# Technological Contributions of the CH-53A Transport Helicopter Development Program

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**The CH-53A helicopter is approximately the same size but has four times the productivity of its predecessor, the H-37. It has an operational speed with payload on a par with the existing world's records set by the MIL Mi-6, an aircraft twice the size. As such, it has demanded the application of many new design and analytical techniques. This paper outlines the general approach to the optimization and selection of the fundamental design parameters and discusses areas of particular interest, such as rotor system, airframe dynamic design and analysis, dynamic aspects of the airframe/powerplant marriage, and the need for inherent stability in high-speed helicopters. Looking ahead, the CH-53A program should advance the state of the art in such areas as: 1) the large scale use of titanium in dynamic components designed for fatigue considerations; 2) the efficient design of large asymmetrical airframes, with extensive cutouts, subjected to relatively low-frequency vibration excitation; 3) the operation of high solidity rotors at high disk loadings, advance ratios, and tip Mach numbers; and 4) stability, control, and failure reaction requirements of high-speed rotary wing aircraft.**

## Introduction

THE CH-53A helicopter program marks a significant point in the evolution of rotary wing design With the advancing maturity of the helicopter industry, the designer has been forced to dig deeper into the many diverse technologies that make up the rotary wing business to come up with significant improvements. In the conventional helicopter, we are no longer working with major breakthroughs of a fundamental nature, but rather with improvements in many areas that still offer, in their accumulated impact, significant over-all mission benefits. As the newest stepping stone on this path, the CH-53A program illustrates the improvements still possible and will provide the basis for further extension of design efficiency in many areas of rotary wing technology. It is the purpose of this paper to survey some of the most significant of these areas.

The U. S. Marine Corps requirement for a large high-speed assault helicopter reduces to the specification shown in Table 1.

At the time of release of these requirements, the Sikorsky S-64 Crane helicopter was approaching first flight. The S-64, with its JFTD-12 engines, had been designed primarily for low speed and heavy lift over relatively short ranges. However, it was apparent that the rotor system of the S-64, combined with a clean fuselage and the low specific fuel consumption of the General Electric T-64-6 turbine engines, would be a close match to the Marine mission. Figure 1 shows the CH-53A, which resulted from this marriage.

To meet Marine requirements, the performance improvements that the CH-53A had to achieve over earlier aircraft were the conventional ones: more payload and higher cruise speeds for less empty weight. Of equal importance was the necessity for greater mission reliability and lower maintenance burden (in addition to better vibration noise and handling quality environment for personnel).

To illustrate the degree of improvement required, Fig. 2 compares the design efficiency of the CH-53A with its cargo helicopter predecessors in terms of useful to gross weight ratio based on sea level out-of-ground-effect hover ability. It will be noted that the CH-53A is shooting for almost 50% useful load capability; this would be creditable in any aircraft system, but is particularly so in a rotary wing aircraft with amphibious capability, drive-in cargo loading, automatic blade folding, and all-weather instrumentation. For instance, an S-61A without all of these provisions has a useful to gross ratio of about 50% compared to 40% for the S-61R with these provisions.

The productivity comparison, based on cruise speed times payload capability (out-of-ground-effect hover at takeoff) divided by weight empty for a 100-naut-mile-radius mission, makes an even more striking comparison. Figure 3 compares this productivity for the same aircraft as a function of takeoff altitude. It will be noted that the CH-53A produces at sea level about four times the productivity of the CH-37, which it replaces, and, at the 6000-ft design point, the improvement factor is greater than five.

The most interesting way to look at the CH-53A's speed capability is to compare its normal mission requirements with the existing world records for speed with payload. Rotor limitations being what they are, we cannot dissociate gross weight, therefore payload, from speed capability. For example, on a 1000-km course, the larger (approximately twice the weight empty) MIL Mi-6, undoubtedly operating in an underloaded condition, achieved the record of 162 knots with a 1000-kg payload. The normal mission payload/speed goals of the CH-53A (one point, for example, 166 knots with a 1000-kg payload) equal or exceed all four of the MIL Mi-6 1000-km records. The 1963 absolute speed record (no payload or fuel for range) set by the Super Frelon with its Sikorsky rotor system is only 19 knots above the operational top speed of the CH-53A.

## Design Parameter Selection

The speed requirement was the major factor in establishing the parametric configuration. In increasing the speed of rotary wing aircraft, the fundamental problem is the compromise required between retreating blade stall and reverse flow (conventionally defined by advance ratio $\mu$, equal to forward speed divided by hovering tip speed) and advancing blade tip Mach number. Higher tip speeds reduce the reverse flow region and help postpone blade stall, but force higher Mach number operation on the advancing blades. The design point selected for the CH-53A, relative to earlier aircraft, is shown in Fig. 4. Also shown are loci of advancing blade Mach number and rotor advance ratio. The Super Frelon's speed record point is also shown, because this repre-

