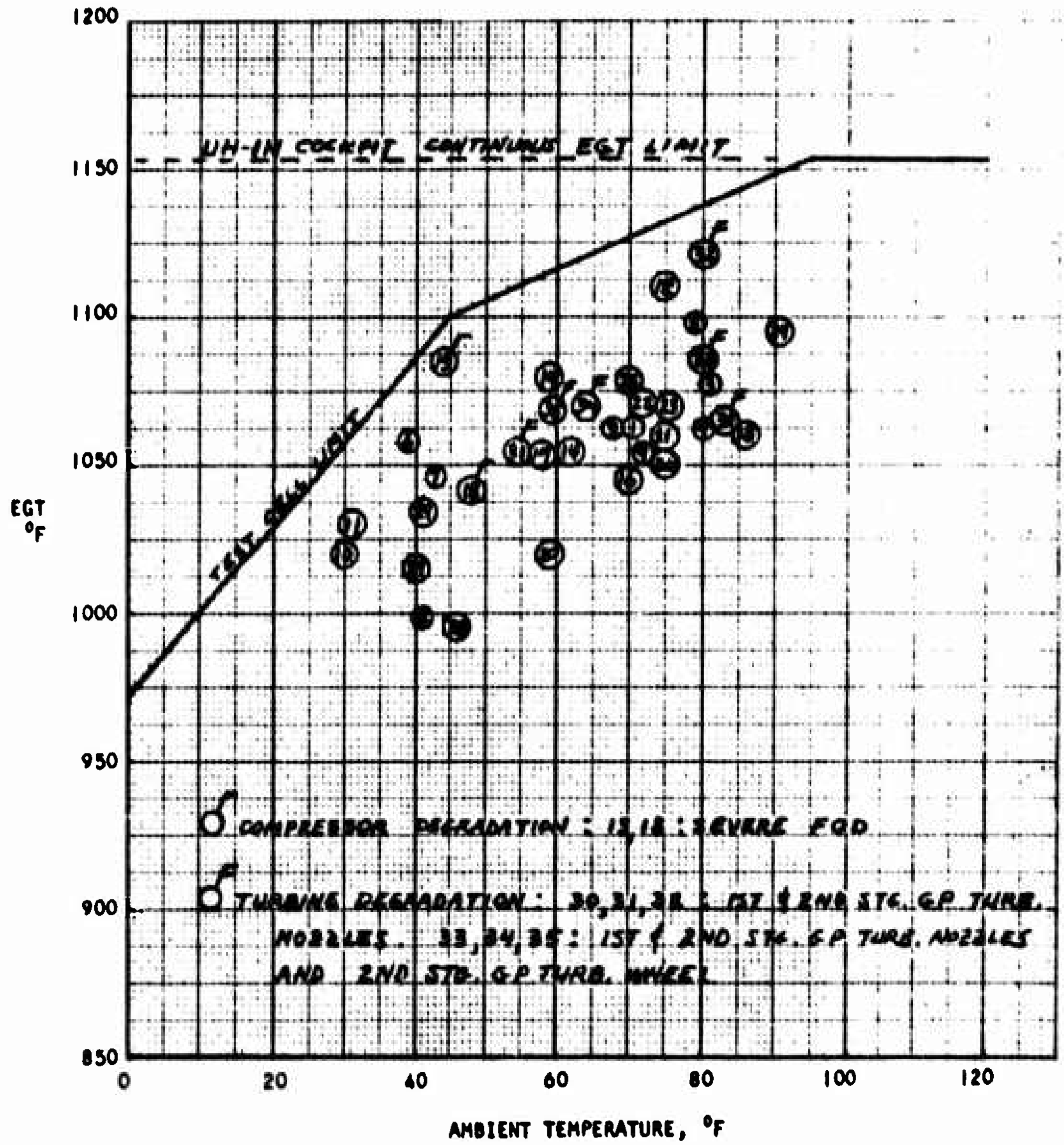

Figure 27. Exhaust Gas Temperature at Military Power for
T53 Engines at Various Stages of Deterioration.

degraded T53 engines. None of the engines exceeds the continuous operation EGT limit, not even the engines with extensive compressor or turbine damage.

The T700 represents a different case from that described above. The T700 hydromechanical fuel control limits rpm for military power in the same manner as for the other engines, but the electrical control unit continuously monitors turbine interstage temperature and keeps it from exceeding a reference value by causing the hydromechanical fuel control to reduce fuel flow as necessary. The pilot can override this function (thereby forcing the control to use only the N_G limit), however, so the definition of maximum available power for the T700 could be subject to debate. If temperature limiting can occur at maximum steady state power under normal conditions, then a corresponding F_T function should be included in the T700 MPA computation. At the time of this investigation, insufficient information is available for the T700 to resolve this particular question.

In summary, for the typical free turbine engine with a hydromechanical control, only the F_N function is needed to describe the action of the control with respect to MPA. For engines with electronic or digital controls that incorporate temperature limiting that can come into play at steady state maximum power levels, both F_N and F_T functions are necessary to characterize the control effects.

The impact of the more complicated representation on LPI design is that an additional calibration constant must be input to adjust the F_N function to the specific engine. In addition, if an engine's MPA is actually being limited by the temperature limit, then its MPA will be much more sensitive to engine deterioration. For this type engine, therefore, there is a much stronger incentive to track engine performance, and this function should be strongly considered for the LPI system.

4.4.2.5 Accuracy Considerations

As described in Section 4.4.2.1, the F_N relationship combines both performance and control characteristics, and therefore, approximates the actual process. This section develops the quality of that approximation by using the T53 engine as a representative example.

The fuel control determines the maximum gas generator speed at which the engine will operate. The individual performance characteristics of the engine, in turn, determine the power that will be developed for that speed. (There is also performance feedback through the spool speed-fuel flow droop relationship.) This process is illustrated in Figure 28 for the T53. The fuel control and gas generator are represented by two functions. The power turbine speed correction is due to the fact that the power turbine is not operated (except by chance) at the (optimum) speed for maximum power turbine efficiency. At military power the loss is only about 1.5 percent at extreme temperature conditions, but at low power levels the mismatch is much larger and the loss can exceed 20 percent (a significant problem for MPA prediction based on measured power at low power levels).

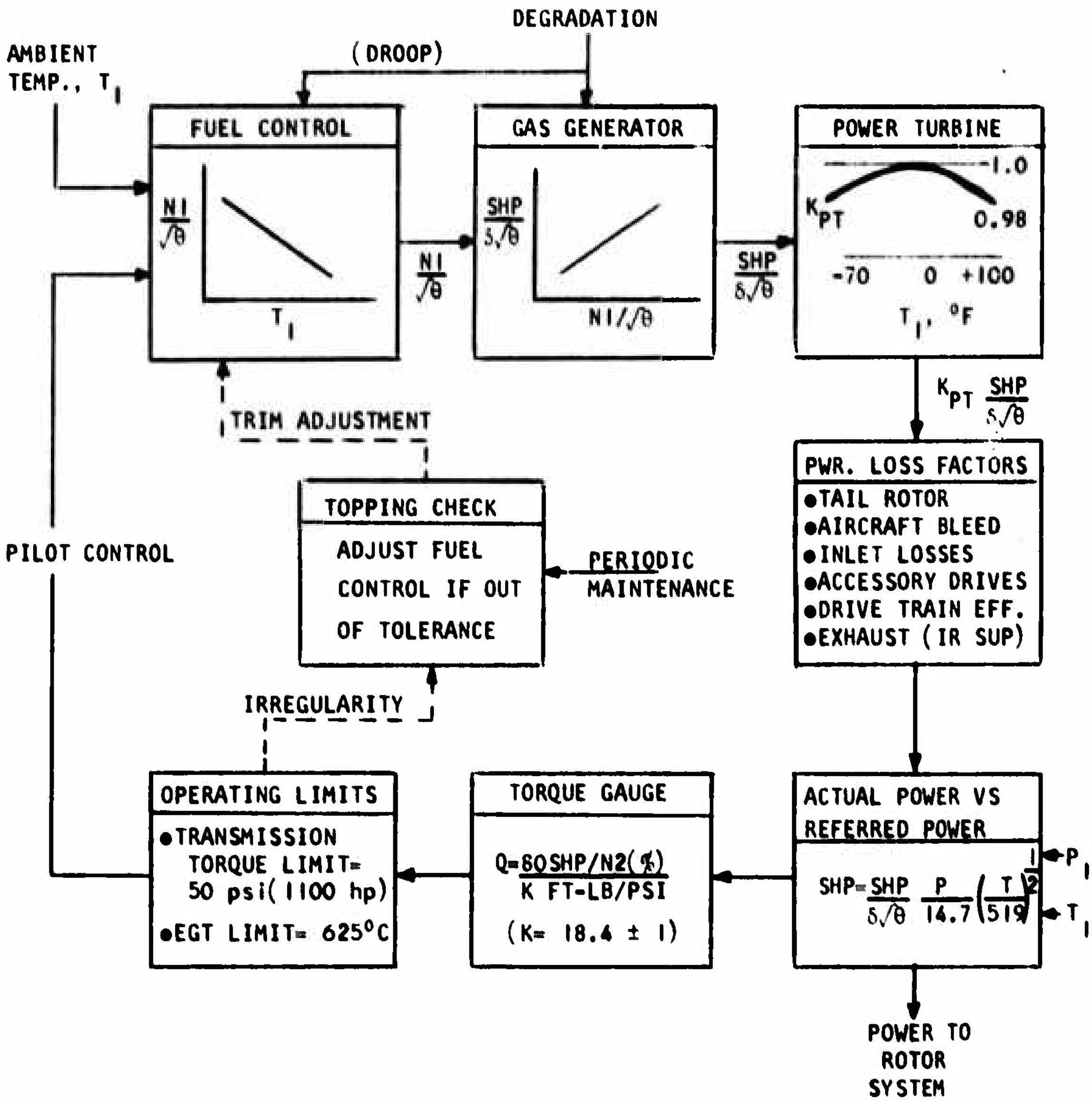

Figure 28. T53-L-13B Maximum Power Available Process.

The power losses listed in the figure are relatively small and tend to remain constant. Transmission efficiency, for example, runs at about 98 percent and does not change much with deterioration.

Actual power is related to the referred power level as shown. The pilot is charged with monitoring the power level by means of the torque gage. The transfer function for this indication illustrates that sizeable error is possible in the power display. Therefore, considerable uncertainty can be attached to the torque limited mode. (In the topping check, an engine torque calibration is employed to avoid this source of error.) The pilot is also responsible for maintaining EGT within limits; but as shown earlier, this does not affect MPA for the T53. The topping check results in adjustment of the fuel control maximum power control if maximum power is not within tolerance.

The F_N function for the T53, combined with the applicable calibration constant, represents the combined effects of the fuel control, gas generator, power turbine, and power loss factors. The potential error in F_N due to variations in gas generator characteristics was investigated by analyzing a collection of SHPC vs NIC characteristics for 35 T53 engine samples. In Figure 29, in addition to an average characteristic and the model specification characteristic, several others are plotted including some of the most widely divergent--considering both slope and offset (same given samples as used for Figures 26 and 27). Note that there is about a 20-percent spread in SHPC and a 3-percent spread in NIC among the characteristics; however, the slopes of the curves are fairly consistent.

Using these gas generator characteristics, the variations in maximum power functions were calculated as described below.

The T53 engine control is constructed so that as ambient temperature varies, the control acts to keep physical speed essentially constant by biasing the N_1 speed command as a function of CIT measured by the control. The bias can be inferred as follows.

At 59°F, the fuel flow (W_f/δ) is about 800 pph and spool speed is 100 percent (all of the conditions quoted are for military power). At 0°F, corrected speed is approximately 106 percent (N_1 = 100 percent, $\sqrt{\theta} = 0.94$) and corrected fuel flow approximately 1000 pph. Then $W_f/\delta = 1000 \cdot \theta^{0.712} =$ 918 pph. Without a CIT bias, the droop characteristic of the control would introduce an error equal to

$$\frac{918 - 800}{800} \; \frac{100}{-6.5} = -2.26 \text{ percent} \tag{24}$$

Therefore, the CIT bias commands (for a temperature reduction) an increase in spool speed to offset the droop error.

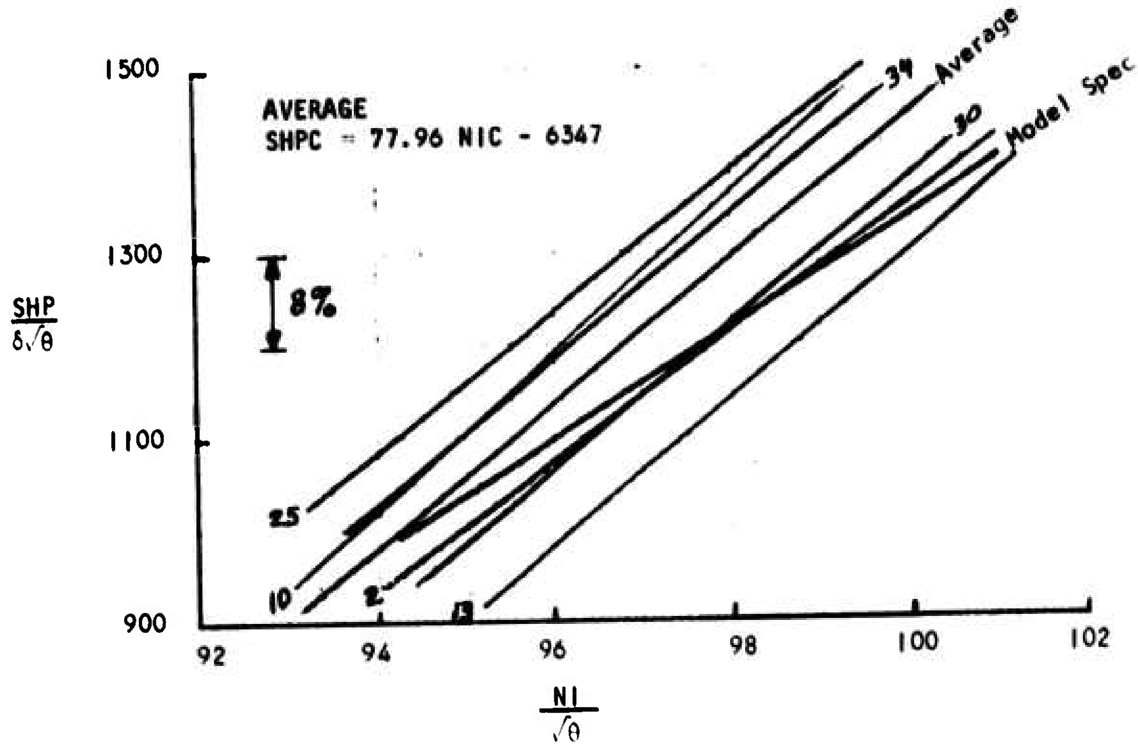

Figure 29. Referred SHP vs Referred Gas Generator Speed for T53 Engines at Various Stages of Deterioration

The CIT bias is set up to cancel the droop error for the nominal engine. For the actual engine, the scheduled speed will be in error proportional to the difference in the change of fuel flow with temperature compared to the nominal engine.

Referred fuel flow vs referred speed:

$$WFC = \frac{W_f}{\delta\theta^{.712}} = M\,\frac{NI}{\sqrt{\theta}} + B \qquad (28)$$

droop characteristic:

$$\frac{(\Delta W_f/\delta)/(W_f/\delta)}{(\Delta NI\sqrt{\theta})/(NI/\sqrt{\theta})} = -6.5 \qquad (29)$$

ΔN = error due to variation of fuel flow characteristic from nominal

$\left.\frac{\Delta W_f}{\delta}\right|_{nom}$ = variation in W_f/δ for temperature change from 59°F to T for nominal engine

$$\Delta N_{0^oF} = \frac{0.0824\left.\frac{W_f}{\delta}\right|_{59^o} + 116.5 - 6M(0.9176)}{6.5\left(\left.\frac{W_f}{\delta}\right|_{59^o}/NI_0\right) + M(0.9176)} \qquad (30)$$

To estimate the characteristic for a particular engine, the nominal schedule can be modified on the basis of the fuel flow effect previously described. The resultant rpm limit function may differ from nominal by a few tenths of 1 percent over a temperature range of 60°F. Combining this function with the SHPC vs NIC characteristic produces the MPA (referred) vs ambient temperature.

