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checking advantage new material offer research present introduction structural design provides study relevant literature aircraft principal char acteristics performance ability collected structural load computed research used several method design using conventional method applying theory elasticity performed using theory farrar allowing u obtain analytical solution problem followed checking obtained result using ansys fem software combined part designed catia furthermore panel calculated using composite material instead conventional aluminum allowing u compare solution research shed light intricate process aircraft structural design material selection calculation methodology highlighting ongoing pursuit new advanced material paper make clear using composite material present several advantage traditional one allowing lighter safer fuel efficient sustainable aircraft use composite material construction airplane structure driven many factor result show chosen composite material reduce weight durable low maintenance quirements reduce noise enhance fuel economy resistant corrosion 1 introduction design aircraft structure complex challenging task requires thorough understanding physic mechanic involved particular calculating structural property rigid panel located wing box beam aircraft real life problem design engineer often face early stage aircraft design problem requires careful consideration aircraft specific characteristic load panel subjected century development material use aviation key area focus within aeronautical industry historic moment wright brother airplane made inaugural flight 1903 engineer scientist corresponding author e mail address kale utku kjk bme hu u kale hva nl u kale http doi org 10 1016 j heliyon 2024 e27403 received 12 july 2023 received revised form 28 february 2024 accepted 28 february 2024 availableonline5march2024 2405 8440 2024 published elsevier ltd open access article cc nc nd license http creativecommons org license nc nd 4 0 p korba et al h e l n 10 2024 e27403 searching material unique characteristic meet rigorous demand aviation 1 2 driven need improve aircraft safety efficiency overall performance characteristic highly sought including among others density mechanical resistance corrosion resistance fatigue resistance first aircraft model made wood fabric first fulfilling structural role second lifting role however due characteristic material affected biological action maintain property aircraft working condition humidity 1 wide range operation aircraft must oscillating 50 c reached summer certain airport cid 0 60 c existing cruise 40 000 ft gave way use metal structural aero nautical manufacturing although steel first metal used fact led problem related low corrosion resistance high density factor mean currently used specific part subjected high load landing gear 2 pursuit material posse desirable characteristic constant challenge aeronautical industry material used aviation must able withstand high stress environment occur flight including extreme tem perature change high altitude condition exposure harsh chemical 3 developing new material meet de mands played critical role advancing aviation industry helped make air travel safer efficient ever 4 thus next stop enter process optimization research order find material complied property described high specific mechanical resistance aluminum starting point subsequent de cades even though aluminum good mechanical property use aluminum alloy adaptable aero nautical sector represents fundamental advance covered much history recent aviation thus duralumin arises higher specific resistance steel improves behavior corrosion 5 however necessary take account addition structural function aircraft component whose main request thermal resistance following point precious metal alloy tungsten titanium used based good mechanical strength thermal property term superalloys mean alloy superior heat corrosion resistance maintain superior property even elevated temperature thus superalloys synonymous high temperature alloy traditionally superalloys classified three type according base element iron based nickel based cobalt based superalloys 6 however broader sense alloy preserve mechanical physical chemical stability high temperature severely corrosive environment called superalloys 6 room temperature however mechanical property su peralloys much different steel much cheaper quickly produced however superalloys stand metallic alloy high corrosion resistance even temperature range making ideal candidate use severely corrosive environment offshore petroleum platform 7 8 nevertheless high cost complex process obtaining machining present mean metal fundamentally limited engine part surprisingly superalloys main application area elevated temperature despite impressive room temperature cryogenic property 9 high temperature superalloys unique mechanical chemical property 8 big leap forward superalloys result intense effort developing gas turbine hot section part meet need military aircraft today military civil aircraft gas turbine still play dominant role developing