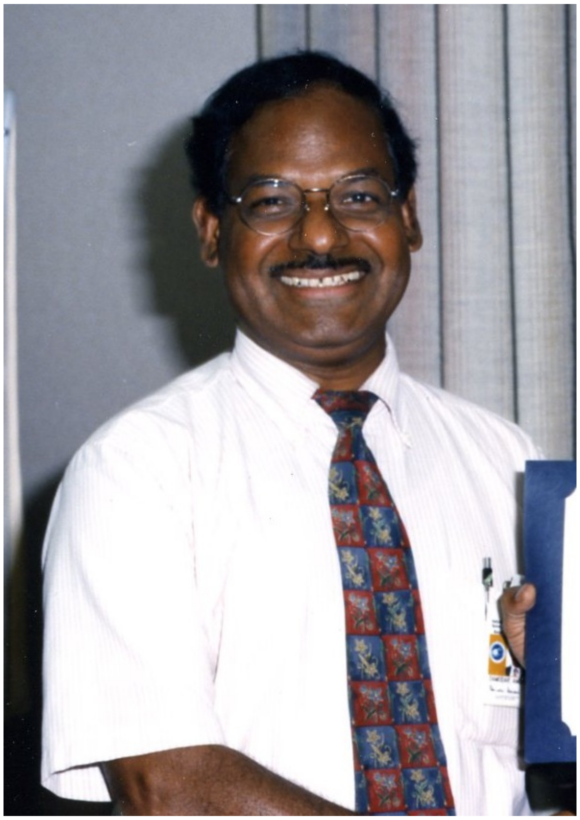

Dr. Damodar R. Ambur

See:
http://0-www.worldcat.org.novacat.nova.edu/identities/np-ambur,%20damodar%20r
http://65.54.113.26/Author/17902878/damodar-r-ambur
http://www.patentgenius.com/inventor/AmburDamodarR.html
http://patents.justia.com/inventor/DAMODARRAMBUR.html
http://openlibrary.org/authors/OL4666815A/Damodar_R._Ambur

NASA Langley Reearch Center, Hampton, Virginia

Selected Publications:

Rouse, M., and Ambur, D.R., "Damage Tolerance of a Geodesically Stiffened Structure Loaded in Axial Compression," Proceedings of the 35th AIAA/ASME/ ASCE/AHS/ASC Structures, Structural Dynamics and Materials Conference, AIAA Paper No. 94-1534, AIAA, Washington, D.C., 1994, pp. 1691-1698.

Ambur, D. A., Rouse, M., Starnes, J.H., and Shuart, M. J., "Facilities for Combined Loads Testing of Aircraft Structures to Satisfy Structural Technology Development Requirements," presented at the 5th Annual Advanced Composites Technology Conference, Seattle, WA, August 22-36, 1994.

Ambur, D. R., Cerro, J. A., and Dickson, J. D., "D- Box Fixture for Testing Stiffened Panels in Compression and Pressure," Journal of Aircraft, Vol. 32, No. 6, Nov.-Dec. 1995, pp. 1382-1389.

McGowan, D. M.; and Ambur, D. R.: Compression Response of a Sandwich Fuselage Keel Panel With and Without Damage. Presented at the Sixth NASA/DoD/ARPA Advanced Composites Technology Conference, Anaheim, CA, August 7-11, 1995.

Ambur, D. R.; Prasad, C. B.; and Waters, W. A.: An Internally Damped, Self-Arresting Dropped Weight Apparatus for Studying the Low-Speed Impact Response of Composite Structures. Journal of Experimental Mechanics, Vol. 33, No. 1, March 1995, pp. 64-69.

Ambur, D. R.; and Cruz, J. R.: Low-Speed Impact Response Characteristics of Composite Sandwich Plates. AIAA Paper 95-1460, April 1995.

Jaunky, N., Knight, N.F., Ambur, D.R., "Buckling of arbitrary quadrilateral anisotropic plates," AIAA Journal, 1995, 3, 938-44.

Jaunky, N., Knight, N.F., Ambur, D.R., "Buckling analysis of general triangular anisotropic plates using polynomials," AIAA Journal, 1995, 33, 2414-7.

Jaunky N., Knight N.F., Ambur D.R., "Formulation of An Improved Smeared Stiffener Theory of Buckling Analysis of Grid – Stiffened Composite Panels," NASA technical Memorandum 110162, June 1995.