Figure 30 shows the estimated variation in the resultant characteristic for the 35 T53 engine samples that would be observed if all the engines were trimmed to produce 1400 SHP at sea level standard conditions. Because of the nature of the droop characteristic and the consistency of the WF vs SHP characteristic among the engines, the variation in the MPA characteristic is reduced by the fuel flow correction to the nominal schedule (sigma of 0.9 percent at 0°F vs 1.2 percent). The scatter due to errors in the test cell data used for this analysis is estimated to be roughly of the same order of magnitude as the scatter shown in Figure 30. (The error in the slope of the

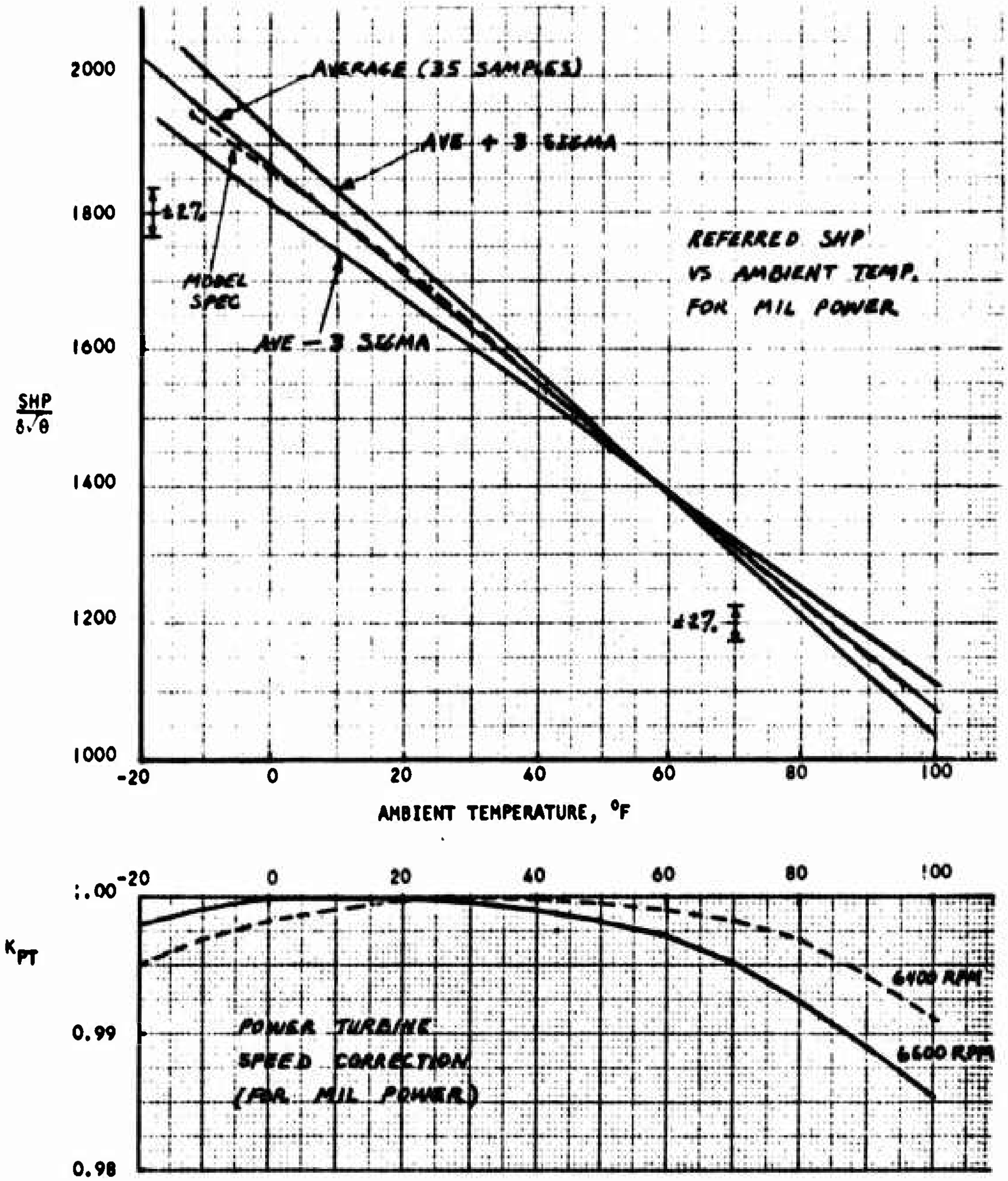

Figure 30. T53 Referred SHP at Military Power and Power Turbine Speed Correction as Functions of Ambient Temperature.

SHPC vs NIC characteristic will be due mainly to errors in N_1 and resolution. A 0.2 percent error in ΔN_1 would cause a 0.8 percent error in the SHPC estimate at 0°F.) It is concluded, therefore, that variations due to individual engine characteristics will introduce errors no larger than about 1 percent at extreme temperature variations.

The magnitude of the error is dependent on the temperature at which MPA is estimated compared to the temperature at which the engine was trimmed (or checked). For the conditions represented in Figure 30, for example, there is no error at 59°F.

The computed average characteristic shows good agreement with the characteristic derived from the T53 model spec. Figure 30 also shows the power turbine speed correction. In application, the two curves would be combined into a single characteristic: essentially the same as the F_N function shown earlier. A calibration constant obtained from the topping check would adapt the function to a specific engine and would account for variations in trim and installation losses. Periodic modification of the constant resulting from periodic topping checks would account for control variations, retrimming, and performance variations.

4.4.2.6 Application to Specific Engines

The above development for the T53 engine is equally applicable to the T55 engine. Therefore, for the T55 and T53 engines (CH-47 and UH-1), MPA would be calculated as the lesser of the spool speed power limit function (F_N), or the tranmission power limitation.

The T73 engine in the CH-54B has the same type of fuel control, but standard operating procedure is for the pilot to look up the maximum permissible value of engine pressure ratio (EPR) for the ambient conditions. MPA is then the power associated with the EPR and the ambient pressure and temperature, provided it does not exceed transmission limitations. The algorithm for the T73, therefore, consists of the SHPC vs EPR relationship for the engine with a calibration constant (also currently measured in the topping check) to account for slight variations among engines.

The algorithm for the T700 would be essentially identical to that for the T53 and T55, with an additional limiting function for the T4.5 and an additional calibration constant to relate the generic limiting function to a specific engine.

4.5 ROTOR PERFORMANCE VARIATION

As shown in Figure 29, helicopter gas turbine engine performance varies substantially among engines of the same model, and the engines are trimmed to operate at different gas generator spool speeds at military power so that each one delivers prescribed maximum power. There are no equivalent adjustments for the main rotors of the helicopters. If rotor performance varies significantly, then some sort of calibration would be required to characterize it in order to achieve high accuracy.

In contrast to the gas turbine engine, the helicopter rotor is an aerodynamically simple system. Available information indicates that calibration of main rotor performance is not required. Initial installed performance levels of main rotors for a given helicopter model are not measurably different. With degradation, main rotor performance can decrease a few percentage points, but such degradation is readily detectable visually and the performance of a serviceable rotor can be restored. Therefore, under routine maintenance, rotor performance is expected to remain effectively constant.

Experienced helicopter pilots report noting differences in performance among helicopters of the same type. It is difficult, however, to quantify this experience and isolate the prime factors. High speed performance tends to degrade with age due to increased fuselage drag (due to added coats of paint, for example).

Flight tests of various models of the UH-1 provide an interesting comparison relative to initial performance levels among helicopters. Figure 17 showed nondimensional HOGE performance curves for three different UH-1 models. The C, D, and M models appear to belong to the same class and have virtually the same performance characteristic at the higher nondimensional power levels, even though the C model apparently has a different main rotor design (different solidity).

The helicopter main (and tail, as applicable) rotors are subject to deterioration due to sand dust effects, for example. The best source of information on the magnitude of rotor degradation effects may be the desert environment tests that have been conducted on most military helicopters. An Army flight test report for one such test on the YOH-6A states that "with the main rotor blades eroded to an unserviceable condition, a 1.2 percent increase in the power required to hover resulted."[17] Equivalently, at maximum power the lift capability of the vehicle would be reduced by approximately 0.8 percent. Considering that the rotor blades were unserviceable, requiring replacement, this indicates a negligible change in performance for lesser degrees of deterioration.

Deterioration effects can be expected to vary among helicoper types, some models showing markedly more tolerance (as in the case of the YOH-6A). For example, leading edge roughness resulting from erosion is known to lead to premature blade stall. For a rotor designed to operate in conditions close to the stall boundary, any significant deterioration of the blade surface might lead to stall flutter in conditions where it previously did not occur.

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[17] "Engineering Test of YOH-6A Helicopter in the Desert Environment," Letter Report STEAV-EN, 15 June 1966, U.S. Army Aviation Test Activity, Edwards AFB, Calif.

Although this situation decreases the maximum lift coefficient, the more troublesome effect is severely increased loading on rotor linkages. The effect is most pronounced at high speed flight, where premature blade stall causes high linkage loading to occur at lower flight speeds.[18] For modern helicopters incorporating cruise guide indicators that measure loads on selected main rotor linkages, the progression of the effect will be seen as a lowering of allowable airspeed and will result in a pilot complaint. Maintenance action would be to inspect main rotor blades and blend out leading edge roughness. The type of deterioration that produces these effects is visible and can be gaged, with experience, by running one's hand over the surface.

Flight tests have shown that rotor blade performance is insensitive to small changes in surface features. For example, the addition of raised tip lights and leading edge nickel-plated abrasion strips to the main rotor of the CH-53D helicopter did not change its lift vs power characteristics.[19]

Thus, flight test results show that (1) the helicopter rotor tolerates significant deterioration without appreciable performance change; (2) where the deterioration is sufficient to alter performance, the onset is likely to be noticed as a lowering of V_{ne} in advance of significant lift loss; (3) the types of deterioration are manually detectable and correctable (for serviceable rotors) at the flight line; and (4) the initial installed performance levels of rotors for the same helicopter models are equal (within tolerance).

As a practical matter, calibration of rotor performance should be avoided if possible because of the uncertainties in the measurement. In his study of helicopter flight test data, Law[20] notes that the nondimensional hovering characteristics data that he has reviewed show a maximum deviation of approximately ± 5 percent in every case. This means that the expected error in a single measurement (with flight test instrumentation) would be about 1.5 percent, and in order to obtain a calibration to within 1 percent accuracy to a 98-percent confidence level, at least 25 independent measurements would be required (100 measurements of 0.5 percent accuracy).

18
P. Brotherhood, "Some Aerodynamic Measurements in Helicopter Flight Research," Aeronautical Journal, October 1975.

19
C.N. Jubeck, "Development and Testing Monitor of the CH-53D Helicopter," NATC Report No. FT-49R-70, Naval Air Test Center, Patuxent River, Maryland, 19 May 1970, AD 883339.

20
Harold Y.H. Law, "Two Methods of Prediction of Hovering Performance," Report No. USAAVSCOM-TR-72-4, U.S. Army Aviation Systems Command, St. Louis, Mo., February 1972, AD 738531.

Calibration should therefore not be considered unless systematic errors in the performance estimate are expected to exceed 1 to 2 percent (either initially or due to deterioration).

4.6 MECHANIZATION

Several earlier efforts serve as partially representative models of both analog and digital approaches to mechanization of LPI functions. A design disclosure for a JANAIR-sponsored analog system for an effective weight margin approach is reported by Edgerton and Williams[21]. Reference 8 reports the results of flight test of that system by the Army. With the addition of weight and c.g. measurement and computation, that system would represent the mechanization of an analog approach for generating one of the recommended LPI information displays, namely, hover capability. With the addition of built-in test circuitry conforming to current standards, that system would cost as much as a more flexible digital system that would provide the full range of recommended LPI functions.

Reference 6 describes a high-performance digital computer mechanization of the thermodynamic model MPA prediction approach (described in Section 4.4.1.1). That system, with a different complement of sensors and corresponding input/output changes and with a modified control/display unit, would meet LPI system requirements.

Reference 16 describes the synthesis of a "helicopter flight performance system" from off-the-shelf LSI microcomputer set components. This system amounts to a fixed program calculator that duplicates the hover capability chart operations using manually input values for ambient temperature and pressure (for an estimated cost of less than $500 per unit). This system is far from representative of recommended LPI capabilities, but it does represent what might be considered the core of the most practical approach for mechanization of the LPI system.

The LPI system will comprise a control-display unit, computer unit, and sensors. Functions of the unit are indicated in broad terms in Figures 1 and 2. The control-display unit (Figure 2) will include an alphanumeric character display, controls for selecting desired display modes, and provisions for manual input of remote-site conditions and wind velocity. The computer unit will consist of circuitry for signal acquisition and conditioning, analog-to-digital conversion, information processing, and output and control interface. The processor will consist of an LSI microcomputer set. Memory requirements are estimated to be approximately 8000 bytes (8 bits/byte) ROM and 512 bytes RAM.

21
Bradford W. Edgerton and Sidney B. Williams, "Final Report for Helicopter Lift Margin System," JANAIR Report No. 731003, December 1974.

Representative sensor selection is as follows:

Ambient temperature: platinum resistance probe (±0.5 to 1°F)

Ambient pressure: quartz variable capacitance (±0.1 percent)

Strut pressures (4 typical): LVDT (±0.5 percent)

Attitude, longitudinal: liquid resistance gage (±1°)

Cargo hook load: strain gage load cell (±2 percent)

System unit cost is estimated at $10,000 in medium scale production quantities, excluding any unsticking devices and installation cost.

5. CONCLUSIONS

5.1 GENERAL

(1) Helicopter lift performance indicator systems that would inform pilots of the likelihood of successful takeoffs and landings before committing the aircraft to flight are feasible for the aircraft addressed in this study, except that in the case of the UH-1, significant redesign of the skid-landing gear would be required to accommodate and facilitate incorporation of weight measurement instrumentation.

(2) Of the helicopters surveyed, the CH-47C shows the greatest need for a lift performance indicator system based on aircraft configuration, load capacities (internal and external), and typical missions and cargo types.

(3) The most logical site for test of a lift performance indicator system in an operational environment is at the 197th Aviation Company, Fort Carson, Colorado, because of the mission and high density altitude.

(4) The lift performance indicator concept is best suited to cargo helicopters because cargo loads (in comparison to personnel and armament) produce larger and more-difficult-to-estimate variations in gross weight and center of gravity.

5.2 PILOT OPINION SURVEY

(1) Present procedures relating to lift performance estimation, particularly weight estimation, are inadequate. In effect, the present operational lift performance criterion is: if the helicopter will lift off the ground, the aircraft is committed to flight. This check is adequate with respect to safety except for the following potential conditions, (1) landing at a higher density altitude, (2) shutdown of one engine, (3) HOGE maneuver for NOE profile, and (4) confined area takeoff (obstacle clearance).

(2) A lift performance indicator system would speed operations and improve safety. Potential benefits are greatest for combat operations where maximum demands are made on speed and vehicle loading.

(3) Utility of the lift performance indicator system will vary with geographic location of the aviation unit and with maximum utility at locations of high density altitudes.

(4) At very low density altitudes where lift performance is at maximum gross weight capability, the most useful potential system outputs are aircraft gross weight and c.g.

5.3 SYSTEM REQUIREMENTS

(1) Due to the variety of takeoff and landing situations, no single lift performance criterion (and corresponding display parameter) can serve as the basis for the decision to commit the aircraft to flight.

(2) Takeoff or landing capability for a given situation can be assessed in terms of the ability of the aircraft to perform one or more of the following maneuvers (for the vehicle weight and prevailing ambient conditions):

(a) Hover out-of-ground effect

(b) Hover in-ground effect

(c) Vertical climb

(d) Climb at best airspeed

(e) Maximum performance obstacle clearance takeoff

(3) The capability to estimate lift performance for landing or takeoff at a remote site is required. This requires provisions for selecting this mode and manually inserting the pressure altitude and temperature for the remote site.

(4) The capability to estimate lift performance for landing while in flight and prior to initiating the landing approach is required. This requires that aircraft gross weight be updated for fuel consumption and cargo hook loads.

(5) The measurement of cargo hook loads is required with accuracy commensurate to required accuracy of vehicle gross weight measurement.

(6) Prior to advanced engineering development of a lift performance indicator system, the scope of potential functions should be broadened to include review of in-flight, real-time functions that might be required to support or facilitate decisions relative to NOE flight.

(7) A correction factor to account for changes in rotor condition is not required. Rotor degradation causing significant variation in lift performance is visually detectable and correctable (if the rotor is serviceable) at the organizational maintenance level. Routine inspections (as currently implemented) therefore obviate the need to calibrate or track rotor performance degradation.

(8) A direct density measurement is not required to achieve a sufficiently accurate estimate of maximum available lift.

(9) Based on review of present operational criteria and performance sensitivities for the helicopters surveyed, it appears that errors beyond ±3 to ±5 percent are unacceptable and that accuracy better than to within ±1 percent is not warranted.

(10) In addition to gross weight, longitudinal c.g. location should be computed and displayed.

(11) Capability of measuring aircraft weight on the ground during dynamic conditions (rotor turning) is desirable, especially in combat operations.

5.4 LIMITATIONS AND CONSTRAINTS

(1) Pilot technique will affect attained lift performance in the following areas:

(a) Maximum performance takeoffs from confined area (obstacle clearance with heavily loaded helicopter), and landings for similar conditions.

(b) Single-engine operation.

(c) Operating in moderate to severe wind conditions.