superalloys aviation gas turbine represent majority application superalloys among application power generation gas turbine classified chemical industry medical industry petrochemical equipment spaceship rocket engine nuclear reactor submarine high temperature industrial furnace various application need high temperature chemical resistance 10 12 today energy sector strategically play even critical role ever escalating fuel price 13 global warming increasing environmental pollution necessitate efficient use petroleum product mandate decreasing fuel consumption exhaust emission aviation industrial gas turbine clearly paramount 14 dictate key positioning new development direction superalloys therefore current technological context compromise solution maintains property close aluminum alloy notable decrease weight structural component developed use composite material fundamental advance last year 15 composite material refer combination macroscopic scale two uniformly distributed material formed fiber generally carbon glass kevlar responsible supporting load providing piece strength rigidity matrix thermosetting thermoplastic resin provide cohesion hold fiber together transmit load 16 fact able obtain property either two constituent achieve independently make composite material versatile solution great future technological sector addition weight reduction implies decrease fuel consumption lower greenhouse gas emission le operating cost composite material better behavior impact corrosion abrasion addition high thermal resistance 17 evolution composite material within aircraft developed last 45 year demonstrates aircraft b 787 a350 contemporary competing within market segment currently exceed 50 composite 3 distribution material used manufacturing b 787 preponderance composite material compared metal aluminum steel despite aircraft designed 15 year ago aircraft fabricated 50 composite particularly steel used highly requested area pylon engine 18 fact airbus boeing two leading company dominate international aeronautical manufacturing dedicate significant resource optimization structural design development composite material show current trend within aeronautical sector another aspect highlight search innovation material process lighten weight aircraft use technique topological optimization additive manufacturing manufacture part without necessarily made 2p korba et al h e l n 10 2024 e27403 advanced material lighter would manufactured designed conventional technique 19 process unfolds starting conventional design iterating removing material mechanically stressed final design mechanical strength significantly lower weight result development process present assignment proposes structural design rigid panel different point view one hand traditional design us aluminum advanced one us composite material allow u compare result able draw justified conclusion trend aircraft structural design manufacturing 20 research atr 72 500 airplane chosen reference aircraft aircraft widely used throughout europe regional medium haul transport abundant information data available airframe structure additionally atr 72 500 simpler construction compared larger aircraft mean easier deal fewer parameter simpler geometry still maintaining rigorous calculation method structural typology atr 72 500 aircraft semi monocoque high wing structurally wing box beam type wing box beam two main longitudinal crossbar beam wing upper lower fairings serving structural part thus aircraft square section beam variable section outer part wing decreasing chord compared middle section chord constant designed panel part discussed fairing wing structure considering section root wing embedding fuselage weakest point structure predominance bending force panel must designed accordingly aluminum would typically made aircraft main secondary tertiary structure many year metal alloy seen represent pinnacle technological advancement however currently sophisticated composite material preferred usage structural aircraft design determine possible use conducted set measurement simulated environment however possible make measurement real environment research contributed excellence aerospace studied increase composite material quality aerospace industry composite material reduce weight material used aerospace industry time increase efficiency performance safety management whole article organized several section firstly main data parameter reference aircraft presented followed section 2 review current state art section 3 methodology material selection described section current technical solution employed aircraft structure specific calculation method used discussed including analytical fem technique section 4 provides step step presentation result obtained method employed calculate rigid panel section 5 present discussion finding article finally section 6 draw conclusion based result obtained 2 related work year aeronautical sector general aircraft design particular experienced substantial growth technological improvement leading technique used today however process related maneuver envelope load calcula tions basic element obtained development problem