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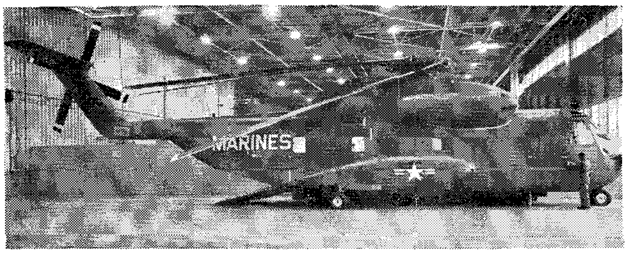


**Fig. 1 Sikorsky CH-53A assault transport.**

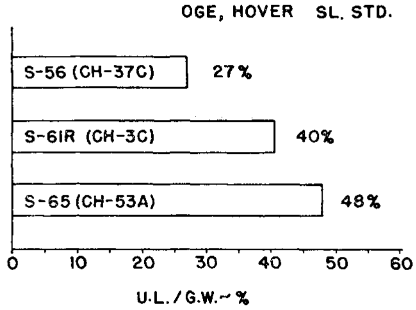


**Fig. 2 Useful load/maximum gross weight.**

sents the $\mu$-Mach frontier on the Sikorsky-type rotor system for which instrumented flight test experience exists. A relatively high tip speed was selected for the CH-53A; blade Mach number as well as advance ratio will exceed previous designs. However, the design point is well within the rotor parametric area investigated by the Super Frelon Record Program. Reference 1 summarizes the test data obtained in this program.

Disk loading selection depends primarily on hovering efficiency, power available, and downwash considerations. Experience with the S-64 Flying Crane had confirmed the acceptability of all types of hovering operations, including external load pickup at gross weights as high as 40,000 lb on a 72-ft rotor, implying a disk loading of 9.83 lb/ft$^2$. The Marine hover requirement can be met with some margin by a rotor of the S-64 size (72-ft diam) and the T-64 engines; a 72-ft rotor aircraft could be designed to fold up within carrier space limitations. Also, the S-64 had provided confirmation of the very desirable vibration excitation reductions achieved with a 6-bladed rotor system. Therefore, the CH-53A was committed from the beginning to a 6-bladed 72-ft rotor to take maximum advantage of the S-64 background.

Blade chord was re-evaluated in the light of the higher speed requirement of the CH-53A. Once the rotor diameter, tip speed, and number of blades have been selected, the blade chord becomes a tradeoff of area required to prevent retreating blade stall, the weight penalty for wider blades, and the profile drag of the extra area. Figure 5 illustrates this tradeoff situation. The Marine mission payload, shown as a function of chord is limited by the gross weight at which each of the following performance requirements can be met: 1) cruise (150 knots without blade stall), 2) hover (6000 ft, out-of-ground-effect), and 3) climb (100 fpm, sea level, 89.6°F). For any chord, the lowest of the payloads associated with these three requirements determines the payload margin this chord will provide.

From Fig. 5 it can be seen that the S-64 chord of 23.65 in. is nearly optimum for meeting the Marine mission with the existing T-64, even with the 150-knot cruise requirement ratings. Originally it was planned to use these blades unchanged. However, at an early stage in the CH-53A detail design, these tradeoffs were re-examined with consideration for T-64 engine growth capabilities. With the more powerful engine, the tradeoffs are revised as shown by the growth engine curves in Fig. 5. For an optimum match with a growth T-64, a 27-in. chord is indicated. With this much chord, however, there is little margin left on the hover ceiling requirement with existing engines. Therefore, the decision was made to go to a 26-in. chord. This would provide most of the benefits of the engine growth while maintaining performance above the requirements with the original engine.

**Table 1 U. S. Marine Corps requirement for a large high-speed assault helicopter**

| | |
|---|---|
| Mission radius | 100 naut miles |
| Payload | 4 tons outbound<br>2 tons inbound |
| Hover ceiling | 6000 ft (out-of-ground-effect, standard day) |
| Single engine rate of climb | 100 fpm (sea level, 89.6°F) |
| Cruise speed | 150 knots |
| Maximum speed | 160 knots |

## Rotor Blade Airfoil Selection

Since, in large high-aspect-ratio rotor blades, centrifugal stiffening primarily determines elastic properties and the deflections under vibratory loads, stresses tend to be proportional to spar thickness. A thinner spar has the advantage with respect to blade stress. For this reason, a great deal of study went into the potential of thinner airfoil sections that would not deteriorate the blade stall characteristics. As a result of this study, a symmetrical 11% section was selected with special leading-edge and thickness distribution. Figure 6 indicates the relationship between the coefficient of lift and Mach number on the CH-53A at 150 knots.