N. Jaunky (1), N.F. Knight Jr (1) and D.R. Ambur (2)
(1) Old Dominion University, Dept Aerospace Engineering, Norfolk, VA 23529-0247, USA
(2) NASA Langley Research Center, Hampton, VA 23681-0001, USA
"Formulation of an improved smeared stiffener theory for buckling analysis of grid-stiffened composite panels", Composites Part B: Engineering, Vol. 27, No. 5, 1996, pp. 519-526, doi:10.1016/1359-8368(96)00032-7
ABSTRACT: An improved smeared stiffener theory for stiffened panels is presented that includes skin-stiffener interaction effects. The neutral surface profile of the skin-stiffener combination is developed analytically using the minimum potential energy principle and statics conditions. The skin-stiffener interaction is accounted for by computing the bending and coupling stiffness due to the stiffener and the skin in the skin-stiffener region about a shift in the neutral axis at the stiffener. Buckling load results for axially stiffened, orthogrid, and general grid-stiffened panels are obtained using the smeared stiffness combined with a Rayleigh-Ritz method and are compared with results from detailed finite element analyses.

Ambur, D. R., and Starnes, J. H., Jr., "Nonlinear Response and Damage-Initiation Characteristics of Curved Composite Plates Subjected to Low-Speed Impact," AIAA Paper 97-1058, April 1997.

David M. McGowan and Damodar R. Ambur (Langley Research Center, Hampton, Virginia), "Damage-Tolerance Characteristics of Composite Fuselage Sandwich Structures With Thick Facesheets", NASA Technical Memorandum 110303, February 1997
ABSTRACT: Damage tolerance characteristics and results from experimental and analytical studies of a composite fuselage keel sandwich structure subjected to low-speed impact damage and discrete-source damage are presented. The test specimens are constructed from graphite-epoxy skins bonded to a honeycomb core, and they are representative of a highly loaded fuselage keel structure. Results of compression-after-impact (CAI) and notch-length sensitivity studies of 5-in.-wide by 10-in-long specimens are presented. A correlation between low-speed-impact dent depth, the associated damage area, and residual strength for different impact-energy

levels is described; and a comparison of the strength for undamaged and damaged specimens with different notch-length-to-specimen-width ratios is presented. Surface strains in the facesheets of the undamaged specimens as well as surface strains that illustrate the load redistribution around the notch sites in the notched specimens are presented and compared with results from finite element analyses. Reductions in strength of as much as 53.1 percent for the impacted specimens and 64.7 percent for the notched specimens are observed.

N. Jaunky (1), N.F. Knight Jr (1) and D.R. Ambur (2)
(1) Old Dominion University, Dept Aerospace Engineering, Norfolk, VA 23529-0247, USA
(2) NASA Langley Research Center, Hampton, VA 23681-0001, USA
"Optimal design of general stiffened composite circular cylinders for global buckling with strength constraints", Composite Structures, Vol. 41, Nos. 3-4, March-April 1998, pp. 243-252,
doi:10.1016/S0263-8223(98)00020-8
ABSTRACT: A design strategy for optimal design of composite grid-stiffened cylinders subjected to global and local buckling constraints and strength constraints was developed using a discrete optimizer based on a genetic algorithm. An improved smeared stiffener theory was used for the global analysis. Local buckling of skin segments were assessed using a Rayleigh-Ritz method that accounts for material anisotropy. The local buckling of stiffener segments were also assessed. Constraints on the axial membrane strain in the skin and stiffener segments were imposed to include strength criteria in the grid-stiffened cylinder design. Design variables used in this study were the axial and transverse stiffener spacings, stiffener height and thickness, skin laminate stacking sequence and stiffening configuration, where stiffening configuration is a design variable that indicates the combination of axial, transverse and diagonal stiffener in the grid-stiffened cylinder. The design optimization process was adapted to identify the best suited stiffening configurations and stiffener spacings for grid-stiffened composite cylinder with the length and radius of the cylinder, the design in-plane loads and material properties as inputs. The effect of having axial membrane strain constraints in the skin and stiffener segments in the optimization process is also studied for selected stiffening configurations.