others using composite material aircraft design fem analysis important advantage developed last decade although research project related aircraft structural design necessary mention aid used software needed solving problem following point important mention reference 21 tutorial catia v5 outline main capability application software well specific project like 22 focused similar work paper working catia v5 nastran patran computer software work engineer furthermore fem calculation processed student version ansys employing reference 23 outline important us ansys mechanical apdl aeronautical structural purpose preliminary design calculation paper included definition maneuvering envelope atr 72 according ref 24 crucial fly condition envelope fatal disaster occur furthermore reference 25 also used method design regional turboprop aircraft reference 26 discus result diagram n v consideration based upon international norm c 25 27 28 defining legal prescription aircraft meet another important consideration dimensioning panel calculation load applied wing structure necessary distinguish load described ref 26 structural aerodynamic engine fuel mass addition contemporary bibliography revised 29 preliminary design wing box structure per formed several numerical cfd load calculation previously cited 22 load calculated introducing fem analysis deeper understanding aerospace structural design book 30 used general reference know common solution adopted purpose specific problem related structure buckling developed ref 31 33 different wing box calculation approach made integrating static buckling fatigue manufacturability requirement different loading condition nonetheless cited 22 reference similar methodology wing calculation performed however cornerstone paper based upon theory farrar first conceived 1949 34 expanded 1953 35 considers design structural element subject mainly compressive load resist buckling main cause failure theory specifically designed metal part revised recently 2017 36 update regarding computer aided numerical fem consideration second part paper based comparison solution using aluminum composite material approach context reference 37 used general purpose book aerospace material well 26 38 39 taken 3p korba et al h e l n 10 2024 e27403 reference paper practical study use composite material similar aerostructures important information regarding atr 72 aircraft technical geometrical characteristic could found airplane manual 40 collection data procedure needed safely operate aircraft well source 41 43 aircraft get depth revision starting point design new similar regional transport airplane finally mathematical model important explanation presented throughout paper corresponding section 3 methodology material object project explained material used throughout history aeronautical structure current trend within sector analyzed section proceeds explain chosen approach way get solution problem fig 1 show process design calculation process firstly analysis situation performed followed data collection related propriety aircraft question calculation load took place form load due structural mass wing load due engine load due aerodynamic effect load due fuel tank following simulation permitted formulation result main parameter characteristic atr 72 500 needed calculating load airframe presented table 1 2 table 1 provides information aircraft component mass well key performance parameter table 2 offer crucial information wing structure geometry information critical accurately determining load aircraft structure must withstand operation additionally necessary assumption made mass lift distribution wing location shear center center gravity presence fuel tank assumption following research normally used reference constant considered assuming standard atmosphere isa sea level γ 1 4 r 287 j kg k ρ sl 1 225 kg m3 gravity take usual value g 9 8 s2 mass distribution wing follows law kc 1 2 lift distribution elliptical cruise flight lift provided wing 105 total aircraft lift htp providing negative 5 shear center located 50 wing chord center gravity 40 lift 25 chord commonly accepted two fuel tank per semi wing considered aft stern crossbar beam 15 60 chord total fuel amount 4189 5 l discussion topic done later commonly used sign criterion bending moment considered throughout whole research objective research summarized two main goal firstly intention validate correctness design rigid aluminum panel comparing analytical result obtained manual calculation fem ansys software analysis permit simulate analyze complex load condition predict structural behavior performance optimize design variation secondly aim evaluate feasibility replacing conventional aluminum panel cfrp composite variant comparing respective fem software calculation part analysis resolution metal wing box composite material one essential analyze maneuvering envelope reference aircraft obtaining flight diagram n v relates load factor speed flight matlab programming software used next critical load suffered aircraft symmetrical steady flight fig 1 flowchart design calculation process 4p korba et al h e l n 10 2024 e27403 table 1 mass