## Stability and Control

Stability and control is another area in which higher cruise speed requirements play a particularly important role. As speed and rotor solidity increase, the load factor dependence on flight path perturbation becomes more critical. At the lower speeds and rotor solidities at which the helicopters of the past have operated, stability and control seldom had serious structural consequences, at least in the case of the single lifting rotor. The reason was simply that very large flight path disturbances were required to produce significant change in steady loads. The CH-37 cruising at 105 knots was exposed to only 0.07-$g$ force per degree. With the CH-53A's 150–170-knot speed range and its $11\frac{1}{2}\%$ solidity, however, one degree will produce more than twice this normal load factor. Fortunately, the increased stabilizing effectiveness of horizontal surfaces aft of a principal lifting

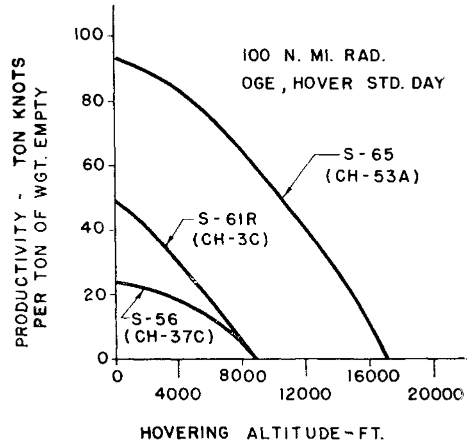


**Fig. 3 Productivity vs altitude.**

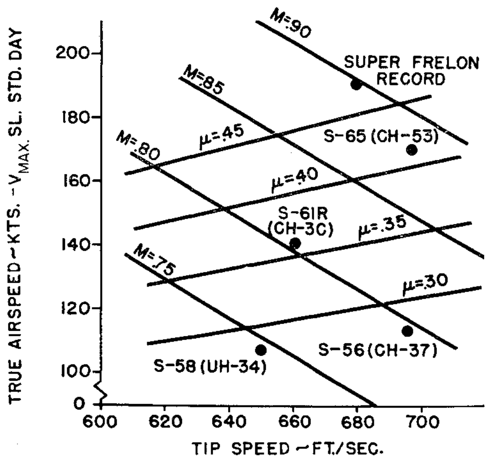


**Fig. 4 Advance ratio/Mach number design gross weight.**

rotor offsets this trend and can easily yield the more stable characteristics necessary to avoid inadvertent overloads. The faster a rotary wing aircraft flies, the more closely its stability problems and their solution resemble those of fixed wing aircraft. Airframe stability and fixed surfaces now have more effect in overcoming the divergent rotor tendencies.

Sikorsky Aircraft has devoted considerable effort toward achievement of the currently accepted utilization of limited authority automatic stabilization equipment for instrument flight rules (IFR) helicopter operation. Our use of artificial stabilization has always been predicated on combining artificial stabilization with inherent characteristics, which permit IFR flight at cruise speed without stabilization. We use stabilization as a very desirable assist in forward flight, but depend on it only for IFR flight at very slow speeds, or in a hover. Although the inherent stability requirement was originally conceived as a get-home-safe provision for helicopters with nonredundant stability augmentation, we find inherent stability has new significance as cruise speeds increase, because of the structural implications of flight path stability.

For this reason, the CH-53A has been designed with a very large horizontal tail even though redundant stabilization is provided. It will have considerably better inherent stability than the SH-3A, which has successfully operated as a fleet IFR helicopter with only a single stabilization system.

The price for this inherent stability is not insignificant, involving as it does the weight of the horizontal tail and the requirement to design the entire rear end of the aircraft for the loads associated with a sizable tail. In the CH-53A, the possibility of depending on redundant stabilization instead of aerodynamic stability was studied. As speed increases and structural considerations enter the picture, such dependence becomes increasingly risky. Rotors tend to become more unstable, and gusts or turbulence more critical. If limited authority auxiliary stabilization is used to offset a large amount of inherent instability, a relatively large amount of authority is required in the stabilization system to prevent unacceptable saturations during turbulence. This larger authority, combined with the greater degree of inherent instability and the load factor sensitivity, can then become unacceptable from the fail-safe point of view.

To illustrate this, Fig. 7 compares the situation for the 150-knot CH-53A and the 105 CH-37, indicating the amount of stabilization system authority necessary to provide satisfactory operation in gusty weather. The data were generated from simulator studies of the stabilized helicopter response to 30-fps gusts. The authority required to avoid unacceptable stability deterioration resulting from saturation is plotted against the combined stabilizer/fuselage angle of attack derivative. By way of reference, the 40-ft$^2$ tail design point shown on the CH-53A line will limit instability at 150 knots to a slow divergence roughly typical of our more advanced helicopters. For the same airframe stability, considerably more authority is required for the CH-53A at 150 knots than for the CH-37 at 105 knots, and the authority requirement increases much faster for the higher speed aircraft as tail is decreased. To allow for trim bias and transient maneuvering, our experience would indicate that the built-in authority should be about 150 to 200% of the amount required to simply stabilize for external disturbances.

A conventional fail-safe demonstration requirement would be to sustain a hardover signal equal to 75% of the built-in authority with a pilot recovery time of 1 sec at $V_{\text{cruise}}$. The hardover load factor increase for an authority adequate to stabilize a neutrally stable CH-53A airframe is about three times what it would be for the CH-37. Although well below the design load factor of three, the factor would be placing the rotor well into blade stall and certainly would be very disturbing under IFR conditions.

There are various ways of arranging dual systems to mitigate the failure implications, but each one has its structural risks if it must cope with a large degree of inherent stability at high speeds. The three-way voting system is the final out, but we have not yet been able to justify this degree of complexity merely to reduce horizontal tail size (any more than have fixed wing designers).