(2) Wind significantly affects helicopter performance capabilities, but due to variations in wind magnitude and direction (e.g., gusts and vehicle orientation), correction of performance estimates for wind effects should not be automatic. (Provided the aircraft is oriented into the wind, performance in wind is always greater than in zero wind.)

(3) Observance of torque limitations can introduce variations up to about ±5 percent with respect to the nominal power available in the torque limited regime due to torque measurement and display accuracy, actual rotor rpm set by the pilot, and the precision with which the pilot observes the torque limitation.

(4) Flight test results for various helicopters show that with flight test instrumentation, maximum scatter in lift vs power measurements amounts to about ±5 percent, indicating that if calibration of a lift performance indicator were required, a single point calibration would be insufficient.

(5) No technique has been advanced for monitoring the maximum power related control functions of the engine controls at reduced power levels (this would amount to a part power trim check of the operational engine).

(6) Typically, where the weight vs lift relationship is near critical (and consequently, the lift performance indication of greatest use), the helicopter will be capable of a maximum performance takeoff (horizontal acceleration IGE before initiating climb), but incapable of HOGE; consequently, any technique calling for information from a HOGE maneuver is of questionable value.

5.5 WEIGHT MEASUREMENT TECHNIQUES

(1) Oleo Pressure Weight Measurement Approach

(a) With the basic, uncompensated technique, accuracy to within approximately ±2 percent is achievable for static conditions (rotors not turning) and ground slopes within ±5 degrees of level.

(b) To achieve accuracy to within ±1 percent and satisfactory operation for all static conditions, some type of antistiction means must be employed.

(c) Potentially the most practical antistiction approach is the zero-friction technique, wherein a simple electromechanical or hydraulic actuator causes the oleo strut piston to momentarily oscillate (angularly) within its cylinder and the resulting pressure is used to compute gross weight.

(d) Strut unsticking can also be accomplished by functional means such as operating at flight idle power, then reducing the power level to ground idle for pressure measurement. Such measures can be easily accommodated in existing preflight procedures, but may be considered an unacceptable constraint.

(2) Strain Gage Weight Measurement

(a) Strain gage weight measurement systems employing shear deflection sensors mounted inside the wheel axles are feasible for helicopters with wheel type landing gear. Development of stable (no drift) and accurate (to within about ±1 percent) systems requires development of sensor-axle combinations that are likely to involve significant experimentation and empirical design in each particular case.

(b) Externally mounted strain gage deflection sensors are also feasible, given the freedom to redesign the landing gear to accommodate them.

(3) Skid-Type Landing Gear

(a) Skid-type landing gear as represented by the UH-1 present unique and individual design cases and no generalized design approach is possible.

(b) Incorporation of load cells into the support points, a substantial modification, could provide the basis for feasible weight measurement for the UH-1. Experimentation would be required to establish feasibility.

(4) Weight Measurement Under Dynamic Conditions (Rotors Turning)

During ground operation, the thrust produced by the rotor operating at minimum collective is significant and cannot be ignored. Attempts with previous experimental weight measurement systems for the CH-47 to accurately measure weight and balance during dynamic conditions (flight idle power levels) have been unsuccessful, although an aerodynamic approach was successful for certain conditions. Test data and measurements in those tests were insufficient to isolate the cause of the errors encountered during the dynamic conditions.

(5) Measurement of Cargo Hook Loads

The feasibility of measurement of cargo hook loads has been demonstrated.

(6) Aerodynamic Weight Measurement Approaches

(a) The technique of measuring the power required to sustain the helicopter in the air and using that to calculate effective weight can be made relatively accurate, but cannot provide information while the aircraft is on the ground.

(b) To obtain reasonable accuracy, low (relative) airspeed measurement is required for airspeed correction. If weight is to be measured in ground effect, input of vehicle height is required (to within about ± 1 ft). High accuracy also requires vertical rate of climb input or maintenance of rate between about ± 0.5 fps.

5.6 PERFORMANCE CAPABILITY ESTIMATION

(1) Definition of Maximum Power Available (MPA)

(a) MPA is power resulting from automatic engine governing (engine fuel control, etc.). On T700 engine, the temperature limiting function can be overridden by the pilot giving rise to question of treatment.

(b) Manually applied operating limitations and standard procedures (principally, transmission torque limitation are involved).

(2) Thermodynamic Model MPA Prediction Technique

(a) Technique is impractical due to complexity and insufficient accuracy.

(b) Analytical studies failed to evaluate effects of engine trimming, topping checks, retrimming, measurement problems in hovering IGE or light-on-skids, and manually applied operating limitations (principally torque limits).

(c) An analysis of the variation of actual MPA vs nominal (performance chart) MPA is needed to evaluate the utility of proposed approaches for estimating actual MPA.

(3) MPA Computation for T53 and T55 Engines (UH-1H and CH-47C)

(a) MPA is equal to the smaller of the transmission torque limit and power computed from spool speed limiting function.

(b) Implementation of temperature limiting and fuel metering limit is not required.

(c) Calibration of the MPA function requires input of SHP reference value for the spool speed limiting function based on presently implemented topping check. Perform periodic recalibration based on topping check results performed under current procedures or in response to pilot squawk.

(4) MPA Computation for T73 Engine (CH-54B)

(a) MPA is equal to the smaller of transmission torque limit and power computed from SHPC vs EPR relationship for the EPR vs ambient temperature schedule (the latter as specified in current procedures).

(b) Implementation of spool speed limiting, temperature limiting, and fuel metering limit is not required (assuming adherence to current procedures for the CH-54B).

(c) Calibration of MPA function requires input of SHP reference value for EPR function obtained from topping check. Perform periodic recalibration based on topping check results performed under current procedures or in response to pilot squawk.

(5) MPA Computation for T700 Engine (UTTAS)

(a) Policy decision relative to temperature limiting function, which can be overridden by pilot, is needed.

(b) MPA is equal to the smaller of the transmission torque limit, power per spool speed limiting function, or power per temperature limiting function.

(c) Calibration of MPA function requires input of SHP reference values for the spool speed and temperature limiting functions with periodic updating.

(d) Due to the temperature limiting function of the T700 controls, the T700 is much more sensitive to engine degradation, assuming that the nondegraded engine is subject to temperature limiting at maximum power.

(6) Power Margin Techniques

(a) Power margin techniques (actual power being used vs maximum available power for present conditions) represent the simplest method of generating lift performance information and introducing the least error; but these techniques have the following disadvantages:

(1) Power margin information cannot be generated before lift-off.

(2) It is difficult to translate an indicated power margin to the power margin that would be obtained for a different flight mode (at a minimum, a relative wind measurement is required).

(b) A manually implemented power margin technique is presently used on the UH-1 as a lift performance criterion.

(7) Effective (Aerodynamic) Weight or Lift Margin Technique

(a) Lift margin is obtained, in effect, from power margin by converting power to an equivalent amount of lift (at HOGE). The technique has good potential accuracy, but suffers from the following disadvantages:

(1) Lift margin information cannot be generated until the helicopter is airborne.

(2) In order that effective weight or lift margin represent actual weight, the conditions under which they are generated must be tightly controlled (a precise flight mode must be achieved, usually HOGE with zero relative wind) or else precise corrections must be introduced.

(b) A lift margin system was recently evaluated by USAAEFA. Based on the test results, USAAEFA concluded that if the requirement to fly a weighing maneuver each flight in order to generate the lift margin indication could be eliminated, and if the weighing maneuver could be made more accurate (with the addition of a low airspeed measurement system, among other things), the lift margin system would be satisfactory for its intended use.

6. RECOMMENDATIONS

The present study has established the functional requirements and feasibility of the LPI system concept. Further effort is required prior to advanced development to develop certain design characteristics, to demonstrate the system concept and performance capabilities, and to evaluate the utility of the concept.

Toward this end, a two-phase program is recommended that would develop the concept for a specific aircraft and test and experimental system in both engineering flight test and operational unit settings. Prime objectives of these programs would be (1) to demonstrate that the required technology is well in hand and (2) to furnish evidence of the utility of the concept.

The CH-47C is recommended as a test vehicle.

6.1 BREADBOARD DEVELOPMENT

The objectives of this phase are to develop a system design for the CH-47C and to implement and demonstrate that design in a breadboard system. Elements of the breadboard system would be as follows:

(a) Laboratory of production version of a flyable general-purpose computer

(b) Interface unit for simulated measurements

(c) Developmental controls and displays

(d) Detailed logic and algorithms

One of the prime uses of the breadboard system would be the development of system controls and displays.

Development of a computer specifically tailored to this application is not recommended for these initial phases of concept formulation. That area of technology is well in hand and need not be demonstrated. To minimize cost, a laboratory or production version of an airworthy computer should be leased to the program. A flightworthy version of this computer would be used in the flight test. Algorithms and logic should be developed specifically for the CH-47C and be published in general form. Documentation of programs for the selected computer can be minimal.

Additional tasks that should be accomplished in this phase are:

(a) Planning for the following phase (test design, in particular).

(b) Installation design (including selection of sensors).

(c) Preliminary design of measurement interface unit (or identification of a suitable off-the-shelf unit that could be obtained for flight test).

(d) Design analysis for zero-friction antistiction device for CH-47C oleo shock struts (see paragraph 6.3).

6.2 FLIGHT TEST

The objective of this phase is to evaluate the design and operating characteristics of an experimental lift performance indicating system designed for the CH-47C. Planning and coordination for this testing should be accomplished in the preceding phase.

Design and fabrication would be accomplished as necessary to primarily adapt off-the-shelf components (computer, sensors, and measurement interface unit) to the system design. The computer logic developed in the preceding phase would also be used in this phase.

Two distinct test phases are recommended. In the first phase, perhaps best suited to the test activity at Edwards AFB, system accuracy would be assessed for a variety of test conditions. A second phase is recommended in which the overall usefulness and operational characteristics of the system would be evaluated in an operational setting at the 179th Aviation Company at Fort Carson, Colorado.

6.3 ZERO-FRICTION ANTISTICTION DEVICE

This technique should be developed for a selected test vehicle (CH-47C recommended). Two distinct phases are recommended:

Design Analysis--Considering (as a minimum) electromechanical and hydraulic approaches, the optimal implementation of this antistiction approach should be identified on the basis of reliability, maintainability, and operational suitability (least impact on operations). This effort would include preliminary design of the selected mechanization approach.

Detailed Design, Fabrication, and Test--Design, fabrication, and test can be conducted concurrently with the LPI system electronics provided that the need for a laboratory test program is not identified.

6.4 DYNAMIC WEIGHT MEASUREMENT MODE

The developmental LPI system should include a selectable dynamic (rotors turning) weight measurement mode where residual rotor thrust effects are estimated by means of the aerodynamic technique (at given rotor rpm, collective, and cyclic the rotor thrust vector is a function of essentially only air density).

To enable evaluation of this operating mode, the means of displaying individual strut load measurements should be included in the system. Additionally, platform scales should be provided to obtain an independent measurement of weight on wheels. Additional filtering may be required to facilitate reading the platform scale outputs.

Planning for this aspect of the flight test phase should be accomplished in the breadboard development phase.

6.5 ADVANCED WEIGHT MEASUREMENT CONCEPTS

Two development efforts should be considered relative to weight measurement systems for future helicopters (including advanced models of existing helicopters):

> Strain Gage Weight Measurement--Developments for large, fixed-wing ircraft (747, DC-10, L-1011) have shown that weight measurement systems based on strain gage shear deflection sensors can be successful and that their development should be initiated during the landing gear design stage to produce an integrated design. This approach merits consideration for future helicopter weight measurement systems.

> Integral Strut Antistiction Designs--The oleo pressure weight measurement approach with the zero-friction antistiction technique is considered competitive with the strain gage approach for future helicopters. It appears, however, that the oleo pressure approach might be considerably improved by integrating the antistiction device within the strut. The potential gain in this area is associated with the conclusion that the oleo pressure approach can be a very simple, accurate, practical weight measurement approach provided that the means of handling stiction is practical. Therefore, a small scale design feasibility study should be made to assess the potential of the integral antistiction concept.

6.6 INVESTIGATION OF RELATED FUNCTIONS

Prior to advanced engineering development of an LPI system, the scope of potential functions should be expanded to include consideration of the following:

(1) Lift performance information needed relative to NOE flight decisions ("altitude margin", for example, has been mentioned)

(2) Energy management information needs (e.g., range and endurance)

(3) Other real time, in-flight needs for lift performance type information

The task effort would include identifying functions that should be considered for integration into the LPI system, and then analyzing those functions to determine mechanization and logic requirements and the resulting impact on the baseline LPI system design.

Relative to a baseline, digital LPI system, addition of most of the above functions requires only an extension of logic and minor increase in controls. It is therefore recommended that they be included in the breadboard version.

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APPENDIX A

AIR DENSITY AND HUMIDITY

1. SUMMARY

Ambient pressure and temperature must be available as separate variables for computation of maximum available power for the gas turbine engines considered in this study. For estimating rotor thrust, however, density is needed. These basic facts prompted the following questions (with conclusions in parentheses):

(1) Does the pressure-temperature model for atmospheric density offer sufficient accuracy relative to lift performance computations? (Yes. Simply ignoring other factors such as humidity introduces errors no larger than about ± 0.4 percent in the lift performance results for extreme, worst-case humidity variations.)

(2) Is a direct density measurement needed? (No.)

(3) Is a humidity correction needed? (No.)

(4) Are any corrections for density or humidity needed relative to engine maximum power available? (No.)

(5) What is the status of nucleonic direct density measurement devices? (Still promising, but developmental units have not demonstrated sufficient accuracy to be considered competitive with indirect techniques.)

2. TEMPERATURE-PRESSURE MODEL FOR AIR DENSITY

The temperature-pressure model for atmospheric density is developed from the ideal gas equation and is as follows:

$$\rho = \rho_o \frac{P}{P_o} \frac{T_o}{T} \tag{1}$$

where

ρ = density

P = pressure

T = temperature

and the subscripted quantities are the values of the variables at sea level standard condition.

Equation 1 is an accurate expression for density as long as the molecular weight of the air remains constant. The only factor that causes any significant variation in the molecular weight (for altitudes below 100,000 ft) is humidity. Dry air and water vapor mixtures behave as two gases, with the water vapor lowering the molecular weight of the mixture since its molecular weight is 18 compared to about 29 for dry air. The amount of water vapor in moist air, however, never exceeds about 4 percent, so that moist air density is never more than about 2 percent lower than dry air at the same conditions of pressure and temperature.

A number of relations have been developed to show the quantitative variation of density with humidity. Two of the most convenient of these are reproduced below.

$$\frac{\rho_{ma}}{\rho_a} = \frac{1 + H_S}{1 + 1.608\, H_S} \tag{2}$$

$$\frac{\rho_{ma}}{\rho_o} = - 0.1434\, \frac{e}{T} \tag{3}$$

where

ρ_{ma} = density of moist air

ρ_a = density of dry air (at same temperature and pressure)

ρ_o = density at sea level standard conditions

H_S = specific humidity (lb moisture/lb dry air)

e = vapor pressure of moisture (mm Hg)

T = absolute temperature (°K)

Maximum values of water content with respect to humidity effects are generally quoted at about 1 in. Hg vapor pressure (specific humidity of about 0.022 lb water vapor per lb dry air). For comparison, Cormier[22] indicates that the coastal regions surrounding the Persian Gulf, Gulf of Aden, and the Red Sea have the world's most severe joint high-temperature, high-humidity environment--the highest atmospheric water content.

[22] René V. Cormier, World Wide Extremes of Humidity with Temperatures Between 85° and 120°F, AFCRL-TR-74-0603, 5 Dec. 1974.

For that area, the following frequency of occurrence of high specific humidity is given:

Specific Humidity (lb/lb)	Probability (percent)	$\frac{\rho_{ma} - \rho_a}{\rho_a}$ (percent)
> .032	0.1	-1.85
> .028	1.0	-1.63
> .024	5.0	-1.41
> .022	10.0	-1.29

The values quoted at 0.1 percent and 1.0 percent probability exceed the high-temperature, high-humidity design envelope specified in MIL-E-38453A. On the other hand, some investigators use values of saturated U.S. Army summer air as a criterion. This results in a maximum water content of 0.043 lb/lb and a corresponding change in density of -2.44 percent compared to dry air. The value 0.032 lb/lb specific humidity will be used here as maximum water content.

The values of specific humidity quoted here are applicable to the sea level pressure range. It is assumed that specific humidity tends to remain constant with changes in altitude (due to, for example, the temperature lapse rate with altitude).

3. EFFECTS ON MAXIMUM AVAILABLE ENGINE POWER

Determining the effects of humidity on maximum available engine power is a relatively complicated task. In addition to affecting density, humidity alters the thermodynamic properties of air; namely, specific heat, the adiabatic index, and the gas constant for the moist air. The end result, however, is that the effect of humidity on maximum available power of a free turbine turboshaft engine is negligible.