performance data atr 72 500 magnitude symbol value unit maximum take weight mtow 22800 kg lb 50625 maximum landing weight mlw 22350 kg lb 49272 maximum zero fuel weight mzfw 20800 kg lb 45856 semi wing mass mwing 1522 5 kg engine mass meng 887 kg cruise speed vc ta 140 max negative lift coef normal ref line cnmax neg 0 715 lift coef normal ref line take cnmax 2 145 lift coef normal ref line landing cnmax l 2 684 lift coef normal ref line landing cnmax l 2 684 lift coef normal ref line clean configuration cnmax 1 511 table 2 wing data atr 72 500 magnitude symbol value unit wing surface 61 m2 chord root cr 2 6 chord tip ct 1 5 span b 27 05 fuselage diameter ϕfus 2 77 m2 fuselage longitude lfus 27 166 one wing longitude bow 12 245 airfoil center gravity cgaf 0 4 c airfoil lift center caf 0 25 c distance motor cg wing leading edge xeng 0 85 distance motor cg wing root yeng 2 665 calculated following c 25 333 standard considering structural weight wing engine aerodynamic load fuel tank load development different hypothesis carried discussed throughout section matlab programming software used item since convenient platform numerical calculation data analysis required research also case ref 44 46 focusing design phase different case case aluminum panel analytical calculation problem approached point view theory elasticity resistance material going load tension using von mi criterion calculate required thickness panel also used study 45 47 48 process theory farrar allows calculating integral panel metallic material use efficiency factor farrar factor semi empirical abacus used mainly theory farrar based experimentally demonstrated fact optimal design one torsion stringer negligible decoupled bending stringer thus pure bending main mode instability stringer quantify instability aforementioned farrar coefficient used rest element involved theory developed result section used matlab programming software main tool calculating developing iterative process allow u get first approximation design helpful second stage research second stage catia three dimensional design software used design panel stringer stage possible make precise measurement analysis mean software allows testing different design option evaluating impact overall structure ansys structural calculation software used verify numerically chosen design valid checking validity theory farrar behavior panel stage permit identification potential issue design suggestion modification improve strength stability selection right material crucial significantly affect aircraft weight durability overall performance therefore final stage research ansys analysis used compare performance different material evaluate benefit drawback material ansys analysis composite material panel carried order compare result material furthermore analysis allows researcher validate support new trend material manufacturing technique presented related work section 5p korba et al h e l n 10 2024 e27403 4 result 4 1 maneuver envelope section maneuver envelope calculated cruise altitude computing first limit load factor accordance norm c 25 337 allowing value 2 5 3 8 giving result next allowed value 2 5 well cid 0 1 limit negative load factor hand dive speed vd calculated conversion mach number real speed obtaining vd 125 18 maneuver speed calculated obtaining va 99 51 necessary calculate maneuver speed negative limit load factor give value vh 91 5 using ncruise 1 speed flap deployed must taken account using following formula consideration considering maximum value among 1 6vs1 mtow flap take configuration 1 8vs1 mlw flap approximation configuration 1 8vs0 mlw flap landing configuration obtaining final value vf 68 14 finally previous step plotting calculation flight envelope diagram necessary convert ta ea true airspeed equivalent airspeed hence result presented table 3 fig 2 flight envelope 4 2 calculation load section computation applying load wing box structure considered purpose necessary take consideration structural weight weight engine load due aerodynamic effect load due fuel stored inside wing tank case study according norm c 25 333 calculation critical load stationary symmetric maneuver given result bending moment shear tensile force torque one semi wing exists plane symmetry along middle aircraft following fig 3 main dimension wing observed along engine position line torque center assumed torque applied section 50 chord hypothesis apart necessary determine exact chord length section distance 50 chord reference straight line whole length semi wing called r50 another hypothesis weight distribution wing structural mass follows law calculated constant k 42 645 obtained equalling total mass integral law along whole wingspan see eq 1 kc1 2 1 preliminary calculation made possible start computing applying load weakest point fuselage embedment wing box beam main fuselage airplane due fact cantilever beam significant part stress result bending moment accumulating tip beam embedment something easily extrapolated model load due structural mass wing weight semi wing considered applied 40 chord section responsible bending moment mf around x axis torque mt around axis shear