Actually, the hardover fail-safe requirement is considerably more complex than this discussion would imply. The material presented merely indicates the increasing importance of a minimum degree of inherent stability as speed is increased. The real problem is complicated by such things as the nonlinear behavior and stall limits of the rotor at high speeds and

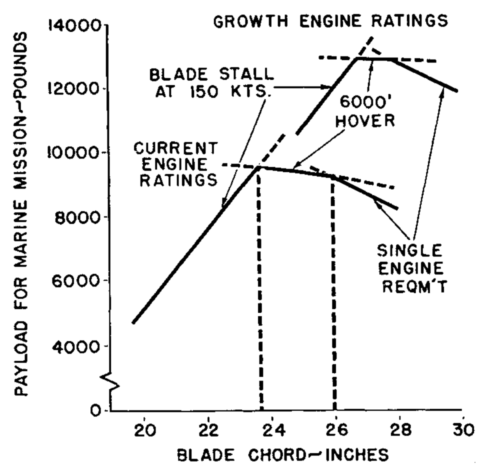


**Fig. 5 Tradeoffs vs blade chord.**

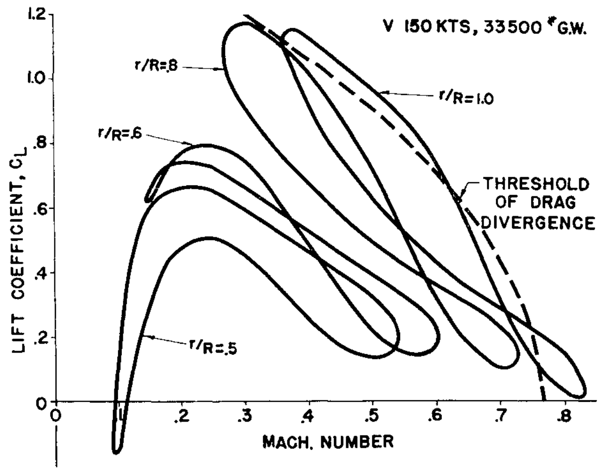


**Fig. 6 Rotor coefficient of lift vs Mach number relations.**

in maneuvers, and the question concerning delay time requirement after malfunctions. The basic point, emphasized by the CH-53A stability studies, is that higher cruise speeds require greater consideration of the degree of inherent stability provided, and dependences on artificial stability assist must be very carefully examined in light of authority and fail-safe requirements.

## Delay Time Criteria

These stability and control studies also pinpoint the importance of resolving the delay time requirement for pilot reaction to an unexpected malfunction. Current specifications for engine or control system malfunction call for from 1- to 3-sec delay, but, in a number of recent situations, rational design resolution has required reducing the delay requirements to about 1 sec. In the course of aircraft endurance testing, Sikorsky Aircraft has conducted stabilization system hardover tests with unalerted pilots as a means of determining pilot reaction time. A 1-sec reaction time is ample provided the failure is of the sort that provides an instantaneous, clear-cut indication to the pilot. Hardover failures of artificial stabilization systems, failures of primary servos which result in force feedback on control, and total engine failures all fall in this category. When the question has come up in the course of design substantiation, Sikorsky Aircraft has generally obtained procuring or certification agency approval, but MIL-H-8501A still calls for 2- to 3-sec delay times for these types. With higher speed helicopters, this question becomes far more critical. The faster an aircraft moves, the faster things happen, and delay time requirements make a tremendous difference in what can be considered acceptable characteristics. Delay time criteria must be resolved once and for all, but this resolution must be based on a soundly conducted experimental program to determine pilot behavior under the types of failures that can occur. The CH-53A program will shed some light on failures at high speed, but an independent program is still needed to obtain statistical answers with service pilots under service conditions.

## Dynamics

Meeting or improving upon generally accepted vibration requirements in a helicopter considerably faster and larger than any other modern production craft, presents a special challenge. The vibration problem originates from the periodic nature of a lifting system with a specific number of blades (a problem not unlike that of a reciprocating engine, but at lower frequency). Speed is also a factor, because the higher the advance ratio, the greater the dissymmetry of blade loading and consequent periodic content of lift and drag forces. In addition, the optimum parametric trend of single lifting rotor helicopters with both size and speed tends to increase the number of blades. This complicates the problem but offers opportunity for improvement in the dynamic characteristics. Size becomes a factor, because we must design for a large flexible airframe reasonably smooth in all locations throughout the cabin, with a variety of loads.