Navin Jaunky (1), Norman F. Knight, Jr. (1), and Damodar R. Ambur (2)
(1) Department of Aerospace Engineering, Old Dominion University, Norfolk, VA 23529-0247, USA
(2) Structural Mechanics Branch, NASA Langley Research Center, Hampton, VA 23681-0001, USA
"Optimal design of grid-stiffened composite panels using global and local buckling analyses", Journal of Aircraft, 1998, Vol. 35, No. 3, pp. 478-486, presented at AIAA 37th SDM Conference, Salt Lake City, UT
ABSTRACT: A design strategy for optimal design of composite grid-stiffened panels subjected to global and local buckling constraints is developed using a discrete optimizer. An improved smeared stiffener theory is used for the global buckling analysis. Local buckling of skin segments is assessed using a Rayleigh-Ritz method that accounts for material anisotropy and transverse shear flexibility. The local buckling of stiffener segments is also assessed. Design variables are the axial and transverse stiffener spacing, stiffener height and thickness, skin laminate, and stiffening configuration, where the stiffening configuration is herein defined as a design variable that indicates the combination of axial, transverse, and diagonal stiffeners in the stiffened panel. The design optimization process is adapted to identify the lightest-weight stiffening configuration and stiffener spacing for grid-stiffened composite panels given the overall panel dimensions, in-plane design loads, material properties, and boundary conditions of the grid-stiffened panel.

Navin Jaunky (1), Norman F. Knight, Jr. (1), and Damodar R. Ambur (2)
(1) Department of Aerospace Engineering, Old Dominion University, Norfolk, VA 23529-0247, USA
(2) Structural Mechanics Branch, NASA Langley Research Center, Hampton, VA 23681-0001, USA
"Buckling analysis of anisotropic variable-curvature panels and shells", Composite Structures, Vol. 43, No. 4,

December 1998, pp. 321-329, doi:10.1016/S0263-8223(98)00118-4
ABSTRACT: A buckling formulation for anisotropic variable-curvature panels is presented in this paper. The variable-curvature panel is assumed to consist of two or more panels of constant curvature where each panel may have a different curvature. Bezier functions are used as Ritz functions. Displacement (C0), and slope (C1) continuities between segments are imposed by manipulation of the Bezier control points. A first-order shear-deformation theory is used in the buckling formulation. Results obtained from the present formulation are compared with those from finite element simulations and are found to be in good agreement.

Young, R. D., Rouse, M., Ambur, D. R., and Starnes, J. H., Jr., "Residual Strength Pressure Tests and Nonlinear Analyses of Stringer- and Frame-Stiffened Aluminum Fuselage Panels with Longitudinal Cracks," Proceedings of the Second Joint NASA/FAA/DOD Conference on Aging Aircraft, Williamsburg, VA, August 31-September 3, 1998. NASA/CP-1999-208982/Part 1, January 1999, pp. 408-426.
ABSTRACT: The results of residual strength pressure tests and nonlinear analyses of stringer- and frame-stiffened aluminum fuselage panels with longitudinal cracks are presented. Two types of damage are considered: a longitudinal crack located midway between stringers, and a longitudinal crack adjacent to a stringer and along a row of fasteners in a lap joint that has multiple-site damage (MSD). In both cases, the longitudinal crack is centered on a severed frame. The panels are subjected to internal pressure plus axial tension loads. The axial tension loads are equivalent to a bulkhead pressure load. Nonlinear elastic-plastic residual strength analyses of the fuselage panels are conducted using a finite element program and the crack-tip-opening-angle (CTOA) fracture criterion. Predicted crack growth and residual strength results from nonlinear analyses of the stiffened fuselage panels are compared with experimental measurements and observations. Both the test and analysis results indicate that the presence of MSD affects crack growth stability and reduces the residual strength of stiffened fuselage shells with long cracks.

James H. Starnes, Jr., Damodar R. Ambur, Richard D. Young, and Charles E. Harris (NASA Langley Research Center, Hampton, VA 23681-2199, USA), “Experimental verification of the analytical methodology to predict the residual strength of metallic shell structure”, Society for Experimental….(1999 –cs.odu.edu)
ABSTRACT: Experimental and analysis results for a curved, stiffened aluminum fuselage panel tested in a combined loads test machine with combined internal pressure, axial compression, and torsonal shear loads are described. The experimental and analytical strain results for the panel with and without discrete source damage are presented. The effect of notch tip geometry on crack growth predictions is addressed. The crack growth trajectory predictions for the panel are presented for the applied loading conditions at failure. (Unfortunately, the pdf file does not permit cutting and pasting the 10 references listed at the end of the 4-page file.)

Ambur, D. R., Rouse, M., Young, R. D., and Perez-Ramos, C., "Evaluation of an Aluminum Panel with Discrete-Source Damage and Subjected to Combined Loading Conditions," Proceedings or the 40th AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics, and Materials Conference, St. Louis, MO, April 12-15, 1999. AIAA Paper AIAA-99-1439, 1999.