force v direction z axis starting calculation bending moment mf see eq 2 ytip f cid 0 gn lim ydy cid 0 gn limk yc 1 2dy cid 0 224706 81nm 2 yemb yemb distance middle plane aircraft embedment root semi wing ytip value coordinate tip semiwing parameter used defined earlier negative sign mean table 3 speed ta ea magnitude vtas veas vc 140 70 108 00 vd 163 09 125 18 va 129 64 99 51 vh 119 20 91 50 vf 68 14 52 30 6p korba et al h e l n 10 2024 e27403 fig 2 flight maneuver envelope atr 72 fig 3 semi wing dimension hypothesis carried wingtip tends bend downwards torque mt calculated occurs embedment wing root see eq 3 cid 0 ytip gn lim cid 0 x tor cid 0 x cg cid 0 r 50 dy cid 0 gn limk c 1 2 cid 0 0 1c cid 0 r 50 dy cid 0 9614 93nm 3 yemb observed order magnitude result much lower compared bending moment negative sign indicate whole aircraft would tend pitch result accordance fact center gravity line wing located advanced position chord center torque line resultant force applied finally value shear force critical section wing see eq 4 ytip v cid 0 gn lim dy cid 0 gn limk c 1 2dy cid 0 33796 03n 4 yemb result nothing vertical force caused weight whole structural component wing load due engine shear force bending moment torque created existence one engine per wing considered mass meng 886 6 kg taking account associated system located 4 653 symmetry plane aircraft 85 cm front wing center line furthermore assumption made engine mass concentrated one point 7p korba et al h e l n 10 2024 e27403 calculated bending eq 5 f cid 0 gn limm engy eng cid 0 101173 67nm 5 similar comment one made load due structural mass made section computation torque shown eq 6 cid 0 c cid 0 gn limm eng cid 0 2eng x eng cid 0 r 50 eng cid 0 46749 06nm 6 torque induces pitch aircraft taking account said position engine finally shear force wing root due engine mass calculated see 7 v cid 0 gn limm eng cid 0 21743 75n 7 load due aerodynamic effect another important source force taken account structural design aerodynamic force flight instead force created opposition displacement mass newton second law like previous case result airflow around wing critical hypothesis consider lift concentrated wing section 25 chord assumption commonly made necessary lift provided wing 105 total lift required airplane type aircraft normally designed trimmed fly horizontal tail plane elevator providing negative lift 5 total mean total lift calculated follows taking consideration limit load factor weight aircraft see eq 8 l 1 05n limmzfwg 535626n 8 lift considered elliptical throughout whole wingspan result would zero wingtips maximum center following next definition give lift section wing see eq 9 4l 2y 2 l 1 cid 0 9 πb b calculation bending moment torque shear force started talking bending moment following result obtained eq 10 ytip 4l 2y 2 f l ydy πb 1 cid 0 b ydy 1513179 56nm 10 yemb positive according expected mean wingtips bend upwards result aerodynamic force applied wing torque calculated following next two expression eq 11 12 first one evaluates torque created result lift applied center torque line whereas second measure torque due coefficient moment cm ytip 4l 2y 2 l x tor cid 0 x aero r 50 dy πb 1 cid 0 b 0 25c r 50 dy 148707 04nm 11 yemb ytip 1 coeff 2 ρ slv e2 asc c 2dy cid 0 45646 05nm 12 yemb obtained result different sign first would make aircraft pitch positive second would opposite according sign criterion moreover observed second value absolute value smaller first one overall torque created aerodynamic effect make airplane pitch positive value finally shear force computed equal total lift provided wing see eq 13 v l 535626n 13 load due fuel tank last important item taken consideration estimate load applying wing fuel tank ac cording geometrical consideration wing structure two fuel tank semi wing height 14 total 8p korba et al h e l n 10 2024 e27403 chord section leading part 15 chord rear 60 wing bow aft main beam wing furthermore length first fuel tank go 12 30 semi wingspan embedment wing engine area whereas second one extends 34 4 89 5 taking part volume non constant section semi wing reaching total necessary fuel volume 4189 5 l tank requisite calculate bending moment torque shear force considering position center gravity 0 975 2 840 x coordinate first tank 1 029 6 011 second starting bending moment eq 14 15 cid 0 f1 cid 0 gn limv ft1ρ f cg1 cid 0 emb cid 0 29566 23nm 14 cid 0 f2 cid 0 gn limv ft2ρ f cg2 cid 0 emb cid 0 95926 48nm 15 bending moment negative create downward displacement wingtips previously necessary compute volume fuel tank torque calculated eq 16 17 cid 0 cid 0 cid 0 t1 cid 0 gn limv ft1ρ f x tor cg1 cid 0 x cg1 cid 0 r 50 cg1 cid 0 6604 14nm 16 cid 0 cid 0 cid 0 t2 cid 0 gn limv ft2ρ f x tor cg2 cid 0 x cg2 cid 0 r 50 cg2 cid 0 3179 04nm 17 torque negative center gravity advanced position compared center torque line wing last last result calculated shear force result fuel tank wing see eq 18 19 v 1 cid 0 gn limv ft1ρ f cid 0 20320 43n 18 v 2 cid 0 gn limv ft2ρ f cid 0 20736 38n 19 total load critical section final result presented following table 4 total load calculated sum value type force 4 3 first