A review of the excitation and transmissibility elements that determine the magnitude of the problem in a rotary wing aircraft provides an excellent summary of the state of our analytical technology available to deal with the problem. The elements in the chain are illustrated in Fig. 8. The analytical problem is one of defining the mathematics of each element and the coupling between them. The techniques of analyzing each of these elements have been developed to varying degrees. The air mass dynamics of the induced flowfield remains the most complex and least clearly defined. Although progress is being made toward understanding this complex aerodynamic phenomenon through research programs supported by Transport Research Command (Trecom) and Bureau of Naval Weapons (BuWeps), we have not quite

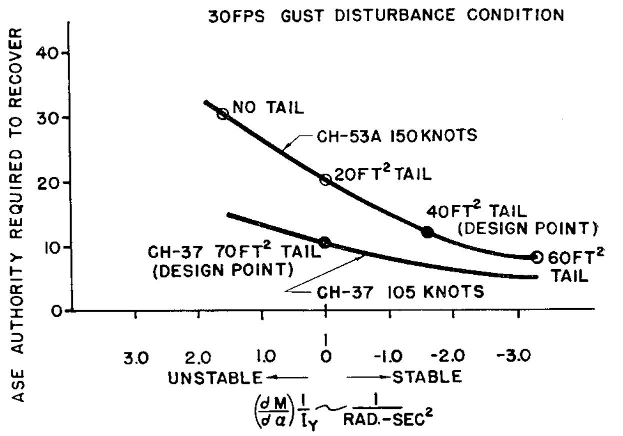


**Fig. 7 American Society of Engineers authority for neutral stability.**

reached a point of having a fully correlated design tool. Reference 2 is the most recent summary of the state of this art. Until this area is completely pinned down, we must settle for constant induced flow field assumptions in our rotor design analysis. Fortunately, such analysis appears to provide adequate estimates of the lower harmonic loadings, which are the major source of blade stresses. However, it fails to yield satisfactory higher harmonic loads, which cause the vibration excitation at blade passage frequencies. The third block in Fig. 8, the dynamics of the flexible blade, is well developed (also reported in Ref. 2). We can calculate the rotating nonuniform beam response with no great difficulty, and we have developed the mathematics to consider the effect of coupling with the fuselage (the impedance of the rotor head). The analysis of $n$ per revolution filtering and coordinate transformations of the blade-fuselage coupling is also straightforward. The airframe response is the final major piece in the puzzle, becoming more difficult analytically as we go to higher frequency excitations with more blades and lower fuselage natural frequencies with larger, relatively more flexible airframes. Our airframe elastic analysis methods appear to be showing good correlation with shake tests in most of the important areas.

At this particular time, our general design approach is as follows: First, we assume that the major lower harmonic blade stresses and rotating push rod loads can be reasonably well predicted from constant induced flow field flexible blade response analysis. Secondly, we make sure we have no significant blade resonances in the operating rpm range with any of the harmonics that can contribute to airframe vibrations; only the $n$, $n + 1$, and $n - 1$ harmonics come through the

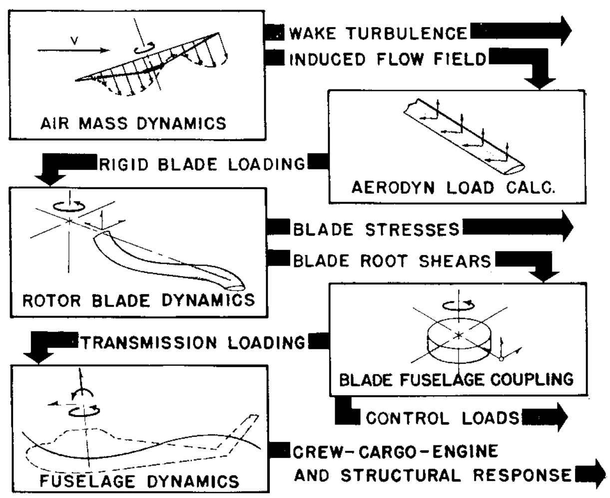


**Fig. 8 Dynamic analysis diagram.**

filtering action of an $n$-bladed rotor system with polar symmetry. We then extrapolate an estimate of the higher harmonic vibratory root shear and push rod loads from flight data on similar rotor systems. Data are available for Sikorsky rotors as a base for extrapolation on two-bladed VS-300, three-bladed S-51, S-55, S-62, four-bladed S-58, S-62, five- and six-blade versions of S-61 and S-56, and six-bladed S-64 rotors, in addition to model rotors with a variable number of blades used in wind-tunnel experiments. We then account for the filtering action and obtain the $n$ per revolution vibratory loads on the transmission and on the nonrotating controls. Finally, these are applied to the airframe elastic model.

The first and second normal modes of the blade are well removed from the fifth, sixth, and seventh per revolution, forcing frequencies that can come through the rotor head, with the exception of the coincidence of the second flatwise bending mode with the fifth harmonic. This coincidence is not expected to present any problem because the flatwise modes are well damped aerodynamically, and the five per revolution flatwise excitation produces only six per revolution hub pitching and rolling moments, which have never been a significant source of fuselage excitation.

The forecast higher harmonic content of the root shears for the CH-53A are shown in Fig. 9. The advantage of six blades becomes evident, because the fifth, sixth, and seventh harmonic excitation is far less than the other possible combinations associated with fewer blades. Figure 10 indicates the resulting six-per-revolution excitation forces, which can be expected to come through the rotor head and act on the fixed coordinates of the airframe, at cruise speed and normal gross weight. To put them in perspective, the steady drag and mass loads are also shown. With the six-bladed rotor, for instance, the vertical excitation at the rotor head is less than 3% of the steady thrust.