Is an analysis performed for an in-house free turbine engine, the results showed negligible change in shaft horsepower output for constant inlet pressure and temperature and constant physical gas generator speed as humidity was varied between extremes. The calculations were performed in a cycle-balancing program using essentially the same technique as Fishbein.[23]

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23 B. D. Fishbein et al, Determination of the Effects of Atmospheric Humidity on the Characteristics of a Turbofan Engine, FTD-HT-23-290-68, 1970.

The fact that a humidity correction is not employed in any of the guaranteed performance check procedures for the engines considered in the study indicates the general applicability of the above result*.

For comparison, the effect of humidity on the thrust of turbojet and turbofan engines was reviewed (J33, J35, J47, J73, J85, J79 and TF41). The average humidity correction amounted to the following for a vapor pressure of 1 in. Hg (equivalent to a specific humidity of 0.022 lb moisture/lb dry air):

Thurst correction: + 0.3 percent

RPM correction: -0.5 percent

Fuel flow correction: -1.5 percent

EGT correction: +0.25 percent

(The humidity effect is the opposite of the correction.)

Considering a thrust/rpm ratio at maximum power of 5:1, the humidity effect on thrust at constant rpm would amount to the order of -3 percent.

4. ROTOR THRUST VARIATIONS WITH DENSITY

Density enters directly into the lift vs power characteristics of the helicopter. Figure A-1 illustrates the effect of density measurement errors on the estimation of lift performance. (All lift performance indication approaches would employ this characteristic.)

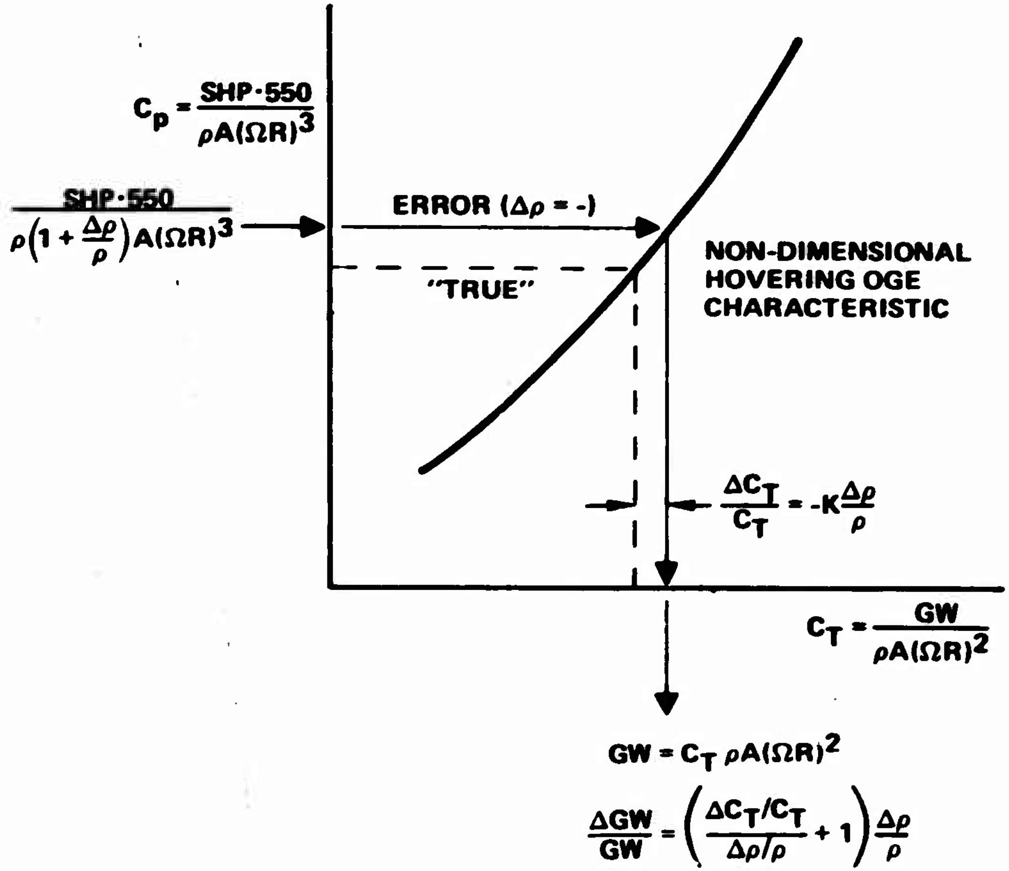
The effect of a negative error in density is illustrated. The error results in the calculated value of the power coefficient, C_P, being higher than actual. The corresponding thrust coefficient, C_T, is commensurately high. When multiplied by the lower-than-actual density, the resulting gross weight estimate is in error on the low side. It is seen that partial compensation of the density error takes place so that the resulting influence coefficient is less than one.

Some investigators have offered an analytical derivation of the density error effect based on an equation of the following form:

$$C_P = \frac{C_T^{3/2} + K}{\sqrt{2}}$$

K in the above equation is not constant, however, but is a function of C_T so that differentiation is not readily possible.

*One could expect a correction to be employed if the effect exceeded about 0.5 percent at its extreme.



APPROXIMATION

$$M = 0.707 \frac{C_T^{3/2}}{C_p}$$

$$GW = (550 \cdot M \cdot SHP)^{2/3} (2A\rho)^{1/3}$$

ASSUMING M CONSTANT
EFFECTS OF ERRORS ON
GROSS WEIGHT CAPABILITY
ESTIMATE ARE

$$\frac{2}{3}\frac{\Delta SHP}{SHP}, \frac{1}{3}\frac{\Delta\rho}{\rho}, 0\frac{\Delta\Omega}{\Omega}$$

INFLUENCE COEFFICIENTS

	UH-1H	CH-47C	CH-54B
SHP	0.72	0.66	0.63
ρ	0.27	0.36	0.38
Ω	-0.15	0	0.11

Figure A-1. Effect of Density Error on Estimate of Gross Weight Capability.

The most satisfactory way of determining the density error effect is to duplicate the process of Figure A-1 using the actual C_P vs C_T characteristic of the helicopter.

Figure A-2 shows the nondimensional HOGE performance characteristics for the UH-1H, CH-47C, and CH-54B helicopters. In the case of the CH-54B, two curves are shown because of the significant difference between data derived from the operator's manual and flight test data.[24] Using the method shown in Figure A-1, the influence coefficients for density errors were calculated and the maximum errors due to humidity were determined. The results are as follows:

Aircraft	$\frac{\Delta GW}{GW} / \frac{\Delta\rho}{\rho}$	$\frac{\Delta\rho}{\rho}$ humidity (percent)	$\frac{\Delta GW}{GW}$ (percent)
UH-1H	0.27	1.85	0.50
CH-47C	0.36	1.85	0.67
CH-54B	0.38	1.85	0.70
(Op. Man.)	0.51	1.85	0.94

The error due to neglecting humidity is positive, since the calculated density of dry air would be higher than actual. Note that the maximum error due to humidity could be limited to half the above values by assuming a midrange value of humidity.

It is concluded that humidity exerts a relatively negligible influence on lift performance, except possibly in the case of the CH-54B (where the effect can still be reduced to less than 0.5 percent).

A relatively simple refinement can be employed where the effect is considered marginal. This would amount to a "climate correction". Just as the moisture content of saturated air is a function only of temperature, the moisture content of air for a particular climate is a function of temperature. In this respect, climates could be classified as dry, moderately humid, and humid. To illustrate, we have found that the specific humidity at sea level pressure can be expressed as a function of temperature and relative humidity, as follows:

$$H_S = \frac{H_R}{730}\, e^{0.0346\, T} \qquad (5)$$

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24 John N. Johnson et al, Limited Performance Tests, CH-54B (Tarhe) Helicopter, U. S. Army Aviation Systems Test Activity, Edwards AFB, Ca., February 1973, AD 910263.

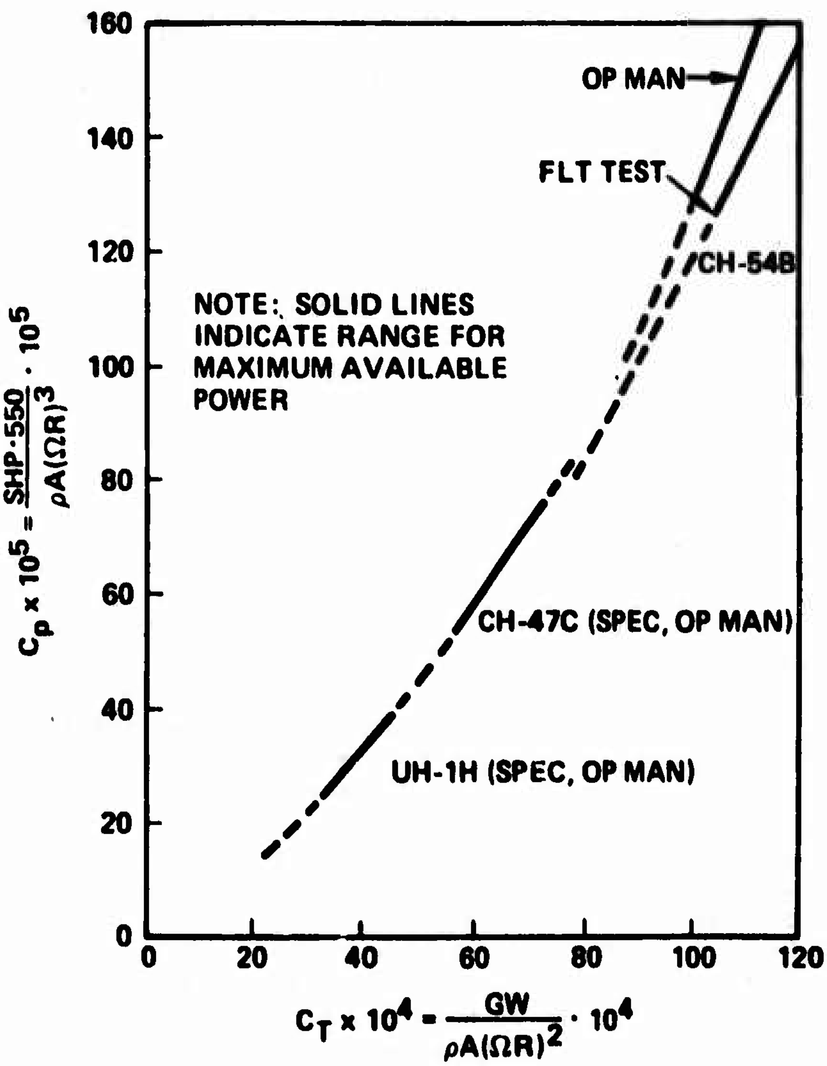

Figure A-2. Nondimensional Hovering OGE Performance Characteristics for UH-1H, CH-47C, and CH-54B.

where H_S = specific humidity (lb/lb)

H_R = relative humidity (percent/100)

T = temperature (°F)

A climate could be characterized as having a particular value of H. Then density would be calculated as

$$\rho = \rho_o \frac{P}{P_o} \frac{T_o}{T} \frac{1 + H_S}{1 + 1.608\, H_S} \tag{6}$$

where P and T are the measured values of pressure and temperature, and ρ_o, P_o, and T_o are sea level standard values.

Equation 5 is not a recommended approach; it is offered only to illustrate the method of correction. The point is that with a three-climate selection feature, errors due to humidity could be held to within about ±0.1 percent. A single point (fixed) correction would limit errors to about ±0.4 percent maximum.

It is concluded that a humidity measurement is not required for accurate lift performance indication.

5. SHAFT HORSEPOWER AND ROTOR SPEED INFLUENCE COEFFICIENTS

The development shown in Figure A-2 can be used to derive the influence factors for shaft horsepower and rotor speed errors. There is another development, however, that may allow better visualization of the error effects.

The rotor figure of merit, applicable to hovering OGE, is defined as

$$M = 0.707 \frac{C_T^{3/2}}{C_P} \tag{7}$$

The numerator is the (approximate) induced power developed by the rotor while the denominator is the total power required to hover (ideally, the induced power plus profile drag power). In the above definition, power is defined as that at the rotor; however, in actual helicopters, power is measured at the engine output so that the figure of merit applies to the whole helicopter instead of just the rotor. Since the nondimensional power coefficient (C_P, the denominator) includes additional power train losses, the FOM that is calculated will tend to be smaller and less variable with increasing levels of lift than the actual rotor FOM.

Substituting the expressions for the nondimensional coefficients and solving for gross weight, GW, the following is obtained:

$$GW = (550\ M\ SHP)^{2/3}\ (2A\rho)^{1/3} \tag{8}$$

Since M does not change much as power is varied, it is seen that the influence coefficient for density (ρ) is equal to approximately 1/3, the influence coefficient for shaft horsepower is equal to approximately 2/3, and the influence coefficient for rotor speed (Ω) is equal to about zero.

Numerical values for the shaft horsepower and rotor speed influence coefficients were calculated using the actual performance characteristics in Figure A-2. The results are as follows:

INFLUENCE COEFFICIENTS

Aircraft	$\frac{(\Delta GW/GW)}{(\Delta SHP/SHP)}$	$\frac{\Delta GW/GW}{\Delta\Omega/\Omega}$
UH-1H	0.72	-0.15
CH-47C	0.66	0.00
CH-54B (flt. test)	0.63	0.11
CH-54B (op. man.)	0.49	0.52

Considering only the rotor power, M would be expected to increase with increasing C_T; therefore, influence coefficients for shaft horsepower would be expected to be larger than 0.667, and those for rotor speed would be expected to be negative. It is seen that using engine output power introduces power train losses (including tail rotor power, as applicable) and results in somewhat unexpected values of influence coefficients.

The derivation of the influence coefficients using the figure of merit approximation appears in Figure A-1 in summary form. Typical influence coefficients for shaft horsepower, rotor speed, and density are also tabulated in the figure.

6. DIRECT AIR DENSITY MEASUREMENT DEVICES

6.1 TWO DEVELOPMENTAL NUCLEONIC METHODS

The Navy has sponsored development and evaluation of an absorption approach[25] to nucleonic direct air density measurement, while the Army has similarly sponsored a scatter approach.[7] (The AEC was also involved in

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[25] S. Kalatucka, Laboratory Evaluation of a Nuclear Air Density Gauge, Report No. NADC-AM-7132, Naval Air Development Center, Warminster, Pennsylvania, October 1971, AD 888342L.

earlier efforts.) Both techniques employ X-ray sources and scintillation detectors (with photomultiplier (PM) tubes). The basic difference is that in one case, the attenuation or absorption of radiation by an air sample forms the basis of the density measurement; while in the scatter approach, detection of the backscatter of radiation from the surrounding air is used to measure density.

The cited references are reports of the evaluations of experimental models for the two different approaches. Further literature on the development of the devices is included in the bibliography.

6.2 GENERAL COMMENTS

6.2.1 Choice of Detectors

The feasibility of using solid-state detectors should be determined. PM tubes are relatively fragile and large compared to solid-state detectors. Although solid-state detectors are not particularly well suited to detecting soft radiation (without cryogenic cooling), they may have adequate resolution (in energy) and adequate efficiency for this task.

6.2.2 Sampling Interval vs Radiation Intensity

Both approaches have rather long measurement times (i.e., compared to pressure and temperature measurements). In both cases, the measurement time is on the order of 5 to 10 sec, and longer sampling times (or stronger sources) appear necessary to reduce data scatter and achieve 1 percent accuracy. This is due to the statistical nature of nucleonic measurements: random error is a function of the number of events counted. Because of this inherent characteristic, errors (e.g, variation with temperature) were quoted in terms of the deviation of data points for a fixed condition from each other (i.e., for five or ten data points at fixed ambient conditions, the error was expressed in terms of the deviation of these points from their average in terms of σ or average deviation). This reflects short-term scatter rather than repeatability or absolute accuracy.

6.3 EVALUATION OF ABSORPTION TECHNIQUE

The absorption technique requires modification and further testing. An automatic self-test, data credibility (and exclusion) test, or a design modification is needed to eliminate the infrequent random gross errors in density measurement. Flight testing would be required to reveal any potential new problems. As recommended in Reference 25, calibration curve modifications and expansion of the sensor (Carbon 14) temperature operating range would be required. This approach has the advantage that the air sample being measured is enclosed and thereby not subject to errors due to the presence of external objects. However, if the air is pumped into the measurement cavity, the possibility of contamination (dirt, water, oil, etc.) exists.

6.4 EVALUATION OF SCATTER TECHNIQUE

The scatter approach was tested under simulated conditions and on a UH-1H. In the former test, a temperature input was utilized to enable estimating lift available. It was explained that with the addition of gross weight and fuel quantity, this system could be considered a candidate lift margin indicating system. The flight tests did not utilize these features. A serious deficiency of the flight test system as it was implemented was that the worst performance (5 to 6 percent error) was realized with the helicopter on the ground. The best performance ($\pm$0.4 percent) was at altitudes of 10,000 to 14,000 ft. Although the large errors on the ground were not fully explained, relocation of the sensor might solve the problem. The sensor was mounted inside the tail boom and "looked out" horizontally through the aircraft skin. It was stated that an obstruction-free diameter of 10 ft minimum was required. If the scatter angles involved were large (30°), the detector would see the ground which was about 5 ft below it. Relocation of the sensor (e.g., to point up rather than horizontally) should solve this problem. However, tail boom mounting would not be appropriate since the sensor would then see the engine exhaust. Further flight testing would be needed to resolve these questions, since inconsistency was noted in the small amount of test data taken.