case aluminum panel structural calculation addressed allow give dimension different element make wing box atr72 especially wing panel stand structural load thus simple method used comprises calculation general parameter wing box using classical result theory elasticity strength material passing load stress normal tangential thickness computed using von mi criterion second step farrar theory design stiffened integral panel applied widely used aviation stronger lighter reinforced bolted screwed small longitudinal beam purpose beam stabilize panel subject compression load avoid buckling important hypothesis simplification taken account preliminary design applying farrar theory panel modeled plain plate constant width straight stiffening beam parallel real world many requirement must met may exactly case 4 3 1 load flux calculation design wing box first calculation needed design wing box made assuming monocoque structure thin wall mean made several panel comprising upper lower panel one object design two lateral beam turn make wing box structural part first required compute inertia first cross inertia zero due symmetric nature wing box rectangle origin center point apart one also necessary know ixx also calculated following eq 20 21 table 4 flight data atr 72 500 load type v n mf nm mt nm aerodynamic 535626 00 1513179 56 148707 04 cm cid 0 45646 05 fuel tank 1 cid 0 20320 43 cid 0 29566 23 cid 0 6604 14 fuel tank 2 cid 0 20736 38 cid 0 95926 48 cid 0 3179 04 structural cid 0 33796 03 cid 0 224706 81 cid 0 9614 93 engine cid 0 21743 75 cid 0 101173 67 cid 0 46749 06 total 439029 41 1061806 37 36913 81 9p korba et al h e l n 10 2024 e27403 xy 0 20 1 h 2 xx 2 12h3t h lt l 2 21 normal stress first calculation pas load stress done normal stress σ computed follows using classical formula computing normal stress section see eq 22 23 σ xy fy 22 xx xx h σ max f 2 23 xx shear stress shear stress necessary compute torque shear force load first torque produce even distribution shear stress quantified following formula see eq 24 q 0t 2at 2ht l 24 hand shear force create shear flux given expression eq 25 q cid 0 iy tyds 25 xx 0 using symmetry anti symmetry result two following equation obtained quantify flux shear stress middle panel upper left corner wing box point middle left main beam wing box respectively see eq 26 27 h q 12 q 1 cid 0 iyt l2s 26 xx h q 23 q 2 cid 0 iyt h 2 cid 0 2 27 xx obtaining value shear stress flux converted pure shear stress using following expression necessary use maximum value shear stress flux add flux coming torque resulting flux divided thickness main beam wing box total pure shear stress computed see eq 28 29 q max q 0t q max 28 q τ max tmax 29 h finally von mi criterion applied determine minimum width pair beam need order stand load obtained done using interactive process matlab thickness changed von mi criterion converges see eq 30 σ2 max 3τ2 max σ e 30 4 3 2 theory farrar part focus calculation needed define geometric parameter stiffened integral panel purpose first required define safety margin 50 give following result previously computed load see eq 31 33 f ult 1 5 f 1592709 56nm 31 ult 1 5 55390 87nm 32 v ult 1 5 v 658544 12n 33 10p korba et al h e l n 10 2024 e27403 last section necessary consider shear force torque flux methodology one explained result shear stress flux value qb however needed calculate load per unit length panel would stand called nb defining model two parameter nb computed eq 34 n b hf lult 34 farrar theory work compressive load necessary find way transform shear flux equivalent compressive load called nmax order parabolic interaction curve considered normal load shear flux correlated defined safety margin qa na 10 m definition instead qb nb non linear equation system defined next four mathematical expression order obtain nmax see eq 35 38 q 2 n 2 1 35 q n max max q 2 n 2 q 2 n 2 1 m 2 b b 36 q n q n max max max max q q b 37 n n b q2 τ2 l max max 38 n max f2 1 r br 2 e first equation represents definition parabola second one show relationship variable without 10 security margin qa na versus qb nb third equation derived application similarity triangle help solve system whereas last one show relationship q2 nmax according max farrar theory result addition fundamental define parameter necessary farrar design stiffened integral panel parameter relationship rb rt farrar factor efficiency f shown equation necessary designing process maximize efficiency optimal value according theory chosen see eq 39 r b 0 65r 2 25 f 0 81 39 value applied chart shown fig 5 entering rb rt value give value f noted dotted line represent isolines efficiency structural part discussed value apply start obtaining geometric parameter stiffening small beam integrated panel first thickness t calculated knowing thickness proper panel followed computing width spacing beam see eq 40 41 r tt 40 r bb 41 order perform calculation following set two equation used needing input distributed load n applying integral panel see eq 42 43 course process iterative thus done help matlab software convergence whole system attained nl f σ 42 e n σ 43 1 r r b combining last two equation final