Structural design tradeoffs become particularly complicated for the airframe response to these vibratory inputs. The manner in which size affects the problem of fuselage response is indicated in Fig. 11, which compares the relative locations of blade passage excitation frequency with the major fuselage mode frequencies for the CH-53A and the HH-52A (a 7000-lb helicopter). With size, the optimum number of blades tends to increase so that excitation frequencies rise. At the same time, the larger airframes are more flexible, so fundamental modes are lower. As a result, in smaller helicopters the designer must be concerned primarily with fundamental mode response to blade passage frequencies; in larger ones, which operate well above the fundamentals, the principal risk is of exciting a number of less easily predicted higher modes, particularly those associated with the main rotor pylon.

The response at the pilot's seat is of fundamental importance, but the cargo compartment response, particularly as it may vary with load, may often demand conflicting structural flexibility characteristics. The stress content in forced response, which in the past was seldom a problem, must be carefully checked for modal content that will overstress key areas, such as tail cone joints and transmission support structure. Finally, the response of engines and the local transmissibility of electronic and heavier accessories require analytical checks to varying degrees.

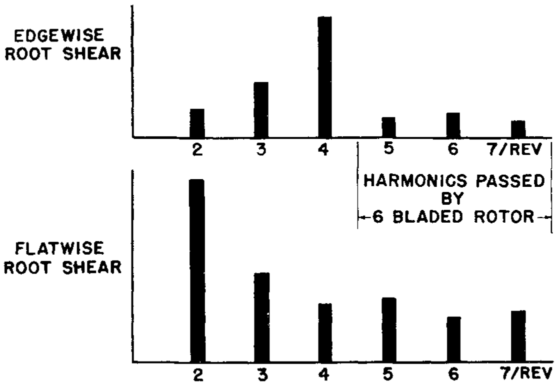


**Fig. 9 CH-53A root shear harmonics.**

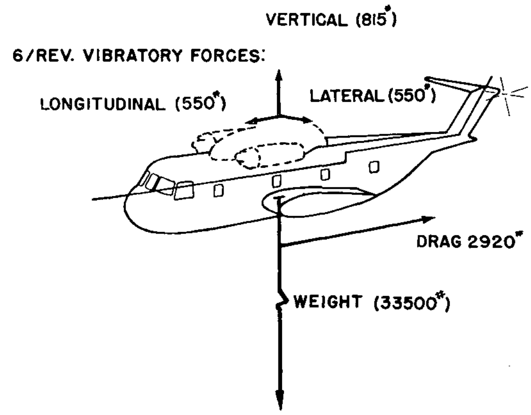


**Fig. 10 CH-53A in-flight forces.**

Obviously, the first step in any design is to avoid any resonance with primary exciting frequencies. Usually the one-per-revolution excitation is not significant in a well designed rotor, but experience with the S-64 has shown that on large helicopters, where the fundamental mode can approach one per revolution, the aircraft will be hypersensitive to blade track and pilot induced oscillations if the fundamental resonant is at one per revolution. In the CH-53A we have chosen to keep the first mode above one per revolution, but this has required careful attention to tail cone rigidity. In the six-per-revolution region on an aircraft of this size, the main rotor pylon modes are most apt to produce trouble. For this reason the careful analysis of the redundant structure in the center section becomes particularly important. The accurate evaluation of influence coefficients in this region has been the subject of considerable study. In the CH-53A, special care had to be exercised in designing transmission support structure to avoid having the main pylon modes in close proximity to six per revolution. The transmission support structure must be designed for crash loads, and the rigidity that follows tends to result in pylon mode $n$ per revolution coincidence. By careful selection of the structural arrangement, it was possible to reduce the stiffness of the pylon in the pitch and roll directions. The pitch and roll natural frequencies of two and five-tenths per revolution and four and one-tenth per revolution should actually provide force attenuation rather than amplify the inplane shears and moments transmitted to the fuselage. Lower response to inplane force has been achieved without the detrimental shift of the vertical mode. This was retained at a frequency sufficiently above six per revolution so an increased transfer of vertical forces would not result. Variation of the predicted modes with load is a matter for concern; but with the major structural stiffness and dynamic mass coincident with the center of cargo loading, the CH-53A modes are relatively insensitive to cargo loading. Figure 12, showing the calculated response of the airframe to the excitation of Fig. 10, gives some further insight into the complex modal content of the forced response.

There is in this analysis a considerable vulnerability to the uncertainties of predicting accurately the elastic characteristics of redundant structures. For this reason, airframe shake tests become particularly important, and a sizable effort has gone into development of shake test techniques for helicopter airframes at Sikorsky Aircraft. Because of the