Damodar R. Ambur (1) and Navin Jaunky (2)
(1) NASA Langley Research Center, Hampton, VA 23681-2199, USA
(2) Eagle Aeronautics Inc., Newport News, Hampton, VA 23606, USA
“Optimal design of grid-stiffened panels and shells with variable curvature”, Composite Structures, Vol. 52, No. 2, May 2001, pp. 173-180, doi:10.1016/S0263-8223(00)00165-3
ABSTRACT: A design strategy for optimal design of composite grid-stiffened structures with variable curvature subjected to global and local buckling constraints is developed using a discrete optimizer. An

improved smeared stiffener theory is used for the global buckling analysis. Local buckling of skin segments is assessed using a Rayleigh–Ritz method that accounts for material anisotropy and transverse shear flexibility. The local buckling of stiffener segments is also assessed. Design variables are the axial and transverse stiffener spacing, stiffener height and thickness, skin laminate, and stiffening configuration. Stiffening configuration is herein defined as a design variable that indicates the combination of axial, transverse and diagonal stiffeners in the stiffened panel. The design optimization process is adapted to identify the lightest-weight stiffening configuration and stiffener spacing for grid-stiffened composite panels given the overall panel dimensions, in-plane design loads, material properties, and boundary conditions of the grid-stiffened panel or shell.

Camanho P.P., Dávila C.G., Ambur D.R., 2001, Numerical Simulation of Delamination Growth in Composite Materials, NASA TP-2001-211041.

Jaunky, N., Ambur, D. R., Davila, C. G., and Hilburger, M. W., "Progressive Failure Studies of Composite Panels with and without Cutouts," NASA/CR-2001-211223, September 2001.

Vinay, K. G., Jaunky, N., Johnson, E. R., and Ambur, D. R., "Intralaminar and Interlaminar Progressive Failure Analyses of Composite Panels with Circular Cutouts," Proceedings of the AIAA/ASME/ASCE/AHS/ASC 43rd Structures, Structural Dynamics, and Materials Conference, Denver, CO. AIAA Paper No. 2002-1745, 2002.

Reeder J., Kyongchan S., Chunchu P.B., Ambur D.R., 2002, Postbuckling and growth of delaminations in composite plates subjected to axial compression, in: Proceeding of the 43rd AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics, and Materials Conference, Denver, Colorado, Paper No. AIAA-2002-1746.

Donald J. Baker (1), Damodar R. Ambur (2), Jack Fudge (3), and Christos Kassapoglou (4)
(1) Vehicle Technology Directorate – ARL, NASA Langley Research Center, Hampton, VA 23681
(2) NASA Langley Research Center, Hampton, VA 23681
(3) Hexcel Corporate Research and Technology, 11711 Dublin Blvd, Dublin, CA 94583
(4) Sikorsky Aircraft, 6900 Main St., Stratford, CT 06615
"Optimal Design and Damage Tolerance Verification of an Isogrid Structure for Helicopter Application", 44th AIAA Structures, Structural Dynamics and Materials Conference, AIAA-2003-xxxx, 2003
ABSTRACT: A composite isogrid panel design for application to a rotorcraft fuselage is presented. An optimum panel design for the lower fuselage of the rotorcraft that is subjected to combined in-plane compression and shear loads was generated using a design tool that utilizes a smeared-stiffener theory in conjunction with a genetic algorithm. A design feature was introduced along the edges of the panel that facilitates introduction of loads into the isogrid panel without producing undesirable local bending gradients. A low-cost manufacturing method for the isogrid panel that incorporates these design details is also presented. Axial compression tests were conducted on the undamaged and low-speed impact damaged panels to demonstrate the damage tolerance of this isogrid panel. A combined loading test fixture was designed and utilized that allowed simultaneous application of compression and shear loads to the test specimen. Results from finite element analyses are presented for the isogrid panel designs and these results are compared with experimental results. This study illustrates the isogrid concept to be a viable candidate for application to the helicopter lower fuselage structure.

Damodar R. Ambur (1), Navin Jaunky (2) and Mark W. Hilburger (1)
(1) Mechanics and Durability Branch, NASA Langley Research Center, Hampton, VA 23681-2199, USA
(2) National Institute of Aerospace, Hampton, VA 23666, USA

“Progressive failure studies of stiffened panels subjected to shear loading”, Composite Structures, Vol. 65, No.2, August 2004, pp. 129-142, doi:10.1016/S0263-8223(03)00153-3
ABSTRACT: Experimental and analytical results are presented for progressive failure of stiffened composite panels with and without a notch and subjected to in-plane shear loading well into the postbuckling regime. Initial geometric imperfections are included in the finite element models. Ply damage modes such as matrix cracking, fiber-matrix shear, and fiber failure are modeled by degrading the material properties. Experimental results from the test include strain full-field data from a video image correlation system in addition to other strain and displacement measurements. Results from nonlinear finite element analyses are compared with experimental data. Good agreement between experimental data and numerical results is observed for the stitched stiffened composite panels studied.