The test device included an autocalibration feature that involved rotating the source electromechanically after every sample. This feature is required, it was stated, to lengthen the time between recalibrations (to 2 years for 0.5 percent accuracy). However, possible reliability problems associated with its mechanization were not addressed.

The basic source was Krypton-85 utilizing a uranium foil target and a copper container to create Bremsstrahlung X-rays. Autocalibration was incorporated because of the short half-life of Kr-85 (10.76 years). A secondary source, Cs 137, was used for automatic gain control of the PM tube in the detector. The 2-year calibration requirement arises because of the 30-year half-life of Cs 137.

Since the test data indicate a consistently positive error for altitudes below 8000 ft, the need for additional calibration (and data) is indicated. It is also noted that the error in the data taken during ascent of the helicopter ranged from about +1 to 0.05 percent until touchdown. A further testing requirement is indicated to determine the cause of this difference (e.g., perhaps a warmup problem as in the case of the absorption device).

APPENDIX B

MATHEMATICAL MODELS FOR GROUND EFFECT, WIND EFFECT, AND VERTICAL CLIMB CAPABILITY

1. INTRODUCTION

For several flight conditions the performance of a helicopter can be predicted by applying corrections to its basic hovering OGE performance characteristics. In particular, ground effect and relative wind effects can be represented in this fashion. Additionally, vertical climb capability can be calculated from the HOGE weight margin expressed as a percentage of vehicle gross weight. These representations are of interest in examining sensitivities and in mechanizing performance computations in an LPI system.

In the course of examining the influence of parameter variations on helicopter performance capabilities, models for ground effect, wind effect, and vertical climb capability were derived from applicable flight test data and performance charts. The derivations and forms of the models are described below.

2. GROUND EFFECT

Ground effect is significant within about one rotor diameter of the ground. Due to the effect, the amount of weight that can be supported at a given power level is significantly increased. Very near the ground, the increase in weight capability is about 20 percent.

Various mathematical models have been devised to represent and predict the performance influence of ground effect. In this study, an empirical model was derived that was found particularly useful. As shown in Table B-1, the model relates percentage increase in gross weight capability to an exponential function of the height of the helicopter's wheels or skids above the ground. Constants applicable to several types of helicopters are tabulated in Table B-1 and are based on fitting flight test results to the model equation. Curve fits for the UH-1, CH-47, and CH-54 are illustrated in Figure B-1.

Reference 20, containing selected flight test data applicable to ground effect for a wide assortment of helicopter models, was the source of data used for all the curve fits except the one for which "no solution" is listed in Table B-1 (CH-54B). Reference 24 was the source of data for the latter.

In the case of the above referenced CH-54B test data, the model did not offer a good representation. However, it appeared that no other model would fit that data. Because this is the only example of a bad fit, it suggests either a discrepancy in the data or an irregular case. The test data was for a CH-54B tested without the cargo/troop-carrier pod installed. For the other CH-54 data, it is not known whether the pod was installed or not. Absence of the pod may produce an unusual case. In any event, it is

TABLE B-1. GROUND EFFECT MODELS

Form No. 1: $(T_1-T_o)/T_o(\%) = K_1e^{-K_2\,Z/D}$

$(T_1-T_o)/T_o$ = percentage increase in lift capability at same power level due to ground effect

Z = wheel or skid height above ground plus distance from wheel or skid to center of rotor hub

D = rotor diameter (for tandem rotor helicopter, diameter of circle with area equal to the area projected by the rotors)

K_1,K_2 = constants

Form No. 2 (Derived from Form No. 1):

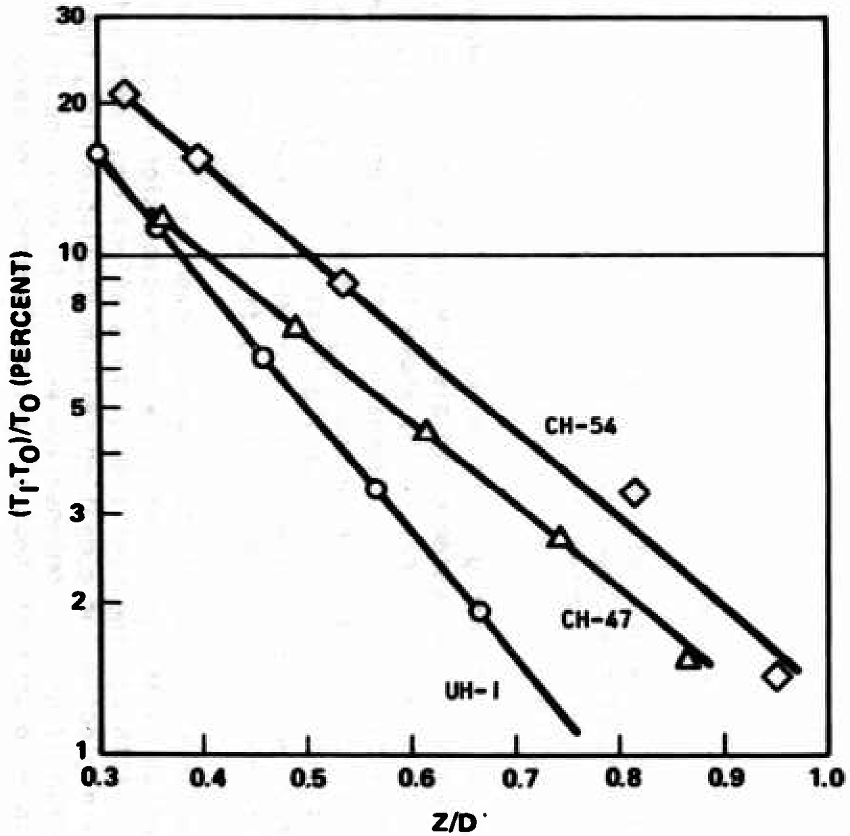
$(T_1-T_o)/T_o(\%) = \Delta_o e^{-K_2H/D}$

H = height of wheels or skids above ground

$\Delta_o = K_1e^{-K_2Z_o/D}$, Zo = Z for H = o.

NO.	AIRCRAFT	D Ft	Zo Ft	K1 Pct	K_2	Δo Pct	ERROR COEF
1	UH-1D	44	11.96	54.1	3.75	19.5	-0.085
2	UH-1D	48	11.96	89.6	5.80	21.1	-0.121
3	UH-1H	48	~12.0	103.0	5.70	24.8	-0.119
4	CH-54A	72	18.58	76.9	4.08	26.8	-0.057
5	CH-54B	72	18.7	No Solution			
6	CH-54B	72	18.7	53.6	4.08	18.6	-0.057
7	CH-47A	79.6*	18.6	54.9	3.83	22.4	-0.048
8	CH-47C	79.6*	18.95	49.0	3.93	19.2	-0.049

*Equivalent Diameter



SEMI-LOG GRAPH GIVES LINEAR FIT FOR

$Y = be^{mx}$

		ERROR COEF
UH-1H:	$24.8\,e^{-H/8.40}$	(2.95)
CH-47C:	$19.2\,e^{-H/20.2}$	(0.95)
CH-54B:	$18.6\,e^{-H/17.65}$	(1.05)

WHERE H = WHEEL OR SKID HEIGHT

Z = ROTOR HEIGHT ABOVE GROUND

D = ROTOR DIAMETER

NOTE: GROUND EFFECT NOT INDEPENDENT OF RELATIVE AIR VELOCITY

Figure B-1. Ground Effect Models (For Zero Wind Velocity).

likely that the pod exerts a measurable influence on the CH-54 HIGE characteristics. Further, it is possible that certain types of suspended loads (e.g., large surface areas) may measurably alter the hover performance characteristics of all the helicopters. No material has been found that addresses this question.

The constants applicable to the equations for the UH-1H, CH-47C, and CH-54B are tabulated in Table B-1. The increased thrust is equivalent to increased gross weight capability; that is,

$$\frac{\Delta GW}{GW} = \frac{T_I - T_0}{T_0}$$

The increased gross weight capability (in percent) due to ground effect for the three study aircraft are

$$\frac{\Delta GW}{W}_{g.e.} (\%) =$$

$$24.8\ e^{-.119\ H} \qquad \text{(UH-1H)}$$

$$19.2\ e^{-.0494\ H} \qquad \text{(CH-47C)}$$

$$18.6\ e^{-.0567\ H} \qquad \text{(CH-54B)}$$

The relationship for UTTAS would, of course, be similar to the above. In the case of the CH-54B, if the above relationship were to be employed by a lift performance indicator, two equations might be required: one applicable to the pod installed, the other for the pod not installed.

The above type of relationship could be employed in the lift performance system. For example, the system could display the weight capability of the vehicle for a nominal, typical height above the ground (e.g., 10 ft for the CH-47C and CH-54B). It is of interest to note the error in that predicted capability for a 1-ft error in the height assumption. At the nominal IGE takeoff heights, the error for the CH-47C and CH-54B is approximately the same, ±0.6 percent. The error for the UH-1H is much larger, ±2.3 percent, due to the lower height (2 ft vs 10 ft) and greater sensitivity.

The data shown in Figure B-1 is for maximum power conditions. There is a further variation of the ground effect with nondimensional power level (or gross weight). Although the magnitude of this effect can reach a few percent, it can be neglected for computing maximum available lift, but could require consideration if ground compensation were to be employed at lesser power levels, as in some sort of check of lift performance indicator accuracy.

The next section shows that wind velocity also influences the magnitude of ground effect.

3. WIND VELOCITY EFFECTS

Empirical equations for the wind effect on hover performance of the UH-1H, CH-47C, and CH-54B were derived through analysis of data extracted from performance charts and flight test data for those vehicles (from previously cited references).

The form of the equations is

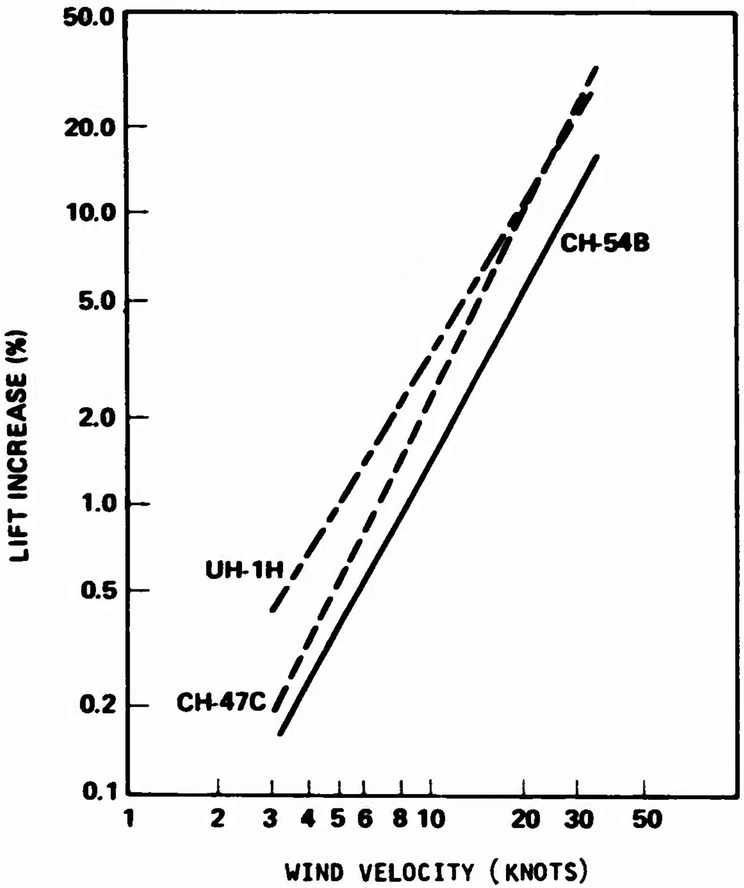
$$\frac{\Delta GW}{GW}_{wind} = A\ V_{wind}^{B}$$

where $\frac{\Delta GW}{GW}$ is the change in gross weight capability, V is wind velocity (TAS), and A and B are constants applicable to the particular aircraft. Figure B-2 illustrates the manual curve fits and resultant constants. For comparison, UH-1M flight test results are also plotted. The difference between the UH-1M and UH-1H data suggests the possibility that the UH-1H data may have been conservatively represented in the operator's manual. (The resolution of questions of this type can be postponed until actual design and implementation of an LPI system.)

As described in Reference 8, wind direction alters the wind velocity effect by a small but detectable amount. The effect can be visualized by considering a wind at 90 deg to the longitudinal axis of a UH-1; that is, a wind that produces a moment on the tail boom of the aircraft. This moment will either oppose or aid the moment produced by the tail rotor, and therefore, will result in the need to apply more or less power to the tail rotor.

The relationships illustrated in Figure B-2 are applicable to operation OGE. The situation is much more complex in proximity of the ground, because as wind velocity increases, the helicopter tends to move off the "bubble" with a consequent reduction in ground effect which tends to balance the effect of increased thrust due to the wind velocity increase.

The end product of this interaction is illustrated in Figure B-3. Maximum available engine power is fixed. Negative excess power in the figure means that the power available is insufficient to support the helicopter. As relative wind velocity becomes larger, ground effect diminishes. Note that there is a range of vehicle heights above ground where the combined effect of wind and ground proximity remains relatively constant up to a fairly substantial wind velocity. The relationship illustrated suggests that for the CH-47C the combined effect could remain relatively constant up to 20 or 25 knots.



LOG-LOG GRAPH GIVES LINEAR FIT FOR

$$Y = b\,X^{m}$$

UH-1H: $0.0647\ V^{1.70}$

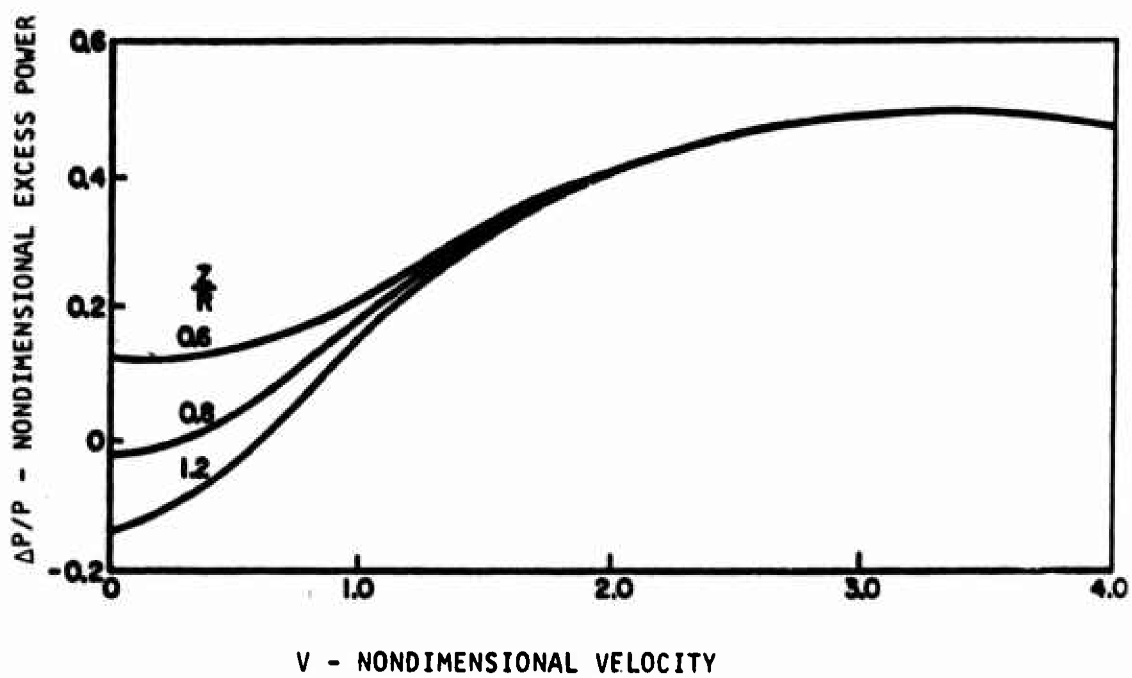
CH-47C: $0.0170\ V^{1.92}$

CH-54B: $0.0182\ V^{2.10}$

NOTES:

1. MEASURABLE VARIATION WITH DIRECTION
2. SHOULD BE CONSIDERED IN CONJUNCTION WITH GROUND EFFECT

Figure B-2. Increased Lift Due to Wind Velocity (Out of Ground Effect).