formula deduced relationship l n et obtained optimal efficiency case study see eq 44 1 nl l 0 501 e 44 f 1 r r e ne b finally next set equation allow computing b value spacing one stiffening beam next mathematical formula obtained farrar theory combined optimal case way straightforward relationship l n et obtained like previous case see eq 45 47 σ 2 σ b σb3 62e b 45 0 11p korba et al h e l n 10 2024 e27403 f 1 3136 σ b 1 4 r3 br 4 r br 1 4 46 σ 1 r r 0 b f 1 r r nl3 1 4 nl3 1 4 b 1 103 b b 1 33 47 r3 br 4 r br 1 2 e e already obtained b total amount stiffening small beam computed quotient width panel l spacing stiffening beam b see eq 48 l n l b 48 lastly value average stress flowing panel obtained shown eq 49 n e σ f lax 49 equation conveniently programmed system solved matlab necessary iteration following final result table 5 stiffened integral panel subject compressive load given 4 3 3 ansys fem study calculated analytically dimension integral panel dimension integral panel analytically highly advised perform fem analysis finite element method validate result purpose static structural analysis ansys performed defined previously three dimensional body catia cad model imported ansys taking shape seen fig 4 meshing using structured pattern already implemented could better instead student license full one used allowing finer meshing next step definition boundary condition load fixed support set boundary condition embedment wing root impeding rotation displacement apart lateral face stand main beam wing box established model make displacement z direction whereas face one load applied displacement along axis allowed regarding load remote force applied evenly whole surface loaded face multiplying value obtained distributed load nmax width panel l consideration shown fig 5 another critical definition made calculation done establish material case aluminum alloy 6061 thermal treatment t6 considered widely used material aviation industry particularly common manufacturing piece like integral panel see fig 6 parameter problem defined turned fem solving giving result following picture among thing first second one fig 7 8 represent stress field panel obtaining maximum stress value yield strength material thus validating previously done calculation complement stress field fig 9 10 show strain field obtained obviously high correlation field observed significant stress mean material suffer bigger deformation thus higher value strain end fem analysis fully validate result eigenvalue buckling analysis performed ansys demon strated fig 11 12 important farrar theory widely based instability may occur panel define minimum resistance mean would uncommon buckling happen smaller load plas tification weakest section important goal buckling analysis obtain load factor relation maximum load considered static structural problem buckling occurs value one buckling happen exhaustion portant capacity panel whereas would happen value factor one interesting obtain factor one much mean efficient use material course factor around two obtained fully validates table 5 result obtained farrar theory magnitude value na 2 646 106 n qa 4 008 105 n nmax 4 578 106 n qmax 6 172 105 n σm 487 5 mpa 3 8 mm t 7 58 mm b 110 66 mm 71 92 mm nl 8 12p korba et al h e l n 10 2024 e27403 fig 4 mesh generation fem analysis ansys fig 5 boundary condition load definition fem analysis ansys calculation made 4 4 second case composite material panel current trend material science show composite material advantageous compared normal one many case due fact composite much higher specific strength compared example aluminum make ideal application weight saving paramount like aircraft construction presented beginning article cfrp carbon fibre reinforced composite widespread material 13p korba et al h e l n 10 2024 e27403 fig 6 property aluminum alloy considered defined ansys fig 7 stress field obtained fem analysis ansys upper part construction airframe aeronautical structure general often comprising 50 weight total structural mass study fem done previously implemented using epoxy woven carbon fibre prepreg 230 gpa commonly used composite structure done analysis aluminum panel result consist fig 13 showing stress field fig 14 showing strain field fig 15 performing eigenvalue buckling analysis interesting thing approach composite factor 3 decrease maximum stress consequently panel even near level stress would make fail lighter aluminum counterpart presented density aluminum around ρ al 2700 kg m3 density cfrp ρ cfrp 1750 kg m3 35 decrease mass attained changing one material addition cfrp resistant aluminum decrease thickness part could done reducing weight part even although analysis considered reach assignment know weight saving worth higher cost new material mean analysis one becoming common necessary eigenvalue buckling analysis seen value load factor higher case aluminum validates previous result piece would prone instability 14p korba et al h e l n 10 2024 e27403 fig 8 stress field obtained fem analysis ansys lower part fig 9 strain field obtained fem analysis ansys upper part 5 discussion result obtained calculation presented paper using