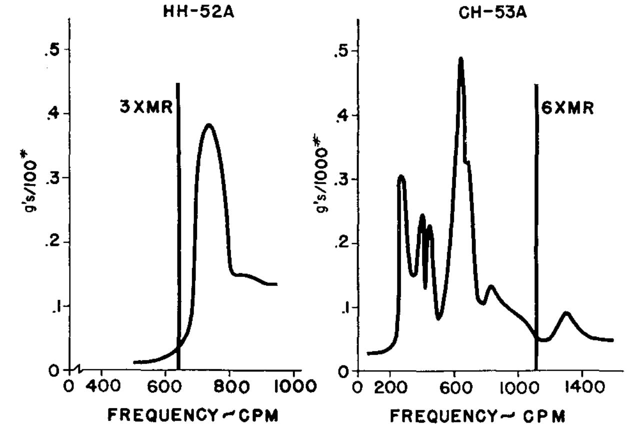


**Fig. 11 CH-53A and HH-52A response relationship.**

complex interaction between modes, it is essential to develop the ability to measure the excitations independently in each coordinate. With the development of a high-capacity unidirectional shaker and relatively elaborate data acquisition instrumentation, the productivity of airframe shake tests has been markedly improved. It is now possible to obtain reasonable correlation between flight measured blade excitation, flight measured response, and the ground shake tests—a problem that has in the past frequently defied logic. The importance of this cannot be overemphasized in dealing with very large airframes, where higher mode response is all important and cut and try fixing is expensive and often unproductive.

One additional dynamic problem deserves particular mention. Experience on the SH-3A vividly demonstrated that the new lightweight turbine engines are no longer merely concentrated weights to be considered as easily isolated sources of potential high-frequency excitation. They are complex, relatively delicate structural entities with a vast number of modes of their own which may respond to fundamental rotary wing excitations and also may couple with the elasticity of the installation structure to produce unacceptable lack of tolerance to their own balance limitations. On the SH-3A, the installation was thought to be trouble free for approximately the first 100 aircraft until a minor design change put the spotlight on an incipient dynamic response problem that could get out of hand on certain aircraft. The result at this late stage in the program was very disappointing and required extensive investigation, culminating in a compromise fix that added an unwelcome and unnecessary weight penalty. To insure against repetition of this situation, the Bureau of Naval Weapons has wisely supported an engine/airframe dynamic investigation, as a collateral program to the CH-53A, to thoroughly explore all possible problem areas at this vital interface. Added to the SH-3A experience, this should greatly enhance both the airframe and engine manufacturers' knowledge of methods to provide reliable control over this new problem area.

Dynamics is one of the most significant areas in which the CH-53A program should contribute a major technological step forward. The flight test program, which will take advantage of the full capabilities of tape data acquisition and processing, and the shake test program, which will double check the most advanced methods of redundant structure elastic analysis, will provide important analytical correlation. In addition, Research and Development programs to measure higher harmonic blade loads, stresses, and root shears in greater detail and at higher speed conditions than previously possible will provide an ideal check point for the more elaborate analytical techniques that are just emerging from the Research and Development phase for predicting the effects of nonuniform inflow.

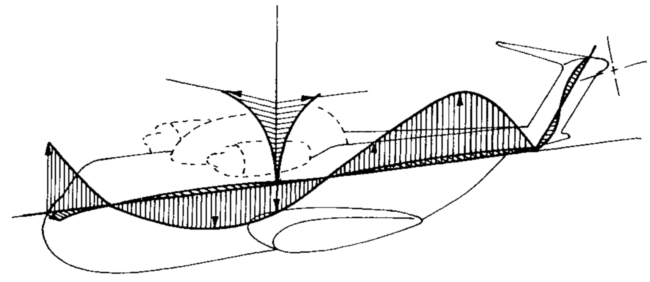

**Fig. 12 CH-53A in-flight response shape.**

## Titanium Utilization

In the materials area, the most interesting subject to report on is the extensive use of titanium in dynamic components. To achieve the significant improvements previously discussed requires taking advantage of all possible areas of technological advancement. The increased knowledge of titanium is clearly one of these areas.

Figure 13 shows the current distribution of titanium in the CH-53A. We are applying titanium primarily in the dynamic components, where low stress and high cycle loading capitalize on its excellent fatigue characteristics and where its ability to withstand corrosion is welcome. The benefits of making this substitution for steel are appreciably greater than in applications where the primary advantage is in the ultimate tensile strength to density ratio. Careful consideration has to be given to the fretting problem and the difference in modulus of elasticity, but in many of our dynamic component applications a direct one-for-one substitution of titanium for steel is entirely feasible.

The effect of titanium utilization on weight efficiency of a major dynamic component is indicated by the weight reduction achieved with titanium in the CH-53A main rotor head. This component makes a particularly good example, since the head was originally designed in steel, and both the steel and titanium heads have been carried through to fabrication. Of the 1900 lb of structural steel originally used in the rotor head, 70% could be converted to titanium on the basis of our current titanium design and fabrication know-how. The direct one-for-one substitution of titanium for steel would have resulted in a 560-lb savings on the simple basis of density ratio; the actual savings was 500 lb. This indicates the very small amount of redesign required to account for the change in modulus of elasticity of titanium or special provisions for wear or fretting. The savings of 26% on this one component alone is dramatic; on the CH-53A, the total savings from the use of titanium amounts to approximately 820 lb. Some idea of the size of machined part we are dealing with is illustrated in Fig. 14, a photo of the main rotor hub plate.