Z = Height of Rotor above Ground Plane

R = Rotor Radius

V = Relative Wind Velocity/Rotor induced flow velocity = v/υ

$\upsilon = \sqrt{W/2\rho A}$

For the CH-47C at sea level, 46,000 lbs, and on the ground:

Z/R = 0.48, υ = 26 knots

Figure B-3. Nondimensional Excess Power Curves of a Helicopter Operating In Ground Effect at Various Constant Altitudes versus Nondimensional Velocity (with Rotor Plane Linear Angle-of-attack Effects).

This characteristic could be of use to the LPI system for obtaining a rough check of the accuracy of the LPI indications. That is, it suggests that power margin should remain fairly constant with moderate wind velocity variation for IGE operation. Power margin is a recommended LPI display variable. Actual power margin is one of the few performance characteristics that is directly observable by the pilot. Comparing actual to predicted power margin can provide verification of correct LPI operation.

4. VERTICAL RATE OF CLIMB

Vertical ascent or descent (or a not-quite-stable hovering condition) alters the lift capability of the helicopter. This factor required consideration both from the point of view of lift performance display possibilities and in relation to any scheme involving the actual measurement of lift performance (for example, as might be required in the calibration of lift performance functions).

An equation for estimating the vertical rate of climb effect was derived from momentum relationships for the helicopter rotor. Using this equation, gross weight capability variations as a function of vertical rate of climb for the UH-1H, CH-47C, and CH-54B were calculated for mean maximum power states for the three aircraft. The results compared very favorably to performance data for the three aircraft contained in the respective operator's manuals.

Table B-2 shows the development of the relationship. Equation 1 is the rotor performance equation for hovering OGE; Equation 2 includes the effect of vertical velocity on the power required. In both Equations 1 and 2, C_{P0} is the profile drag power plus power train losses. The second term in Equation 1 is the induced power, and the second and third terms in Equation 2 are the induced power and climb power. Considering the case of operation at fixed power level, Equations 1 and 2 can be equated to calculate the change in power level. For this condition, it has been assumed that the sum of power train losses (including tail rotor power as applicable) and profile drag power remains relatively constant and can therefore be cancelled in the resulting equation. Carrying through the operations, an equation for the percent of value change in the thrust or weight coefficient with vertical velocity is obtained. This is equivalent to the change in gross weight capability.

Estimated and actual vertical rate of climb effects are plotted in Figure B-4. The estimated effects compare very favorably with the data extracted from the operator's manuals.

TABLE B-2

VERTICAL RATE OF CLIMB CAPABILITY
AS A FUNCTION OF WEIGHT MARGIN

Symbols:

GW = gross weight

C_W = weight coefficient = $GW/\rho A(\Omega R)^2$

C = power coefficient = $SHP*550/\rho A(\Omega R)^3$

H = hovering condition

v = vertical rate of climb condition

V = vertical rate of climb, ft/sec

B = tip loss factor

C_T = thrust coefficient = C_W for hover and vertical climb

Nondimensional power required for hover and vertical climb:

$$C_{P_H} = C_{P_o} + \frac{C_T^{3/2}}{\sqrt{2}\, H_B} \tag{1}$$

$$C_{P_v} = C_{P_o} + \frac{1}{2} C_{T_v} \left(\left(\frac{V_v}{\Omega R}\right)^2 + \frac{2C_{T_v}}{B^2}\right)^{1/2} + \frac{1}{2}\frac{V_v}{\Omega R} C_{T_v} \tag{2}$$

For constant maximum available power, $C_{P_H} = C_{P_V}$, and Equation 1 and 2 may be equated:

$$\frac{C_{W_H}^{3/2}}{\sqrt{2}\, B} = \frac{1}{2} C_{W_v} \left(\left(\frac{V_v}{\Omega R}\right)^2 + \frac{2C_{W_v}}{B^2}\right)^{1/2} + \frac{1}{2}\frac{V_v}{\Omega R} C_{W_v} \tag{3}$$

Since $\left(\frac{V_v}{\Omega R}\right)^2 \ll \frac{2C_{W_v}}{B^2}$, Equation 3 can be simplified to

$$C_{W_H}^{3/2} = C_{W_v}^{3/2} + \frac{B}{\sqrt{2}} \frac{V_v}{\Omega R} C_{W_v} \tag{4}$$

Considering small changes in V_v from hover (so that $V_V = V_V$), and noting that the left side of Equation 4 is constant, differentiation yields the following simplification:

$$\frac{\Delta C_W}{C_W} = -\frac{\sqrt{2}}{3} \frac{B}{C_W^{1/2}} \frac{V_v}{\Omega R} = -\frac{B}{3}\left(\frac{2\rho A}{GW}\right)^{1/2} V_v \tag{5}$$

Percent weight margin, $\Delta GW/GW$, equals the negative of the above.

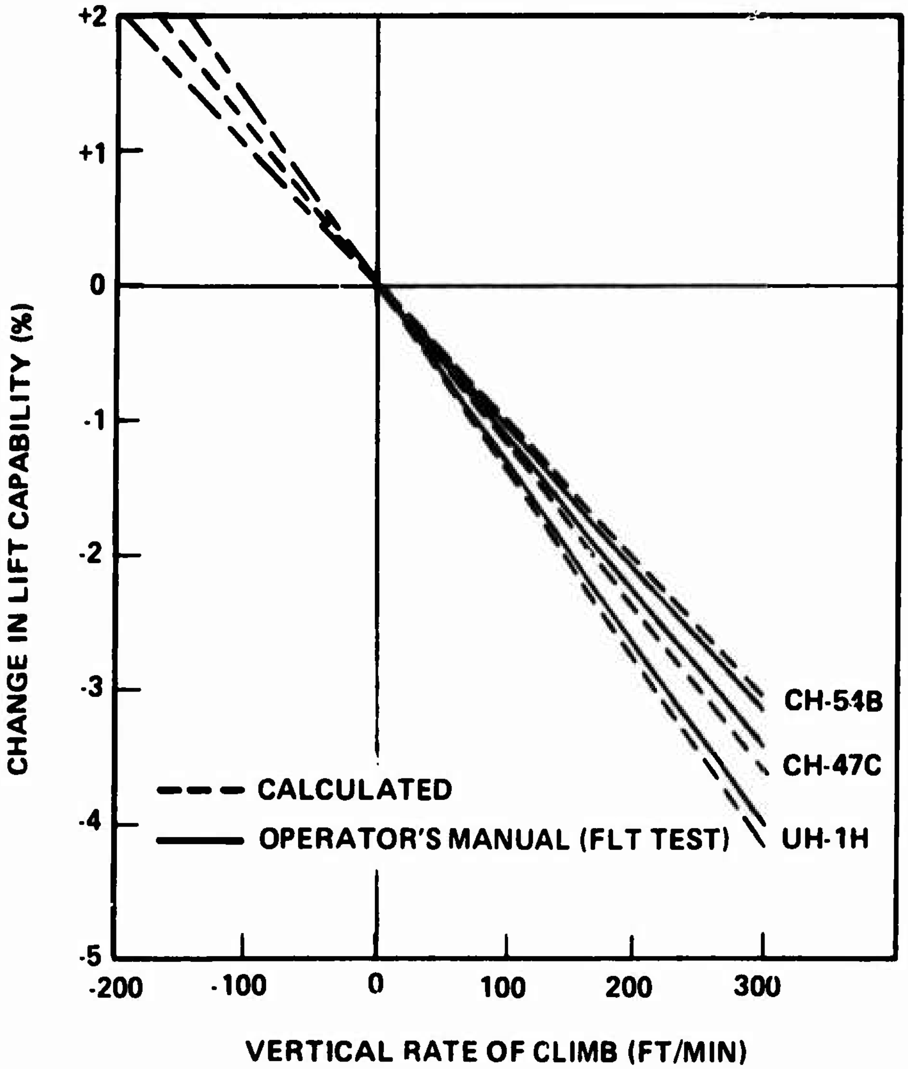


Figure B-4. Change In Lift Capability Due to Vertical Rate of Climb.

The figure illustrates that excess lift capability can be converted to vertical climb capability; or alternately, that vertical climb rates (positive) reduce the weight capability of the vehicle.

APPENDIX C

SLIDE-RULE-TYPE LIFT PERFORMANCE COMPUTERS

1. INTRODUCTION

No lift performance indicators or computers are found on any military helicopter. Among commercial helicopter manufacturers, Aerospatiale is unique in providing a manually operated collective pitch indicator-computer that provides a degree of lift performance information. Operation of that device is described here.

Additionally, another type of manually operated computer-indicator is described below. This simple lift performance computer-indicator was designed in this program for the UH-1H to illustrate some of the lift vs power relationships. Although simple, it is quite accurate and is a manual version of the helicopter lift margin system described in Reference 8. It is also functionally similar to a torquemeter computer designed by Floyd L. Dominick and CW3 John D. Thompson of the U.S. Army Aviation Engineering Flight Activity at Edwards Air Force Base, California.[26] The latter was a demonstration unit consisting of a torque pressure indicator surrounded by circular slide-rule-type scales for computing maximum power available and maximum gross weight capability. The latter scale is for a 2-ft hover and also provides a means for estimating gross weight at that condition.

2. MANUAL LIFT PERFORMANCE COMPUTER FOR THE UH-1H

Figure C-1 is a "working" model of lift performance computer for the UH-1H. This model is intended only to aid in demonstrating the sensitivities of certain lift vs power relations, applicable both to the UH-1H and to all of the helicopters considered in this study.

To learn how the computer in Figure C-1 operates, follow the instructions for the following sample problem:

Sample Problem

OAT = 25°C Pressure altitude = 5000 ft.

Solution from UH-1H operator's manual performance charts (see Figures C-2, C-3, and C-4).

Maximum available torque = 41 psi
Gross weight capability at HOGE = 7730
Gross weight capability at HIGE = 9150

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[26] Floyd L. Dominick and CW3 John D. Thompson, "Recommendation for Torquemeter Computers to be Installed on Turbine Powered Helicopters", U. S. Army Aviation Engineering Flight Activity, Edwards AFB, California, 7 June 1971.

1. For each of the 2 sliding scales, line up the outside air temperature (°C) (from the cockpit OAT gauge) with the pressure altitude (from altimeter set at 29.92 in. Hg.).
2. Pointer on left hand sliding scale points to maximum available engine torque (but not to exceed transmission limitation of 50 psi).
3. Lift capability at HOGE and HIGE (2 ft.) are lined up with torque capability.
4. Current weight for HOGE and HIGE (2 ft.) can be read opposite of torque indication for those conditions.

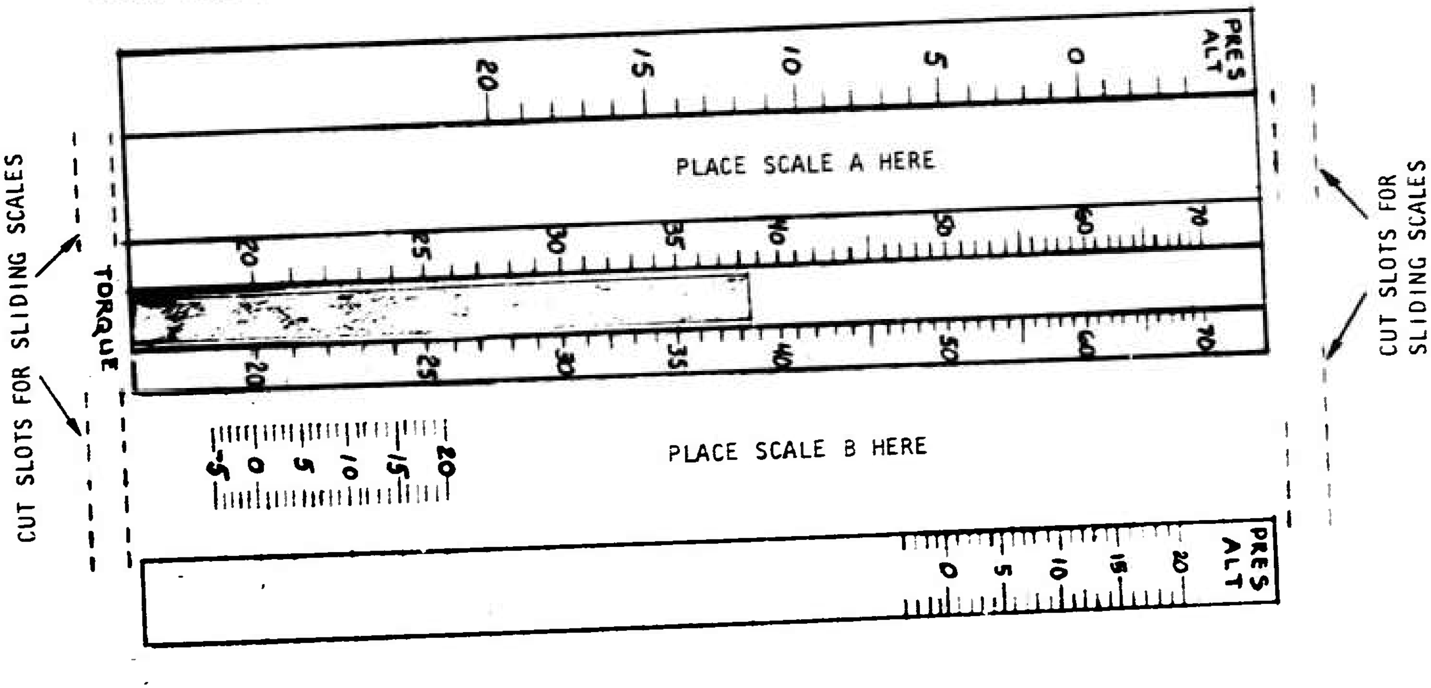
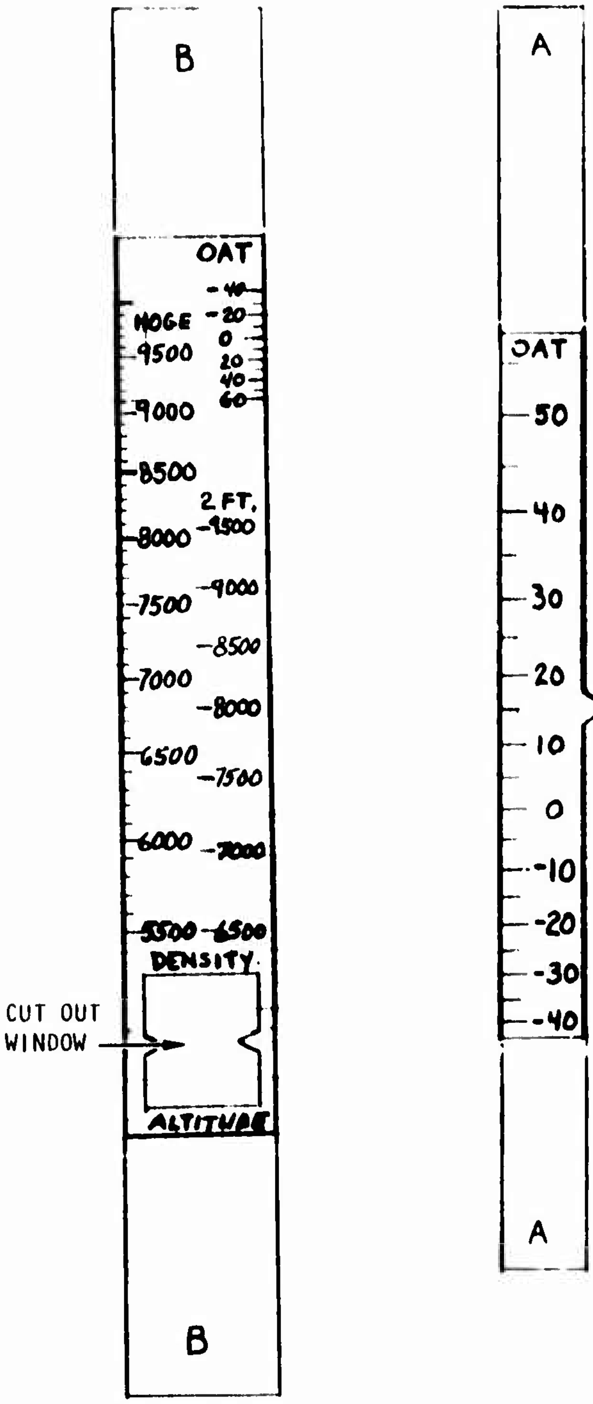


Figure C-1. Illustrative Manual Lift Performance Computer (Sheet 1 of 2).



(CUT OUT SLIDING SCALES FOR USE WITH FIXED
SCALES IN SHEET 1 OF FIGURE C-1.)

Figure C-1. Illustrative Manual Lift Performance Computer (Sheet 2 of 2).

Computer Solution

1. Line up left-hand OAT scale so that 35°C is opposite 5000-ft pressure altitude. The pointer on the temperature scale points to maximum available torque of about 41 psi.