theory farrar help matlab computing environment rigid panel 91 cm width top bottom part wing box root following 8 stiffening integral beam height 7 2 cm width 7 6 mm together main panel 3 8 mm thick made machining aluminum piece average stress 487 5 mpa flow part mainly due compressive load critical scenario result validated using ansys mechanical apdl applying boundary condition load set match reality closely possible giving value yield strength every element mesh well critical buckling factor around 2 good value low enough prove material wasted safe enough buckling would never occur failure part due plastification similar analysis performed us cfrp instead aluminum meant reduction mass around 35 retaining good resistance compared aluminum part maximum stress buckling factor last fem study use cfrp instead aluminum limitation term definition composite work could done precise result could attained defining layer structure material apart limitation may also commented definition load boundary condition fem analysis designed whole wing structure would resulted precise solution something could improved future term numerical calculation design phase part assumption model aircraft could made using complex method however author paper believe one used appropriate even though always 15p korba et al h e l n 10 2024 e27403 fig 10 strain field obtained fem analysis ansys lower part fig 11 eigenvalue buckling analysis strain field due buckling critical load upper part room improvement minor problem arose calculation solution using matlab performed numerical instability detected paper met main goal set beginning research goal designing calculating structural part existing aircraft fulfilled furthermore comparing aluminum cfrp composite part acknowledged weight reduction new material imply traditional metallic material aerospace industry increasingly used today practice result may high value people want perform similar analysis even contribution science implies use novel material eventually mean reduction greenhouse effect gas always beneficial society higher volume scientific work kind sign sustainability objective form 2030 agenda way met comparison recent paper ref 39 similar problem studied using comparable technique use catia ansys computer based analysis similar wing structure methodology similar result term advantage composite material offer present lower strain level reduced weight 6 conclusion conclude variety tool engineering process used paper software like catia cad ansys fem analysis matlab calculation among others design via traditional numerical method fem analysis stiffened rigid panel wing box structure atr 72 aircraft main objective paper twofold one hand real problem 16p korba et al h e l n 10 2024 e27403 fig 12 eigenvalue buckling analysis strain field due buckling critical load lower part fig 13 stress field obtained fem ansys cfrp panel sought close possible engineer face professional career adapted part work academic field using hypothesis explained section hand author used wide range tool achieve robust solution lot time spent research document reference paper processing information development equation expression shown document could obtained also many interesting currently relevant topic world aeronautical engineering worked throughout paper new material design optimization main technique aircraft manufacturing paper main goal design real part real airplane kind work commonly performed enterprise airbus atr part analysis following point carefully considered maneuver envelop aircraft n v diagram calculation load according c 25 333 analytical design according farrar theory 3d design using catia v5 implementation comprehension fem analysis result firstly necessary information study assembled studied model established maneuver envelope according c 25 conceived critical situation structure wing would 17p korba et al h e l n 10 2024 e27403 fig 14 strain field obtained fem ansys cfrp panel fig 15 eigenvalue buckling analysis strain field due buckling critical load cfrp panel higher stress could defined parameter possible define different applying load according regulation main characteristic reference aircraft obtaining critical section wing embedment knowing structural load theory farrar design stiffened rigid panel applied geometry dimension structural part determined validation design performed use fem analysis ansys comparison designed aluminum panel equivalent one made cfrp composite material including weight reduction new material offer compared traditional one simple see composite material currently preferred metal design manufacturing modern aircraft although metal always used certain case completely replaced composite composite provide range design option metal technology alone match composite industry seems bright

future ahead expected improve ensure sustained long term growth always innovative newer exciting
product application horizon furthermore paper help get deeper insight topic aircraft structural calculation
design well validate growing trend use composite material comparison traditional material aluminum
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