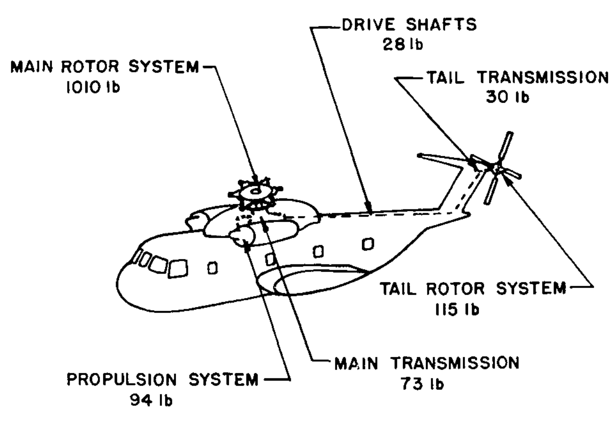


**Fig. 13 CH-53A distribution of titanium.**

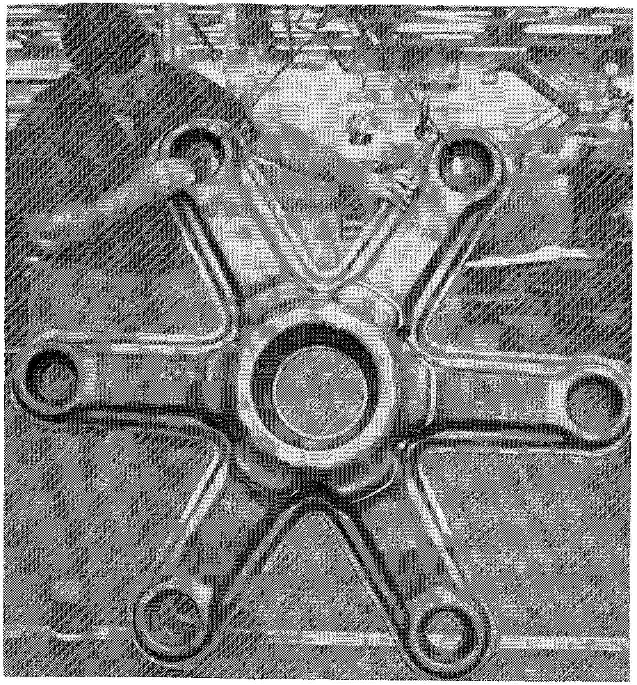

**Fig. 14 CH-53A titanium main rotor hub.**

For our helicopter dynamic components, we have selected the 6 Al-4V annealed titanium alloy because of its wide use and the amount of accumulated experience. To report on all that has been learned in the application of titanium to helicopter dynamic components would far exceed the scope of this paper. Reference 3 has already reported on our machining and fabrication experience, and a technical paper on the design and substantiation aspects of titanium utilization is planned for another technical session in the near future. Much of our effort has been directed toward pinning down the fatigue characteristics, particularly as determined by production finishes. Prior to the CH-53A program, over 600 fatigue specimens had been tested. Finishes on which information has been obtained include the following: as forged; chem milled; surface finished to 15AA, 50AA, and 100AA; and shot peened.

A good illustration of the similarity of steel and titanium fatigue characteristics of a typical helicopter part is indicated by the comparison of the fatigue test results on an S-61 main rotor cuff in which titanium was directly substituted for steel on a one-for-one basis. In the tests, there was no discernible difference in the fatigue strength between titanium and steel, and the mode of fatigue failure was identical. In the CH-53A program, about 70 component specimens are scheduled for fatigue testing; this will greatly increase the statistical depth of data on fatigue characteristic of actual parts.

In the CH-53A, we currently have about 1350 lb of titanium. Although this represents only 6% of the total empty weight, it is a major increase over any previous model and foreshadows even greater application in the future. As more experience is gained on the titanium characteristics and as the fabrication techniques are more fully developed, it is easy to visualize a 10–12% titanium content in empty weight, in the near future.

## Conclusions

The discussion has necessarily been limited to a representative cross section of the more interesting and unique areas of technical development in which the CH-53A is contributing. Many other efforts, such as the Drag Control Program, the Reliability Program, and the Integrated Maintenance Management Program, also deserve mention as major contributors to efficiency of the final design and further improvements in rotary wing technology. The CH-53A has had its share of formalized design control of all the "abilities" and the similar "technologies," such as human engineering and value analysis. The program has been very useful in extending our practical applications of these design specializations and the philosophy of integrating them into the organization. Any adequate discussion of these areas will have to be covered in a separate paper.

The CH-53A, with four times the productivity of the helicopter it replaces, will dramatically increase the reliability and efficiency of the support helicopter in vertical assault operations. The maturing of rotary wing technology, which the CH-53A program has furnished the opportunity to demonstrate, is equally important for its potential to provide greater reliability, speed, and efficiency in all types of rotary wing aircraft.

## References

[1] Legrand, F. L., "Structural behavior of the super frelon in high speed flight," *Proceedings of the 20th Annual National Forum* (American Helicopter Society, Inc., 1964), pp. 109–116.

[2] Gerstenberger, W. and Wood, E. R., "Analysis of helicopter aeroelastic characteristics in high-speed flight," AIAA J. **1**, 2366–2381 (1963).

[3] Bredin, H. W., "Sikorsky turns to titanium forgings for high-performance helicopters," Machinery Magazine **70**, 88–93 (July 1964).