2. Line up right-hand OAT scale so that 35°C is opposite 5000-ft pressure altitude. The weight capability scales are now opposite the corresponding power required. Thus, 41 psi (maximum available power) corresponds to a weight capability of about 7730 lb at HOGE or about 9150 lb HIGE (2 ft). Density altitude is also indicated at 8300 ft.

The computer in Figure C-1 provides the same information as in the operator's manual performance charts reproduced in Figures C-2, C-3, and C-4. Additionally, it provides density altitude and aerodynamic weighing capability.

The functions implemented by the computer are as follows.

Maximum Available Power

$$Q_{max} = (3.651 - 2.651\ )\delta$$

where Q = torque pressure

θ = (OAT °C + 273.2)/288.2

δ = ambient pressure/14.7

Gross Weight Capability

$$GW_{HOGE} = 574.13\ \ Q\ 0.72\ \ \sigma^{0.285}$$

$$GW_{HIGE} = 1.18\ \ GW_{HOGE} \qquad (2\ \text{ft})$$

σ = density ratio = δ/θ

Density Altitude (Density Ratio)

$\sigma = \delta/\theta$

The accuracies of the gross weight and power equations are commensurate with the scales used in Figure C-1. The equations for maximum power and gross weight were obtained by curve fitting.

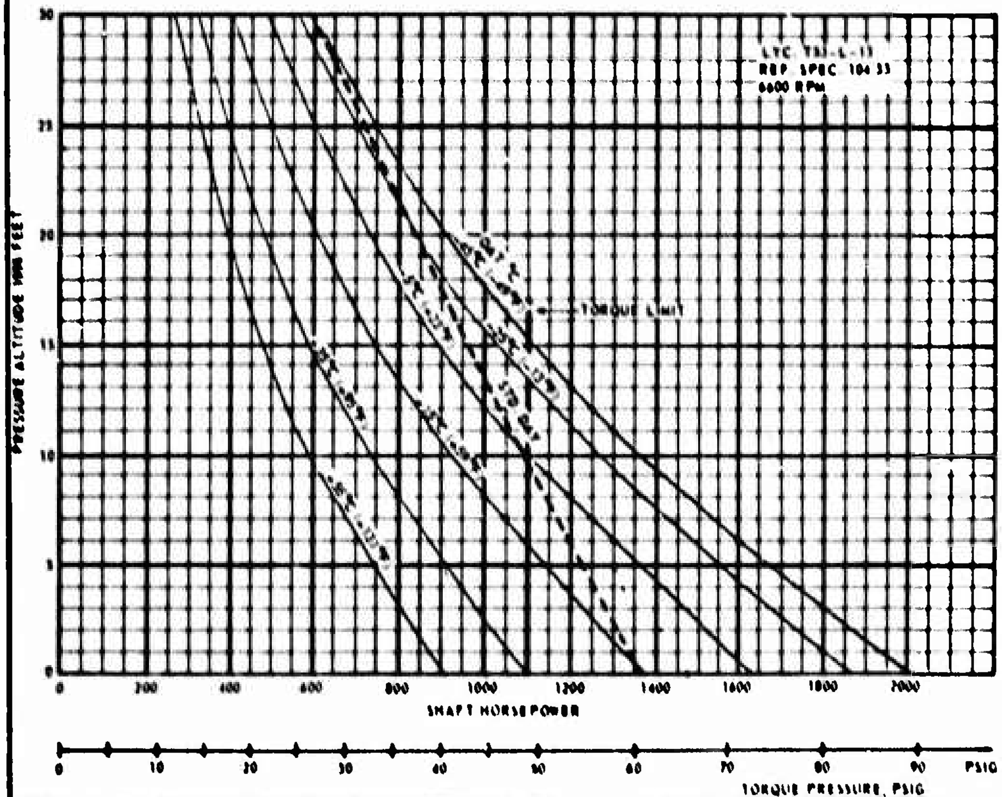
Imagining a vertical scale indicator representation of torque in Figure C-1, the approximately 38.5 psi torque indication is equivalent to 7400 lb at HOGE or about 8750 lb at HIGE (2 ft) at 35°C and pressure altitude of 5000 ft. Note the large displacement between the HOGE and HIGE 2-ft scales. This is also about the size of the difference between hovering OGE in zero knots

ENGINE OPERATING LIMITS

MILITARY POWER AVAILABLE
2°C INLET TEMPERATURE RISE

ARMY MODEL: UH-1H
DATA AS OF: AUGUST 1966
DATA BASIS: CALCULATED Lycoming Engine Spec No. 104.33 (Engine Installed)

ENGINES: Lycoming T53-L-13
ENGINE SPEED: 6600 RPM
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL.



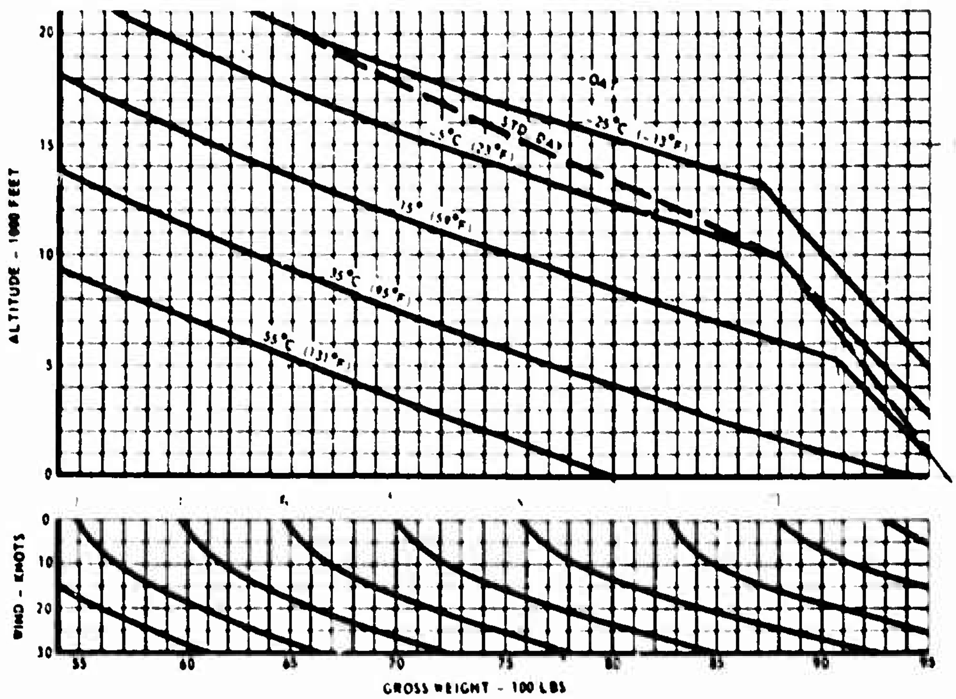
REMARKS:

Figure C-2. Engine Operating Limits--Military Power.

HOVERING

MAXIMUM GROSS WEIGHT FOR HOVERING OUT-OF-GROUND EFFECT WITH MILITARY POWER
2°C INLET TEMPERATURE RISE

Model(s): UH-1H	Engine(s): Lycoming T53-L-13
Data as of: NOVEMBER 1964	Engine RPM: 6600
DATA BASIS: ARMY FLIGHT TEST	Fuel Grade: JP-4
	Fuel Density: 6.5 LB/GAL.



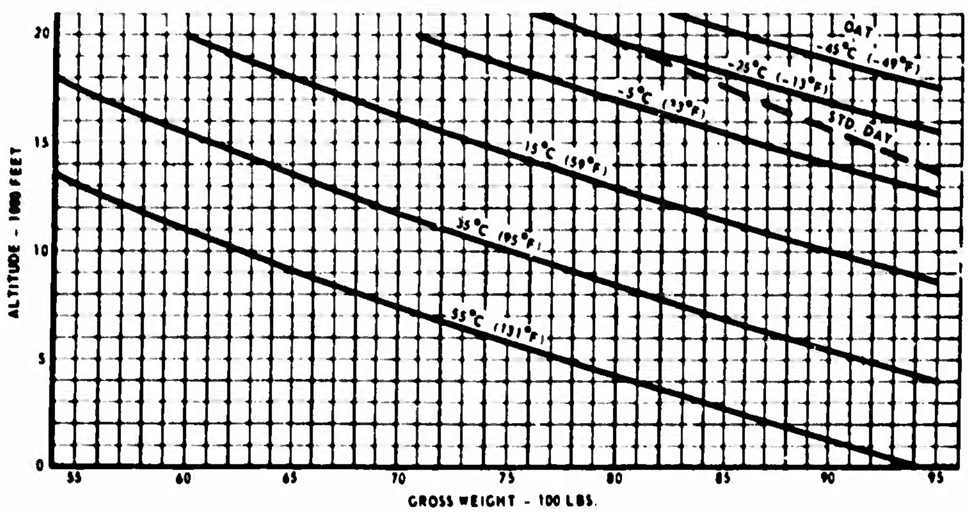
REMARKS:

Figure C-3. Hovering Out-of-Ground Effect Chart.

HOVERING

MAXIMUM GROSS WEIGHT FOR HOVERING IN GROUND EFFECT
WITH MILITARY POWER
2°C INLET TEMPERATURE RISE

Model(s):	UH-1H	Engine(s):	Lycoming T53-L-13
Data as of:	NOVEMBER 1964	Engine RPM:	6600
DATA BASIS:	ARMY FLIGHT TEST	Fuel Grade:	JP-4
		Fuel Density:	6.5 LB/GAL.



REMARKS:

Figure C-4. Hovering In-Ground-Effect Chart.

relative wind vs 30 knots. Visualizing this difference dramatically illustrates the problem of aerodynamic weighing. (Wind effects and ground effect combine at low level HIGE in such a way as to greatly reduce the effect of wind.)

Figure C-1 illustrates the effects of temperature and pressure variations. Comparing the two different pressure and temperature scales, it is seen that the weight capability vs power available relation is only slightly affected by temperature and pressure variations, while the maximum power available is a strong function of pressure and temperature.

The relative sizes of the scales in Figure C-1 are typical for all the helicopters surveyed, except for the temperature scale applicable to maximum power available (i.e., the left temperature scale). Of the engines surveyed, the T53 installed on the UH-1 had the most sensitive temperature scheduling: an approximate 1 percent change in power per degree Celsius vs about 0.5 percent per degree for the other engines. Thus for the other engines, the left temperature scale would be about half the size of that in Figure C-1.

The manual computer in Figure C-1 could also incorporate the two principal performance calibrations considered for the LPI system. The first calibration would account for differences between engines and their torque output. That is, the maximum torque at sea level standard conditions would vary among aircraft depending on topping tolerance and check accuracy and on the calibration of the torque indicating system. To account for this difference in the manual computer would require only displacing the left pressure altitude scale so that at sea level and 15°C, the pointer would indicate the correct torque pressure. (For example, for the UH-1H the within-tolerance indication for maximum torque at standard sea level conditions would lie between 57 and 65 psid.) The second calibration would account for lift vs power differences. The required adjustment would be to match the correct weight with a given torque for given pressure and temperature conditions by movement of the right pressure altitude scale.

The manual devices can be made to accurately represent the performance charts in the operator's manual. Compared to actual engine and aircraft performance, the largest source of error is likely to be in the computation of maximum available engine power. In the case of the UH-1H, for example, the performance chart representation of maximum power reflects an average or specification engine. Actual engine performance, even though trimmed to a within-tolerance value at the timing of a topping check, can depart significantly from assumed performance with variations in temperature from the value at which the engine was checked or trimmed.

3. <u>AEROSPATIALE COLLECTIVE PITCH COMPUTER-INDICATOR</u>

Aerospatiale Helicopter Corporation was contacted regarding lift performance and/or power available instrumentation offered as standard or optional on Aerospatiale helicopters. It was found that three types of instrumentation are used: (1) simple torquemeter, (2) collective pitch indicator-computer, and (3) torquemeter/thermal load indicator (TTLI). Usage of these displays on Aerospatiale helicopters is as follows:

SA-360 Dauphin: TTLI

SA-318 Alouette II, SA-316 Alouette III, and SA-315 Lama: Collective pitch indicator-computer

SA-330 Puma: Collective pitch indicator-computer and torquemeter

SA-341 Gazelle: Torquemeter

3.1 OPERATION OF COLLECTIVE PITCH INDICATOR-COMPUTER

Operation of the collective pitch indicator-computer is as follows. The indicator display of collective pitch is driven by the position of the collective lever. The computer operation is manual and consists of a circular slide-rule device mounted to the outside diameter of the display. The computer, together with the various scales and collective pitch display, allows the following information to be derived:

(a) Air density (density altitude)

(b) Maximum permissible available collective pitch

(c) Maximum HOGE weight capability (permissible)

(d) Current gross weight (estimated based on amount of collective used to hover)

The operation of the indicator-computer is illustrated in Figure C-5. Instructions for using the computer are repeated below from the SA-315B Lama helicopter flight manual.

1.0 Density altitude:

1.1 By means of the rotating circle, line up:

- The O.A.T. value (scale B) transferred from the O.A.T. indicator.

- The pressure altitude (scale A) transferred from the altimeter set at 1,013 mb.

1.2 Read the corresponding density altitude value on scale I opposite arrow C.

2.0 Maximum permissible collective pitch for hovering, whether in ground effect or out of ground effect.

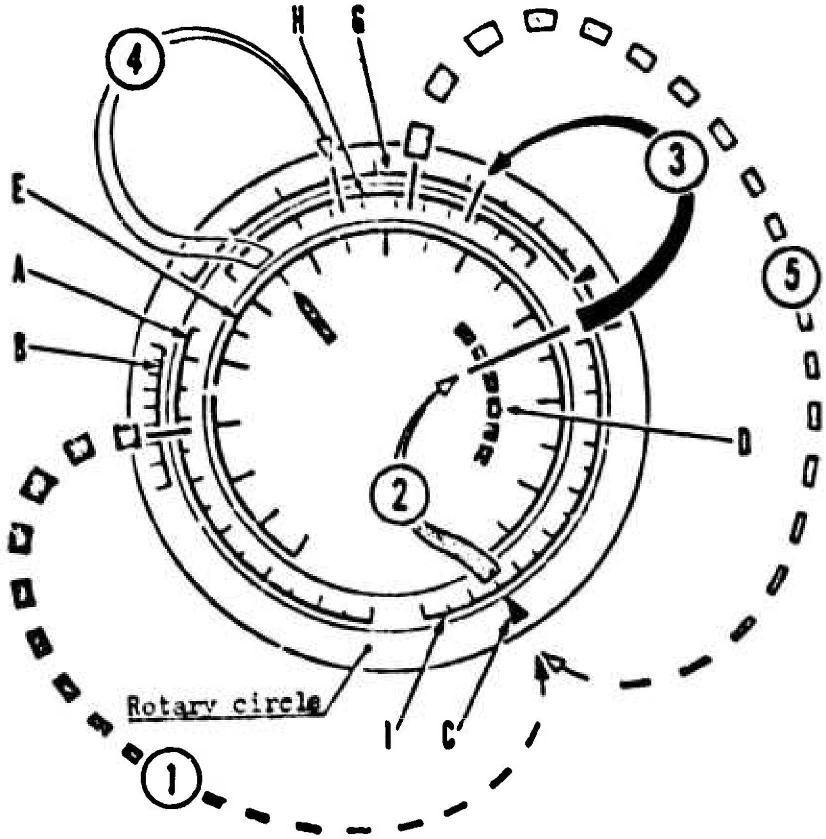


Figure C-5. Aerospatiale Collective Pitch Indicator-Computer.

2.1 Determine density altitude as specified in para. 1 above.

2.2 Convert the density altitude figure into km (or thousands of feet), transfer the result to scale D, then read opposite, on scale E, the maximum permissible collective pitch for this density altitude.

3.0 Maximum hovering takeoff weight out of ground effect:

3.1 Determine maximum permissible collective pitch as specified in para. 2 above.

3.2 Transfer the maximum permissible collective pitch figure to scale G and read opposite, on scale H, the approximate maximum permissible weight.

4.0 Current gross weight during hover in ground effect:

4.1 Carry out step (1).

4.2 Read on scale E the pitch angle indicated by the needle.

4.3 Transfer the above pitch angle to scale G and read on scale H the weight corresponding to this pitch angle.

It also appears possible to estimate the HOGE ceiling through the following iterative procedure (not specified in the flight manual instructions):

(1) Guess a value of the maximum density altitude and adjust the arrow C to that value

(2) Read the corresponding permissible collective pitch (scale E) and then find the corresponding permissible HOGE weight (scale H)

(3) If the permissible weight is greater than the estimated actual weight, guess a new higher density altitude; if lower, guess a new lower value

(4) Repeat until permissible weight matches estimated actual weight

3.2 OPERATION ON THE TORQUEMETER/THERMAL LOAD INDICATOR

This indicator is used on the SA-360 Dauphin. It is important to note that the engine used on this vehicle (as well as the engines used on the Alouette II and III, the Lama, and the Gazelle helicopters) is a single shaft engine as opposed to a free turbine engine. With conventional hydromechanical controls acting to provide constant rpm output, the pilot has considerably more opportunity to overstress the engine and transmission. In a free turbine engine as the pilot advances the collective, the gas generator section of the engine increases the energy output to the free turbine in order to maintain constant rpm. As this happens, the gas generator rpm increases. The extent of this increase is limited by the engine control. In a single spool engine, however, advancing the collective merely increases the energy input to the turbine--turbine inlet temperature increases while turbine rpm remains constant. With control maintained only on engine rpm, there is no built-in limitation and the engine is more susceptible to abuse.