Damodar R. Ambur (1) Navin Jaunky (2), Mark Hilburger (1) and Carlos G. Dávila (3)
(1) Mechanics and Durability Branch, NASA Langley Research Center, Hampton, VA 23681-2199, USA
(2) National Institute of Aerospace, Hampton, VA 23666, USA
(3) Analytical and Computational Methods Branch, NASA Langley Research Center, Hampton, VA 23681-2199, USA
“Progressive failure analyses of compression-loaded composite curved panels with and without cutouts”, Composite Structures, Vol. 65, No. 2, August 2004, pp. 143-155, doi:10.1016/S0263-8223(03)00184-3
ABSTRACT: Progressive failure analyses results are presented for composite curved panels with and without a circular cutout and subjected to axial compression loading well into their postbuckling regime. Ply damage modes such as matrix cracking, fiber-matrix shear, and fiber failure are modeled by degrading the material properties. Results from finite element analyses are compared with experimental data. Good agreement between experimental data and numerical results are observed for most part of the loading range for the structural configurations considered. Modeling of initial geometric imperfections may be required to obtain accurate analysis results depending on the ratio of the cutout width to panel width.

Vinay K. Goyal (1), Navin R. Jaunky (2), Eric R. Johnson (1) and Damodar R. Ambur (3)
(1) Department of Aerospace and Ocean Engineering, Virginia Polytechnic Institute and State University, 215 Randolph Hall, MS 0203, Blacksburg, VA 24061-0203, USA
(2) NASA Langley Research Center, MS 190, 8 West Taylor St., Hampton, VA 23681-2199, USA
(3) NASA Langley Research Center, MS 188E, 2 West Reid St., Hampton, VA 23681-2199, USA
“Intralaminar and interlaminar progressive failure analyses of composite panels with circular cutouts”, Composite Structures, Vol. 64, No. 1, April 2004, pp. 91-105, doi:10.1016/S0263-8223(03)00217-4
ABSTRACT: A progressive failure methodology is developed and demonstrated to simulate the initiation and material degradation of a laminated panel due to intralaminar and interlaminar failures. Initiation of intralaminar failure can be by a matrix-cracking mode, a fiber-matrix shear mode, and a fiber failure mode. Subsequent material degradation is modeled using damage parameters for each mode to selectively reduce lamina material properties. The interlaminar failure mechanism such as delamination is simulated by positioning interface elements between adjacent sublaminates. A nonlinear constitutive law is postulated for the interface element that accounts for a multi-axial stress criteria to detect the initiation of delamination, a mixed-mode fracture criteria for delamination progression, and a damage parameter to prevent restoration of a previous cohesive state. The methodology is validated using experimental data available in the literature on the response and failure of quasi-isotropic panels with centrally located circular cutouts loaded into the postbuckling regime. Very good agreement between the progressive failure analyses and the experimental results is achieved if the failure analysis includes the interaction of intralaminar and interlaminar failures.

John T. Wang, Clarence C. Poe, Jr., Damodar R. Ambur and David W. Sleight (NASA Langley Research Center, Hampton, VA 23681, United States), “Residual strength prediction of damaged composite fuselage panel with R-curve method”, Composites Science and Technology, Vol. 66, No. 14, November 2006, pp. 2557-2565, Special Issue in Honour of Professor C.T. Sun, doi:10.1016/j.compscitech.2006.01.011
ABSTRACT: This study applies the crack growth resistance curve (R-curve) method to predict the residual strength of a composite fuselage panel with discrete source damage. The R-curve is constructed from the energy release rates computed from cracked tensile plate test specimens at their failure loads. To predict the residual strength of a curved fuselage panel with discrete source damage using the R-curve method, G-curves of the energy release rate of the damaged fuselage panel need to be established. These G-curves are generated for various fuselage pressures over a range of crack lengths. Since this study found that the membrane stiffening effect could significantly reduce the energy release rate at the crack tip of a pressurized fuselage, geometrically nonlinear behavior was considered in the calculations. The residual strength of the damaged fuselage panel was determined by comparing the energy release rates with the “allowable-like” R-curve. The correlation between the crack growth predictions and the damage progression results obtained from experiments is very good. Therefore, the R-curve method has the potential to be a practical engineering method for predicting the residual strength of damaged fuselage panels.