The TTLI provides an integrated display of two variables that must be limited to avoid abuse of the engine and transmission. At low altitudes, the torque output of the engine must be limited to avoid damage to the transmission. At higher altitudes the torque output is reduced for a given power level and the engine temperature level must be limited. The limits on these two variables are fixed and can be expressed as percentages:

$$\text{Torque percentage} = \frac{\text{Actual torque}}{\text{Max torque}} \times 100$$

$$\text{Temperature percentage} = \frac{\text{Actual temp.}}{\text{Max temp.}} \times 100$$

The above percentages are calculated continuously and the larger of the two is displayed in percent on a needle-type indicator. The indicator also includes two lights that advise which variable is being displayed, and pushbuttons to allow overriding the display to show either of the two variables or fuel flow.

APPENDIX D

WEIGHT AND BALANCE EQUATIONS

1. STATIC CASE (ROTORS NOT TURNING)

Figure D-1 illustrates this case. The helicopter body axes are drawn with the origin at the point of contact of the front wheel with the ground plane. The X-axis parallels the aircraft water lines and the Y-axis parallels the aircraft station number planes. Other variables illustrated and to be developed below are as follows:

X_{CG}, Y_{CG}: Location of center of gravity (CG) with respect to helicopter body axes.

X^*_{CG}, Y^*_{CG}: Apparent or measured values of X_{CG} and Y_{CG} if no correction for helicopter inclination from ground plane ("pitch" angle) is introduced.

α: Inclination of X-axis (and ground plane) with respect to gravimetric horizontal plane.

W: Gross weight of helicopter.

W^*: $F_F + F_R$ = apparent gross weight of helicopter = W (cos α) ($W^* = W$ if $\alpha = 0$).

F_F, F_R: Forces normal to the ground plane acting on the front and rear wheel pairs (forces in the plane of the ground plane can be ignored in this treatment).

X_{FR}: Distance between the front and rear wheels.

The apparent longitudinal location of the c.g. is

$$X^*_{CG} = \frac{F_R\ X_{FR}}{W^*} = X_{CG} + Y_{CG} \tan \alpha$$

The true values of weight and c.g. are therefore

$$W = W^*/\cos\alpha$$

$$X_{CG} = X^*_{CG} - Y_{CG} \tan\alpha$$

The above assumes that the helicopter water lines and the ground plane are always parallel. If that is not the case (as for the CH-47), then another correction must be introduced. Figure D-2 illustrates this situation for the CH-47. Here, where the ground plane is level, the helicopter's water lines are pitched upward by 2 degrees. The previous equations for c.g. are

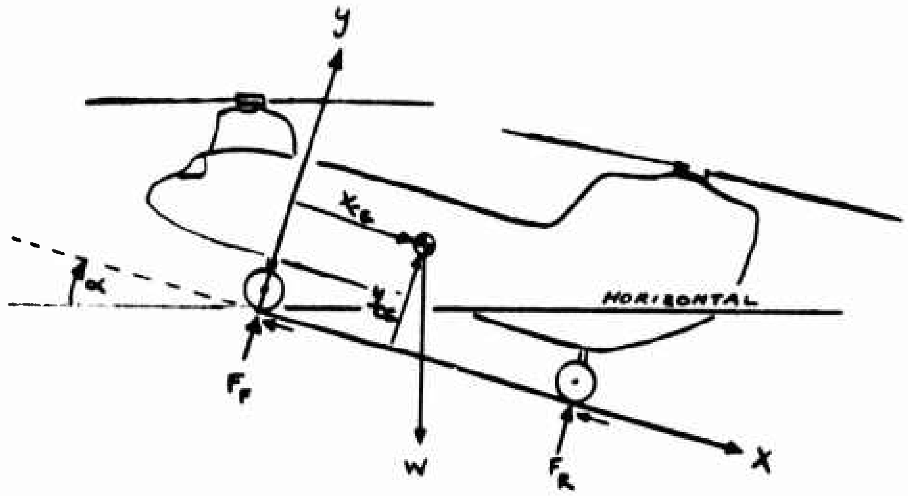


Figure D-1. Forces Acting in Static Case.

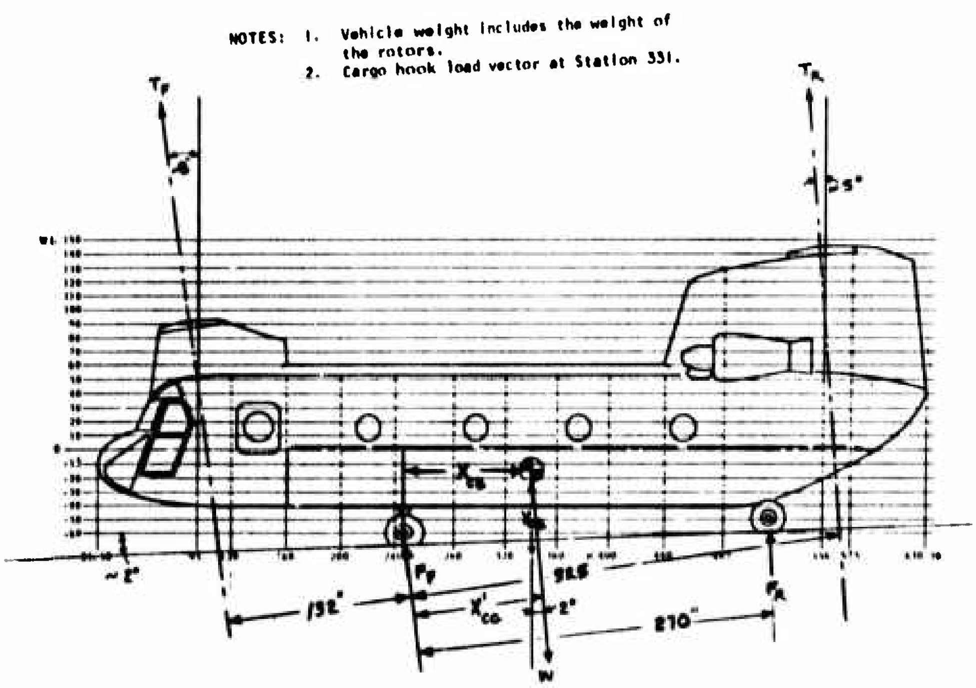


Figure D-2. Forces Acting on CH-47C in Dynamic Case (Rotors Turning at 100 Percent and Minimum Collective).

applicable to an X-axis lying in the ground plane. Denoting that as the X' axis, the transformation to the X-axis is

$$X_{CG} = X'_{CG} \cos \theta - Y'_{CG} \text{ sine } \theta$$

where equals the angular offset of the X'-axis with respect to the X-axis (2 for the CH-47).

For small angles, cos θ = 1, sine θ = θ, and tan $\alpha = \alpha$. Therefore, the equation for c.g. modified for the body axis offset is

$$X_{CG} = X^*_{CG} - Y'_{CG} (\alpha + \theta)$$

If the inclination of the helicopter's longitudinal axis is measured with respect to horizontal, and denoted by ϕ, then $\phi = \alpha + \theta$ and

$$X_{CG} = X^*_{CG} - Y'_{CG}\phi$$

Y'_{CG} is the vertical location of the c.g. If this is unmeasured and can vary by ± 5 inches (internal loads only), then the error introduced by using a nominal value of Y'_{CG} is (for $\phi = \pm 10°$) ± 0.9 in.

Only the longitudinal inclination (or pitch) of the helicopter has been considered. The lateral inclination (or roll) of the vehicle has no theoretical effect on longitudinal c.g. If roll is limited to ± 5 degrees (considered as worst case in previous studies), then the maximum effect on the gross weight reacted along a line perpendicular to the ground plane is -0.4 percent. This effect could therefore be adjusted to remain within ± 0.2 percent of the true gross weight.

2. DYNAMIC CASE (ROTORS TURNING)

In an operational situation, it is possible that unloading and loading the helicopter may take place with the rotor(s) turning at 100 percent rpm and minimum collective. Under this "residual thrust" condition, the rotor(s) may develop 10 to 15 percent of maximum thrust, or about 30 percent of the empty weight of the vehicle (numbers applicable to the CH-47C).

Given the measurement or approximation of the residual thrust, the measured gross weight and c.g. can be corrected. Tests of developmental weight and balance systems for the CH-47C (References 2 and 4) indicate that direct measurement of residual thrust is very difficult. Figure D-2 illustrates the relation of residual thrust forces to vehicle weight loads and measurements.

Based on analysis and test results reported in Reference 2, it is concluded that the residual thrust can be adequately approximated provided three conditions are met: (1) collective is set at minimum, (2) cyclic is set to neutral, and (3) rotor rpm = 100 percent. For a given collective angle, the magnitude of the residual thrust vector is a function only of density altitude (ignoring wind) and is not affected by the position of the cyclic. However, the angular position of the residual thrust vector is affected by the cyclic.

Consequently, the apparent location of the longitudinal c.g. is affected by cyclic position. To avoid an additional measurement (needed only in regard to c.g. measurement), it is required that the cyclic be located at its neutral position during system operation.

Equations for the CH-47C and CH-54B follow. For the CH-47C, it is assumed that the thrusts of the two rotors are equal, with the quantity T representing the total residual thrust. The nominal value of T at sea level standard conditions is 6000 lb; it varies directly as the ratio of ambient air density to sea level standard density.

CH-47C Weight and Balance Equations

Residual Thrust (T):

$T = T_o\sigma \quad \sigma = \frac{\rho}{\rho_o} \quad \rho$ = air density

ρ_o = sea level standard value of air density

Gross Weight (W):

$W = W^* (1 + 0.5\,\alpha)$ $\quad \alpha$ = ground slope = aircraft pitch -2°
(α in radians)

$W^* = F_F + F_R$

$F_F = F_F^* + 0.641\ T$ $\quad F_F^*$ = measured vertical force on front landing gear

$F_R = F_R^* + 0.357\ T$ $\quad F_R^*$ = measured vertical force on rear landing gear

C.G. Location (X_{CG}):

$X_{CG} = 245 + X_{CG}^* - 70\,\phi$ $\quad$ (ϕ = helicopter "pitch" in radians)

$X_{CG}^* = \frac{270\ F_R}{W^*}$ $\quad$ (nominal coefficient for ϕ)

Sensitivity Coefficients

For W = 36,000 lb, X_{CG} = 336.1, T = 6000

$\Delta X_{CG} = -2.46\ \Delta F_F^*$ (per 1000 lb) = 1.22 $\Delta\phi$ (per degree)

(in.) $+4.84\ \Delta F_R^*$ (per 1000 lb)

$+0.15\ \Delta T$ (per 1000 lb)

$$\text{Approximation: } \Delta X_{CG} = \frac{(515-X_{CG})(X_{CG}-245)}{(270)2} \frac{\Delta F}{F} \approx 30 \frac{\Delta F}{F}$$

where F is the force on an individual strut (multiply by -1 for front struts).

Errors have a one-to-one effect on gross weight.

CH-54B Weight and Balance Equations

Residual Thrust (T):

$$T = T_o\sigma \quad \sigma = \rho/\rho_o$$

Gross Weight (W):

$$W = W^* (1 + 0.5\,\alpha^2) \qquad \alpha = \text{ground slope} = \text{pitch } (\phi)$$

$$W^* = F_F + F_R$$

$$F_F = F_F^* + 0.157\ T \qquad F_F^* = F_R^* \text{ are measured values}$$

$$F_R = F_R^* + 0.842\ T$$

C.G. Location (X_{CG}):

$$X_{CG} = 100 + X_{CG}^* - 100\,\phi \qquad \phi = \text{pitch (nominal coefficient for } \phi)$$

$$X_{CG}^* = \frac{293\ F_R}{W^*}$$

Sensitivity Coefficients

For W = 36,000 lb, X_{CG} = 335, T = 6000

$\Delta X_{CG} = -6.35\ \Delta F_F^*$ (per 1000 lb) = 1.75 $\Delta\phi$ (per degree)

(in.) $+1.57\ \Delta F_R^*$ (per 1000 lb)

$+0.32\ \Delta T$ (per 1000 lb)

$$\text{Approximation: } \Delta X_{CG} = \frac{(393-X_{CG})(X_{CG}-100)}{(293)2} \frac{\Delta F}{F} \approx 45 \frac{\Delta F}{F}$$

where F is the force on an individual strut (multiply by -2 for the front strut).

Errors have a one-to-one effect on gross weight.

3. CH-47C AFT LANDING GEAR

For the helicopters considered in this study, the typical landing gear strut is a vertically oriented oleo strut assembly consisting of an air-oil shock strut mounted in a telescopic cylinder. For this configuration, the oleo strut pressure is linearly related to the load supported by the strut (ignoring friction).

The CH-47C aft landing gear departs from the usual design. Figure D-5 illustrates the configuration of the aft gear and summarizes the relation between the force, F, applied by the shock strut to balance weight, W. The ratio, F/W, is larger for smaller weights. Over the range of possible gross weights, the ratio varies by about 11 percent. Considering identical errors for both aft gear, this could cause about 3.5 percent error in gross weight and about 5.5 in. error in c.g. location. That is, these errors would occur if the value of the F/W ratio for maximum weight were used at minimum weight, and vice versa.

Fortunately, the deflection of the aft gear is a function of applied weight, so the F/W ratio variation can be compensated. The nominal weight on gear function for the CH-47C is as follows:

$$W = 4.94P \qquad P \leq 665 \text{ psig}$$

$$W = (4.94 + (P-665)/1000)\,P \qquad P > 665 \text{ psig}$$

W = weight on one aft gear

P = aft gear oleo pressure

The tolerance on strut deflection is determined by the check performed on strut air pressure by organizational maintenance. This tolerance averages about +0.25 in. yielding an error of about +0.5 percent.

Laboratory tests of a CH-47B aft landing gear as reported in Reference 4 may fail to confirm the above oleo pressure vs applied weight behavior. (Interpretation of data in reference is subject to error due to required assumptions.)

$$\theta = \cos^{-1}\frac{X^2 + 33.3^2 - 19.7^2}{66.6\,X} \qquad \alpha = \sin^{-1}\frac{33.3 - X\cos(\theta)}{19.7} - 8.76^{\circ}$$

$$\frac{F}{W} = \frac{23.5\cos(\alpha)}{33.3\sin(\theta)} \qquad X_{\theta} = 30.8 = 38.53$$

Strut Compression (in.)	(deg)	(deg)	F/W	Nominal Gross Wt (1000 lb)	Weight/Press. Coefficiency
0	30.8	- 8.2	1.37	21.0	4.94
1	31.6	- 4.9	1.34	24.3	5.03
2	32.4	- 1.6	1.32	27.2	5.12
3	33.1	1.5	1.29	30.8	5.22
4	33.7	4.7	1.265	35.9	5.34
5	34.3	7.7	1.24	42.5	5.46
6	34.8	10.8	1.215	51.7	5.58
7	35.3	13.8	1.19	54.9	5.71

Variation in F/W ratio determines error due to using fixed sensitivity.

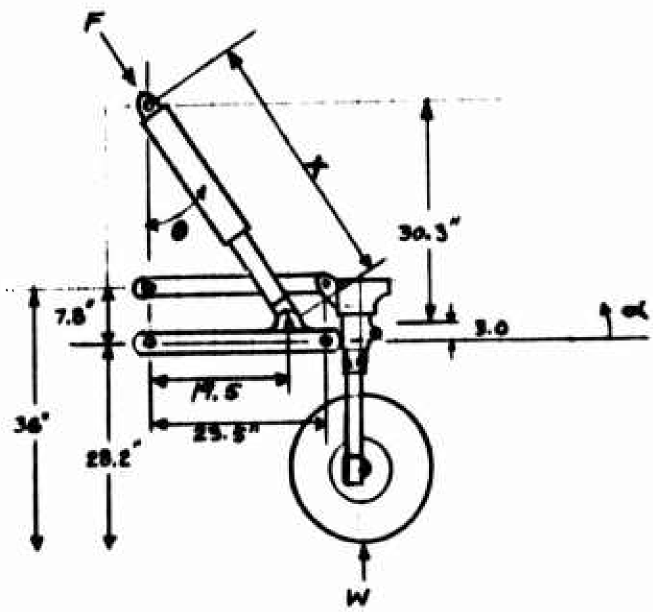


Figure D-3. CH-47C Aft Landing Gear Strut Pressure vs Loading.

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